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The Copernicus Project

Proposal for AAE 241
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Executive Summary

The Copernicus spacecraft, to be launched on May 4, 2009, is designed for scientific exploration of the planet Pluto. The main objectives of this exploration is to accurately determine the mass, density, and composition of the two bodies in the Pluto-Charon system. A further goal of the exploration is to obtain precise images of the system.

The spacecraft will be designed for three axis stability control. It will use the latest technological advances to optimize the performance, reliability, and cost of the spacecraft. Due to the long duration of the mission, nominally 12.6 years, the spacecraft will be powered by a long lasting radioactive power source. Although this type of power may have some environmental drawbacks, currently it is the only available source that is suitable for this mission.

The planned trajectory provides flybys of Jupiter and Saturn. These flybys provide an opportunity for scientific study of these planets in addition to Pluto. The information obtained on these flybys will suppliment the data obtained by the Voyager and Galileo missions.

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ACRONYMS

AACS: Attitude, Articulation, and Control System

ASTROS: Advanced Star/Target Reference Optical System

CCC: Command, Control, and Communication

CCD: Charge-Coupled Device

COM: Center of Mass

FDC: Flight Data Subsystem

FORS: Fiber Optic Rotation System

HECP: High Energy Charged Particle

HDA: Harmonic Drive Actuator

ISS: Imaging Science Subsystem

IIS: Infrared Interferometer Spectrometer

kbps: Kilobits Per Second

LECP: Low Energy Charged Particle

MITG: Modular Isotopic Thermoelectric Generator

MMPC: Mission Management, Planning and Costing

NAC: Narrow Angle Camera

PP: Plasma Particle

ACRONYMS (cont.)

RCS: Reaction Control System

RTG: Radioisotope Thermoelectric Generator

TCM: Trajectory Correction Maneuver

S/C: Spacecraft

UVS: Ultraviolet Spectrometer

ΔV: Delta Velocity

WAC: Wide Angle Camera

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INTRODUCTION

Introduction

The Copernicus Project proposal describes a Phase A design for an unmanned mission to Plutoian space for the purpose of scientific inquiry. This paper proposes that the spacecraft be designed, built, and launched in an effort to increase our knowledge of the outer Solar System and, in particular, the Pluto-Charon system. Thus far Pluto is the only planet that has not been visited and investigated by a space probe.

In order to insure an efficient and successful spacecraft and to bring focus to the overall mission, the Copernicus Project proposal will adhere to various mission guidelines and design requirements. The following is a list of the spacecraft primary design requirements.

- The spacecraft must be unmanned.
- The spacecraft must be launched in the first decade of the twenty-first century.
- The spacecraft should be reliable and easy to operate.
- The spacecraft should use off the shelf hardware whenever possible.
- The spacecraft should not use materials or techniques expected to be available after 1999.
- On-orbit assembly should be identified and minimized.
- The launch vehicle to be used must be identified and the interfaces must be compatible.
- The design must be flexible enough to perform several possible missions.
- The design lifetime must be sufficient to carry out the mission plus a reasonable safety margin.
- The spacecraft must use the latest advances in artificial intelligence.
- The design will stress reliability, simplicity, and low cost.
- Four spacecraft will be built.
- Give an implementation plan for production of a final product.

In an effort to adhere to these design requirements and to create an original and unique proposal, the project is divided into six subsystems. Each subsystem is responsible for the design of a specific area of the mission and the identification of any interactions between the subsystems. An additional responsibility of each subsystem is to optimize the performance, weight, and cost of the individual subsystem in order to optimize those parameters for the overall mission design. A list of the subsystems and their major responsibilities follows.

Structures: Responsible for material selection for major spacecraft components, component placement, thermal control for the spacecraft, calculation of spacecraft inertia and center of mass, and production planning.

Mission Management, Planning and Costing: Responsible for mission type selection, trajectory planning, launch vehicle selection, mission timeline planning, and mission costing.

Command, Control, and Communication: Responsible for the quality of the spacecraft computers, the information storage capability of the spacecraft, and insuring that the communication link with the spacecraft is available at all times.

Power and Propulsion: Responsible for providing adequate power supplies to the spacecraft components during all mission phases, propellent selection, and propulsion unit selection and sizing.

Science Instrumentation: Responsible for planning the mission science objectives, planning the mission science timeline, and scientific instrument selection.

Attitude and Articulation Control: Responsible for attitude control of the spacecraft, maintaining antenna pointing requirements, trajectory correction maneuvers, science maneuvers, and stability throughout the mission.

STRUCTURES

Structure Subsystem: Introduction

The responsibility of the structure subsystem for the Pluto project is to stress reliability, simplicity, and low cost in the areas of material selection, thermal control, and overall spacecraft design. Subjects to consider in fulfilling this responsibility are minimizing the spacecraft weight, minimizing the amount of on-orbit assembly of the spacecraft, and insuring a design lifetime sufficient to carry out the mission plus a safety margin. An additional responsibility is to provide an implementation plan for production of the final product. To meet these requirements the structure subsystem is divided into the following areas of consideration:

1. Drawings of the spacecraft
2. Placement of the spacecraft components to meet requirements
3. Mass and inertia of the spacecraft
4. Material selection
5. Thermal control
6. Launch vehicle compatibility
7. On-orbit assembly
8. Production of the final product
9. Interactions with other subsystems

Drawings of the Spacecraft

Drawings of the spacecraft are provided to enhance the reader's conception of the component placement and the overall spacecraft design. The major spacecraft components included in the drawings are the bus, propellent tank, main propulsive unit, three boom extensions, RTG, scan platform, and antenna unit. Major spacecraft dimensions are provided in meters. Two views of the spacecraft will provide the reader with a clear idea of the spacecraft configuration.

The spacecraft axis was selected such that the origin coincides with the geometric center of the bus. The Z-axis points out along the antenna mast, the X-axis points out along the magnetometer boom, and the Y-axis points out along the science boom to form a standard righthanded coordinate system.

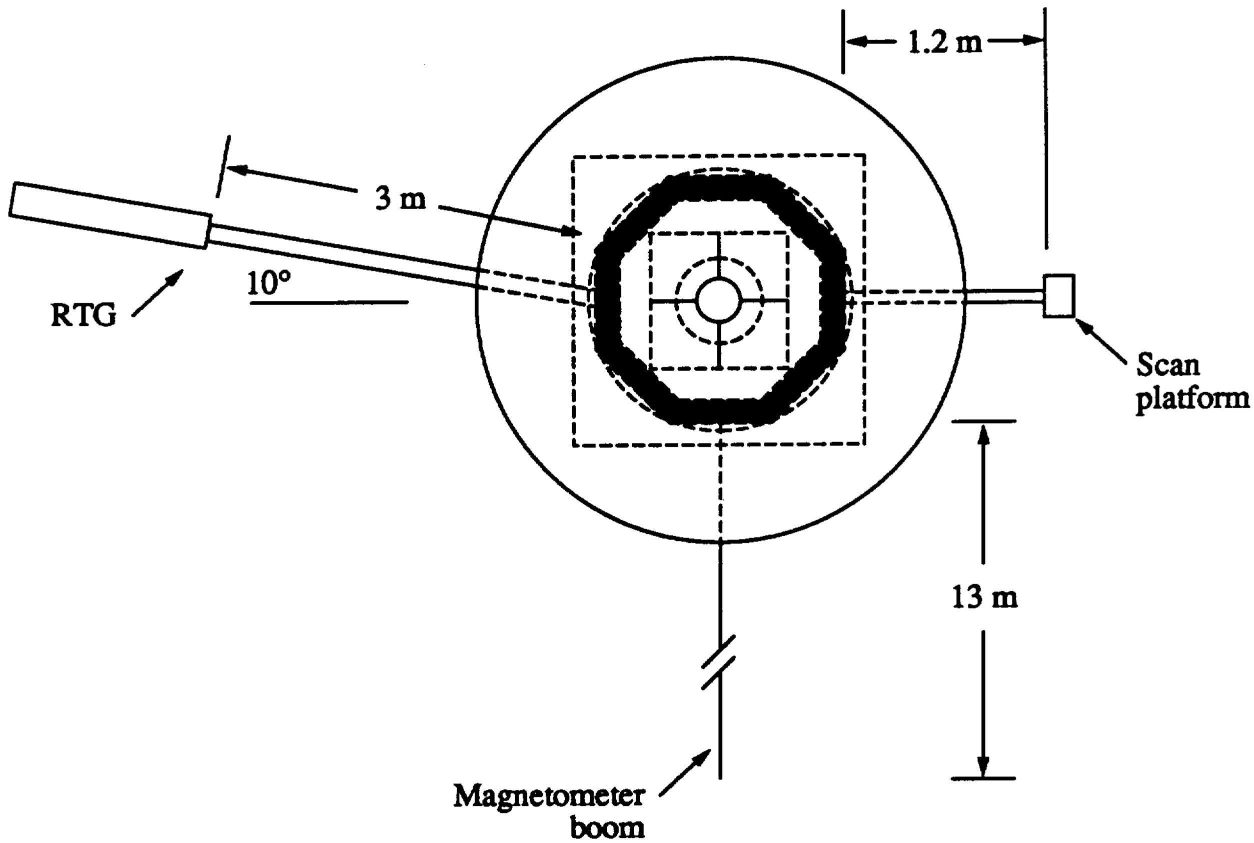

Figure 1A

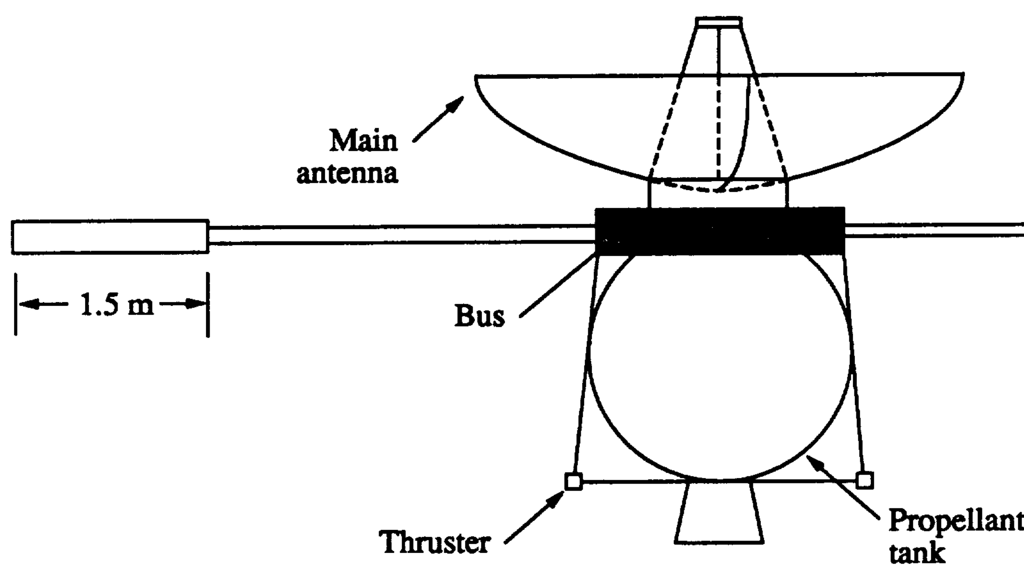

Figure 1B

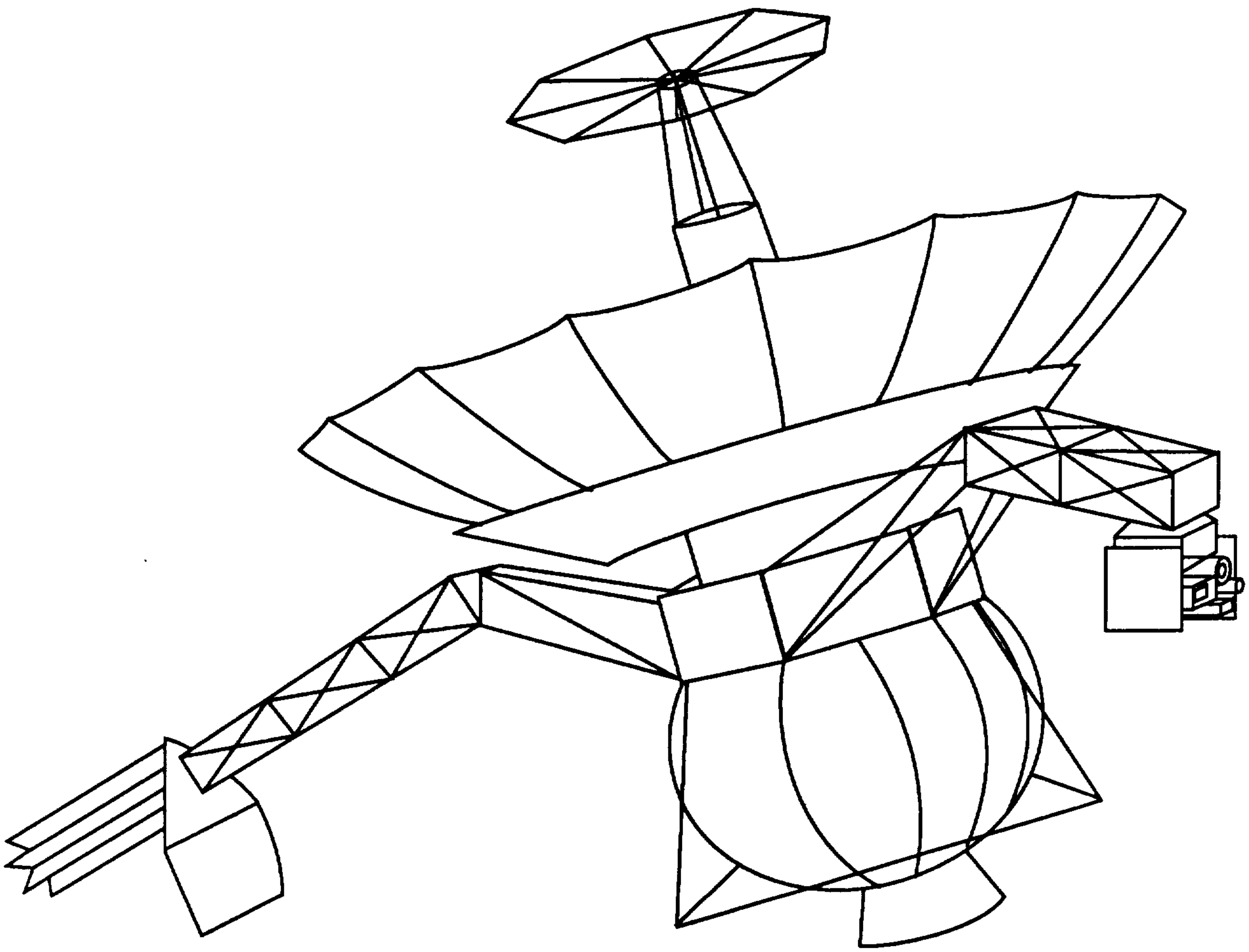

Figure 1C
Copernicus

Placement of Spacecraft Components to Meet Requirements

The driving requirements of this mission are reliability, simplicity, and low cost. From these primary requirements come several derived requirements that influence the positioning of the individual spacecraft components. These derived requirements are: radiation protection for all spacecraft components, the scientific instruments must have a clear field of view, no component that would disrupt communications should be placed near the antenna, the main propulsive unit should create a line of force through the spacecraft center of mass (COM), the components in the bus must be compact to aid in thermal control of the bus, and the magnetometer must be isolated from the interference of other spacecraft components.

The radioisotope thermoelectric generator (RTG) emits radiation that is damaging to other spacecraft components. To minimize this radiation damage the RTG should be placed as far as possible from all other spacecraft components. The distance that the RTG can be placed from the main spacecraft structure is limited by the strength of the RTG boom and spacecraft COM considerations. In an effort to keep the COM on the Z-axis for attitude control simplification, the RTG will be placed approximately 3 meters from the bus and at an angle of 10° off the negative Y-axis in the XY plane. For additional radiation protection, a metal shield will be placed at the end of the RTG boom between the RTG and the main body of the spacecraft.

The placement of the scan platform must provide an adequate viewing range for the scientific instruments. This is one of the most important placement requirements. If this requirement is not met, then the success of the mission will be limited. The scan platform will be placed on a 1.2m boom that extends 0.3m beyond the rim of the antenna. This placement was achieved by a tradeoff of field of view and the previously mentioned COM restriction. Also, the scan platform will be placed such that the spacecraft main body is between the platform and the RTG for redundant radiation protection.

Communication is also essential for the success of the mission. In an effort to increase reliability, any components that are placed within the antenna's field of transmission or reception should be transparent to the antenna. A better placement technique is to leave this area of the antenna free of any components at all. The second technique is simpler than the use of antenna transparent components and it was therefore selected.

To prevent any unwanted torques while the main propulsive unit is in operation, the unit will be oriented so that its line of force coincides with the Z-axis of the spacecraft. As previously stated, all spacecraft components will be positioned so that the spacecraft COM lies on the Z-axis.

The components housed within the bus will be placed in a compact manner. This technique reduces the overall volume of the bus and therefore the volume that requires the most thermal control. The method in which this reduction in thermal control cost is achieved will be discussed in a later section. The compact placement of the components within the bus helps to reduce the mission cost and thereby helps to fulfill a primary mission requirement.

A final placement requirement involves the magnetometer. The magnetometer must be placed as far as possible from the other spacecraft components to reduce the amount of interference encountered from the other components. Again, the distance that the magnetometer can be placed from the main spacecraft assembly is restricted by COM placement, the strength of the magnetometer boom, and the cost per unit length of the boom.

Mass and Inertia of the Spacecraft

Mass estimates are provided only for the major components of the spacecraft. The following mass estimates are derived from other subsystem requirements, considerations, and calculations.

Table 1-A. Component Masses

Component:	Mass (kg):
Antenna	5
Antenna Base	45
Bus (includes Structure, Thermal Control, and Cabling)	270
Computers	100
Science Platform	111
Science Boom	35
Magnetometer Boom	5
RTG Boom	5
RTG	60
Propulsion Unit Tank	120
Propellent	1500-2000

Total spacecraft mass (unfuelled): 756 kg

The inertia of the spacecraft is calculated with the aid of a computer program. The inertia and COM of individual components are calculated by hand and these results are input into the program which calculates the overall spacecraft inertia and COM. The individual components are idealized into geometric shapes to simplify the inertia calculations as described in the structure section appendix.

In an effort to simplify the placement of the attitude thrusters and the main propulsive unit, the spacecraft COM should lie on the Z-axis and as close to the geometric center of the bus as possible. Several trials were performed in which the lengths of the science boom and the RTG boom were varied. An additional variable was the angle between the RTG boom and the negative Y-axis in the XY plane. On the ninth trial the spacecraft COM was within approximately 0.5 cm of the Z-axis and approximately 11 cm below the geometric center of the bus. This result was obtained with the unfuelled configuration. This position of the COM is adequate for the purposes of this preliminary design report.

The inertia and COM for the unfuelled configuration of the ninth trial is:

Table 1-B. Inertia of Copernicus

Body Name: Copernicus

Inertia Matrix:	2334.0560	-155.8637	-.3384
	-155.8637	700.0375	-.4187
	-.3384	-.4187	2724.9290
Body COM:	.0039	.0048	-.1147
Body Mass:	756.0000		
Number of Bodies:	9		
Principal Inertia Matrix:			
	2348.7910	.0000	.0000
	.0000	685.3029	.0000
	.0000	.0000	2724.9290
Eigenvector Matrix:			
	.9956	.0941	-.0008
	-.0941	.9956	-.0001
	.0008	.0002	1.0000

Material Selection

There are several factors to consider in the material selection process. First, to comply with the primary mission requirements, the materials should be light weight, low cost, and should reliably fulfill their design function. Additional material selection considerations include: radiation damage threshold, contamination resistance, thermal characteristics, strength, stiffness, and general structural qualities. These characteristics must be carefully considered when selecting materials for the spacecraft.

The main purpose of this mission is scientific exploration of Plutoian space. Therefore it is essential that the science instruments be kept operational. Contaminants such as atomic oxygen, outgassed materials, and cosmic debris will accumulate on instrument surfaces over time and impede their performance. Since the mission is of such long duration, contamination protection of the instruments must be a major factor in material selection.

One method of protecting the instruments is by installing a permanent cover which is transparent to the instrument. A second means of protection is the retractable cover design. This design involves moving parts and should be used only where absolutely necessary in an effort to enhance simplicity. If the retractable cover should fail to open, then the success of the mission would be limited.

A redundant method of radiation protection is achieved by placing a metal shield between the RTG and the main spacecraft body. An aluminum shield was selected due to its low cost, light weight, and high radiation damage threshold. Composites should not be used for this application due to their susceptibility to radiation damage.[1]

An application that is well suited for composite materials is the main antenna. The composite can be easily molded into the unique antenna shape. Also, because of their low coefficient of thermal expansion and high thermal conductivity, composites can be used in systems which require high thermostructural stability like the antenna dish.[2]

For the main structural supports of the spacecraft, titanium should be used where strength and thermal stability is important. Graphite epoxy can be used in secondary truss supports and stiffeners. Aluminum is attractive for its strength to weight ratio, availability, low cost, and because it is space proven.

In situations where the stiffness of a structural member is crucial, beryllium will be used instead of titanium. The modulus of elasticity of beryllium is 2.5 times that of titanium and beryllium is considerably lighter in weight. Although beryllium is more costly to produce than titanium, beryllium's weight savings makes it less costly than titanium to put into orbit.[3]

The use of cosmic ray resistant parts for the computer's electronic components will depend on their performance on the Galileo probe.

Sandia National Laboratories developed these components in an effort to reduce the number of single event upsets in the computer's logic and memory.[4] If these components prove successful in reducing the number of computer sequence failures and if the cost is reasonable, then cosmic ray resistant parts should be incorporated into the Pluto probe's computer for enhanced reliability and performance.

Thermal Control

Thermal control will insure that each part of the spacecraft will have an appropriate thermal environment for operation. The different components will require significantly different thermal environments so that temperature gradients will be present throughout the spacecraft. Thermal control will be further complicated by the changing thermal surroundings as the mission progresses. The three most significant phases are: thermal control on Earth and during launch, thermal control in space close to the sun (0.5-3 AU), and thermal control in the outer solar system.

The problem of thermal control is best solved by examining the major components of the spacecraft.

Bus: The major considerations for thermal control of the bus are isolation from solar heating, internal coupling to prevent temperature gradients, and heat rejection at external bus surfaces.[5] A very cost and weight efficient method of preventing solar heating in the bus is by the use of multilayer insulation blankets. This passive thermal control technique makes use of the unique insulation properties of multilayer designs. Redundancy is also achieved by using multiple layers. The material is selected for minimum heat transmission except for a few layers of very tough material such as Teflon for micrometeoroid protection. The internal coupling is achieved by positioning the internal components as compactly as possible. This technique produces a smaller volume to be thermally controlled and thus the cost of thermal control is reduced. This helps meet the low cost mission requirement. The heat rejection phase is accomplished by transporting waste heat from the interior of the bus to the external bus surfaces via a system of thermal switches. At points along the external bus surface are heat radiators in the form of thermostatically controlled louvers. There will be several of these louver sites for redundancy.

RTG: The RTG produces large amounts of heat to be converted into electrical power for the spacecraft. Due to radiation protection considerations, the RTG is relatively isolated from all other spacecraft components. This isolation also serves as an excellent thermal barrier between the RTG and the spacecraft. Any waste heat produced by the RTG can easily be rejected into space by an array of metal fins that act as passive heat radiators.

Thrusters: The hydrazine thrusters will be thermally controlled by strip heaters constructed of printed heating element circuits imbedded in Kapton film.[6] These heaters will be placed on the catalyst bed of the thrusters to produce temperatures well above 500K. The hydrazine fuel lines will be heated by wrapping wire heating elements around the fuel line.

Science Instruments: The great design flexibility of the printed circuit strip heaters mentioned above will allow them to provide thermal control to the science instruments as well as the thrusters. The design temperature for the science instruments is approximately 140K which is well within the thermal range of the strip heaters. To help meet the requirement of redundancy in all spacecraft systems, two strip heaters will be provided for every science instrument and every thruster. This increase in thermal control should not produce a drastic increase in overall spacecraft weight due to the very small mass of the strip heaters.

Launch Vehicle Compatibility

The spacecraft must be compatible with the selected launch vehicle. This means that all interfaces between the spacecraft and the launch vehicle must be selected for compatibility. Also, the dimensions of the spacecraft cannot exceed the payload dimensions of the chosen launch vehicle.

The launch configuration is approximately cylindrical in shape. The approximate dimensions of this cylinder are: width=3.7 m and length = 4.5 m. The width c[illegible]ponds to the antenna diameter. The antenna is a one pie[illegible] and is similar in design to the antenna used on th[illegible]t. The fact that the antenna is not foldab[illegible] inch vehicle selection at all. The leng[illegible] ate any problems with the payload a[illegible]

what about interface?

The [illegible] launch vehicle must not create any [illegible]e interfaces include thermal contr[illegible] assistance in thermal control while [illegible]nch pad and while the launch ve[illegible] ther interfaces may include power [illegible] interfaces, and the mountings that h[illegible] position inside the launch vehicle. All these [illegible] be compatible.

ORIGINAL PAGE IS
OF POOR QUALITY

On-Orbit Assembly

The Copernicus will be a complete unit in its launch configuration. No assembly will be required while in orbit. However, there will be several boom deployments and general transformations of the spacecraft from its launch configuration to its cruise configuration while in orbit. Separation from the launch vehicle and upper stage will be achieved by pyrotechnic methods such as explosive bolts.

These deployments will be made while the spacecraft is in LEO. This will enable a repair and/or rescue attempt in the event of a deployment failure. If the deployments are made in GEO or on route to Pluto and a deployment failure occurs, then repair attempts would be much more difficult to engineer. Deployment of the booms in LEO will help improve mission reliability.

Production of the Final Product

The production of the Pluto probe will be a multistep process of design, parts construction, system integration, and possible redesign. In each of these phases testing for quality and reliability is essential. A series of testing procedures has been described that helps insure the production of reliable spacecraft.[7] The following is a description of that testing procedure.

Test Objectives:

Development Test: Establish a fundamental behavior pattern upon which a design can be based.

Qualification Test: Verify that the equipment and associated software will meet all specified requirements.

Acceptance Test: Verify workmanship and demonstrate that the equipment functions properly over the range of correctly selected operating conditions.

Prelaunch Verification Test: Performed at the launch site to verify that the spacecraft has sustained no shipping damage and has been properly mated to the launch vehicle.

Interactions with other Subsystems

Mission Planning: The dimensions of the spacecraft in launch configuration limits the mission planner's selection of launch vehicle. Also, the mission planner has selected a flyby mission which greatly simplifies the overall spacecraft configuration.

Science: The scanning and pointing requirements of the scientific instruments requires that the scan platform be positioned in a clear field of view. The scientific instruments must also be provided with shielding from the contaminating space environment. Thermal control must be provided.

Attitude and Articulation Control: The spacecraft inertia and COM, determined by component masses and positions, affects the placement of attitude thrusters and thruster force selection. Thermal control must be provided.

Command, Control, and Communication: The antenna size and placement places restrictions on the placement of the scan platform for clear viewing. The massive computers housed in the bus significantly affect the spacecraft inertia and COM. Also, the computers generate heat that must be rejected from the bus by radiating louvers.

Power and Propulsion: The propellent tank, when fuelled, is the most significant factor in determining the spacecraft inertia and COM. The main propulsive unit must be oriented so that its line of force acts through the spacecraft COM. Thermal control must be provided.

Appendix 1A: Inertia Calculations

The calculation of the individual component inertia's is simplified greatly by idealizing those components into simple geometric shapes. This assumption yields results which are adequate for the purposes of this preliminary design report. Of course this simplified methodology is in no way appropriate for actual inertia calculations of the later stages of design. Another simplifying assumption is that each component is homogeneous in density. Also, for the purpose of this calculation the mass of the bus includes the bus structure, command and control computers, thermal control, and cabling. The following is a list of component idealizations, component inertias, and component COMs. All dimensions are in meters. All inertias are in units of $kg\text{-}m^2$.

Bus (370 kg): Hollow cylinder. L=.35 Ro=.95 Ri=.65 COM=(0,0,0)

$Ix=Iy=M[(Ro^2+Ri^2)/4+(L^2)/12]=126.3$

$Iz=M(Ro^2+Ri^2)/2=245$

Propellent Tank (120 kg empty): Spherical shell. R=1.0 COM=(0,0,-.94)

$Ix=Iy=Iz=2MR^2/3=80$

Antenna (5 kg): Flat disk. R=1.85 COM=(0,0,.7)

$Ix=Iy=MR^2/4=4.3$

$Iz=MR^2/2=8.7$

Antenna Base (45 kg): Solid cylinder. L=.12 R=.95 COM=(0,0,.5)

$Ix=Iy=M[(R^2)/4+(L^2)/12]=10.2$

$Iz=MR^2/2=20.3$

Magnetometer Boom (5 kg): Thin rod. L=13 COM=(7.45,0,0)

$I_x=0$

$I_y=I_z=ML^2/12=70.4$

RTG Boom (5 kg): Thin Rod L=3 COM=(-.5,-2.45,0)

$I_y=0$

$I_x=I_z=ML^2/12=3.7$

Science Boom (35 kg): Thin rod. L=1.2 COM=(0, 1.55,0)

$I_y=0$

$I_x=I_z=ML^2/12=4.2$

Scan Platform (111 kg): Prism. L=.5 W=.3 H=.3 COM=(0,2.2,0)

$I_x=M(W^2+H^2)/12=1.7$

$I_y=I_z=M(W^2+L^2)/12=3.1$

RTG (60 kg): Cylinder. R=.1 L=1.52 COM=(-.53,-4.71,0)

$I_y=MR^2/2=.3$

$I_x=I_z=M[(R^2)/4+(L^2)/12]=11.7$

Appendix 1B: References

1. "Configuring the Grand Tour Spacecraft." Advances in the Aeronautical Sciences, vol. 29, part 2. p. 447.

2. "Structures and Materials Technology." NASA Space Systems Technology Model, June, 1985. p. 2-12.

3. "Beryllium." Machine Design, June 7, 1984. p.59.

4. "Galileo to Employ Cosmic Ray-Resistant Parts." Aviation Week and Space Technology, May 7, 1984. p. 20.

5. Dumas, L. N. "Temperature Control of the Mariner Class Spacecraft: A Seven Mission Summary." Progress in Astronautics and Aeronautics, vol. 35: Thermophysics and Spacecraft Thermal Control, 1974. p. 538.

6. Taylor, Charles H. "Precise Touches of Warmth in Cold Space." Aerospace America, October, 1984. p. 28.

7. Krausz, A. "The Effectiveness of Spacecraft Testing Programs." Journal of Environmental Science, January, 1980. p. 14-18.

MISSION MANAGEMENT, PLANNING, AND COSTING

Introduction

Mission Management, Planning and Costing (MMPC) has several responsibilities regarding the unmanned mission to Pluto. A mission timeline, outlining such features as the launch date, impulse points, encounters with planets, arrival at Pluto, and the proposed end of mission date must be furnished. MMPC must also determine a trajectory system so that time and ΔV are optimized. Another responsibility is the selection of the launch vehicle. A vehicle which minimally satisfies the spacecraft's dimensions at launch as well as the mass of the launch configuration is necessary. Furthermore, MMPC must also select the type of mission to be performed at Pluto. The mission should stress simplicity, reliablity and low cost. Lastly, a total costing analysis for the project must be furnished.

The remainder of the MMPC section contains a detailed analysis of the requirements previously mentioned, including trade studies and mission planning effects on other subsystems. The requirements are treated as separate categories where applicable, and each will be discussed individually.

Mission Timeline

On May 4, 2009 (day 0) NASA will launch the spacecraft Copernicus into a low earth orbit (LEO) of 270 km and an eccentricity of 0.00. The spacecraft will then leave the Earth's orbit via an upper stage and begin it's voyage to Pluto. On March 1, 2010 (day 300.4) Copernicus will fire an impulse to prepare for its gravity assist at Jupiter. This gravity assist at Jupiter will occur on February 18, 2012 (day 1019.8). The spacecraft will then be on a trajectory for the planet Saturn, arriving on July 29, 2015 (day 2276.6). Once again, a gravity assist will be made. Copernicus will then travel uninterrupted for about six years until it reaches its target destination, Pluto. The spacecraft will fly by Pluto on December 14, 2021 (day 4607.0). It will then continue on, leaving our solar system, not to return. The end of the mission will occur after the encounter with Pluto on December 14, 2021 (day 4607.0).

During it's flight, Copernicus will be performing correction maneuvers (see Attitude and Articulation Control) when necessary. As they cannot be predicted, no mention of it is included in this time schedule. Figure 2.A shows a timeline view of the mission, from the launch date to the end of mission date.

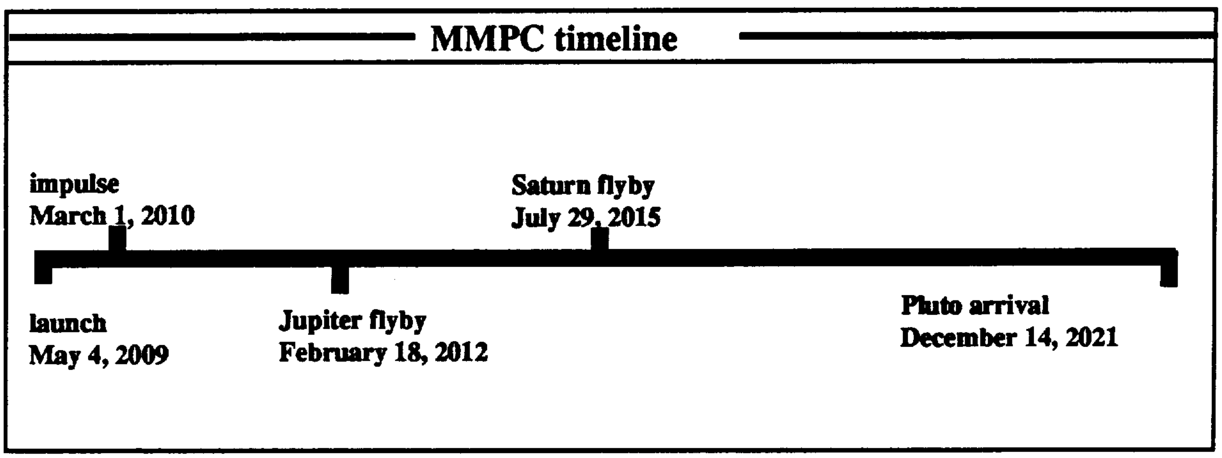

Figure 2A. Mission Timeline

The overall duration of the mission is 12.613 years (4607.0 days). During this time a management program will be in effect. The structure of this program will include a management, control, administration and support staff as well as division representatives[1]. Also, the duration time pertains only to flight of the spacecraft and does not include the planning, research and development and the assembly and testing.

Trajectory Systems

The selection of a trajectory system is perhaps the biggest task for the MMPC subsystem. The spacecraft ideally should arrive at Pluto in a minimum amount of time, while using a minimum amount of fuel. This immediately produces a conflict. A compromise which effectively minimizes both is desired.

The analysis of a trajectory system was performed with computer software. The spacecraft had the requirement that it must be launched sometime in the first decade of the twenty-first century. The spacecraft would have to travel about 33 AU's. A direct flight to Pluto was on the order of 28 years[2]. This was double the desired flight time so efforts to use gravity assists were employed. The first system consisted of using Jupiter as a gravity assist. Much work cut the flight time down considerably to about 15-16 years[2]. However, more planet gravity assists to shorten the flight time were still necessary. The next project involved using Jupiter, Saturn and Neptune for gravity assists. The project was named EJSNP (Earth-Jupiter-Saturn-Neptune-Pluto). The project was aborted in one week. Neptune could not line up properly in conjunction with the other planets, and was requiring too large a ΔV to correct it. So project Pluto began, consisting of an "Earth-Jupiter-Saturn-Pluto" configuration. This cut the flight time down on the order of 13-13.5 years. However, one problem was that it is desirable to leave before or after the first decade of the twenty-first century for Saturn and Jupiter to align properly, preferably early. Another problem is that Jupiter is not to be approached closer than 10 body radii due to large radiation output and Saturn should not be approached closer than 2.4 body radii due to it's rings. The Jupiter restriction was not a problem but Saturn continually required an approach of less than two body radii. The project was switched to "Longshot", using the same bodies as project Pluto but using a launch time at the end of the decade. This allowed Saturn's restriction to be satisfied and produced a flight time of about 12.8-13.2 years. The following graph (Figure 2B) depicts a trade off between ΔV required and time for Operation

Longshot. The final trajectory selection was then determined (Figure 2C.).

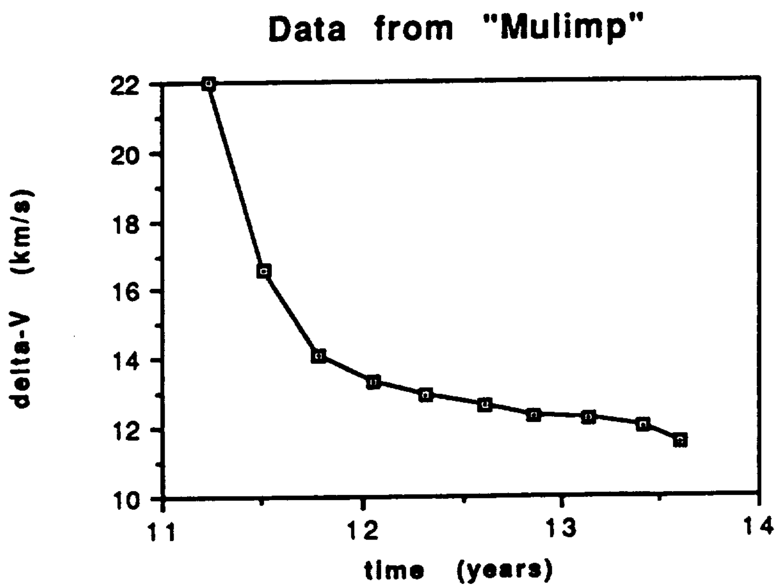

Figure 2B. Mulimp Data

The final selection was optimized to produce an impulse to assist the gravity assist at Jupiter. The mission flight time was finally reduced to 12.613 years. The complete analysis of this mission can be found in the appendix after this subsystem, including but not limited to launch time, ΔV required, and the coordinates of the specific events. The final trajectory is mapped in Figure 2C. Note that planet sizes are not to scale but are shown for illustration purposes.

Another problem is the solar system's asteroid belt. To avoid any possible collision that might result in a mission failure, the impulse fired after departure will provide a ΔV of 0.267 km/sec in the negative z-direction (see Appendix). Another advantage with this trajectory is that it uses all of its fuel (not including the safety factor

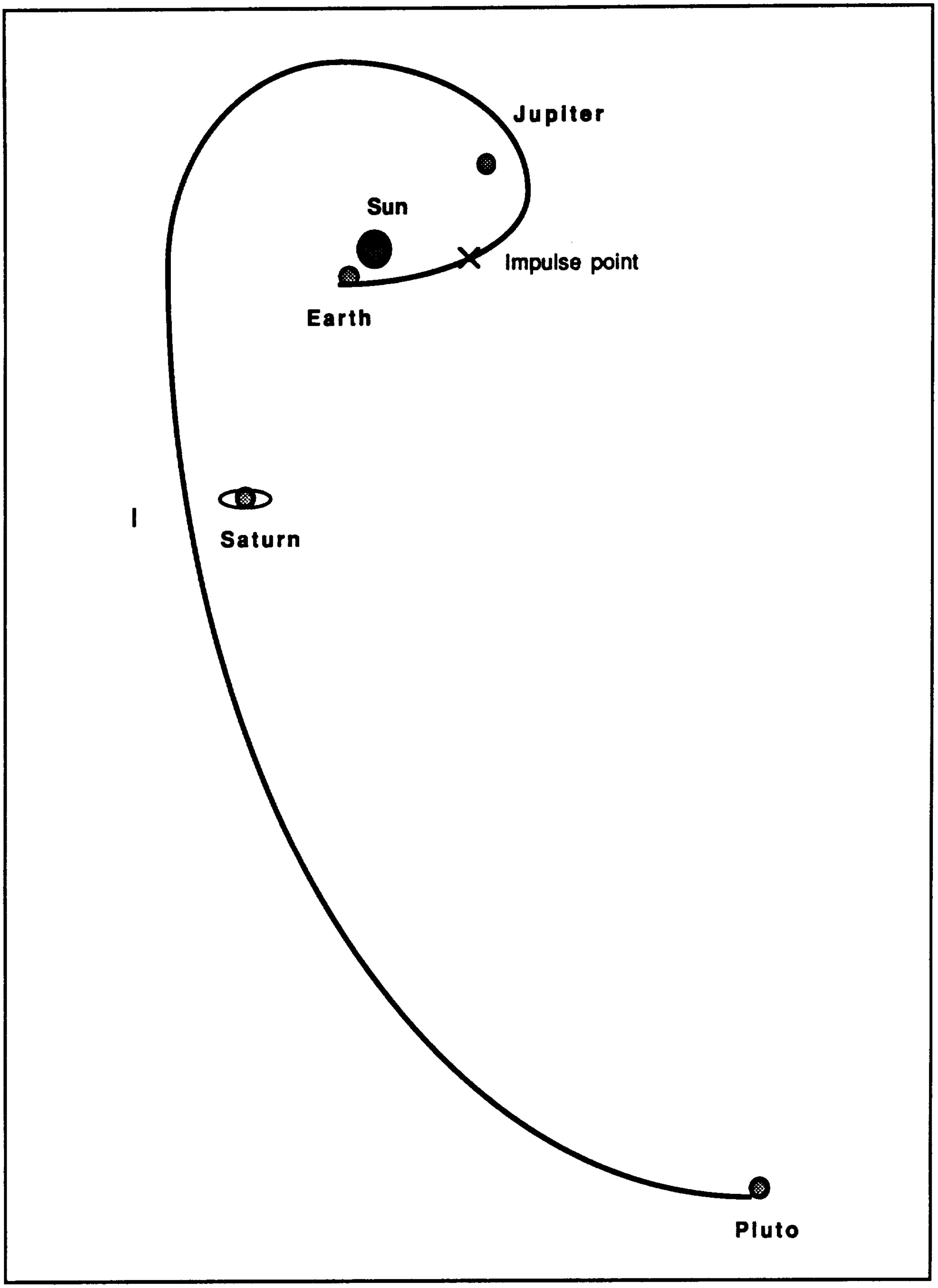

Figure 2C. Copernicus Trajectory

of fuel) early. After 2.798 years, Copernicus will have made its last burn and will travel the remainder of its ten years with the weight of the spacecraft only (excluding attitude control fuel). This is also responsible for its short flight time. The total ΔV required for the mission is 12.371 km/sec. This includes the departure from the Earth's orbit. The ΔV required from the spacecraft's propulsion module is 6.123 km/sec. This can also be found in the Appendix following this subsystem.

Launch vehicle selection

The spacecraft Copernicus requires a launch vehicle to insert it into earth's orbit. The selection of vehicles was limited to United States launch vehicles. The launch vehicle would have to be able not only to reach orbit, but it was also desired to use a configuration that would let the spacecraft escape the earth's gravity and to begin it's mission.

The launch vehicle must satisfy the spacecraft's weight including fuel and launch packing. A factor of safety of at least 10 percent was also desired. The companies that were considered were Martin Marietta, General Dynamics (GD)/Space Systems, McDonnell Douglas, and Boeing. Initially, GD/Space System's Atlas G was selected. However, as more fuel was added, the spacecraft's weight increased and the minimum performance payload necessary became 2730 kg and the Atlas G could no longer meet the requirement[4]. The spacecraft's pre-launch configuration had the dimensions of a cylinder of radius 3.7 m and a height of 5.0 m (see "Structures"). These dimensions could be employed on most launch vehicles and was not a primary concern.

The launch vehicle finally selected consisted of the vehicle and an upper stage. The launch vehicle selected is the Titan T-34D, by Martin Marietta. This, used in combination with the Centaur D1-T, could handle a payload weighing up to 5910 kg[4]. This exceeds the minimum requirement easily. However, the Centaur upper stage is built by GD/Space Systems and the Centaur D1-T is a modified version of the Centaur, designed specifically for the Titan T-34D.

The Titan T-34D uses a solid propellant while the Centaur D1-T uses LOX/LH_2. Both vehicles are environmentally safe and pose no conflicts regarding the safety of the launch.

Mission Type

The type of mission selected is the result of a lengthy trade analysis. The types of missions were divided into three categories: flyby, lander and orbiter. A flyby class mission was identified as any mission which did not perform any thrusting at Pluto. A lander mission was defined as the landing of any item on Pluto's surface. Lastly, an orbiter mission involved using a burn to obtain an orbit about the planet for a given length of time. Of the three classes, only the flyby and the orbiter missions were highly analyzed.

A lander mission involved sending a spacecraft to a planet of which there is little knowledge of. Historically, a lander mission follows an initial study of the planet. For a lander mission to be effective, an accurate idea of what is to be accomplished should be known. It would be senseless to send a lander to Pluto without first knowing what areas of the planet interest us. Also, the difficulties of uncertain areas including the gravity, composition, surface conditions and temperatures possess too high a risk factor for such a mission. Furthermore, the cost of carrying out a lander mission to Pluto might as much as double that of a flyby.

Initially, ideas for an orbiter mission were assembled. An orbiter could perform many experiments, and would also allow a longer encounter time at Pluto. Also, the mission was to incorporate a needle probe to penetrate the surface of Pluto and to examine samples. However, the ΔV required was high (a burn of 9.0-11.5 km/sec was required to insert the spacecraft into orbit[2]). Also, further research posed yet a bigger problem: Pluto's moon, Charon.

This was of no major concern at first. However, since Charon's mass exceeds 4 percent of Pluto's mass[6], the two bodies behave as a binary system. This system would make an attempt to orbit Pluto very difficult. Essentially, a three body problem must be solved. Another idea would be to orbit in a "figure eight" configuration. Also, while Charon's sphere of influence is estimated at 7000 km , Pluto can retain a satellite up to an estimated 5000000 km[6]. The mission tended to lean toward flyby class at this point. To orbit Pluto, burns would likely be needed for stable equilibrium. This suggests that the orbit duration would be short (finite), and a finite orbit duration did not warrant the increased cost of fuel required for orbit insertion. The mission to send an orbiter to Pluto was finally aborted. A mission to flyby Pluto was decided.

A flyby mission is the least expensive to build, test and fly. The components needed for the mission are considerably less than that of an orbiter class mission, making it a simpler design and more reliable. Furthermore, a flyby mission has an attractive ΔV (see "trajectory system"). While a flyby mission has less of an opportunity to gather information, it still provided adequate instrumentation, including imaging equipment to make an initial survey of the planet. Lastly, the spacecraft would ideally leave the solar system permanently. The spacecraft will have drawings on it's buss including a picture of man, as well as the location in our solar system in the Milky Way galaxy in the event of an encounter with any intelligent life. Only a flyby mission would allow this to occur.

Costing

The costing of the spacecraft includes the cost of not only the design and research leading to the construction of the vehicle, but the ground support operations of the lifetime of the mission. A detailed analysis of the costing can be found in the appendix following the end of this subsystem. The costing estimation used in this report is the "model estimation" method. This primarily involves assigning a number of labor hours to each section of the spacecraft. The labor hours are in turn converted into labor cost and the labor cost is finally related to the total cost. The total cost of the spacecraft is $999,443,600 dollars in terms of the 1977 fiscal year.

Another estimation technique is the concept of inheritance. The model estimation technique uses the masses of the individual systems but gives no consideration to the design and research development of the systems. Inheritance involves assigning each system to one of five classes:

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

Any components from class one will benefit from the previous design while class five receives no benefits whatsoever. By incorporating inheritance into the model estimation technique, the final cost will be effectively estimated. Assume that four spacecraft will be built for costing purposes.

Appendix 2A.

Costing-

Section1:

This contains the determination of direct labor hours (DLH) and recurring labor hours (RLH). The standard format is either $x*(N*M)^y$ or exp(x+y*N*M), where N is the number of spacecraft and M is the mass in kilograms. Note DLH and RLH are given in thousands of hours.

NRLH = DLH - RLH

	Inheritance Class
<u>Structure and Devices</u>	
DLH = 1.626*(2*285)^0.9046 = 947.1	
RLH = 1.399*(2*285)^0.7445 = 264.0	
NRLH = 683.1	3
<u>Thermal Control, Cabling & Pyrotechnics</u>	
DLH = exp*(4.2702 + 0.00608*4*30) = 148.4	
RLH = 3.731*(4*30)^0.6082 = 68.6	
NRLH = 79.8	3
<u>Propulsion</u>	
DLH = 56.1878*(4*120)^0.4166 = 735.6	
RLH = 1.0*(4*120)^0.9011 =260.7	
NRLH = 474.9	3
<u>Attitude & Articulation Control</u>	
DLH = 21.328*(4*49)^0.7230 = 968.8	
RLH = 1.932*(4*49) = 378.7	
NRLH = 590.1	2

Telecommunications

DLH = 4.471*(4*20)^1.1306 = 633.9

RLH = 1.626*(4*20)^1.1885 = 297.1

NRLH = 336.8 2

Antennas

DLH = 6.093*(4*5.1)^1.1348 = 186.6

RLH = 3.339*(4*5.1) = 68.1

NRLH = 118.5 2

Command & Data Handling

DLH = exp (4.2605 + 0.02414*4*49.7) = 8600.1

RLH = exp (2.8679 + 0.02726*4*49.7) = 3972.6

NRLH = 4627.5 3

RTG Power

DLH = 65.300*(4*60)^0.3554 = 458.0

RLH = 7.88*(4*60)^0.7150 = 396.6

NRLH = 61.4 3

Line-Scan Imaging

DLH = 10.069*(4*36.5)^1.2570 = 5291.5

RLH = 1.989*(4*36.5)^1.4089 = 2228.4

NRLH = 3063.1 2

Particle & Field Instruments

DLH = 25.948*(4*39.0)^0.7215 = 991.8

RLH = 0.790*(4*39.0)^1.3976 = 917.8

NRLH = 74.0 2

Remote Sensing Instruments

DLH = 25.948*(4*44.5)^0.5990 = 578.2

RLH = 0.790*(4*44.5)^0.8393 = 61.2

NRLH = 517.0 2

Section 2:

This section analyzes the Development Project - Support Functions and the Flight Project. PPL is in units of pixels/line. MD is the mission duration in months and ED is the encounter duration in months.

PPL = 1024
MD = 151.2
ED = 4.0
Σ DLH(hardware) = 19540
NRLH = DLH

<u>System Support & Ground Equipment</u>

DLH = 0.36172(ΣDLH)^0.9815 = 5887.3

<u>Launch + 30 Days Operations & Ground Software</u>

DLH = .09808(ΣDLH) = 1916.5

<u>Imaging Data Development</u>

DLH = 0.00124(PPL)^1.629 = 99.4

<u>Science Data Development</u>

DLH = 27.836(non-imaging science mass)^0.3389 = 124.7

<u>Program Management/MA&E</u>

DLH = 0.10097 (ΣDLH all categories)^0.9670 = 602.5

<u>Flight Operations</u>

DLH = (ΣDLH/3100)^0.6*(10.7*MD + 27.0*ED) = 5208.8

<u>Data Anlalysis</u>

DLH = 0.425*(DLH Flight Operations) = 2213.7

Section 3:

Total Costing:

This section incorporates inheritance into the costing. Costing for class 2 =1.00(RLH) + 0.2(NRLH). Costing for class 3 = 1.00(RLH) +0.75(NRLH). Since both equations represent labor hours, they must be converted to dollars.

LH = labor hours = (1.0-Z)*NRLH + RLH
Z = percent cost reduction
LC = labor cost
TC = total cost

Cost Category	LH	LC to TC
Structure & Devices	776.3	26975.0
Thermal Control, Cabling & Pyrotechnics	128.4	4369.8
Propulsion	616.9	23511.7
Attitude & Articulation Control	496.7	17671.9
Telecommunications	364.5	12205.8
Antennas	91.8	3169.1
Command & Data Handling	7443.2	227894.7
RTG Power	442.7	13375.4
Line-Scan Imaging	2841.0	108225.8
Particle and Field Instruments	932.6	33624.8
Remote Sensing Instruments	164.6	5760.3
System Support & Ground Eq	5887.3	191053.5
Launch+30 days Ops & Ground S/W	1916.5	65969.6
Image Data Development	99.4	3565.5
Science Data Development	124.7	6344.0
Flight Operations	5208.8	176571.4
Data Analysis	2213.7	79155.3
Totals	29749.1	999443.6

Total Cost of the Copernicus mission: $ 999,443,600

Equations Pertaining to MMPC-

TC = total cost = (100%-Z) NRC + RC
see costing section of appendix for individual component equations.

Final Trajectory Orbital Elements-

On the following page is an excerpt containing the orbit elements for the final design trajectory. This contains various data, including but not limited to flight time, ΔV required, and the Cartesian coordinates of significant encounters.

Final Trajectory Orbital Elements

```
***************************************************************************************
CASE TITLE: OPTIMIZATION REPEATS WITH ADDED IMPULSE(S)

FLIGHT TYM:            4607.0 DAYS          12.613 YEARS
      MODE: TOTAL DV OPTIMIZATION
            TERMINAL FLYBY(UNCONSTRAINED)

            CENTRAL BODY IS SUN         EARTH ECLIPTIC & EQUINOX [illegible]
    LAUNCH:     C3=   75.827   C3PRIME=   75.827
               VHL=    8.708        DLA=  -15.726      LONG=  [illegible]
  IMPULSES:  NO.    TYM(JD)        BODY    DVX(KPS)  DVY(KPS)  [illegible]  [illegible]

             1  2454956.192       EARTH          ORBIT DEPARTURE           [illegible]
             2  2455256.552          NO      0.000    -0.004   [illegible]  [illegible]
             3  2455976.040       JUPIT     -2.264    -5.279   [illegible]  [illegible]
             4  2457232.910       SATUR      0.000     0.000   [illegible]  [illegible]
             5  2459563.192       PLUTO      0.000     0.000   [illegible]  [illegible]
                                                               --------------------
                                                               TOTAL [illegible]

IMP POINTS:  NO.      DATE         DAYS      X(AU)     Y(AU)    [illegible]  [illegible]

             1  2009 MAY  4         0.0    -0.7313   -0.6944   [illegible]  [illegible]
             2  2010 MAR  1       300.4     0.1707   -0.2191   [illegible]  [illegible]
             3  2012 FEB 19      1019.8     2.5519    3.4712   [illegible]  [illegible]
             4  2015 JUL 29      2276.4     4.5891   -8.9752   [illegible]  [illegible]
             5  2021 DEC 14      4607.0    14.3309   30.0007   [illegible]  [illegible]
  AUX DATA:  NO.      VHP(KPS)  DEC(DEG)   RA(DEG)  PHAI(DEG)  [illegible]  [illegible]

             1*       8.708   -15.726    213.049    37.001    [illegible]  [illegible]
             2        0.000     0.000      0.000     0.000    [illegible]  [illegible]
             3*       6.024    -1.008    145.406   102.800    [illegible]  [illegible]
             4*      15.746   -21.422    237.073     0.774    [illegible]  [illegible]
             5*      13.796    64.001     21.330     1.971    [illegible]  [illegible]
              * = BODY EQUATORIAL VHP COORDINATES
  S/A DATA:  NO.      VHP(KPS)    RP(BR)   DAM(DEG)  FCI(DEG)  [illegible]  [illegible]

             3( 0)   17.214     10.000     77.941    77.740   [illegible]  [illegible]
             4( 0)   15.765      2.454     53.507    52.571   [illegible]  [illegible]
  RUN DATA:  NO.      TYM GRAD  POSXGRAD  POSYGRAD  POSZGRAD

             1     0.777E-02 0.000E+00 0.000E+00 0.000E+00
             2     0.927E-02-0.112E+01-0.217E+00-0.265E 02
             3     0.127E-01 0.000E+00 0.000E+00 0.000E+00
             4    -0.209E-01 0.000E+00 0.000E+00 0.000E+00
             5     0.000E+00 0.000E+00 0.000E+00 0.000E+00
           NF=       1    1    7    0    0   20
          NRI=       0    0    0    0
```

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Appendix 2A. References

1. "Planet", McGraw-Hill Encyclopedia, V.13, New York: McGraw-Hill, 1987. pp. 566-569.

2. Kepes, Paul, mulimp files,1990. Files are furnished at the end of report. enc.

3. "Jupiter",McGraw-Hill Encyclopedia, V.9, New York: McGraw-Hill, 1987. pp.482-488.

4. Fink, Donald E.,"Specifications", Aviation Week, March 19, 1990, pp.163-64.

5. Binzel, Richard P., "Pluto-Charon Mutual Events", Geophysical Research Letters, Vol.16, No. 11, November, 1989, pp.1205-1208.

6. Dobrovolskis, Anthony R., "Dynamics of Pluto and Charon", Geophysical Research Letters, Vol. 16, No. 11, November, 1989, pp.1217-1220.

COMMAND, CONTROL, AND COMMUNICATIONS

Computer Control

Copernicus, like most spacecraft, must perform a variety of functions at precise times with unerring accuracy. In order to do this, an on-board computer system is necessary. The computer system must control three main areas, the attitude and articulation subsystem (AACS), the flight data subsystem (FDS), and the computer control subsystem (CCS). A schematic layout of the computer system is shown in Appendix 3A. The computer will be made of three separate, freestanding but interacting computers, controlling the three areas mentioned. This system is modeled after the system on board the Voyager spacecraft.

The FDS computer is responsible for all of the flight data received during the lifetime of the spacecraft. All of the data from the science platform as well as all the periodic status reports of the spacecraft are fed into this computer, where it is assimilated, reduced and passed on. The FDS computer will be a 16 bit x 8192 word computer, as on the Voyager, and will interact with the rest of the computer system as well as the science platform and most other instruments for status reports.

The AACS computer is responsible for keeping Copernicus going in the right direction, with the correct orientation in space. All tracking data is fed into the AACS computer and it decides if a readjustment burn is necessary to correct its trajectory. Every reorientation of the spacecraft, to allow burns or communications, is timed and the AACS computer knows when to command the burns and precisely how long to burn. The AACS computer will be an 18 bit x 4096 word computer. This provides ample room for all of its programming needs.

The CCS computer is also an 18 bit x 4096 word computer. Most of the permanently stored programs are kept in this computer. If necessary it can completely reprogram both the AACS and the FDS. This provides a vital redundancy factor for the spacecraft computer system. Should the CCS need reprogramming, that would need to be done from Earth. All information to be sent to Earth and all incoming information from Earth goes through the CCS computer before moving on to the other computer subsystems, the antenna, or other areas of the spacecraft.

The three components of the overall computer system interact fully and all feed into a central storage unit, as shown in Appendix 3A. The data storage unit has a 400 kilobits per second(kbps) record rate, which will be able to handle all of the incoming data from the various computer subsystems. It also has five different playback rates, 100, 50, 25, 12.5, and 6.25 kbps. This wide range will handle all of the needs of the computers, the science platform and telecommunications.

Communications

Communications back and forth between Copernicus and Earth is essential for proper mission accomplishment. Copernicus needs to relay information such as status reports, scientific data and imagery back to Earth, while the command center on Earth needs to be able to send commands to the spacecraft to have it perform certain functions such as execute a burn, change course or take a picture. While most of the necessary commands for Copernicus will be stored in the computer system it is still necessary for communications to be able to reach the spacecraft.

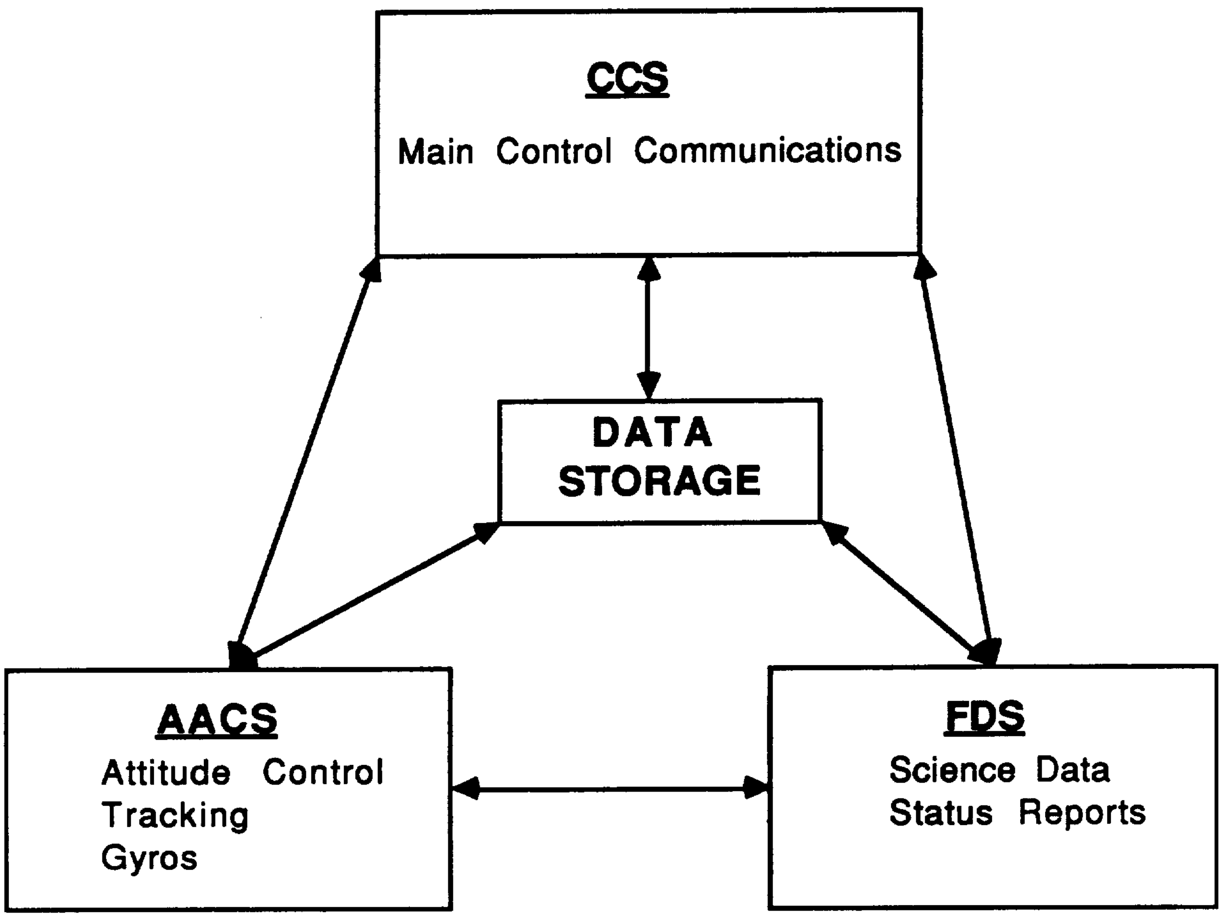
An antenna is the instrument used to perform the necessary transmission and collection of data. Copernicus' antenna is a standard parabolic dish that focuses the radio waves it intercepts to a central receiving unit, or broadcasts the radio waves onto the dish which sends them back to Earth.

There are two general radio wave frequencies used in deep space telecommunications, S-Band and X-Band. The X-Band is generally preferred due to its higher frequencies, which have less interference problems, and it will be used for Copernicus. The X-Band uplink (Earth to space) frequency is 7.161 GHz while the downlink (space to Earth) is 8.414 GHz. There are many factors that affect the transmission energy before it reaches its destination. These factors are summed up by the equation in Appendix 3B. Most of these factors are losses that reduce the energy from transmission to reception.

The size of the antenna is the driving factor in the calculation of necessary power. Large antenna sizes have larger gains, so less power is needed to achieve a required receiving power. Our antenna has a significant mass and keeping the mass to a minimum is important, so we can not allow our antenna to become too large. Another factor involved in the sizing of the antenna is the fact that it must fit within our launch vehicle. This means that the antenna must either be kept small or be collapsible, and much more complicated. In order to keep the configuration of Copernicus simple and less costly a solid antenna was chosen. It will be 3.7 meters in diameter. This provides Copernicus with a small, lightweight antenna that fits within the launch vehicle but is still capable of making necessary transmissions with little energy (app. 25 W).

The positioning of the antenna is vital in mission accomplishment. The antenna must point towards Earth if communications between Copernicus and Earth are to occur. Generally, though, the propulsion for the spacecraft points out of the back of the spacecraft, towards Earth. The antenna and the propulsion package will be on opposite ends of the spacecraft. For most of the beginning of the voyage, the antenna will be useless because of the fact that Copernicus will be in its burn stage. After the primary burn stage is complete, Copernicus will rotate 180°, allowing full communications. During the flight, if a burn using the main propulsion is needed, Copernicus must again be rotated 180°.

Appendix 3A. Control Flowchart



Appendix 3B. Communications Equation

$$P_R = P_T L_T G_T L_{TP} L_S L_A L_P L_{RP} G_R L_R$$

P_R=Power Received
P_T=Power Transmitted
L_T=System Losses in Transmitter
G_T=Transmitting Antenna Gain
L_{TP}=Pointing Loss of Transmitter
L_S=Free Space Losses
L_A=Atmospheric Attenuation
L_P=Polarization Loss Between Antennas
L_{RP}=Pointing Loss of Receiver
G_R=Receiving Antenna Gain
L_R=System Losses in Receiver

Appendix 3C. References

1. Deep Space Telecommunications Systems Engineering
   JPL (Joseph H. Yuen) , 1983, Plenum Press, New York & London.

2. Draper, R. F. (1988). The Mariner Mark II Program.
   Paper AIAA-88-0067, presented at the AIAA 26th Aerospace Sciences Meeting, Reno, Nevada, January 11-14, 1988.

3. Draper, R.F. (1984). The Mariner Mark II Program.
   Paper AIAA-84-0214, presented at the AIAA 22nd Aerospace Sciences Meeting, Reno, Nevada, January 9-12, 1984.

4. Heacock, R.L. (1980). The Voyager Spacecraft.
   Paper Proc Instn Mech Engrs Vol 194

POWER
AND
PROPULSION

Overview

The power system aboard the vehicle utilizes inherently reliable components. Only materials and techniques available before 1999 are to be used in the final fabrication of the system. The system design lifetime is sufficient to carry out the mission, allowing for a reasonable safety margin. Under normal mission conditions, the power system is fully autonomous. If necessary, new commands can be transmitted from the ground station on Earth. Performance, simplicity, and low weight and cost are stressed in design tradeoffs.

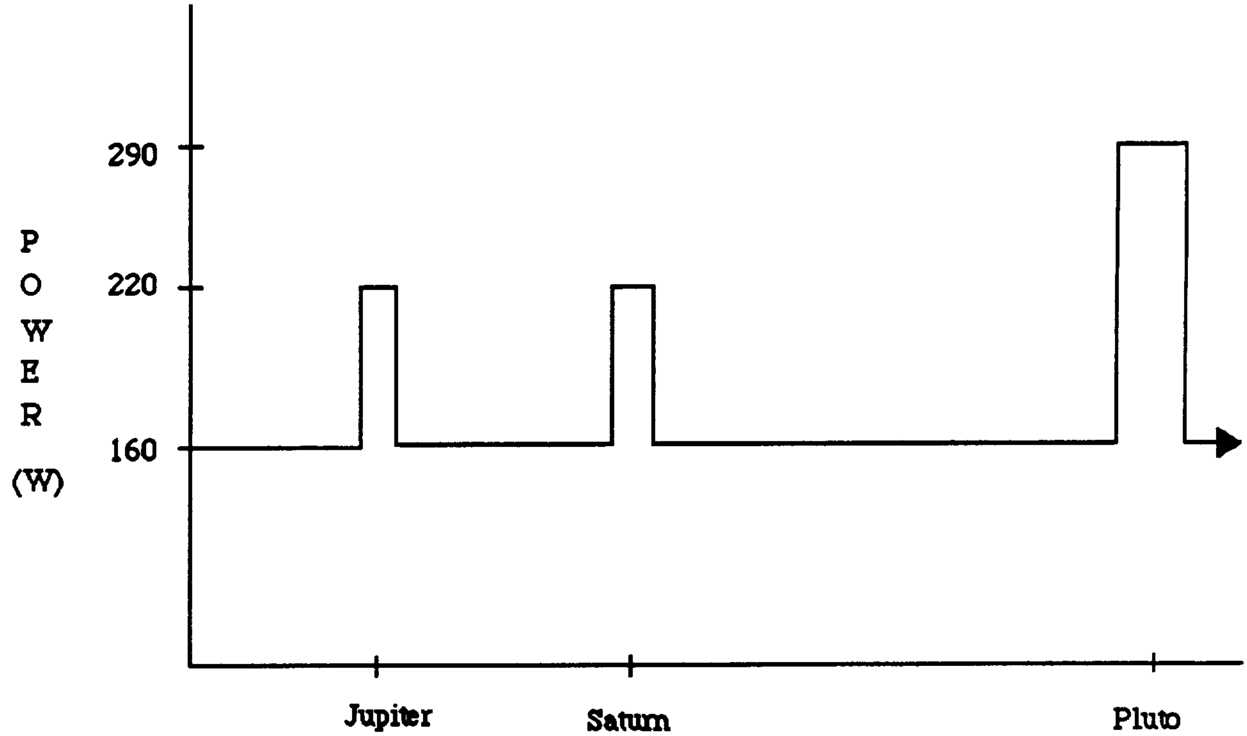
The main power source is a Modular Isotopic Thermoelectric Generator (MITG). With the flyby of several planets, the power requirements will change with respect to the mission timeline. The modularity of this component makes it ideal for use in this mission. Releasing power in small scaled amounts, this unit efficiently meets the power needs of the spacecraft at all times during the mission.

There exist socio-ecological problems in the use of the MITG, problems shared with all isotopic thermoelectric generators. Containing plutonium oxide, debris from these units would be extremely dangerous in the event of launch mishap. These are legitimate concerns and have been taken into consideration of the overall design. For a mission of this duration, however, it is infeasible to incorporate any other type of system.

Table 4-A. Power Requirements

system/component		power requirement
AACS		40 W
Science		130 W
Structure		
	thermal control	19.6 W
	pyrotechnics	2.4 W
CCC		
	computer	24.7 W
	data storage	23.2 W
	antenna	25 W
Power		25.2 W
	Total:	289.9 W

The maximum power required by the system is approximately 290 W. The total power supplied by the MITG is approximately 310 W, sufficient for the load requirements. The maximum power levels will only be reached during planetary flyby. Here the bulk of the scientific instrumentation will consume approximately 60 W of power. The imaging equipment will only be utilized at the encounter with Pluto, requiring an additional 70 W. The modification of power supplied will be autonomously controled by the computer.



PLANETARY FLYBY ENCOUNTERS
Figure 4-A.

The earlier planetary encounters require power increases for only a few days centered about the flyby date. In the case of Pluto, the imaging process requires weeks of the increased power level. An insignificant power of 2.4 W is needed for pyrotechnics at separation of the vehicle from the upper stage.

Component Selection

The MITG design was conceived by Fairchild Space and Electronics Company. They have developed several unit sizes ranging from output levels of 260 W to approximately 300 W. Satisfying the power requirement for the spacecraft , the 13 slice generator has been selected.

A redundant circuit design for both the dual busbars and network has been selected to decrease the chances of failure due to micrometeorite impact. Parallel fuses are incorporated on each load to provide redundancy. The electric circuit is located outside the generator housing, minimizing the probability of shorts-to-ground problems. Incorporating field-cancelling circuit modules, scientific instrumentation on the spacecraft will not be affected by induced magnetic fields from the MITG.

The generator consists of 13 independent slices each supplying approximately 24 W at 28 V. Each thermoelectric slice contains four plutonium oxide pellets supplying a total of 250 W of thermal power. A series of eight thermoelectric modules per slice convert the thermal power, given off by the fuel pellets, into electric power for the spacecraft. The plutonium oxide is contained in an iridium clad surrounded by an impact shell. Thermal insulation, consisting of carbon bonded carbon fibers, protects the fuel pellets from under or over-heating. The whole assembly is protected by an aeroshell, designed to maintain its structural integrity at extremely high temperatures. This design uses four radiator fins situated at the corners of the unit, optimizing heat dissipation as well as weight.

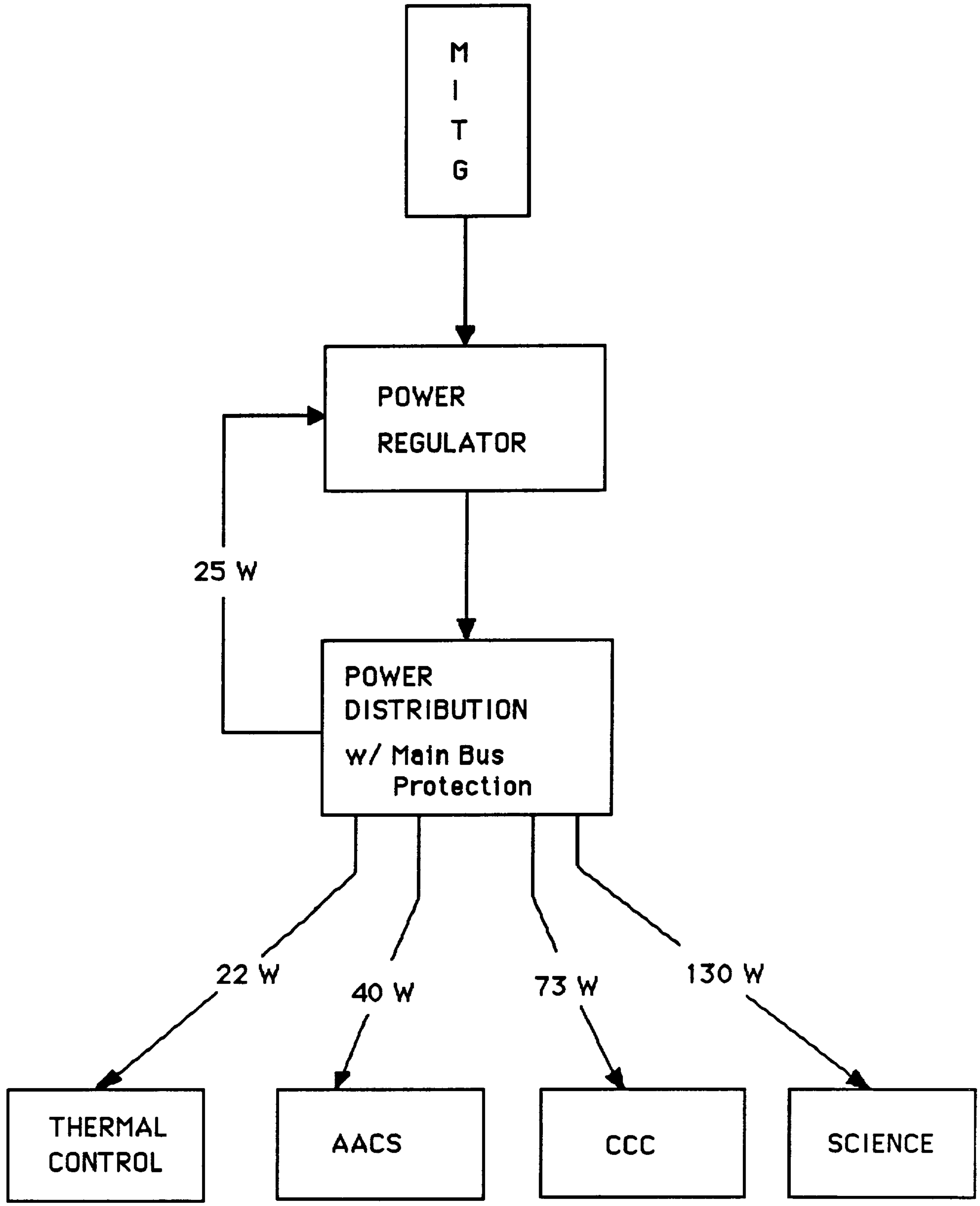


Figure 4-B. Power Breakdown

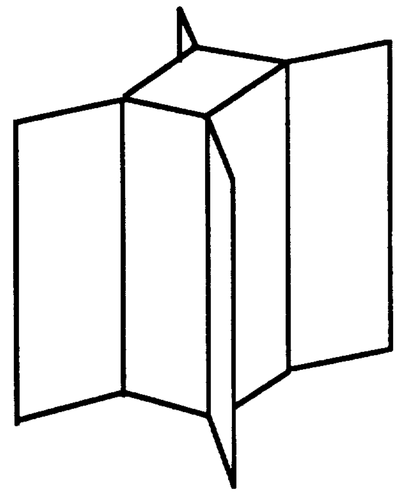


Fig. 4-C. Modular Isotope Thermoelectric Generator

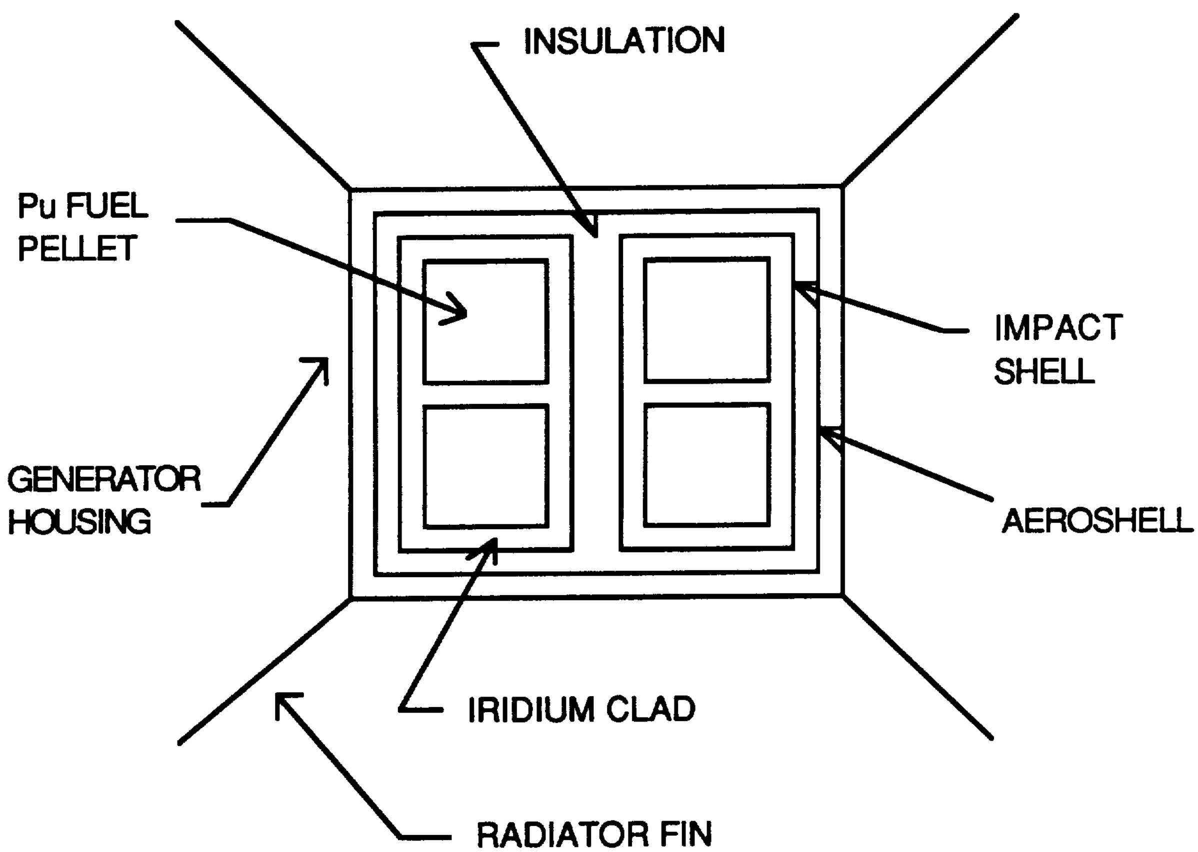


Fig. 4-D. Thermoelectric Slice

Overview

The propulsion system for the vehicle is characterized by simplicity and reliability. Components incorporated in the system have been flight tested extensively, meeting with proposal requirements on availability before the year 1999. The propulsion unit as well as the fuel storage have design lifetimes sufficient to carry out the mission, allowing for use of the thrusters for unexpected mid-course maneuvers. The system relies on autonomous control by the onboard computer. Performance, weight, and cost have been optimized in design tradeoffs.

The fuel used in this system is augmented hydrazine. Similar to conventional hydrazine, it is space storable for long periods of time. Considering the longevity of this mission, storability is essential. Because it is a monopropellant fuel, oxidation systems are not needed, lowering cost and weight. Generally systems of this type are capable of specific impulses of 200 to 250 seconds. With the use of augmented hydrazine, values of 300 seconds specific impulse can be obtained. Advantages of augmented hydrazine include low plume contamination and no surface contamination, problems which could interfere with the normal operation of the spacecraft and scientific instrumentation on board.

The main thrusters will burn twice during the mission. These two burns will provide the spacecraft with a total ΔV of 6.1 km / s. The first burn required is a small mid-course impulse, taking place approximately ten months after launch. The next burn is at Jupiter flyby, approximately two years later. This schedule provides for a smaller probability of error in the propulsion system since all the major burns occur in the first three years. The remaining amount of fuel, used by the attitude and articulation thrusters, will be approximately 5 % that of the initial supply.

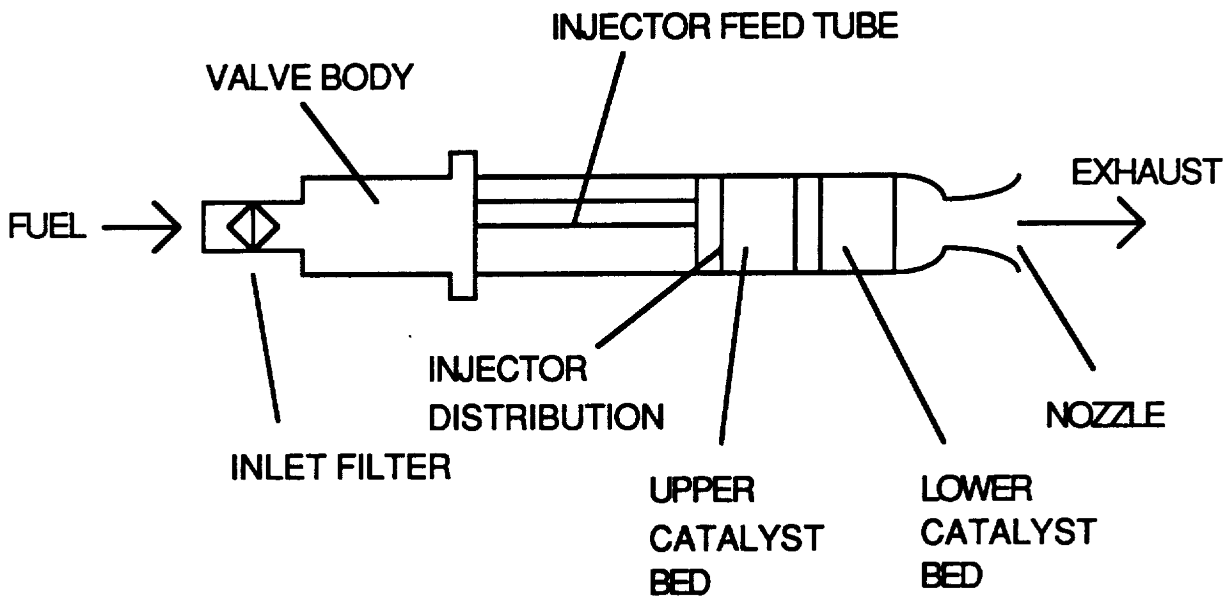


Figure 4-E. Hydrazine Thruster

The fuel storage tank is characteristic of the bladder design, eliminating the need for a pressurizing system. As the fuel is consumed the bladder folds in on itself providing the thrusters with a steady supply of fuel during burns. Since the attitude and articulation thrusters also use the hydrazine fuel, the storage tank can be shared between the two systems. Fuel from the tank travels through an inlet filter, which removes all foreign particles from the fuel stream. From there, it is driven through an injector feed tube and into the injector distribution element. The fuel then passes through the catalyst bed where it is ignited chemically. Heaters are situated around the catalyst bed for the chemical reaction to be carried out properly. Exhaust gasses then escape out of the nozzle providing the spacecraft with the necessary thrust.

Appendix 4A. Equation for Propulsion Subsystem

$$\Delta V = g_o \; I_{sp} \; \ln (m_i / m_f)$$

ΔV = change in velocity

g_o = constant for gravity

I_{sp} = specific impulse

m_i = initial mass

m_f = final mass

Appendix 4B. References

"Thermoelectric power - the road behind and ahead "
Author: Hall, W. C.
10th International Telecommunications Energy Conference Vol.3

"Rover Power System for Mars Sample Return"
Author: Bents, D. F.
Journal of Space Nuclear Power

"Electrical Power System Architectures for Future Aerospace Vehicles"
Authors: Dige, Mark W.; Harold, Neal C ; Mehdi, Ishaque S.
Boeing Advanced Systems

"Modular Isotopic Thermoelectric Generator"
Author: Schock, A.

"Advanced Propulsion Systems for Space Application"
Author: Knapp, David Edwin

AAE 241 Class Notes ~ Power Subsystem

AAE 241 Class Notes ~ Propulsion Subsystem

SCIENCE INSTRUMENTATION

Introduction

This section describes the scientific subsection of project Copernicus. This includes a science time line, the planned experimentation, and the equipment needed to complete the mission. The selection of experiments was based on present day scientific objectives for information gathering of the outer planets. Individual instrument systems were compared and selections were made based on experimental need. In addition, the requirements and constraints of NASA's Request For Proposal (RFP) were obeyed.

Voyage to Pluto and Charon

The long voyage to Pluto and Charon will allow an excellent opportunity for Copernicus to gather information on the galaxy. This time will not be wasted. During every phase of the journey, experimentation will take place.

Earth-Jupiter Cruise Phase

After initial Earth orbit and spacecraft deployment have been established, the science mission will begin in earnest. Once out of Earth orbit the scientific equipment will be tested and calibrated through relay with mission scientists on Earth. Later in the journey such fine tuning will not be possible. Copernicus will spend the majority of its time in interplanetary space, at these times science will act in cruise mode. During cruise phases, fields and particles experiments will be employed. Distant stars will be targeted for observation and data recording. Information will be gathered and relayed to Earth approximately every 0.5 AU.

Jupiter Encounter Phase

As Jupiter nears the instrumentation and experimentation will convert to encounter mode. The scan platform will be turned to focus directly on Jupiter. Approximately 80 days and 80,000,000 km before the closest approach to Jupiter, Copernicus's imaging equipment will come to life. Over the next seven weeks, the narrow angle camera will take visual information of the whole planet. A series of color filters on the camera will also be employed. At this time, the infrared and ultraviolet spectrometers along with the photopolarimeter will be taking whole planet data.

As Copernicus approaches 30,000,000 km from closest approach, the transmitter will begin sending information at encounter data rate. At this time, the wide angle camera and its color filters will be engaged. The fields and particles experiments will also be placed in encounter mode. Specifically, they will investigate the transition from the region of space dominated by the solar wind to that of Jupiter's magnetosphere.

As closest approach nears, the equipment on the scan platform will take advantage of the change in phase angle, from low phase angles to high, to observe any differences in information due to the phase angle change. During Jupiter pass by, the Earth will be eclipsed from Copernicus which will allow an excellent opportunity for mission scientists observe the effects of the Jovian atmosphere on the communications signal. This radio science information could be used to draw conclusions about the composition and height of the Jovian atmosphere.

As Copernicus leaves it will pass through Jupiters shadow which will allow ultraviolet inspection of the atmospheric upper layer composition. Also, long exposure imaging of Jupiters night side will take place. As the probe continues out the fields and particles experiments will investigate the extended tail of the magnetosphere. Transmission will return to cruise data rate 40 days after closest approach.

Jupiter-Saturn Cruise Phase

Upon entering the Jupiter-Saturn cruise phase, science investigations will return to primarily fields and particles. Special attention will be paid to the gradual changes in the character and temperature of the solar wind. Particles experiments will emphasize the cosmic ray environment. During this phase, the annual solar conjunctions allow radio science the opportunity to investigate the solar corona. As communication signals transverse the solar corona mission scientists can measure the coronal electron density.

Saturn Encounter Phase

The Saturn encounter will progress as did the Jupiter encounter. The only difference being the emphasis on Saturns rings. Imaging will begin 80 days out, fields and particles experiments and transmission rates begin encounter mode 30 days out, and Copernicus returns to cruise mode 40 days after closest approach. The information gathered from the Jupiter and Saturn encounters can be compared to data obtained from the Voyager missions. Any differences found could be very useful in understanding our changing planets and galaxy[4].

Saturn-Pluto Cruise Phase

In the final interplanetary cruise phase Copernicus will investigate the proton component in the distant solar wind plasma. It will also measure the intensity, composition, and differential energy spectrum of galactic cosmic rays. These experiments are very important, as no other spacecraft has taken this final route.

The power and data rate requirements of the science subsection are shown in time line format in Figure 5A. This clearly portrays the distinct peaks of power use and transmission requirements during the planetary encounters. The power capabilities and communication needs are adequately met by the Copernicus spacecraft. Figure 5B indicates the individual instruments used in each phase of the mission. The instruments were selected for each phase to maximize the data gathering and to minimize the power drawn and the data transmitted.

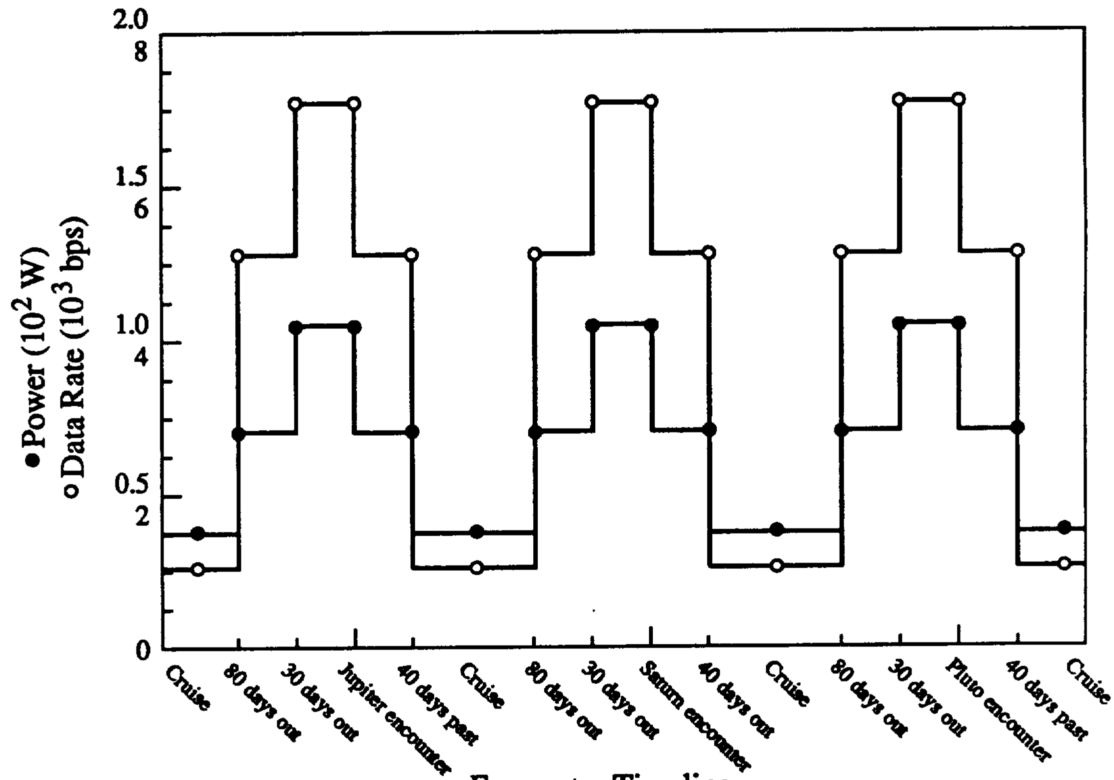


Figure 5A Power and Data Transmission Timeline

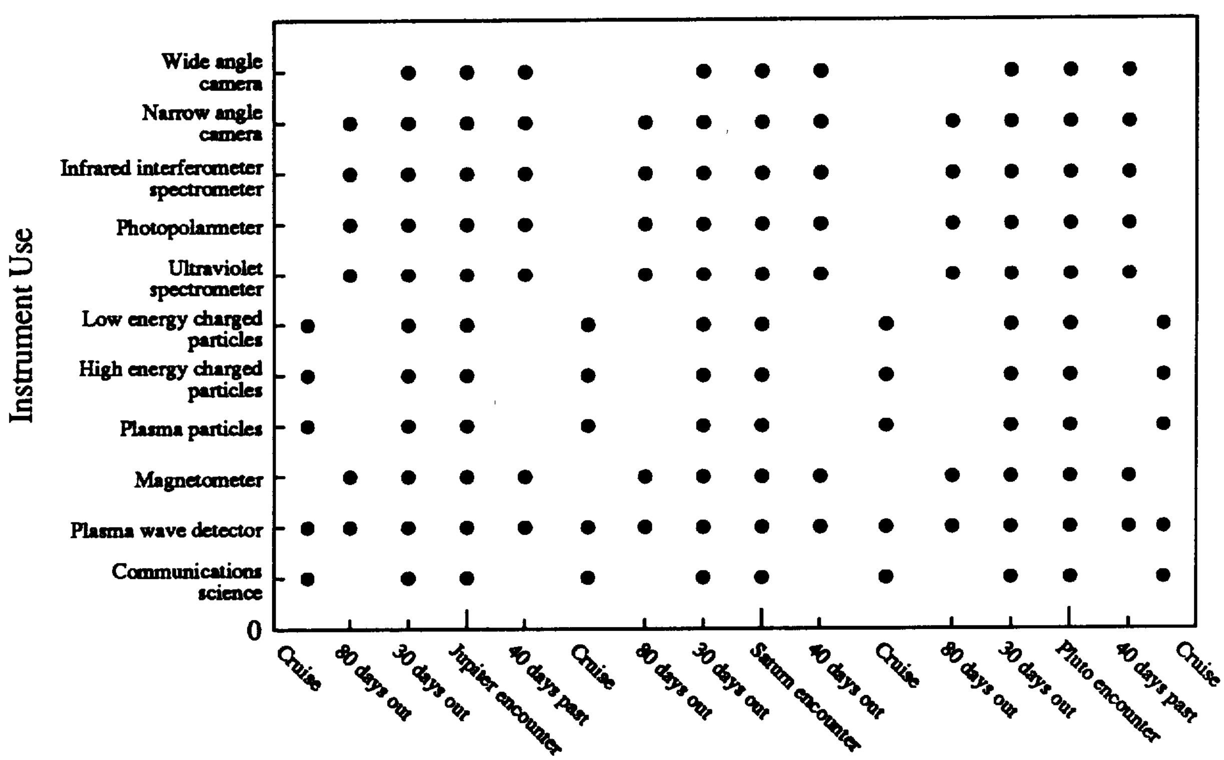


Figure 5B Instrument Use Timeline

Pluto and Charon Encounter

The mission culminates with the investigation of Pluto and its satellite Charon. Scientific objectives for the two bodies were based on those from the National Academy of Sciences objectives for the outer planets[5]. Experiments and investigations specific to Pluto and Charon were developed that would fulfill the needs of the scientific community. These experiments in approximate order of importance can be seen in Table 5A[2,6,7]. Many of the investigations have specific subexperiments.

Experiments in Approximate Order of Importance
1. Total Mass and Density • Map the surface albedo distribution • Investigate ice to rock ratio • Investigate composition and hydration state
2. Radius and Oblateness • Find global maps of Pluto and Charon • Investigate hydrostatic shape changes • Map solid body shapes
3. Atmospheric Composition • Investigate atmospheric induced limb darkening effects
4. Gravitational Harmonic Coefficients
5. Shape and Strength of magnetic Field at Several Radii
6. Pattern and Magnitude of Heat Flux, Surface Temperature, and Heat Balance at Various Latitudes and Phase Angles
7. Shape and Intensity of the Tail of the Magnetosphere or of the Cavity in the Solar Wind
8. Local Anomalies • Investigate possible dark spots and rings

Table 5A Experimental Listing

The experiments will proceed in a similar manner to the encounters of Jupiter and Saturn. However, because the system being investigated is a two body system, care will need to be taken with respect to time management. The scan platform will need to be rotated to allow adequate time to gather data from both Pluto and Charon.

As Copernicus approaches Pluto and Charon, the imaging equipment will begin the investigations of radius and oblateness. It will begin to compile images that will be used to create global and solid body maps, and to investigate any hydrostatic shape changes. These maps will be used to help determine the radius and oblateness of both bodies. As the probe nears, the infrared interferometer spectrometer will be used to investigate thermal emissions, composition of thermal structure, and heat balances. This data will be collected over a variety of phase angles. The information, along with the imaging data will help to map the surface albedo distribution, investigate the ice to rock ratio, find the pattern and magnitude of heat flux, surface temperature, and heat balance at various latitudes and phase angles of both Pluto and Charon. While still on approach, Copernicus will accumulate data with its magnetometer. Information from the magnetometer will aid in determining each bodies gravitational harmonic coefficients and the shape and strength of their magnetic fields at several radii. The photopolarimeter will investigate the physical and chemical properties of Pluto and Charon. This information, along with data from the infrared interferometer spectrometer and the imaging equipment, will help to determine the composition, mass, and density of both bodies.

As the spacecraft passes through its closest approach, the particles experiments will convert to encounter mode. In this mode they can gather a variety of important information. The high energy particle detector will measure electrons and cosmic rays, while the low energy particle detector investigates particles in the planetary magnetosphere. The plasma particle detector will determine plasma flow direction and the plasma wave detector will study the wave and particle interaction in the dynamics of the magnetosphere. All this information will be used to model the shape and intensity of the tail of the magnetosphere or of the cavity in the solar wind.

While the probe is eclipsed from Earth, additional investigations will be made. The imaging equipment will focus on Plutos limb and terminator region. Data acquired can be used to determine the atmospheric induced limb darkening effects. Communications tracking of the probe can aid in finding Plutos gravitational harmonic coefficients and the strength of its magnetic field.

As Copernicus sails into the outer galaxy its investigations will not end. Possibly it could investigate the heliopause. It will continue to send data from our galaxy back to Earth.

Equipment Selection

The design features a wide variety of imaging, spectroscopy, and fields and particles instruments. All equipment was selected from existing hardware used on the Cassini, Galileo, and Voyager missions. This was done to minimize cost while keeping a high level of information accuracy and reliability.

Imaging Science Subsystem (ISS)

The Copernicus probe will encounter a wide variety of targets and range of observing distances. Therefore, two separate cameras will be used in the ISS, a Narrow Angle Camera (NAC) and a Wide Angle Camera (WAC). In this way, Copernicus can provide two different scales of image resolution and coverage.

The two cameras are framing Charge Coupled Device (CCD) imagers. The charge couple device design is a square array of 1024×1024 pixels, each pixel is 12 μmeters on a side. They differ primarily in the design of the optics: the NAC has a focal length of 2000 mm and the WAC has a focal length of 250 mm. Both cameras have a focal plane shutter of the Voyager/Galileo type, and a two-wheel filter changing mechanism derived from the Hubble Space Telescope. Both cameras have deployable dust covers. To minimize mass, power, and cost, the two cameras will not be completely independent - they will share a common electronics module. This module services both cameras, and contains the digital part of the video signal chain, power supplies, mechanism drivers, command and control logic, and the digital data compressor[3].

Key parameters of the ISS:

	Narrow Angle	Wide Angle
Camera Type	Framing CCD	Framing CCD
Optics Type	Ritchey-Chretien	Refractor
Focal Length	2000 mm	250 mm
Focal Ratio	f/10.5	f/4.0
Resolution per pixel	6 μrad	48 μrad
Field of View	0.35° square	2.8° square
Spectral Range	200-1100 nm	350-1100 nm
Spectral Filters	22	14
Heater Unit	Strip heaters	Strip heaters

Infrared Interferometer Spectrometer (IIS)

This instrument consists of an infrared radiation telescope, two Michelson interferometers for evaluating spectral data, and a radiometer for measuring total body reflection. The IIS will be used to measure planetary thermal emissions, surface composition, and thermal structure. It will accomplish this by measuring reflected solar radiation and heat balances[1,4].

Photopolarimeter

The photopolarimeter gathers information on surfaces or particles by observing how they scatter light. To accomplish this the photopolarimeter must take measurements over a variety of phase angles. This data can be evaluated to find the physical and chemical properties of planetary atmospheres and surfaces. The intensity and polarization of light are measured in 10 narrow bands from 0.41-0.945 microns, including areas where methane and ammonia strongly absorb radiation[1,4].

Ultraviolet Spectrometer (UVS)

The ultraviolet spectrometer operates in two distinct modes: airglow and solar occultation. During Copernicus's cruise phases, the UVS will operate in airglow mode. It will observe the sources of extreme ultraviolet radiation in the galaxy. As the probe enters an encounter phase and passes by a planet, the ultraviolet spectrometer will convert to solar occultation mode. In this mode the instrument will study solar light and the effects a passing planets atmosphere has on it. The UVS covers a 0.115-0.43 micron spectrum and views with a 0.1° slit width. The ultraviolet spectrometer can detect nitrogen, sulfur, and atomic hydrogen and oxygen. Microprocessor control provides flexibility. The UVS can fix at one wavelength and look for intensity changes during a scan, or it can rapidly step through wavelengths for a full spectrum over a broader area - or some combination in between[1,4].

Particles Investigations

The particles studies consist of three distinct instrument investigations. They are a Low Energy Charged Particle (LECP) detector, a High Energy Charged Particle (HECP) detector, and a Plasma Particle (PP) detector. The LECP detector operates with two objectives: measure particles in planetary magnetosphere and to detect low energy charged particles in interstellar space. It accomplishes its objectives by measuring particle source, composition, energy spectra, flux intensity, and favored particle direction. The HECP detector is similar to the low energy charged particle detector, however it measures particles by charge, mass, energy, and arrival direction. The LECP and HECP work with a combined range of 0.020-55 million electron volts for ions and 0.015-11 million electron volts for electrons.

The plasma particle detector consists of two Faraday cup plasma sensors and three mass spectrometers. Its objective is measuring the plasma in the solar wind and in planetary magnetospheres. It is also responsible for finding the plasma flow direction. The PP detector studies plasma by detecting its velocity, density, and pressure. This device measures the energy range of electrons and positive ions from 1.2-50,400 electron volts. The

Faraday cup plasma sensors collect the plasma data, while the three mass spectrometers are included to identify the composition of ions[1,4].

Fields Investigations

The instruments that fall under the fields category are the magnetometers and the plasma wave detector. The magnetic fields investigations employs four magnetometers. This investigation uses two sets of two triaxial fluxgate magnetometers. One set is of low field, the other high field. These magnetometers measure planetary magnetic fields. They measure with a range of 0.00032-0.16384 gauss.

The plasma wave detector will be used to study wave and particle interaction in the dynamics of a planets magnetosphere. The detector measures changes in electric and magnetic fields. The electric and magnetic fields can be measured separately over ranges of 5 Hz. to 5.6 MHz. and 5 Hz. to 160 KHz., respectively[1,4].

Table 5B shows the scientific mission at Pluto/Charon of each instrument Copernicus will be carrying. All equipment will be heated with a combination of strip heaters and passive athermalization with invar and aluminum structures.

Instrument Layout

Instruments will reside in one of three locations aboard the spacecraft. The magnetometer boom, the scan platform, or the scan platform boom. The scan platform and its boom, along with the magnetometer boom were located so as to maximize their distance from each other and from the Radio Isotope Thermal Electric Generator (RTG).

Scan Platform

The scan platform will house the instruments that specifically need to be pointing at the target they are investigating. It will be extended out from the Copernicus by a folding boom. The platform itself will have two axis of freedom about which to rotate. This will

EQUIPMENT	INVESTIGATION CONCERNS	
Imaging	Radius Oblateness Global maps Solid body maps Limb darkening	Map surface albedo Ice to rock ratio Mass Density Terminator region
Infrared Interferometer Spectrometer	Thermal emissions Composition of thermal structure Heat balances Heat flux Mass	Map surface albedo Ice to rock ratio Surface temperature Composition Density
Magnetometer	Harmonic coefficients	Magnetic fields
Photopolarimeter	Physical, chemical properties Mass	Composition Density
HECP Detector	Measure electrons Tail of magnetosphere	Measure cosmic rays Cavity in the solar wind
LECP Detector	Particles in magnetosphere Cavity in the solar wind	Tail of magnetosphere
Plasma Particle Detector	Plasma flow direction Cavity in the solar wind	Tail of magnetosphere
Plasma Wave Detector	Particle interaction Cavity in the solar wind	Tail of magnetosphere
Ultraviolet Spectrometer	Atmospheric composition	

Table 5B Instrument Investigations

minimize the maneuvering required from the spacecraft. The instruments on the scan platform include the narrow angle and wide angle cameras and their electronics, the infrared interferometer spectrometer, the photopolarimeter, the ultraviolet spectrometer, and the plasma wave detector. The equipment will be placed together and bore sighted with the narrow angle camera. By placing the instruments in a cluster, the strip heaters can serve more than one instrument, thereby minimizing power use and cost. Because the equipment will be bore sighted on the narrow angle camera, mission scientists will have an image corresponding to data collected from the other scan equipment.

Requirements are placed on the movement of the scan platform by the science instrumentation, specifically the imaging equipment. The platform can rotate with a maximum slew rate of 0.33° per second. At this rate the instruments with the exception of the imaging equipment can be accurately used after a settling time of 45 seconds. However, if the cameras are to be employed a settling time of 288 seconds is required. The equipment on the scan platform also places a limit to the maximum maneuver rate of the spacecraft. The maximum allowable maneuver rate of Copernicus while performing experiments, except imaging is 0.033° per second. The maneuver rate while imaging drops to 0.00972° per second. Another requirement for the scan platform is its pointing accuracy. The platform must be high precision with pointing accuracy of at least 2 mrad with 1 mrad knowledge and stability of 10 mrad in 0.5 seconds and 100 mrad in 100 seconds. Figure 5C represents a view of the scan platform and its equipment[3,8].

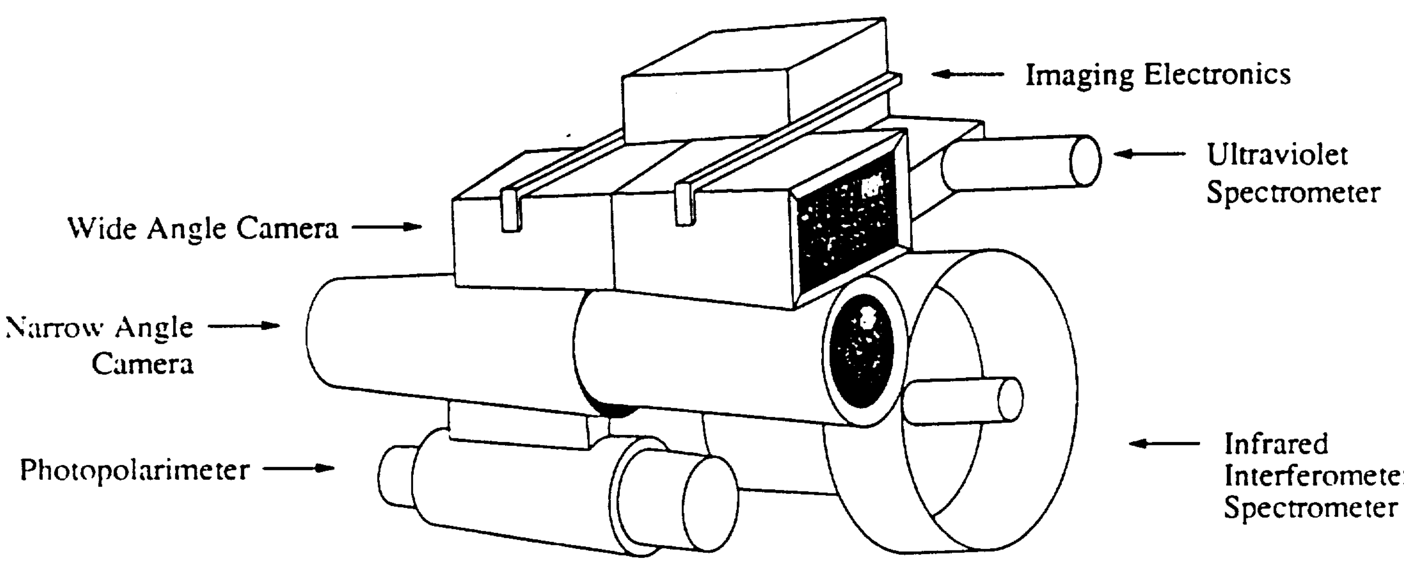


Figure 5C

Scan Platform Boom

The scan platform boom is a convenient location to place the particles instruments. It is away from the spacecraft and allows undisturbed flow through of the interstellar environment. The boom will house the low and high energy charged particle detectors and the plasma particle detector.

Magnetometer Boom

This 13 meter long boom will remove its low field magnetometers from interference with the other science equipment. The magnetometers will be the only instruments placed on this boom. The high field magnetometers will be place on the boom near its attachment to the spacecraft. One low field magnetometer will be located half way down the boom, the other placed at the farthest end.

Table 5C is a listing each instrument and its mass, power requirement, data transmission rate, and location on the probe[4,8].

INSTRUMENT	MASS (kg)	POWER (W)	DATA RATE (bps)	LOCATION
Imaging	36.5	29.0	3850	Scan Platform
Infrared Interferometer Spectrometer	18.5	12.0	500	Scan Platform
Photopolarmeter	13.0	13.0	450	Scan Platform
Ultraviolet Spectrometer	13.0	13.0	450	Scan Platform
LECP Detector	9.0	16.0	450	Scan Boom
HECP Detector	13.8	16.5	450	Scan Boom
Plasma Particle Detector	9.9	8.1	450	Scan Boom
Magnetometer	4.9	5.8	400	Magnetometer Boom
Plasma Wave Detector	1.4	1.6	200	Scan Platform

Table 5C Instrument Data

Conclusion

The mission to Pluto and Charon can only be completed cost effectively by a spacecraft whose science section maximizes accurate data gathering and the number of target investigations, while minimizing mass, power consumption, and complexity. The Copernicus probe meets these requirements.

APPENDIX 5A
References

1. Aviation Week & Space Technology, Galileo Represents Peak in Design Complexity, P77. (October 9, 1989).

2. Croswell, Pluto: Enigma on the Edge of the Solar System, P8-22. Astronomy (July 1986).

3. Jet Propulsion Laboratory, Vol. VI: Imaging Science Subsystem Description; Cassini Mission: Saturn Orbiter Proposal Information Package, P1-8. JPL D-6464 (October 10, 1989).

4. Jet Propulsion Laboratory, Voyager Neptune/Interstellar Mission; Flight Science Office; Science and Mission Systems Handbook, P1-248. California Institute of Technology PD 618-178 (1987).

5. National Academy of Sciences, Conference on Outer Planetary Exploration, P1-12. (1978).

6. Reynolds, The Interiors of Pluto and Charon: Structure, Composition, and Implications, P1209-1212. Geophysical Research Letters Vol. 16 No. 11 (November 1989).

7. Trafton, Pluto's Atmosphere Near Perihelion, P1213-1216. Geophysical Research Letters Vol. 16 No. 11 (November 1989).

8. Science Working Group, Report of the Comet Rendezvous. NASA Technical Memorandum 87564 (1985).

ATTITUDE AND ARTICULATION CONTROL

Introduction

The task of the Attitude and Articulation Control System (AACS) is to control the attitude of the spacecraft. This requires pointing the high gain antenna toward the Earth and/or Sun, pointing the trajectory correction thrusters in any direction, providing control authority during the rocket engine burns, performing science maneuvers, and pointing the scan platform.

These control requirements are very challenging because of the complex and time changing parameters the Copernicus will encounter. Initially, there is the change in mass at separation from the launch vehicle, and then the changes in mass during mid-course correction and orbit burns. Propellant slosh is and wobble amplifications are also factors.

These requirements and the time-varying parameters dictate a complex set of AACS sensors and actuators controlled by a high performance computer, and that a great deal of on-board autonomy be present in the AACS. Also there are weight and power constraints that put stringent requirements on the electronic components. A mission objective is to prevent single-point failures from jeopardizing the mission. This forces redundancy of the critical components and requires internal fault protection logic to control that redundancy.

Without doubt, accurate attitude control of the Copernicus is imperative to mission success. This section describes the attitude control of the Copernicus spacecraft during the entire mission, giving detailed descriptions of the components and methods used in designing the AACS.

Attitude Control Modes

The attitude of the Copernicus is achieved through the use of a set of celestial sensors, a set of inertial sensors, an onboard digital computer, and a set of hydrazine thrusters. The Copernicus will be three-axis stabilized due to the science requirement for a scan platform and the lower cost compared to a dual spin design. Three-axis stabilization also permits extended viewing of selected targets, thus permitting a larger number of individual measurements or a longer integration time for increased sensitivity per measurement than can be achieved with a spin stabilized spacecraft unless it has a de-spun platform.

On account of the length of the mission, the Copernicus must be able to function autonomously for a large amount of its travel time. A basic guideline is that the spacecraft (S/C) be able to operate for at least one week without ground intervention without loss of more than one science instrument or loss of more than one-half the engineering telemetry and the S/C must be left in a commandable state. Therefore it is imperative that the control computer have various fault detection and correction actions when the S/C subsystems experience certain failures, and be able to maintain correct attitude control during these times[6].

A software estimation process has been derived to determine the best spacecraft position, rate, and acceleration estimates in the presence of noise and disturbance processes. Based on these estimates the attitude of the spacecraft is corrected by activating the appropriate hydrazine thrusters. The algorithm for determining the best spacecraft position and rate is described in Appendix 6A [1].

During cruise, the normal response to a fault is to "safe" the S/C in a specifically oriented attitude. However, during critical mission phases, the on-board systems must reconfigure the Copernicus in such a way as to maximize the probability of completing critical sequences (such as burn and science maneuvers). To accomplish various maneuvers necessary in reorienting the Copernicus, a commanded turn capability is implemented. A turn in any of the three axes is accomplished by the insertion of a bias in the control loop during inertial cruise.

Scanning Platform and Pointing Control

The mounting of a science scan platform at the end of a science boom permits the physical tie-down of its mass during launch, provides for mass balancing of the RTG's for spacecraft center of mass control, and maximizes the unobstructed solid angle through which the remote sensing instruments can be pointed. This platform holds all of the science instrumentation and sensor and control components, which have accurate pointing requirements, thereby eliminating many sources of error that have existed on prior spacecraft. Clearly, the pointing performance of this platform is critical to the success of the mission.

Typical pointing requirements for a high precision scan platform (HPSP) are shown in Table 6-A. These requirements are primarily driven from the requirements of the cameras, and apply to each of

Table 6-A. Pointing Requirements

High Precision Scan Platform Requirements	
Inertial Pointing Control	2.0 mrad (0.11°)
Inertial Pointing Knowledge	1.0 mrad (0.06°)
Inertial Pointing Stability (during 0 to 17.5 mrad/sec slew)	10 μrad/0.5 sec 100 μrad/100sec

the two required axes of articulation. These requirements fall well within the requirements for the entire Copernicus mission. The dynamics of the platform boom can be excited by both basebody motion and platform slews. The choice of an appropriate scan actuator which controls this platform, and compensates for disturbances, will be described next.

Scan Actuator

A key element in the mission is the high precision scan platform. On this platform a number of instruments are mounted, including several cameras and the star tracker and gyro used for S/C attitude control. Clearly, the pointing performance of this platform is critical to the success of the mission. The central consideration of a scan actuator can have an impact on the design of the entire spacecraft.

A direct drive actuator with a platform mounted momentum compensation wheel is selected for the Copernicus. This actuator is selected on the basis of net effect on spacecraft mass, required power, cost, expected pointing performance, necessary control complexity, suitability to mission, operational considerations, and ability to accomodate changes in the mission or spacecraft. It is assumed that all actuators considered met the spacecraft reliability and lifetime requirements.

Table 6-B compares four models of possible actuators, including a momentum compensation harmonic drive (MCHD), direct drive, harmonic drive (HDA), and two-motor actuators. It can be seen that

Table 6-B. Scan Actuator Comparison

Criteria	Direct Drive	Two-Motor	MCHD	HDA
Reliability	Least Risk	Acceptable	Acceptable	Unacceptable
Mass	27 KG	51 KG	50 KG	31 KG
Total Power Peak/ Steady State	8W/6W	17W/12W	11W/8W	10W/6W
Performance	1 μrad	16 μrad	7 μrad	N/A
Heritage	Galileo	Pathfinder	Breadboard	Halley Intercept

overall the direct drive actuator is the best choice, with the bonus that it's been space tested on the Galileo.

The reason for the momentum compensation wheel is that a savings in attitude control propellant can result in an overall savings of spacecraft mass for missions requiring a large number of platform slews, such as Copernicus. Thus when the scan platform accelerates in azimuth, the motor-mounted wheel with the required inertia ratio will accelerate in the opposite direction. The elevation axis works the same way. So ideally the spacecraft body will not sense the platform articulation disturbance torques.

The direct drive actuator is the simplest of the configurations considered. It consists of a brushless DC motor mounted at the gimbal joint. Torque is applied directly by the motor to the platform and a reaction torque is applied directly to the basebody[5].

Star Tracker

The development of charge-coupled device (CCD) optical sensors has made it possible to construct high-performance star and target trackers for spacecraft. They offer high resolution, dimensional stability, and both geometric and photometric linearity. The ASTROS-II (Advanced Star/Target Reference Optical Sensor) tracker currently being developed at the Jet Propulsion Laboratory is scheduled to be launched on the Comet Rendezvous Asteroid Flyby mission. This tracker uses the RCA 501 DX CCD, has integral microprocessors to control the data acquisition, make image position calculations, and provide an effective interface to the pointing control computer.

Table 6-C compares available star sensors. The ASTROS-II is based on the ASTROS built for flight on a series of shuttle-based ultraviolet astronomy missions. The revised design will be tailored to requirements of the Copernicus mission. The ASTROS-II has the following capabilities:

a) Tracks several stars simultaneously for attitude reference (up to 5 stars per field).

Table 6-C. Star Tracker Comparison

Characteristic	CS-203	Canopus	ASTROS	ASTROS-II
Mission	VRM	Voyager	Shuttle	Copernicus
Field of View	4.6° wide	9° x 36°	2.2° x 3.5°	11.5°x11.5°
Drift Rate (°/sec)	0.2-1.0	N/A	<.1	<.5
Internal Redundancy	Yes	No	No	Yes
Dimensions (cm)	17x24x18	29x13x11	50x25x20	25x16x16
Mass (kg)	5.5	4.3	2.8	8
Power (w)	7	4.5	38	11

* VRM - Venus Radar Mapper

b) Follows rapidly moving , time-varying , extended targets during a close flyby or rendezvous.

c) Determines the limb position and orientation of a nearby target.

d) Develops image data for ground-based target searches during target approach.

e) Tracks both stars and extended targets and provides optical navigation data for the mission.

f) Mass, power, volume, and environmental compatibility with the Copernicus mission.

These qualities make the ASTROS-II an optimal choice for the Copernicus mission. The unit will be internally redundant and therefore the specifications listed in Table 6-C make it a substantially better choice than all others. The tracker will be located on the HPSP along with the scientific instrumentation[2].

Laser Gyro

The attitude of the Copernicus in three-space is measured by a new technology gyro based on fiber optics, Fiber Optic Rotation Sensor (FORS). Nearly 100 years ago, it was discovered that light, along with conventional gyroscopes, could provide gyroscopic information. The time it takes light to traverse a circular pathway depends on whether the pathway is stationary or rotating. The time difference can measure the amount of rotation[7].

The FORS design uses a single 5 mW GaAlAs laser to input light, divided and injected, into both ends of a 3 to 20 km long fiber waveguide wrapped around an 18 cm coil. After the light has passed through the fiber waveguide, it is recombined and detected. This concept is based on the Sagnac interferometer principle. The phase angle between the two light beams is dependent upon coil rotation rate, direction, number of turns of the fiber, and area enclosed[3].

There will be two sets of three of these gyros for redundancy. The use of this type of gyro results in a planetary gyro with ten times improved drift rate over today's conventional gyros. With the absence of moving parts, no gas discharge tube, and no short term wearout mechanisms, the operating lifetime is well within the mission requirements for Copernicus.

The fabrication processes are relatively inexpensive. The absence of moving parts and close similarity to electronic microcircuit fabrication allow this. The recurring cost of these new planetary gyros is less than one-third of today's conventional gyro cost. The mass, power, and volume will also be

Table 6-D. Gyro Comparison

Unit	Drift Rate(°/sec)	Angular Resolution	Power (w)	Mass (kg)	Volume (cm^3)
FORS	2×10^{-4}	0.005 arcsec	10	10	16400
DRIRU-II	3×10^{-3}	0.05 arcsec	22	11	16236
CG-1300 Laser	7×10^{-3}	1.4 arcsec	18	18	5740

less than present gyros. Table 6-D compares the FORS and two other currently available gyros. The entire gyro component will be placed on the science scan platform for optimal accuracy[3].

Reaction Control System (Thrusters)

The Reaction Control System (RCS) of the Copernicus consists of twelve 1N thrusters located in four clusters about the center of mass of the spacecraft, illustrated in Figure 1-C (Structures Section). The RCS is a monopropellant hydrazine system which has fuel supply lines running from the main propellant bladder. The thrusters are similar to the Voyager design and act as couples. They provide attitude control torques and thrust for small engine maneuvers and trajectory correction maneuvers (TCM), but the main propulsion engine provides most of the control thrust during impulse burns and large maneuvers. The use of four clusters with three thrusters each provides redundancy, designating main and backup sets of thrusters which can be used for control about specific axes. An example of the designated control setup is shown in Figure 6-A and Table 6-E.

The thruster is designed to provide 0.95N thrust and 300s specific impulse at propellant inlet pressure of 24.6 kgf/cm^{2}a to meet the requirements for Copernicus. The thruster has a 60:1 expansion ratio conical nozzle. Thrust level is adjusted by controlling the flow rate of propellant with valves located on the fuel lines. The amount of propellant reserved for attitude control is estimated to be about 5% of the total fuel for the mission. This estimate takes into account the longer duration and therefore many more TCM's which will take place compared to previous missions, but also realizes the greater mass of fuel which is being carried for this mission (compared to other missions)[8].

Thermal design of the thruster cluster uses three catalyst bed heaters and valve heaters to maintain the catalyst bed above 200° C prior to firing. The cluster is designed to be thermally isolated from the spacecraft and minimize heat transfer to the cluster or propellant valve to keep the catalyst bed hot. The thrusters are designed to be

Figure 6-A. Thruster Cluster Configuration

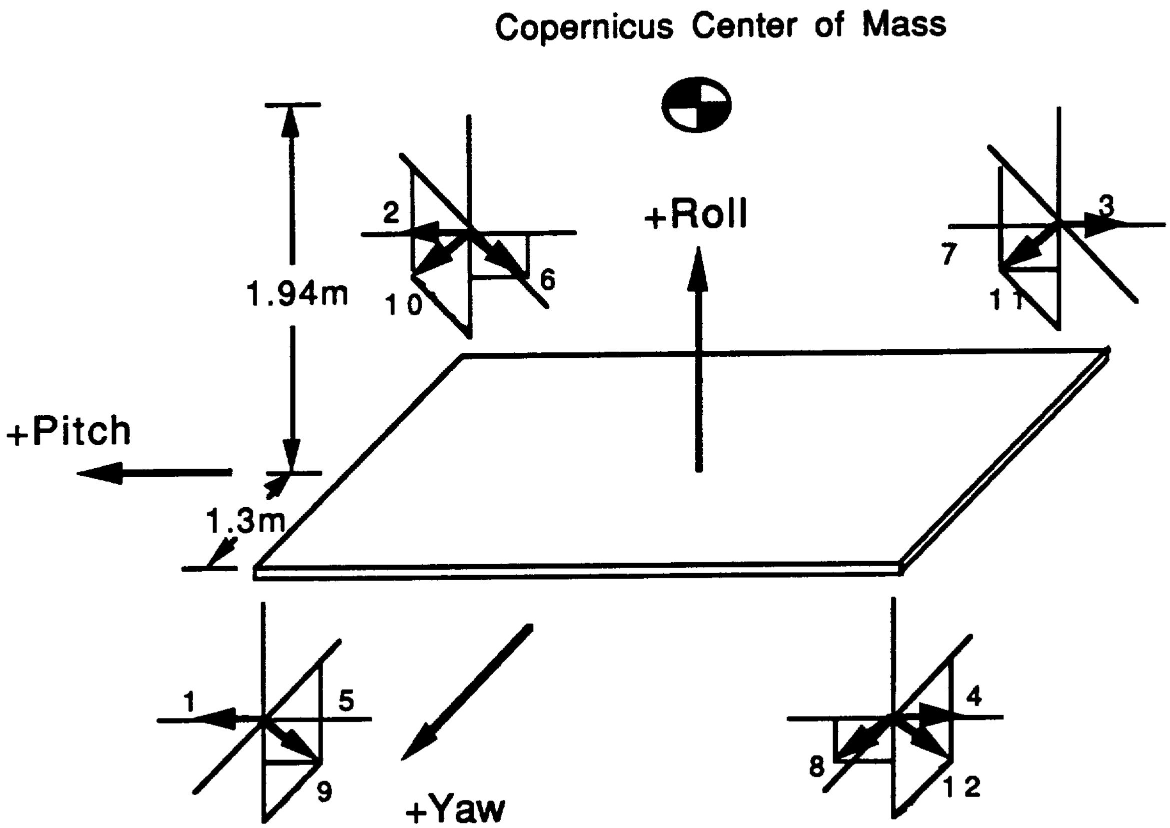


Table 6-E. Thruster Location and Function Matrix

Control Mode	Thruster Number											
	1	2	3	4	5	6	7	8	9	10	11	12
+Yaw					B	B	M	M				
-Yaw					M	M	B	B				
+Roll	M	B	M	B								
-Roll	B	M	B	M								
+Pitch									M	B	B	M
-Pitch									B	M	M	B

capable of achieving mission requirements even in the case of one heater failure.

Algorithms within the main computer control the thrusters to provide three-axis control and to perform closed-loop turns of the spin axis. Such turns may be required up to four times daily to keep the high gain antenna pointed toward Earth, and to orient the spacecraft for TCM's[1].

Conclusion

The Copernicus spacecraft is 3-axis stabilized, using a digital onboard computer, a set of fiber optic gyros, a star tracker, and hydrazine thrusters. Attitude control of the spacecraft is based on measuring spacecraft orientation, estimating spacecraft states, and actuating the thrusters for attitude correction.

The orientation of Copernicus is measured by FORS. The position is calculated using the ASTROS-II. A direct drive actuator with momentum compensation wheel will be used to operate the scan platform. The attitude of the spacecraft will be adjusted with 1N thrusters, located on a structure which surrounds the propellant bladder.

This configuration for the AACS will provide the best control for the long journey the Copernicus will undertake to Pluto.

Appendix 6A. Inertial Control Single Step State State Predictor in Cruise

$$\hat{\underline{x}}_p\,(K/K) = \hat{\underline{x}}_p\,(K/K-1) + \underline{K}_p[\,Mp(K) - \underline{H}\hat{\underline{x}}_p(K/K-1)]$$
$$\hat{\underline{x}}_p\,(K+1/K) = \phi(K+1,K)\hat{\underline{x}}_p\,(K/K) + \underline{\Gamma}_p(K+1,K)T_p(K)$$

$$\phi\,(K+1,K) = \begin{pmatrix} 1 & \Delta T & .5\Delta T^2 \\ 0 & 1 & \Delta T \\ 0 & 0 & 1 \end{pmatrix} \qquad \underline{\Gamma}\,p(K+1,K) = \underline{\Gamma}_p = \begin{pmatrix} .5\Delta T^2/J_p \\ \Delta T/J_p \\ 0 \end{pmatrix}$$

The decision to turn the appropriate thruster on at K+1 is based on:

$$E_p(K+1) = (1 \quad K_{rp} \quad 0)\,\hat{\underline{x}}_p(K+1/K)$$

$\hat{\underline{x}}_p(K/K)$ is best estimate of spacecraft pitch statres at K given measurements $M_p(K)$.

$\hat{\underline{x}}_p(K+1/K)$ is the best one-step prediction of S/C pitch state based on $M_p(K)$.

$\underline{K}_p$ is the Kalman gains.

$\underline{T}_p$ is the estimate of torque developed by pitch thrusters.

Process is sequentially repeated in real time.

For yaw and roll axes, the subscripts p are changed to y or r.

Appendix 6B. References

1. Bahrami, K., "Inertial Attitude Control of Voyager Spacecraft Using Dry Tuned Rotor Gyro", AIAA Guidance and Control Conference, Hollywood, CA, August 8-10, 1977 pp. 520-529.

2. Dennison, E.W., Stanton, R.H., Shimada, K., "Astros: High Performance CCD Tracker for Spacecraft", International Symposium on Space Technology and Science, 15th, Tokyo, Japan, May 19-23, 1986 Vol.2 pp. 1131-1142.

3. Draper, R.F., "The Mariner Mark II Program", AIAA Aerospace Sciences Meeting, 22nd, Reno, NV, January 9-12, 1984, 34 pgs.

4. Heacock, Raymond L., "The Voyager Spacecraft", Proceeding of the Institution of Mechanical Engineering", Vol. 194, 1980, pp. 211-221.

5. Schier, J. Alan, Bell, Charles E., Agronin, Michael, Socha, Michael, "The Mariner Mark II High Precision Scan Actuator: Results of an Options Trade Study Covering Point Performance and System-Wide Impacts", AIAA Navigation, Guidance, and Control Conference, Minneapolis, MN, August 15-17, 1988, pp. 1053-1060.

6. Stephenson, R. Rhoads, "The Galileo Attitude and Articulation Control System: A Radiation-Hard, High Precision, State-of-the-Art Control System", IFAC Automatic Control in Space, Toulouse, France, 1985, pp. 83-89.

7. Stuart, Dr. James P., White, Thomas W., Coffey, Randall E., "Space System Design and Tradeoffs of Ring-Laser Gyroscope Guidance and Control", AIAA Aerospace Sciences Meeting, 25th, Jan. 12-15, 1987

8. Yoshiyama, Iwao, Iida, Chisato, Kon-No, Akira, Takiyama, Tetsuma, Mori, Takashi, Yabuhara, Eiji, "Design and Flight Performance of the Hydrazine Reaction Control Subsystem for the ETS-III", International

Symposium on Space Technology and Science, 13th, Tokyo, Japan, 1984, pp. 221-231.

C-3

CONCLUSION

Conclusion

This proposal for an unmanned mission to Pluto calls for the spacecraft Copernicus to be launched on May 4, 2009 on a 12.6 year journey through the outer Solar System with flybys of Jupiter and Saturn before it reaches its (possible) final destination of Plutoian space.

The proposed design adheres to the previously stated mission requirements and special emphasis was put on optimizing performance, reliability, and mission cost.

This proposal is only a Phase A design report, but it does provide the initial research necessary for later more detailed mission concepts and designs.

[redacted]

Correlation of Primary Design Issues

Primary Design Issue	Related Design Requirements	Options Considered	Rationale for Option Selected
Power Source	Power Requirements Environmental Concerns Interaction with other Subsystems	Only RTG's (MITG)	Length of Mission Modularity increases efficiency over conventional RTG's.
Propulsion	Type of Fuel Used Configuration of Propellant Unit Tank Sizing Interaction with other Subsystems	Electric Propulsion Nuclear Propulsion Solid Propellant Liquid Monoprop. Liquid Biprop.	Flight tested, space storable, simple
Mission Type	Maximize Science Data Minimize Cost	Flyby Orbiter Lander	Attractive delta-V, reliable, simple, low cost
Launch Vehicle	Meets Requirements of Copernicus Reliable, Safety Low Risk	Atlas G-Centaur Titan 34D, Centaur DIT	Satisfies minimum requirements
Orientation Measurement	Reliable, Low Cost Accurate	FORS Laser Gyro Conventional Gyro	Low Cost, Weight Very Accurate Flight Tested By 2000
Star Tracker	Reliable, Low Cost Accurate	ASTROS-II Canopus	Low Cost, Weight, Power Very Accurate Flight Tested By 2000
Scan Actuator	Simple, Reliable Low Cost	Direct Drive Harmonic Drive Two-motor	Simple, Reliable, Low Cost Worked well on Galileo
Attitude Thrusters	Reliable Center of Mass Considerations Low Cost	1N 3N 10N	Length of Mission Simple, Reliable Allows for Redundancy

Correlation of Primary Design Issues (cont.)

Primary Design Issue	Related Design Requirements	Options Considered	Rationale for Option Selected
Material Selection	Micrometeoroid Protection Contamination Protection Radiation Protection	Composites Aluminum Titanium Beryllium	Combination of all to fulfill the variety of requirements.
Thermal Control	Provide Proper Thermal Environ. for Components	Printed Circuit Strip Heaters Multilayer Insulation Blanket Radiating Louvers	Combination of all three for redundancy.
Placement of Components	Scan Platform must have clear view Antenna must be clear COM must be on line of force from main thruster	Nine different combos of components are considered	Ninth configuration was selected because its COM was on the line of force of main thruster. It also fulfilled other requirements.
Antenna Sizing	Proper Communications with Earth	<u>3.7m diameter</u> Other various dia.	Minimum diameter to meet requirements of mission
Accurate Video Imaging	Pointing Accuracy Cost Reliability	Voyager Cameras Galileo Cameras <u>Cassini Cameras</u> New Design	High Tech Data Compressing will be readily available
Type of Science Experiments	Accurate information requested by science	Penetrator/Sample Gathering Atmospheric Investigations <u>Accurate Measur. of planet data</u>	With no previous mission to Pluto it seemed prudent to make general studies and not send unnecessary equipment.