SMALL SPACECRAFT ACTIVITIES AT JPL

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ABSTRACT

This paper presents a brief technical description of some of the small spacecraft concepts prepared by JPL for various sponsors. Some of JPL's work in microspacecraft is presented. The paper contains brief technical descriptions of the following four small spacecraft conceptual designs: 1) Lunar GAS, 2) Polar Mesoscale Explorer, 3) DARPA SHF and 4) Discovery. Since 1986, JPL has studied more than 10 small spacecraft including those to be presented here.

INTRODUCTION

The purpose of this paper is to review some past and present activities at JPL in small spacecraft. In this paper; standard, small and microspacecraft are defined to have a dry mass (without propellant) of approximately 1000, 100 and 10 kg respectively.

JPL has been involved in small spacecraft since the start of the "space age". JPL's first spacecraft in 1958, Explorer 1, had a mass of only about 5 kg. The trend since Explorer 1 has been to larger spacecraft as launch vehicle capability increased. This trend is clearly presented in figure 1. Figure 1 presents the dry mass of all planetary spacecraft launched by NASA versus their date of launch. Figure 1 includes the early Pioneer planetary spacecraft; the series of Mariner spacecraft designed and built by JPL including Voyager, Galileo and the most recent of the Mariners CRAF and Cassini that are presently being designed at JPL. figure 1 also includes the recently launched Magellan and soon to be launched Mars Observer spacecraft both designed and built by JPL contractors Martin Marietta and General Electric respectively.

There are two clear trends shown in figure 1 i.e., the dry mass of planetary spacecraft has increased by over a factor of ten and the launch frequency has dramatically decreased with time.

Certainly, there are many factors responsible for these two trends. One factor is that as spacecraft grow in mass (and inevitably capability and complexity) their cost also grows. Even though the ratio of cost to capability may go down, the absolute cost of the spacecraft goes up and has gone up faster than the financial resources available to support such programs. The situation of more costly spacecraft programs and a relatively fixed amount of resources leads to less frequent programs. This situation is viewed with alarm by some people. Dr. Freeman Dyson of the Princeton University Institute for Advanced Study has independently stated a similar view as follows, "I do not believe that a fruitful future for space science lies along the path we are now following, with space missions growing larger and larger and fewer and fewer and slower and slower as the decades go by."[1]

One obvious approach to counter this situation is to plan and carry out less costly programs. To the extent that spacecraft mass and cost correlate, less costly programs imply smaller spacecraft.

The attractive features of small compared to large spacecraft were recognized at JPL at least as long ago as 1979[2]. The term "microspacecraft" was used in 1981[3] on a study of a small spacecraft (about 50 kg) intended for observations of the Sun. In 1987, JPL began a study of using the Shuttle Get-Away-Special canister as a "launcher" for a small spacecraft to study the Moon. This study was called Lunar GAS. The Lunar GAS study evolved into a more traditional

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system design called Lunar Prospector[14] which with a dry mass of 265 kg was hardly a small spacecraft anymore.

In 1988, this writer re-invented the term "microspacecraft" and applied it to very small spacecraft that would use advanced technology being developed by the Strategic Defense Initiative Organization. Also in 1988, JPL contributed a spacecraft conceptual design to proposal made by UCLA[10] to NASA for a Polar Mesoscale Explorer. Four small spacecraft (40 kg) would be placed into polar orbit around the Earth by a Scout launch vehicle.

During 1990, JPL performed a study for DARPA of a small spacecraft to be used for SHF communications. Within the past year, the concept of small spacecraft, (not necessarily microspacecraft), has become more accepted within JPL and NASA and some serious studies have been sponsored. The Discovery Program is being proposed as a new initiative by NASA's Solar System Exploration Division. The current mission objectives of the Discovery program are to investigate near Earth asteroids and comets.

The remainder of this paper will present a short summary of JPL's work in microspacecraft and our conceptual spacecraft designs for: 1) Lunar GAS, 2) Polar Mesoscale Explorer, 3) DARPA SHF and 4) Discovery. Since 1986, JPL has studied more than 10 small spacecraft including those to be presented here.

MICROSPACECRAFT

Technology developments sponsored by the Strategic Defense Initiative Organization (SDIO) led this writer to assert that such technology could be employed to enable a microspacecraft whose mass would be about 10 kg. In July 1988, NASA and SDIO sponsored a workshop at JPL titled "Microspacecraft for Space Science". The results[4,5] of the workshop are presented below.

1) Microspacecraft (1-10 kg) are technically feasible.

2) There is a class of scientific and exploration missions that can be enabled by microspacecraft. This class of missions requires many simultaneous measurements displaced in position, as on the surface of a planet or small body or in a region of space. The enabling feature of microspacecraft is the assertion that using many microspacecraft (1 - 10 kg) will cost less (spacecraft and launch costs) and involve less risk than using large (500 - 1000 kg) spacecraft for such missions.

3) Other missions enabled by the microspacecraft concept are those that require very high mission delta-V's.

4) While useful and perhaps enabling for the types of missions mentioned above, microspacecraft are not applicable to all types of space exploration and science and should not be viewed as a panacea.

The primary source of microspacecraft technology is the SDIO work in small kinetic energy projectiles. These projectiles are envisioned as being space based and launched by chemical rockets. The SDIO projectile concepts include power, propulsion, guidance, structure, and command and control components. These projectiles have remote sensing instruments that enable them to carry out their mission. While these projectiles are really "space capable missiles" not microspacecraft, they do contain many typical spacecraft components in miniature form. The reader is referred to references 6 through 10 for more information on microspacecraft missions, systems and technology.

Before microspacecraft can become a reality for long lived space-science missions, the SDIO developed projectile technology will need to be augmented to include a long duration power source, the ability to return data over interplanetary distances and, most importantly, micro science instruments.

Conceptual designs for microspacecraft power and telecommunications subsystems have been completed at JPL[7]. Presently the Advanced Spacecraft System Concepts Group at JPL is creating a microspacecraft concept for an asteroid flyby mission.

LUNAR GAS[11]

During 1987, JPL created a proposal to NASA to launch a spacecraft to the Moon using a shuttle Get Away Special

(GAS) can. The spacecraft was to use xenon ion thrusters for propulsion to the Moon and carry a gamma ray spectrometer as the only science instrument. The mission starts with ejection of the spacecraft from the GAS canister. Two fixed and opposed xenon ion thrusters are used alternately on opposite quadrants of the orbit to raise the orbit altitude, transfer to the Moon and obtain a low, polar orbit at the Moon. The final orbit is circular at 100 km, with a 95 degree inclination. The trip time to this final orbit is 2 years. One year of orbit operations was planned for this small, simple spacecraft. During operations at the Moon, gamma ray spectrometer data is recorded on board in a small solid state memory when the spacecraft is out of sight of the Earth. When a tracking station is available, real time data is transmitted to the ground at 500 bps along with the data previously recorded for a total downlink data rate of about 1 kilobit/second.

The major challenges to the spacecraft designer were the mass and volume limits of the GAS canister, the incorporation of solar electric propulsion and the high radiation exposure in the Van Allen belt. A goal of 150 kg or less was imposed very early in order to keep the total trip time to the Moon about two years.

The Lunar GAS spacecraft mass summary is shown in Table 1. The spacecraft is spin stabilized with the spin axis pointed generally toward the Sun so that the solar panels are continuously illuminated except for occultations. Figure 2 illustrates the fully deployed spacecraft. Precession of the spin axis to follow the Sun is achieved by modulating the thrust of the two opposed ion engines which are parallel to, but offset from the spin axis. Cold gas thrusters provide impulses for spin up and initial Sun acquisition.

Power processing, command, control and data electronics are integrated on a set of boards in a single electronics box. This box also acts as a major system structural element. Power processing is done by a single set of electronics and processed power is distributed to spacecraft users. The only exceptions to this are the ion engines, which have a dedicated power processor due to the required high voltages, and existing subsystem designs such as the transponder which have internal power supplies. Figure 3 illustrates the internal arrangement of spacecraft elements in the stowed configuration.

The ion propulsion system is based upon the SERT II designs flown in the 1970s. Necessary design modifications of the flight proven SERT II, 15 cm engines include changing the propellant feed system from mercury to xenon, incorporating high current density, dished accelerator grids and updating lifetime and efficiency of the cathodes and neutralizer. Initial thrust provided by these engines is up to 42 milli-Newton.

The solar array is made up of thin silicon cells on a thin flexible substrate. The array is folded against the spacecraft body while in the GAS canister and then deployed in an accordion-fold manner after ejection from the Shuttle. The array is supported on deployable booms which when stowed, are flat, prestressed metal strips rolled onto a spool. When deployed the strips unroll, forming a tubular boom. Each wing of the solar array is about 0.8 m wide by 8 m long.

The communications link uses two low gain antennas, standard S-band transponder design and the 26 meter subnet of the NASA/JPL Deep Space Network. The spacecraft equipment is contained within a monocoque external shell which provides the most mass efficient approach for a small, tightly integrated structure.

The Lunar GAS mission and spacecraft proposal was innovative, made use of new technology and was low cost. Unfortunately, NASA support for the concept was not received.

POLAR MESOSCALE EXPLORER[12]

In 1988, a small team of JPL engineers contributed a spacecraft conceptual design to a proposal made by UCLA to NASA's Small Class Explorer Program. The title of the proposal was "A Polar Mesoscale Explorer". Four small spacecraft would be placed into polar orbit around the Earth by a Scout launch vehicle to investigate large scale plasma structures.

The mission objective was to operate four spacecraft in a low Earth, near polar orbit for at least 1 year. At least four spacecraft were required in order to support the science investigation. The orbit must precess such that the spacecraft encounter all local Sun times. It is also required, that the four spacecraft be arranged in the cross-track direction. The separation distance between the ends of the cross-track formation of spacecraft was to change over the mission duration from less than 10 km to at least 100 km. Spacecraft separation in the along-track direction should be 10% or less of the cross-track distance. Science observations will be made and the data will be recorded whenever the spacecraft are within about 45 degrees of the either the North or South pole.

Four spacecraft were to be launched by the four stage Scout launch vehicle from the Vandenberg launch site into a nominally circular orbit with an altitude of 500 km and an inclination of 80 degrees. After separation from the Scout, despin "yo-yo's" would be deployed to reduce the spin rate of the undeployed spacecraft. The top pair of spacecraft (see figure 4) would be separated from the bottom pair. The spacecraft sun sensor would be utilized to determine when the two pairs of spacecraft are properly aligned. Separation springs would be used to give the spacecraft cross-track velocity in order to approximately, evenly distribute the spacecraft in the cross-track direction.

It is anticipated that natural orbit perturbation forces will cause the spacecraft constellation to drift apart. This natural drift is consistent with the science objectives of the mission i.e., to investigate mesoscale features up to 100 km in dimension. The baseline approach is to rely mainly on the natural forces to distribute the four spacecraft 100 km cross-track. A small propulsion subsystem is included in the spacecraft design to maintain the along-track separation to no more than 10% of the cross-track and to provide some drag make-up capability.

The spacecraft concept for the Polar Mesoscale Explorer mission was nadir oriented and gravity gradient stabilized. Figures 4 and 5 present the launch and deployed configurations respectively. The basic shape of the spacecraft is a cube 35 cm on a side. The instruments were to be deployed on the end of the 3 meter booms in the nadir and zenith directions.

The basic deployed configuration of the spacecraft provides the attitude control functions, via passive means. The instruments and the booms together provide the spacecraft inertia characteristics that enable gravity gradient stabilization about the two axes perpendicular to the nadir axis. The spacecraft would be stabilized about the nadir axis by the solar array which would be deployed in the anti-velocity direction. The solar array would act as a "weather vane" when acted upon by the small amount of atmosphere. This "weather vane" action as well as the spacecraft rotation, (once per orbit), about the axis perpendicular to both the nadir and velocity directions would keep the spacecraft controlled about the nadir axis. A Sun sensor was to provide knowledge of the direction to the Sun and the angle about the nadir axis.

The power subsystem provided unregulated power. The solar array was sized to produce adequate power to enable a spacecraft energy balance with all spacecraft/Sun attitude combinations. The solar array that trails the spacecraft has several panels that fold out in order to provide power in any spacecraft orientation. The exterior of the spacecraft is also covered with solar cells which will produce a small amount of power at the poles in the noon orbit. The solar array performance was assumed to be 126 w/m^2 and 31 w/kg for silicon cells (both end of life values). A 6 amp hour nickel cadmium battery was included to supply power during the periods of Sun eclipse and at the poles for the noon orbits.

The design included a small nitrogen gas propulsion subsystem to provide thrust for initial attitude acquisition and orbit maintenance. The propulsion system was sized to produce 3 meters per second of velocity. The telecom subsystem provided the capability to receive, detect, acquire and pass on to the command and data subsystem commands from the ground. The telecom subsystem also transmitted the science and GPS data to the ground.

The command and data handling subsystem was designed to have the capability to receive, send, store and execute commands; store data;

execute algorithms pertaining to the operations and health of the spacecraft and process science and GPS data for transmission to the ground. Solid state memory was used to store more than one day's worth of science data at a rate of 5 kbps for at least 12 hours. The data was to be transmitted to the ground during the daily ground station contact period.

The basic spacecraft structure was to be aluminum honeycomb panels that are available as standard products from many aerospace industry suppliers. The instrument booms, canisters and lanyard deployers, while smaller than flown before, were of standard design.

The telecom subsystem was designed to transmit data to the ground station at a rate of 1 Mbps using S band and low gain antennas. The transponder is the NASA standard S band transponder, near Earth version. The GPS receiver and antenna were to be used in order to provide the position of the four spacecraft relative to one another.

The short lifetime requirement and the multiple spacecraft allowed a single string design i.e., no redundancy. A mass summary is shown in table 1. Although the mass margin would be small, four of these small spacecraft could be launched on the Scout whose minimum capability to a 500 km circular orbit at 80 degrees inclination is 178 kg.

A proposal for a Polar Mesoscale Explorer including the spacecraft concept described above was submitted by UCLA to NASA in October 1988 but was unsuccessful.

DARPA SHF[13]

During 1990 JPL was sponsored by the Advanced Space Technology Program of the Defense Advanced Research Projects Agency (DARPA) to create a spacecraft system concept to support real-time, mobile, tactical communications in the SHF frequency band. DARPA directed the use of either the Pegasus or Taurus launch vehicles. Mission analysis determined that a Molniya orbit (12 hour period, elliptical, 63.4° inclination) was most desirable. Taurus could place over 1100 kg into 500 km, circular parking orbits appropriate for injecting the spacecraft into the final Molniya orbit.

The baseline spacecraft design for DARPA was spin stabilized with a despun payload platform. Power is supplied with an array of silicon solar cells that wrap around the cylindrical wall of the spun section of the spacecraft. The power requirements are about 90 W and will be provided by a 4 amp hour nickel cadmium battery during eclipse. The despun section is at one end of the cylinder, and a monocoque structure at the other end supports a Star 24C solid rocket motor that provides the required delta V (2.5 km/sec) for transfer from the parking orbit to the operation orbit. In the operational orbit, four 0.9 Newton hydrazine thrusters provide thrust for trajectory corrections and attitude control purposes. Two thrusters point in essentially opposite axial directions and two thrusters point essentially in the same lateral direction. Command and data handling functions utilize a central computer based on the Generic VHSIC Spacecraft Computer 1750A microprocessor family and an 8 Mbit RAM. The command and data handling functions are linked to the payload. Attitude knowledge is provided by a Sun sensor, steerable horizon sensor, and accelerometer all of which are located on the spun section of the spacecraft. A motor is used to despin the payload, and active nutation control is provided utilizing torques on the despun section of the spacecraft. The hydrazine thrusters are used to reposition the spin axis as necessary.

The SHF payload provides tactical Earth-to-Earth communications. It utilizes a disk shaped phased array antenna that can electronically scan in two dimensions. The antenna is part of the despun section of the spacecraft and, at launch, is flush against another despun disk shaped structure that houses other parts of the payload and is ringed by four small low gain antennas at its perimeter. The spacecraft mechanically provides rough antenna pointing control in two axes. After injection the antenna is deployed to a position appropriate for the target area. The payload provides a pointing error signal to the spacecraft that helps the spacecraft maintain the correct antenna pointing. The payload also provides the engineering telecommunications for the spacecraft.

A one year mission life was required which allowed the spacecraft design to be single string i.e, minimal redundancy. The dimensions of the spacecraft are 1.0 m diameter by 1.3 m high. Figures 6 and 7 are views of the spacecraft flight configuration. The mass estimate of this spacecraft is presented in table 1.

The DARPA SHF spacecraft concept presented here is the baseline. Two other versions using advanced technology were also created. DARPA is presently studying both EHF and SHF concepts for small tactical communication spacecraft.

DISCOVERY

During 1990, the Solar System Exploration Division (SSED) within NASA's Office of Space Science Applications (OSSA) initiated a study of a series of small missions for planetary exploration modelled on the Explorer and Earth Probes programs within OSSA. This new initiative of SSED is called "Discovery". The Discovery program is meant to be "small" primarily in a financial sense where the total program cost for one Discovery mission is to be less than $150M.

The program objectives of Discovery are as follows: 1) to provide science investigations at the small, low cost end of the mission spectrum, 2) to allow for rapid responses to new emerging science opportunities, 3) to provide opportunities for conducting collaborative/cooperative ventures with other agencies, foreign and/or domestic, 4) to give increased opportunities to young researchers in the field of planetary science and 5) to provide a programmatic vehicle for trying and testing new technologies at acceptable risk levels.

In order to set reasonable bounds on the missions to be considered within Discovery and to be consistent with the programmatic realities of securing a place in the OSSA Strategic Plan, the following Discovery program constraints have been established: 1) Discovery mission costs shall be limited to less than $150 million, 2) Discovery missions shall be conducted as a series of small, low cost missions which draw from common designs, experiences, hardware and software inheritances, etc. to form a program, 3) missions would be launched between 1996 and 2006, 4) missions shall be restrained to Delta class or preferably smaller launch vehicles, 5) science investigations shall be basic and focused on addressing the most fundamental questions and 6) congressional approval for a Discovery mission will be sought on an individual basis independent of any other Discovery mission.

Consistent with the objectives and the constraints, the scope of the Discovery Program is necessarily limited relative to past and present planetary exploration missions. Specific missions to be considered include the near Earth bodies (asteroids and/or comets), the moon, Venus, and Mars as possible targets.

The focus for the activities in 1991 has been the near Earth bodies in either the flyby or rendezvous trajectory mode, whichever is achievable within the constraints. JPL has developed conceptual mission and system designs for both near Earth asteroid flyby and a rendezvous missions. The Discovery spacecraft concepts are presented below.

Discovery Asteroid Flyby

For the flyby study the selected target was the large near Earth asteroid, Eros. The mission was constrained to use the Pegasus launch vehicle. There were two minimum science requirements. First, obtain at least 2 images that have at least 100,000 pixels filled by Eros with a resolution of 30 m or better per pixel. Second, obtain images with a resolution of 300 m per pixel or better at approximately 30 minute intervals for the five hours proceeding closest approach and the 5 hours following closest approach.

A conceptual design for a 6 filter camera was created for this study. The characteristics of the optics of this camera were as follows: 1) 1 meter focal length, 2) 12 milli-radian field of view and 3) an effective F number of 10. The camera used a half masked 1024 by 2048 CCD array with 12 micro-meter pixels. The camera concept employed time delay integration. The physical characteristics of the camera were as follows: 1) 3.5 kg, 2) 8 watts and 3) sized to be a cube 7 inches on a side.

The nominal launch date was chosen to

be May 28, 1995. Pegasus would deliver the spacecraft to a 200 km circular orbit. An upper stage was required to inject the flight system onto a flyby trajectory with Eros. The nominal arrival date was March 14, 1996.

The spacecraft conceptual design used the science and mission requirements discussed above and the following derived spacecraft requirements.

The spacecraft and its upper stage shall be consistent with the injection performance of Pegasus to a 200 km orbit (400 kg) and the injection energy requirement of 1.89 $(km/sec)^2$. The injected mass allocation was about 75 kg. The spacecraft lifetime shall be one year. The spacecraft shall have on-board failure protection algorithms for potential system failures that can, if enabled, place the spacecraft in a safe configuration for at least 7 days without ground intervention.

The spacecraft shall be capable of providing 130 m/sec of delta V for all post launch maneuvers. The spacecraft shall be capable of executing all maneuvers in any inertial direction.

The spacecraft shall be capable of simultaneous X band radio metric tracking, telemetry and commanding and shall be compatible with the NASA/JPL Deep Space Tracking Network. The spacecraft shall be generally consistent with the recommendations of the Consultative Committee for Space Data Systems (CCSDS).

The resulting spacecraft conceptual design was compatible with Pegasus and was injected to Eros with a Star 24C solid rocket motor. The spacecraft was spinning at all times at about 10 rpm except for the injection when it was spun up to about 100 rpm. The spacecraft was configured as an oblate cylinder with a diameter of about 1 meter. Figures 8 and 9 present the spacecraft configuration. The structure was aluminum. The post injection delta V requirements were met by a mono-propellant subsystem. The spacecraft was powered by body mounted solar cells and a battery. Cruise attitude control references were the Sun and the star, Canopus. The spacecraft computer used a 1750A micro-processor. Data was stored in a solid state memory. All communications with Earth were supported by an X band subsystem. Table 1 presents the mass summary for the spacecraft.

Discovery Asteroid Rendezvous

For the rendezvous study, the selected target was the near Earth asteroid, Anteros. The mission was allowed to use a Delta launch vehicle. The science requirements were to image the entire surface of the asteroid at a resolution of at least 6 meters per pixel and obtain IR and elemental composition information.

The following instruments were the payload for the rendezvous mission: 1) a visible wavelength camera, 2) an IR point spectrometer and 3) a gamma ray spectrometer. The camera had a 154 milliradian field of view with f/2, 80 mm optics, a 6 color filter wheel and a 1024 by 1024 micrometer CCD detector. The IR spectrometer had a 100 milliradian field of view and covered 0.8 to 2.5 micrometer spectral region. The gamma ray spectrometer had a wide field of view. The mass and power requirements of the camera, the IR spectrometer and the gamma ray spectrometer were 4.5 kg and 8 W; 4.5 kg and 4 W and 18 kg and 12 W respectively.

The nominal launch date was chosen to be May 20, 1997. The Delta 7925 would inject the spacecraft directly onto a trajectory with Anteros. The nominal arrival date was July 8, 1998.

The spacecraft requirements were the same as for the flyby spacecraft presented previously except for the following items: 1) the injection energy of 40 $(km/sec)^2$ from the Delta limits the spacecraft injected mass to about 570 kg, 2) the spacecraft lifetime needed to be at least 3.0 years, 3) in order to rendezvous with Anteros the post launch delta V requirement was 1200 m/sec and 4) the spacecraft was required to be capable of performing the maneuvers in orbit around the asteroid to a precision of 0.001 m/sec or better.

Figures 10 and 11 present the external and internal views of the resulting conceptual design of the Discovery rendezvous spacecraft. Power is provided by a body-mounted, silicon solar array that is designed to allow a very wide range of sun angles. Command and data handling functions are centralized in a computer and mass data storage capability is in excess of 0.1 Gbit. Telecommunications are all X-band, include a 3 W RF power amplifier, and utilize body-mounted antennas. There is one high-gain antenna and three low-gain antennas which allow commandability at any spacecraft orientation. The spacecraft is fully spin stabilized during cruise and then switches over to momentum bias operation after rendezvous. This is accomplished by despinning the spacecraft and spinning up a single momentum wheel instead. In both cases, the boresight of the high-gain antenna is aligned with the momentum vector and thus allows continuous HGA communications when Earth-pointing is selected. In the momentum bias mode, continuous nadir pointing of the instruments is available simultaneously as long as an orbit about the target body is selected that has its normal pointed at Earth. Fine sun sensors and the science [and star] camera provide necessary spacecraft attitude information; no gyros are required. Propulsion is provided with a simple monopropellant hydrazine system with a capacity of 348 kg of usable propellant. The spacecraft was designed to be fully redundant in all the usual components. The spacecraft mass estimate for the Anteros mission is shown in table 1.

The Discovery program is receiving serious attention within NASA SSED. JPL is studying mission and spacecraft options. Due to programmatic reasons, the Discovery spacecraft concepts presented above are unlikely to be built but have been useful in illustrating the potential design solutions to requirements.

SUMMARY

This paper has presented a brief technical description of some of the small spacecraft concepts prepared by JPL for various sponsors. This paper also briefly summarized some of JPL's activities in microspacecraft. Over the past several years, JPL's sponsors have clearly shown more interest in small spacecraft. NASA's Discovery program is the most recent example of this increased interest.

ACKNOWLEDGEMENT

The lead spacecraft system engineer for the Lunar GAS spacecraft was Ron Salazar. The lead spacecraft system engineer for the DARPA SHF and both Discovery spacecraft concepts was Dave Collins. The research described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

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Table 1 Spacecraft Mass Summary by Subsystem (mass in kg)

	Lunar GAS	DARPA SHF	Polar Mesoscale Explorer	Discovery Flyby	Discovery Rendezvous
Payload	10.0	13.0	10.0	3.5	31.5
Telecom	7.0	0.5	9.6	6.7	18.7
Power	40.0	13.2	10.7	6.7	21.8
Att Control	1.0	11.4	0.4	4.4	13.1
Cmd & Data	6.0	2.2	2.4	5.7	6.8
Structure & Cabling	33.0	30.3	6.0	21.7	56.2
Thermal	2.0	3.9	0.8	2.2	10.0
Propulsion	15.0	8.3	0.5	6.5	45.9
Contingency	note 1	24.2	note 1	13.1	33.3
Dry Mass	114.0	107.0	40.4	70.5	237.3
Star Motor	NA	252.4	NA	240.4	NA
Propellant	36.0	10.0	0.5	7.0	223.0
Total	150.0	369.4	40.9	317.9	460.3

Note 1) Mass Contingency was Distributed in the Subsystem Numbers

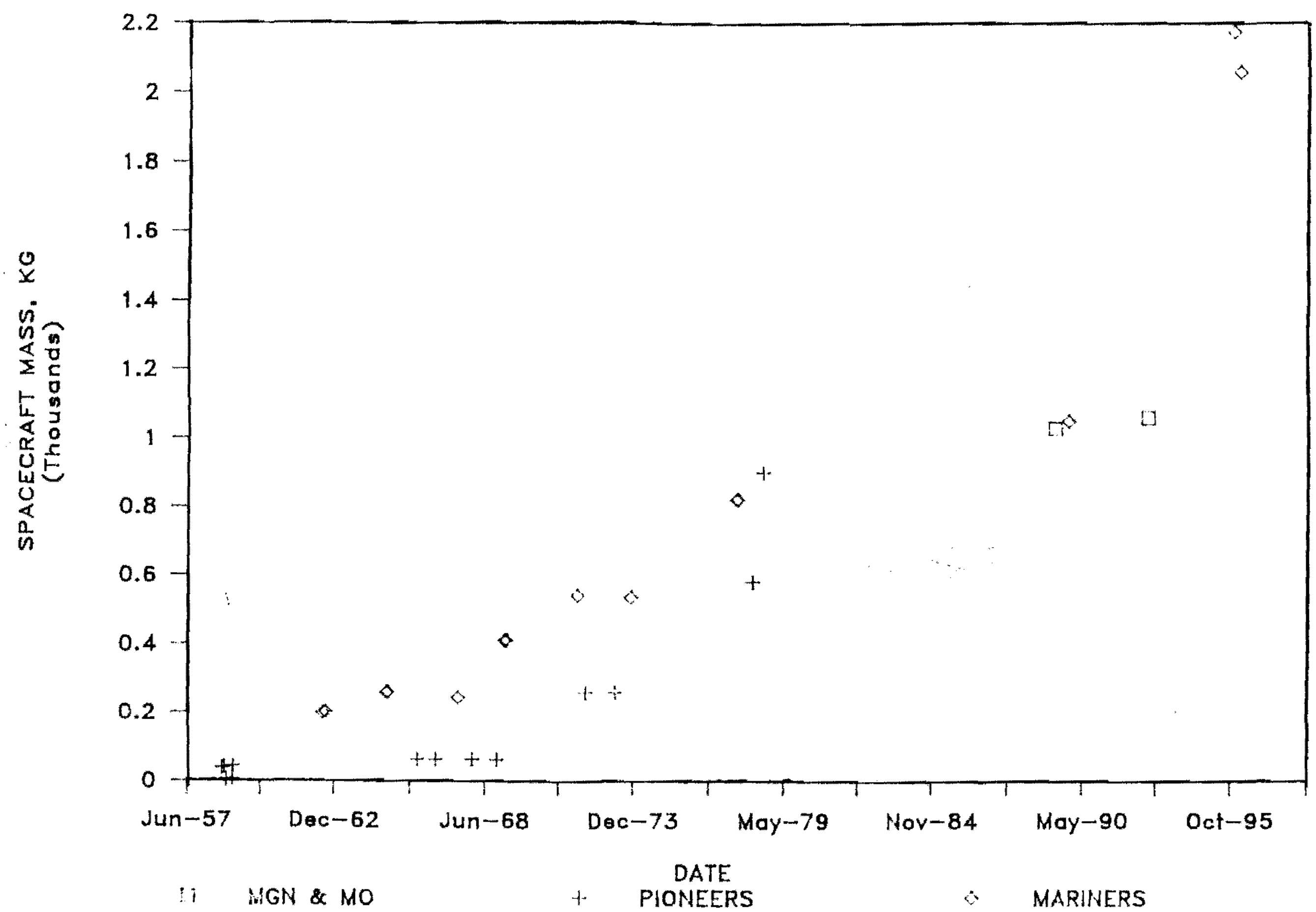

Fig. 1 Dry Mass of NASA Planetary Spacecraft vs. Time.

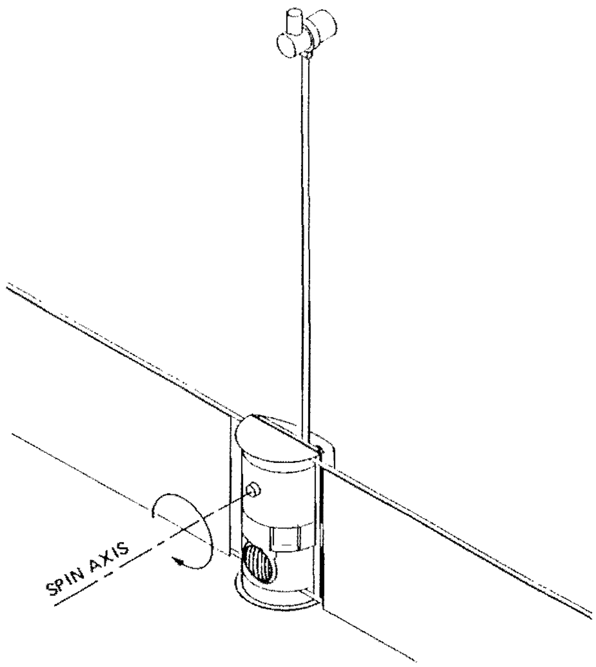

Fig. 2 Lunar GAS Spacecraft Flight configuration.

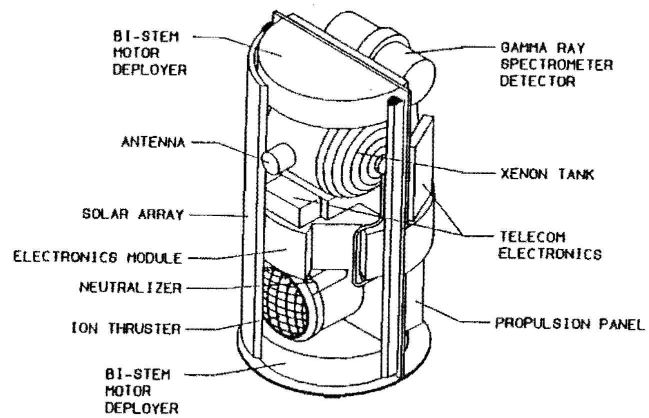

Fig. 3 Lunar GAS Spacecraft Internal Arrangement.

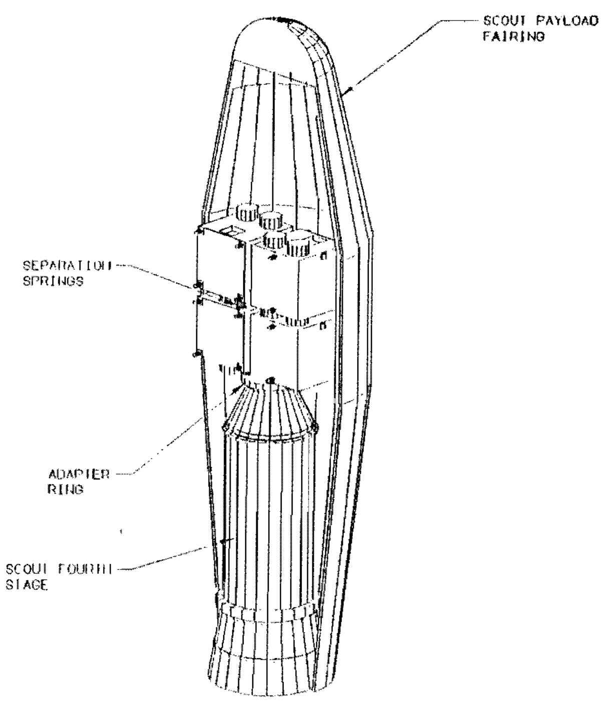

Fig. 4 Polar Mesoscale Explorer Spacecraft Launch Configuration.

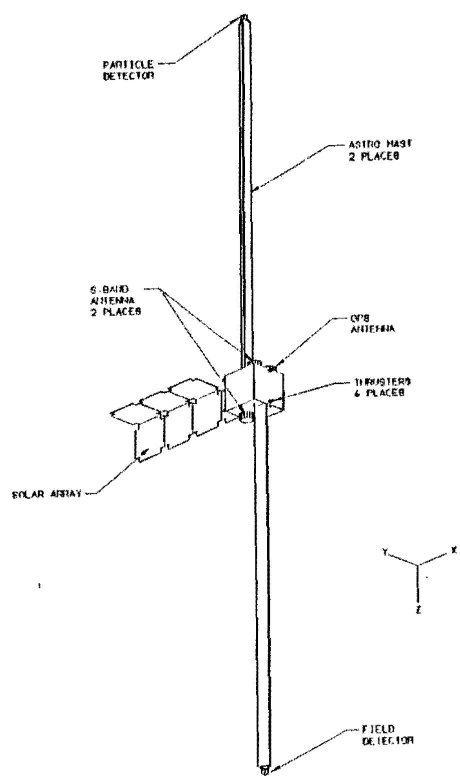

Fig. 5 Polar Mesoscale Explorer Spacecraft Flight Configuration.

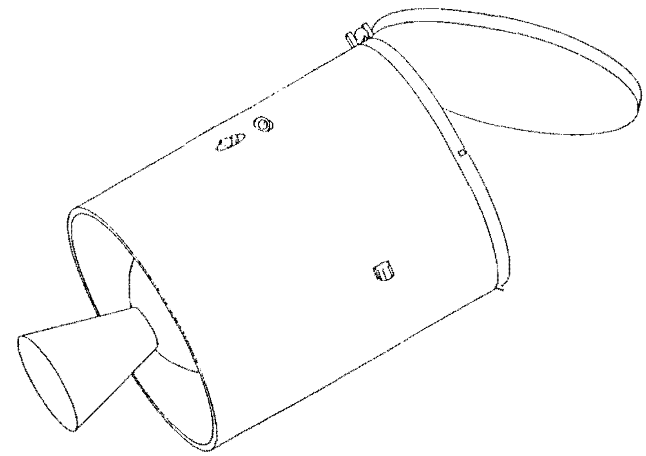

Fig. 6 DARPA SHF Spacecraft Flight Configuration.

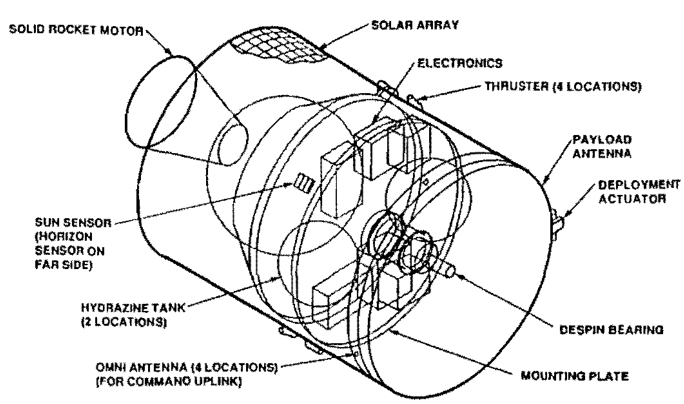

Fig. 7 DARPA SHF Spacecraft Internal Arrangement.

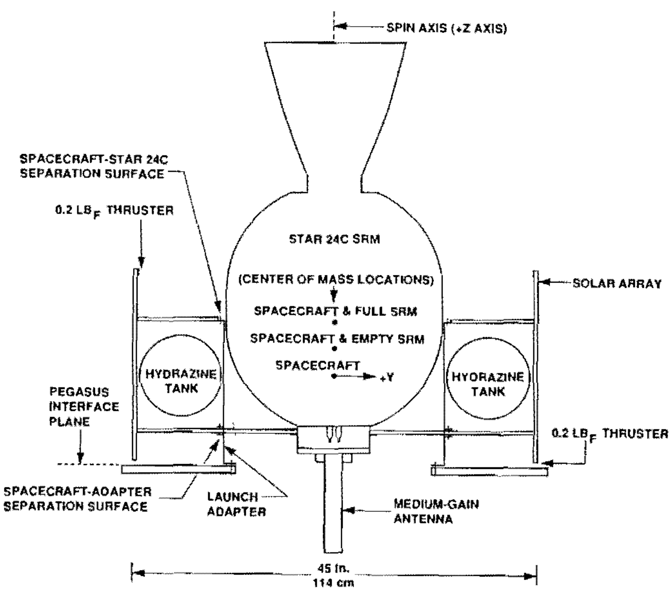

Fig. 8 Discovery Asteroid Flyby Spacecraft Launch Configuration (X-Z Plane Cross Section).

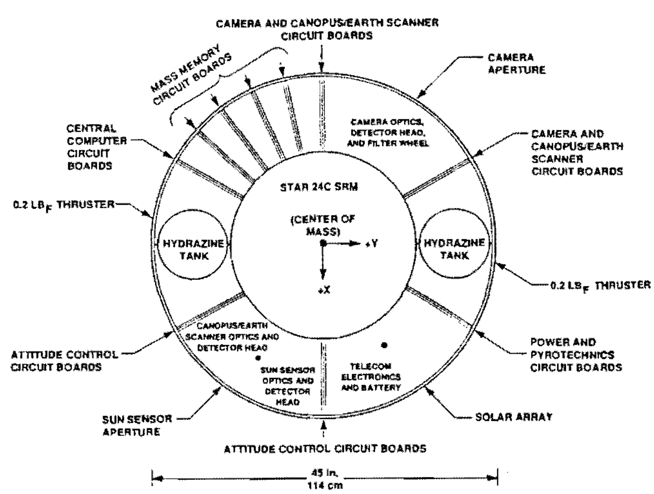

Fig. 9 Discovery Asteroid Flyby Spacecraft Launch and Flight Configuration (X-Y Plane Cross Section).

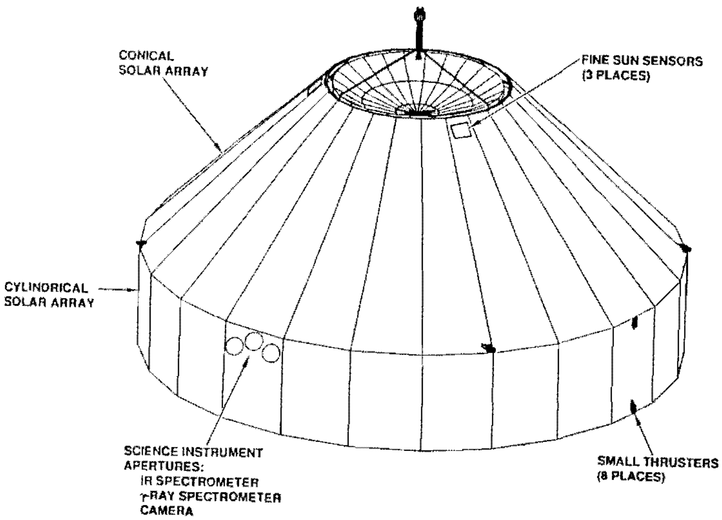

Fig. 10 Discovery Asteroid Rendezvous Spacecraft Flight Configuration.

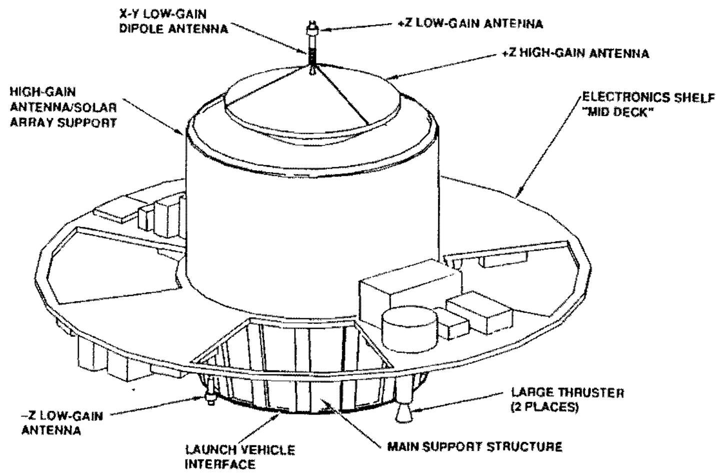

Fig. 11 Discovery Asteroid Rendezvous Spacecraft Flight Configuration (with Solar Panels Removed).