P97

N91-16047

Intrepid: A Mission to Pluto

Aerospace Design Class
AAE 241
April 24, 1990

Group 2

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Introduction

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

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

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

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

SCIENTIFIC INSTRUMENTATION (SI)

Mission Science Objectives

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

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

Science Objectives at Pluto

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

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

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

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

Limitations and Requirements of the Mission .

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

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

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

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

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

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

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

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

Science Instruments

The following science instruments have been selected:

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

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

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

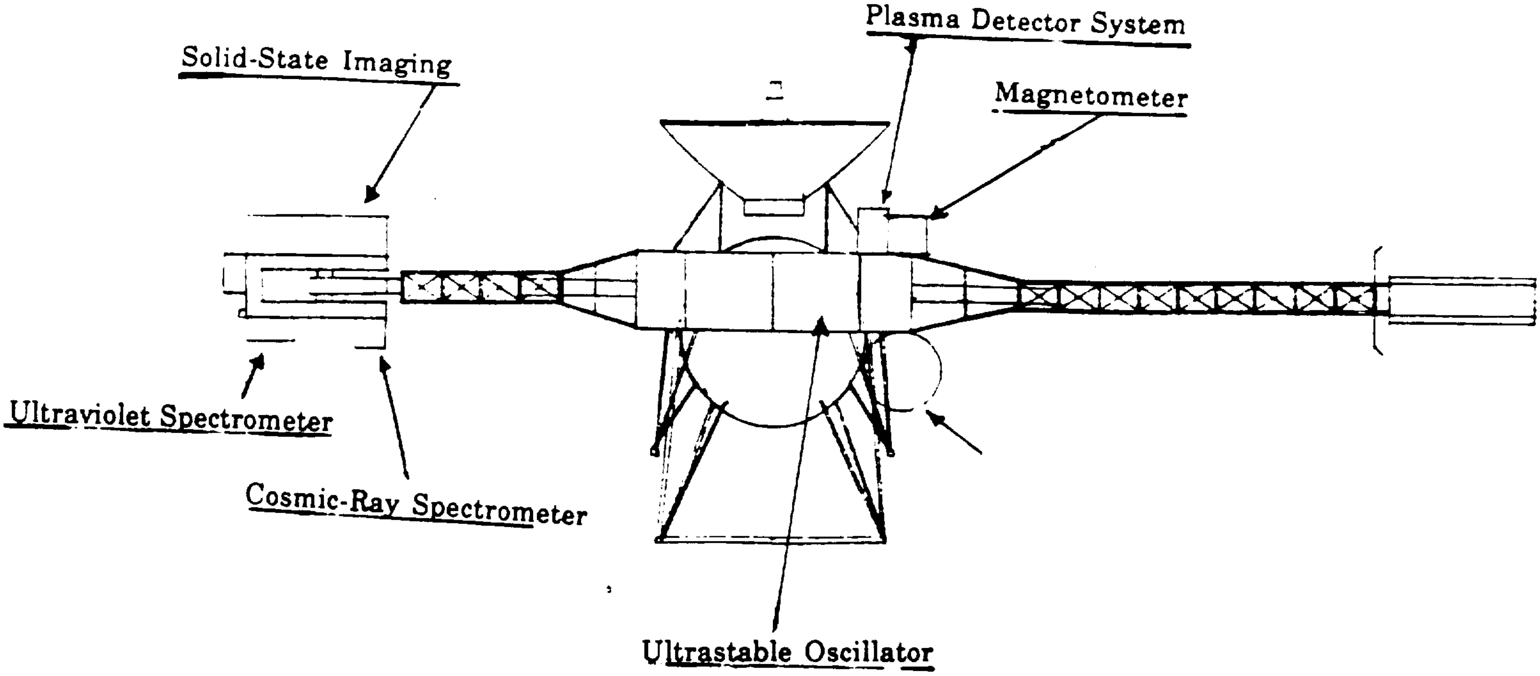

Fig. 1.1 Layout of Scientific Instrumentation

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

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

The Cosmic-Ray Detector System

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

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

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

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

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

The Plasma Detector System

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

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

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

Magnetometer

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

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

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

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

Ultrastable Oscillator (for radio science experiments)

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

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

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

Solid-State Imaging

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

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

Ultraviolet Spectrometer

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

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

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

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MISSION MANAGEMENT, PLANNING AND COSTING (MMPC)

Requirements of the Mission

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

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

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

Type of Mission

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

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

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

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

Trajectory Determination

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

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

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

Table 2.1: Parameters for trajectory consideration

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

* Least expensive compatible launch vehicle was selected

Trajectory Analysis

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

Table 2.2: Launch parameters for the *Intrepid* spacecraft

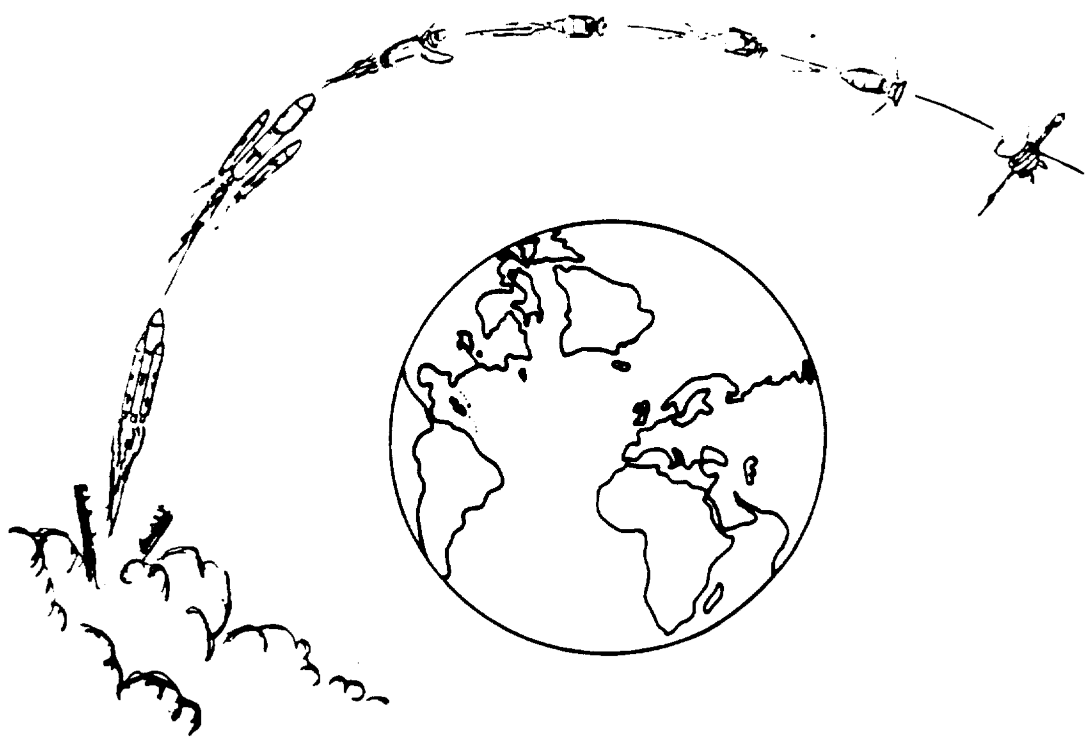
Launch Parameter	Value
Launch Energy	68.4
Launch ΔV	5980 m/s
Launch (wet) Mass	1386 kg
Propellant at Launch	800 kg
Launch Vehicle	Titan IIID/Centaur
Launch Vehicle Cost	$130-140 Million

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

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

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

Figure 2.1: *Intrepid* launch sequence



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

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

Table 2.3: Summary of trajectory encounters

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

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

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

Mission Costing

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

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

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

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

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

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

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

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

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

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

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

Development Project - Flight Hardware

Structure & Devices

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

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

Thermal Control, Cabling & Pyrotechnics

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

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

Propulsion

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

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

Attitude & Articulation Control

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

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

Telecommunications

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

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

Antennas

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

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

Command & Data Handling

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

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

RTG Power

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

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

Landing Radar/Altimeter

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

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

Line-Scan Imaging

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

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

Particle & Field Instruments

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

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

Remote Sensing Instruments

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

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

Development Project - Support Functions

System Support & Ground Equipment

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

Launch + 30 Days Operations & Ground Software

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

Imaging Data Development

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

Science Data Development

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

Program Management/MA&E

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

Flight Project

Flight Operations

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

Data Analysis

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

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

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

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

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

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

Inheritance Class Categories

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

Cost Reduction Algorithm by Inheritance Classes

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

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

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

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

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

Appendix A2.4: *Intrepid* cost estimates

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

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POWER AND PROPULSION SUBSYSTEM (PPS)

Propulsion Subsystem

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

Propellant and Pressurant

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

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

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

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

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

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

Propulsion Feed System

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

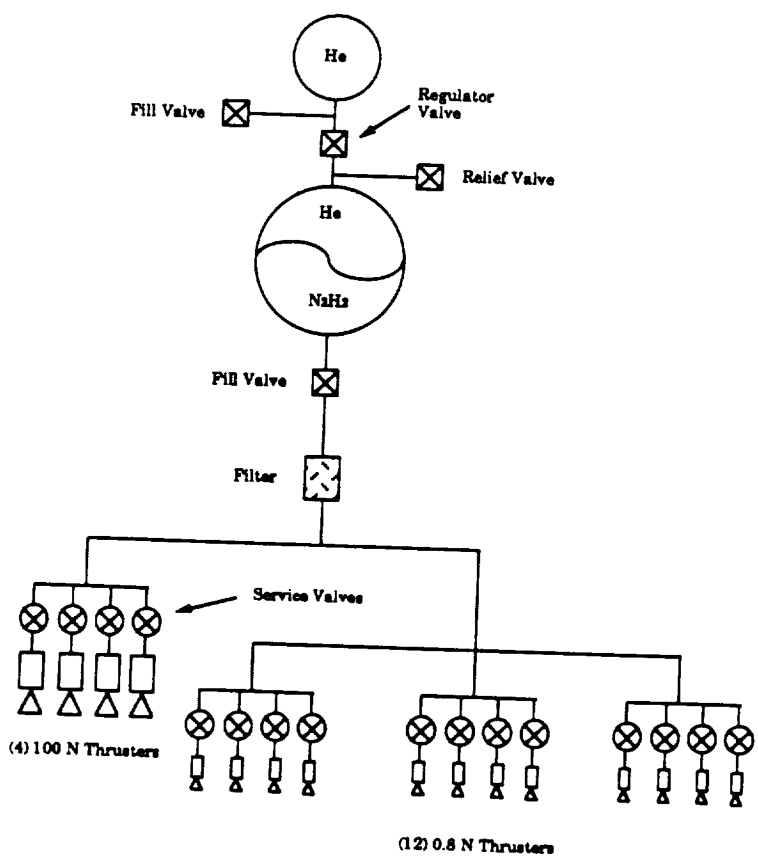

Figure 3.1. Propulsion System Configuration

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

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

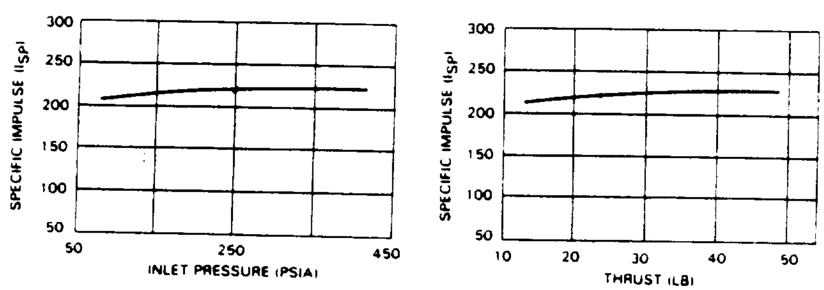

Figure 3.2. Inlet Pressure versus Specific Impulse (TRW)

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

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

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

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

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

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

Thrusters

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

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

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

Further Comments

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

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

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

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

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

Power Subsystem

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

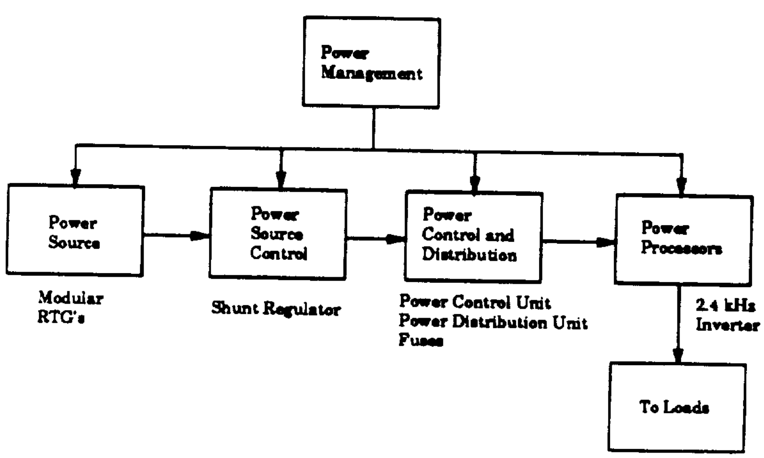

Figure 3.3. Schematic of Power Subsystem

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

Table 3.1. Breakdown of Power Requirements [watts]

Power Generation

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

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

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

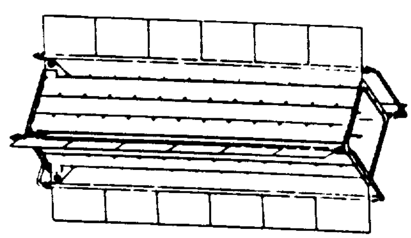

Figure 3.4. Modular RTG Design (Schock, 338)

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

Energy Storage

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

Power Conditioning

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

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

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

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

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

Further Comments

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

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

Launch Vehicle Subsystem

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

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

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

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

Table 3.2. Launch Vehicle Information+

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

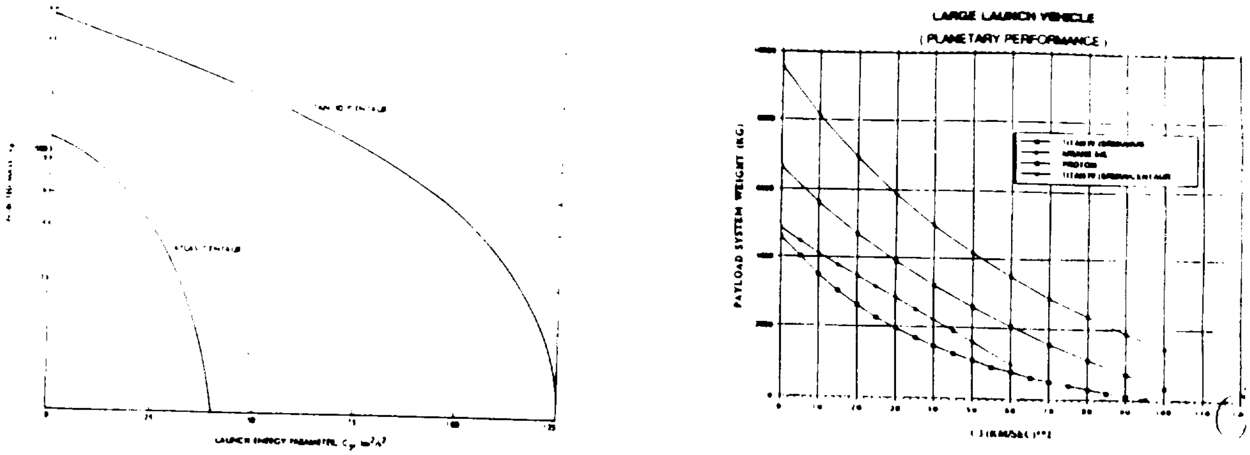

Figure 3.5. Titan IIID/Centaur and Titan IV/Centaur Launch Energies versus Launch Mass (Atkins, 43 and JPL Mission Group)

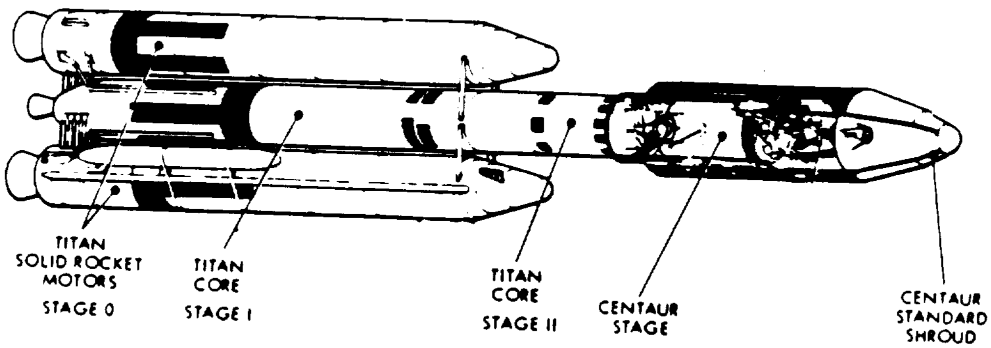

Figure 3.6. Titan IIID/Centaur*

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

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

Further Comments

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

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

Appendix A.3 - Equations

Propulsion Subsystem

- Equations for propellant mass and volume required:

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

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

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

Volume of propellant = mass propellant / density of propellant

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

- Equation for pressurant mass and volume required:

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

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

- Propellant and pressurant tank sizing equations

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

- Thruster sizing

$$\text{acceleration} = {}^{m_{S/C}}/_{thrust}$$

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

4c

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STRUCTURE SUBSYSTEM (SS)

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

Structural Design Subsection

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

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

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

Figure 4.1 Subsystem Necessities

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

Figure 4.2 Structural Design Considerations

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

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

Intrepid *Spacecraft Design*

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

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

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

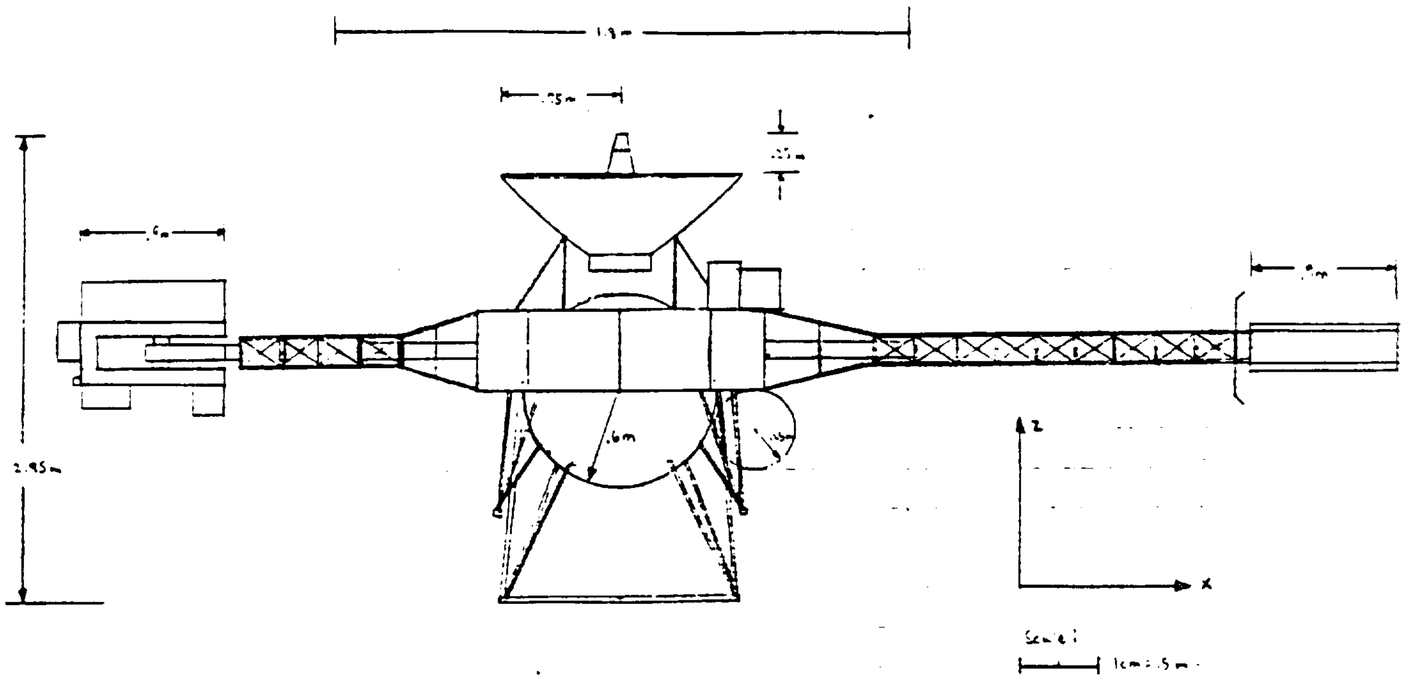

Figure 4.4 *Intrepid* Spacecraft Flight Configuration (x-y plane)

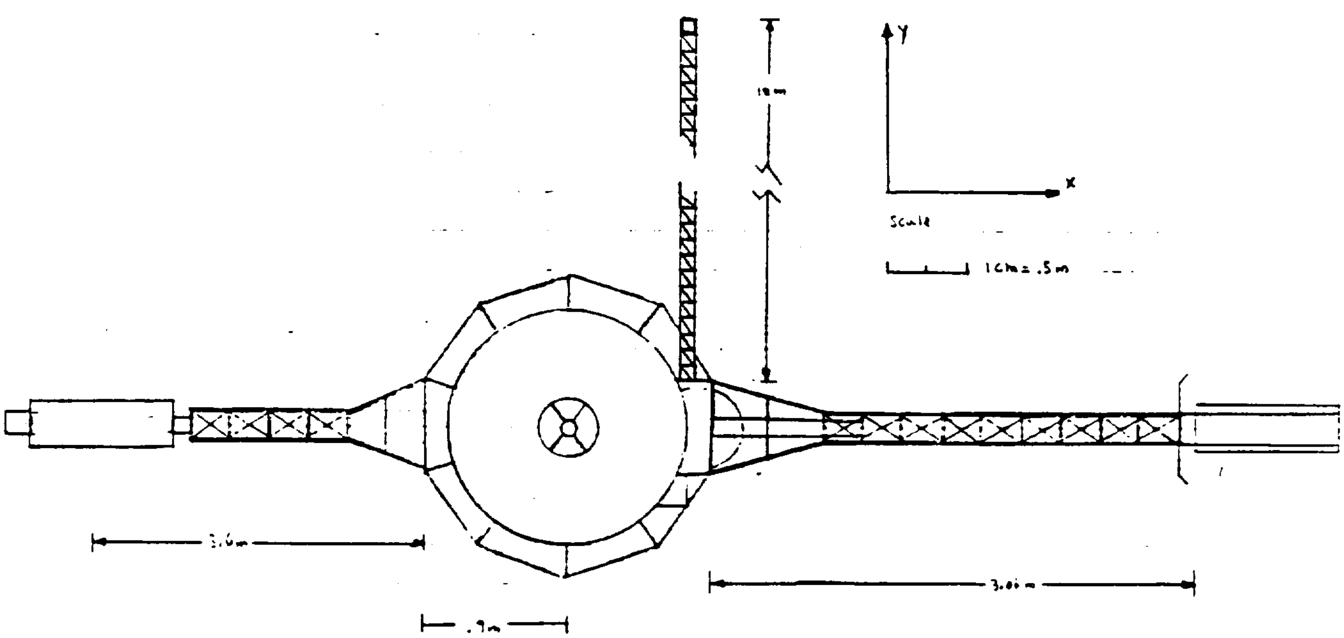

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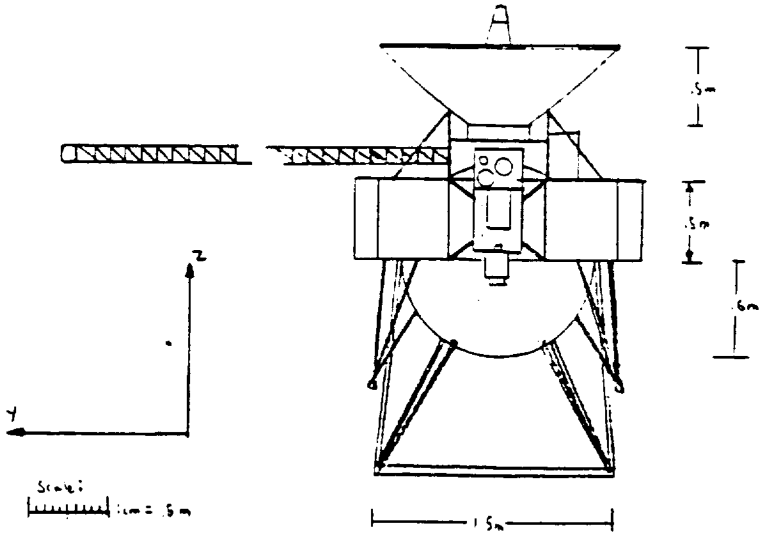
Figure 4.5 Intrepid Spacecraft Flight Configuartion (y-z plane).



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

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

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

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

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

Launch Vehicle Compatibility

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

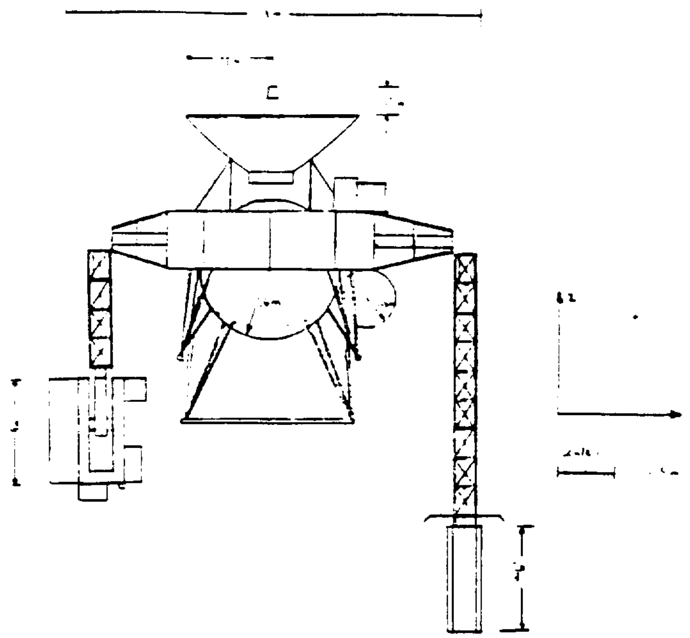


Figure 4.7 *Intrepid* Spacecraft in Launch Configuration

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

Launch Vehicle Survivability

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

On-Orbit assembly

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

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

Thermal Control Subsection

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

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

Table 4.1 Method of Achieving Thermal Control

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

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

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

Bus & Science Thermal Control

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

Propulsion Unit Thermal Control

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

Materials Subsection

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

Table 4.2 Environmental Considerations

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

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

Table 4.3 Materials Considered

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

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

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

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

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

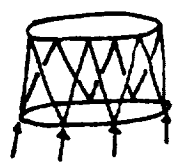
Fiberglass is selected for use in the magnetometer boom because of its unique launch configuration and on-orbit assembly. Fiberglass was also used for the *Voyager* magnetometer boom thus reducing development cost

Appendix A.4.2

Launch loads and axial loads carried by adapter truss.

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

(where $g=9.8\ m/s^2$)



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

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

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

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

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

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

$$59270.4\ N > 52959.2\ N$$

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

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

Micrometeorite Protection

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

Further Comments: Conclusion and Recommendations

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

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

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COMMAND,CONTROL, AND COMMUNICATIONS (CCC)

Communication Subsystem

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

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

Radio Frequency Subsystem

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

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

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

Antenna Subsystem

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

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

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

TABLE 5.1

Performance Parameters for *Intrepid* Radio System

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

Further Comments

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

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

Command and Control Subsystem

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

Command and Data Subsystem

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

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

Computer Command Subsystem

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

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

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

Flight Data Subsystem

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

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

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

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

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

Data Storage Subsystem

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

Microprocessors

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

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

Data Rates

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

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

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

Data Memory

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

Further Comments

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

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ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM (AACS)

Introduction

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

Method of Stabilization

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

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

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

High Precision Scan Platform (HPSP)

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

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

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

Inertial Reference Unit (IRU)

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

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

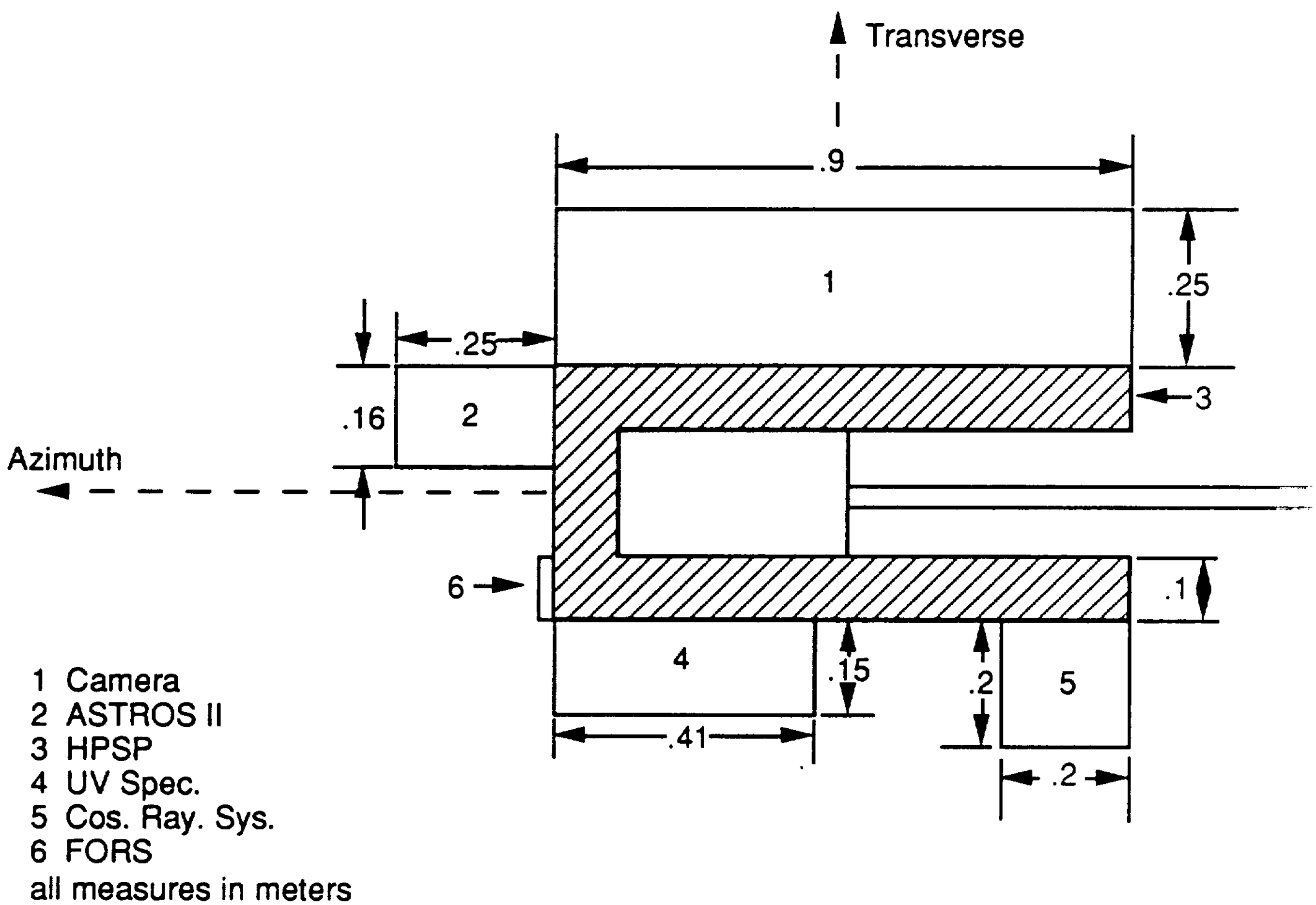

Figure 6.1 HPSP with attitude and science instruments

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

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

Table 6.1 Comparison of rate sensors (Draper, 14)

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

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

Star Sensor

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

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

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

Sun Sensor

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

AACS Computer

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

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

Reaction Control System

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

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

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

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

Mission Modes

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

Separation from launch vehicle

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

Cruise mode

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

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

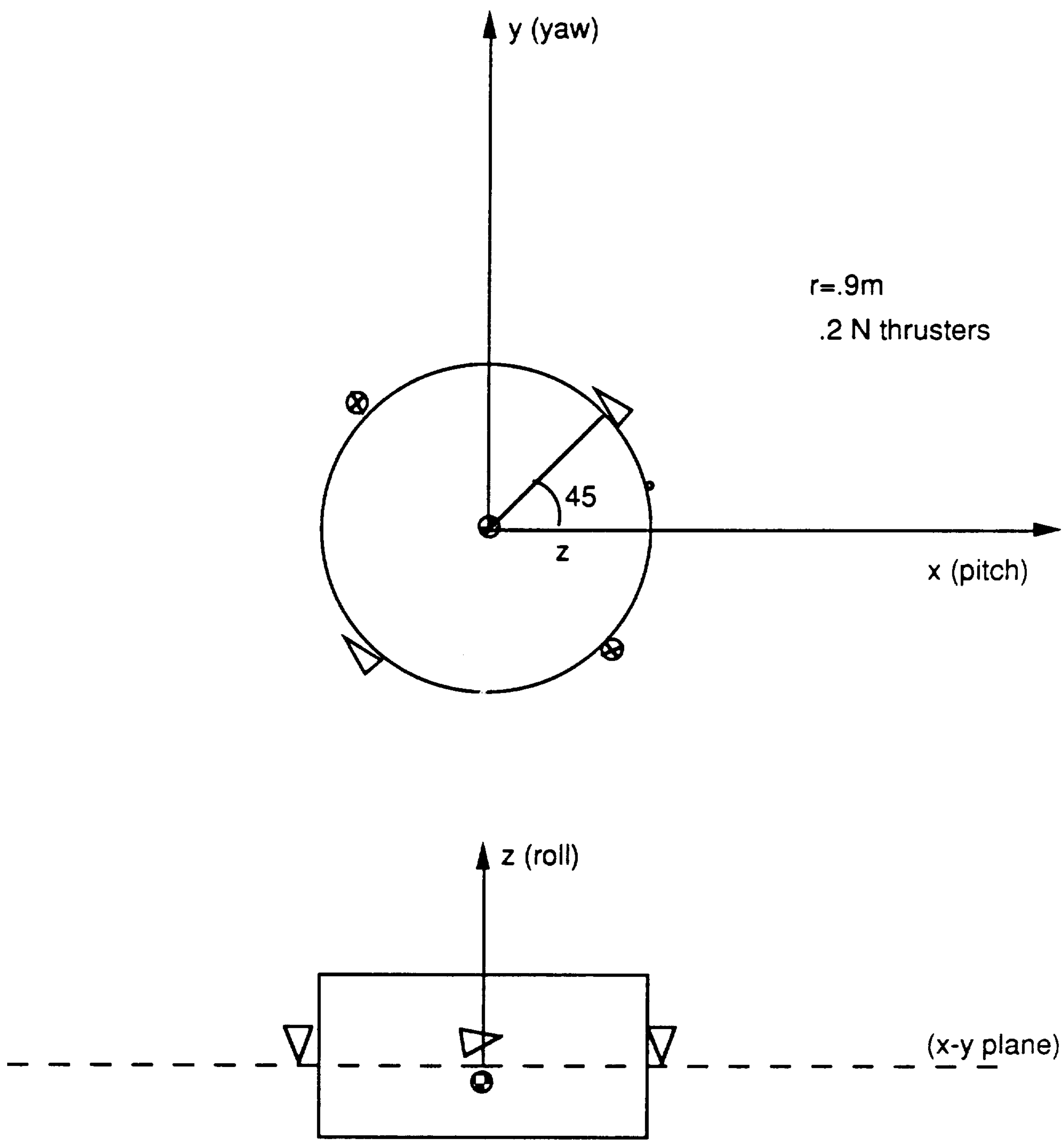

Figure 6.2 Bus thruster layout

Planetary encounter

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

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

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

Delta-v burns

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

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

Figure 6.3 Mission modes

Conclusion

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

Appendix A.6 .1

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

Equation:

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

Solve for F, using:

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

then $F = .0054$ N

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

Appendix A.6.2

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

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

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

then $H_t=1.264$ N m sec/sec

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Conclusion

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

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

Appendix B - Mass and Power Requirements

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

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

Appendix C - Trade Studies

RFP Requirements

Design Choices

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

Appendix D - Acronyms

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