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FINAL REPORT

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NEW VISION SOLAR SYSTEM MISSION STUDY

Use of Space Reactor Bimodal System with Microspacecraft

to

Determine Origin and Evolution of the Outer Planets in the Solar System

by

Jack F. Mondt & Robert M. Zubrin

March 1996

MASTER

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**USE OF SPACE REACTOR BIMODAL SYSTEM WITH MICROSPACECRAFT
TO
DETERMINE ORIGIN AND EVOLUTION OF THE OUTER PLANETS
IN
THE SOLAR SYSTEM**

WRITTEN

by

JACK F. MONDT & ROBERT ZUBRIN

for

**OFFICE OF ENGINEERING AND TECHNOLOGY
OFFICE OF NUCLEAR ENERGY, SCIENCE AND TECHNOLOGY
DEPARTMENT OF ENERGY**

MARCH 1996

NEW VISION SOLAR SYSTEM MISSION STUDY

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NEW VISION SOLAR SYSTEM MISSION STUDY

INTRODUCTION

FUTURE VISION FOR PLANETARY EXPLORATION

**CAPABILITY TO DELIVER "CONSTELLATIONS" OR "FLEETS" OF
MICROSPACECRAFT TO THE OUTER PLANETS**

**FLEETS OF MICROSPACECRAFT ACT IN A COORDINATED MANNER TO GATHER
SCIENTIFIC DATA**

MAKING THE SOLAR SYSTEM ACCESSIBLE TO THE GENERAL PUBLIC

**CONSTELLATIONS COUPLED WITH ADVANCED VISUALIZATION TECHNIQUES AND
HIGH-RATE COMMUNICATIONS PROVIDE BASIS FOR "VIRTUAL PRESENCE"**

**HELP MAKE SOLAR SYSTEM EXPLORATION A PART OF THE HUMAN
EXPERIENCE**

NEW VISION SOLAR SYSTEM MISSION STUDY

INTRODUCTION

The vision for the future of the planetary exploration program includes the capability to deliver "constellations" or fleets" of microspacecraft to a planetary destination. These fleets will act in a coordinated manner to gather science data from a variety of locations on or around the target body, thus providing detailed, global coverage without requiring development of a single large, complex and costly spacecraft. Such constellations of spacecraft, coupled with advanced information processing and visualization techniques and high-rate communications, could provide the basis for development of a "virtual" presence" in the solar system. A goal could be the near real-time delivery of planetary images and video to a wide variety of users in the general public and the science community. This will be a major step in making the solar system accessible to the public and will help make solar system exploration a part of the human experience on Earth.

NEW VISION SOLAR SYSTEM MISSION STUDY

OBJECTIVE

DETERMINE MISSION OPTIONS AND TECHNOLOGY NEEDS FOR FUTURE PLANETARY
"CONSTELLATIONS" MISSIONS, ASSUMING THE PRESENCE OF A SPACE REACTOR
BIMODAL SYSTEM AND SECOND GENERATION MICROSPACECRAFT

NEW VISION SOLAR SYSTEM MISSION STUDY

OBJECTIVE

The primary objective of this study is to determine mission options and technology needs for future planetary "constellations" missions, assuming the presence of a space reactor bimodal system and second generation microspacecraft. NASA's goal is to make the general public and the scientific community a part of Solar System exploration in determining the origin and evolution of the outer planets. By using many microspacecraft throughout the Solar System NASA would bring first time scientific data and visual images of planets, comets, asteroids, etc., into the general public's homes at their request or as a scheduled TV program.

This study examines the potential of Space Reactor Bimodal (SRB) Systems to transport many advanced technology microspacecraft and science instruments from Earth orbit to the outer planets. Then collect, process and transmit scientific data and images from the microspacecraft and science instruments back to Earth. As a result of this study, mission options and new technology needs for future planetary missions will be determined.

NEW VISION SOLAR SYSTEM MISSION STUDY

APPROACH

DETERMINE REPRESENTATIVE SCIENTIFIC TARGETS AND DEFINE THE MISSION GOALS AT EACH TARGET.

DEVELOP SAMPLE MISSION PROFILES FOR EACH SELECTED SCIENTIFIC TARGET.

DEFINE WAYS THE MICRO S/C AND THE BIMODAL CARRIER S/C MUST INTERACT.

PREPARE CONCEPTUAL DESIGN OF SPACE REACTOR BIMODAL SYSTEM, DEFINE POWER, PROPULSION, DATA PROCESSING AND COMMUNICATION SUBSYSTEMS AND DETERMINE OPTIMAL TRAJECTORIES.

PREPARE A FINAL REPORT.

PREPARE AND MAKE A FINAL PRESENTATION TO DOE AND NASA.

PREPARE TECHNICAL PAPER FOR THE 13TH SYMPOSIUM ON SPACE NUCLEAR POWER AND PROPULSION.

NEW VISION SOLAR SYSTEM MISSION STUDY

APPROACH

This study examines the ways in which space reactor power and propulsion might be used to enable key elements of this vision. The capability of a space reactor bimodal (power and propulsion) system, defined for potential Air force Earth orbital missions, will be investigated as a power system and propulsion system for planetary missions using many small sciencecraft or its own science payload. Scientific targets of interest include Main Belt asteroid, comets Jupiter, Saturn, Uranus, Neptune, Pluto, and their satellites.

The Jet Propulsion Laboratory (JPL) will lead the study and define the overall mission goals at each of the scientific targets of interest. Lockheed Martin Corporation will support JPL in determining ways to accomplish the mission goals using a space reactor bimodal system and advance microspacecraft.

JPL and Lockheed Martin Corporation will do the following:

- Determine representative scientific targets and define the mission goals at each target. Conceptual end-to-end missions will be examined and defined. Key microspacecraft and instrument characteristics will be identified and related technology needs will be assessed. Extensive use will be made of ongoing microspacecraft development activities.

- Develop sample mission profiles for each selected scientific target and define ways in which the microspacecraft and the space reactor bimodal carrier spacecraft must interact.

- Prepare a final report and a technical paper for the 13th symposium on Space Nuclear Power and Propulsion in January 1996.

NEW VISION SOLAR SYSTEM MISSION STUDY

APPROACH (CONT.)

PREPARE CONCEPTUAL DESIGN OF NUCLEAR BIMODAL CARRIER S/C BUS FOR TRANSPORTING SEVERAL MICRO S/C TO SELECTED SCIENTIFIC TARGETS.

DEFINE POWER, PROPULSION, DATA PROCESSING AND COMMUNICATION SUBSYSTEMS NEEDS TO INTERACT WITH MICRO S/C WHEN AT SCIENTIFIC TARGET.

DETERMINE OPTIMAL TRAJECTORIES TO THE SELECTED SCIENTIFIC TARGETS.

EXAMINE TRAJECTORIES PARAMETRICALLY FOR DIFFERENT SPACE REACTOR BIMODAL POWER AND PROPULSION LEVELS WITH THEIR ESTIMATED MASSES.

PREPARE AND MAKE A FINAL PRESENTATION TO DOE AND NASA.

NEW VISION SOLAR SYSTEM MISSION STUDY

APPROACH (CONT.)

Lockheed Martin will do the following:

Prepare a conceptual design of the nuclear bimodal powered spacecraft bus that could be used to transport the selected number of microspacecraft from Earth orbit to each of the selected scientific targets. A conceptual design of a bimodal carrier vehicle that includes communication, propulsion, data handling, and power functions will be developed. This will include mission trades and top-level system analysis of the nuclear bimodal carrier vehicle.

For comparison, prepare a conceptual design of a nuclear bimodal spacecraft bus with its own science payload that could perform independent scientific investigations at the selected scientific targets.

The technology base for the nuclear bimodal system will be the Nuclear Engine for Bimodal Application (NEBA) System defined in a recent study by the Air Force Phillips Laboratory. A 1500 kg NEBA system that produces a 1000 N thrust at an Isp of 850 seconds and 10 Kwe of power for ten years will be used as the reference bimodal system for the initial phase of this study. Lockheed Martin will evaluate the nuclear bimodal engine as the propulsion and power subsystems for a carrier spacecraft that transports and services multiple small spacecraft or provide power and propulsion for its own science payload.

Define the power, propulsion, on-board data processing, and telecommunication subsystems required to interact with multiple microspacecraft or its own science instruments, to process the scientific data, and transmit the data to Earth.

Perform trajectory analyses and determine the optimal direct and/or gravity assisted trajectories to the selected scientific targets. The trajectories to the outer planets will be determined using direct thermal and electric propulsion as well as gravity assisted trajectories. The trajectories will be examined parametrically for different bimodal power and propulsion levels with the corresponding masses.

Prepare and present a final briefing to NASA and DOE Headquarters in Washington DC at the conclusion of the study.

NEW VISION SOLAR SYSTEM MISSION STUDY

BACKGROUND

PAST STUDIES:

NEP DELIVERS LARGE PAYLOADS TO THE OUTER PLANETS

NEP DELIVERS PAYLOADS WITH SHORTER TRIP TIMES TO THE OUTER PLANETS

LOW POWER NEP, CHEMICAL PROPULSION, GRAVITY ASSISTS, AND SMALL PAYLOADS

USES HIGH POWER TO INCREASE SCIENCE RETURN FROM THE OUTER PLANETS

REQUIRES EXPENSIVE TITAN CLASS LAUNCH VEHICLES

SPACE REACTOR BIMODAL STUDY SHOWED BENEFITS FOR AIR FORCE MISSIONS

THIS STUDY:

USE THE SPACE REACTOR BIMODAL DESIGN FROM THE AIR FORCE STUDY

INVESTIGATE ITS USE FOR PLANETARY MISSIONS WITH SEVERAL MICROSPACECRAFT

NEW VISION SOLAR SYSTEM MISSION STUDY

BACKGROUND

Traditional studies of the benefits of space nuclear power for planetary missions focused almost exclusively on mission enhancements available from nuclear electric propulsion (NEP).

Result: Increases in useful payload or decreases in launch costs were demonstrated.

Criticisms: Large increases in science payload are not required or affordable.

Saving from use of smaller boosters are overwhelmed by NEP costs.

Using NEP to maximize payload requires excessive NEP burn times (~ 8 to 10 years).

In 1993 Martin Marietta proposed an alternative strategy: Use ballistic trajectories with gravity assists and light-weight S/C technology to minimize NEP burn time. Primary nuclear mission benefit comes from increasing data rate by a factor of 100. Strategy studied for DOE in 1994.

Result: Outer solar system nuclear missions shown with NEP burn times of ~ 2 to 4 years.

Criticism: Mission strategy requires launch vehicles in the Proton - Titan 4 class.

In 1994 a DOE/Phillips Lab team developed design of NEBA-3 bimodal space nuclear power system. In January 1995 DOE proposed that a Lockheed Martin/JPL team study application of NEBA-3 to planetary missions.

Hypothesis 1: Because it can inject itself from LEO onto interplanetary trajectories with direct thrust at 850 s Isp, the bimodal system can eliminate need for large launch vehicles plaguing the Martin Marietta 1994 mission strategy.

Hypothesis 2: Because it combines high communication capability with versatile propulsion capability, the bimodal S/C could serve as an ideal carrier for a group of micro S/C on outer solar system missions.

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

NEPTUNE/TRITON

OVER-ARCHING THEME: DETERMINE ORIGIN AND SUBSEQUENT EVOLUTION OF PLANET

- NEPTUNE

**DEEP ATMOSPHERIC STRUCTURE
DYNAMIC METEOROLOGY**

- TRITON

**VOLATILE AND SURFACE/ATMOSPHERIC INTERACTIONS
CHEMICAL EVOLUTION OF ATMOSPHERE
GEOLOGICAL HISTORY & PRESENT DAY ACTIVE PROCESSES**

- SYSTEM

**RING TEMPORAL VARIATIONS AND COMPOSITION
SMALL SATELLITE COMPOSITION
COLLISIONAL HISTORY/RING INTERACTIONS**

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

NEPTUNE/TRITON

The Scientific interest in exploring Neptune is to determine its origin and based on this information, help determine how it evolved as a planet in the solar system. At Neptune the understanding of the atmosphere, its elements, its structure, and the dynamic meteorology is needed to determine its origin and how it evolved.

Triton is as interesting scientifically as Neptune. The Voyager images showed a geyser-like eruption of dark material shooting almost straight up from the surface. The dark plume rises vertically about 8 kilometers forming a cloud that drifts horizontally 150 kilometers in Triton's wind. The cloud is probably nitrogen carrying dark particles of carbon rich material and possibly ice crystals. The scientific interest in Titan is to determine the chemical evolution of the atmosphere, and the geological history and the present day creative processes.

The Neptune planetary system would be examined for rings, small satellite composition, and collisional history.

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

SATURN/TITAN

OVER-ARCHING THEME: DETERMINE ORIGIN AND SUBSEQUENT EVOLUTION OF PLANET

- SATURN

**DEEP ATMOSPHERIC STRUCTURE
DYNAMIC METEOROLOGY**

- TITAN

**VOLATILE AND SURFACE/ATMOSPHERIC INTERACTIONS
CHEMICAL EVOLUTION OF ATMOSPHERE
GEOLOGICAL HISTORY & PRESENT DAY ACTIVE PROCESSES**

- SYSTEM

**RING TEMPORAL VARIATIONS AND COMPOSITION
SMALL SATELLITE COMPOSITION
COLLISIONAL HISTORY/RING INTERACTIONS**

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

SATURN/TITAN

The scientific interest of Saturn is to determine its origin and how it evolved as a planet of the solar system. Understanding the atmosphere, its elements, its structure, and its dynamic behavior all are needed to determine its origin and evolution.

Cloud covered Titan is of great scientific interest. The cloud layer at Titan's North Pole is denser than that found on Mars. The Titan surface atmosphere density is about 5 kg/m^3 and the gravity is one-half that of Earth. Detailed scientific information about the Titan atmosphere and the Titan liquid or solid surface will help determine Saturn's origin.

The Saturn system of rings and their composites as well as the small Saturn satellites composites is needed to help determine the origin and evolution of Saturn.

**NEW VISION SOLAR SYSTEM MISSION STUDY
SCIENTIFIC TARGETS OF INTEREST**

MAIN BELT ASTEROID

ASTEROIDS

WHAT IS THE SURFACE MATERIAL?

**IS THE ASTEROID FRAGMENTS OF HEATED MATERIAL OR IS IT UNALTERED
PRIMITIVE CHONDRITIC MATERIAL.**

DETERMINE WHAT IS BELOW THE SURFACE

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

MAIN BELT ASTEROID

More detailed scientific knowledge of the main belt asteroid will help determine the origin and evolution of the Solar System. Examining many asteroids to determine their surface material and the variations between asteroids would help understand the evolution of the Solar System.

Knowing if the asteroids are fragments of heated materials or unaltered primitive chondritic material will determine the asteroid belt origin and evolution.

Also, determining the material below the surface will provide further scientific data to determine the main belt origin and evolution.

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

JUPITER AND ITS SATELLITES

JUPITER

ATMOSPHERIC DATA DOWN TO 1000 BAR (GIL PROBE OBTAINED DATA TO 10 BAR)

CLOSE IN POLAR ORBITTER (BELOW THE HIGH RADIATION FIELD)

EUROPA

RADAR/SOUNDER ORBITTER TO DETERMINE IF THERE IS LIQUID WATER
LANDERS TO EXAMINE THE SURFACE FOR LIFE
PENETRATORS TO DETERMINE THICKNESS OF CRUST AND IF THERE IS LIQUID
WATER

IO

WHAT CAUSES THE VOLCANO ?
WHAT IS THE MATERIAL FROM THE VOLCANO?
LANDERS TO EXAMINE THE SURFACE?

ALL OTHER JUPITER MOONS GEOLOGICAL HISTORY AND MATERIALS

NEW VISION SOLAR SYSTEM MISSION STUDY SCIENTIFIC TARGETS OF INTEREST

JUPITER AND ITS SATELLITES

Jupiter and its satellites are scientifically of great interest because it is the closest of the outer planets, and nearly large enough to be a star system. So a good understanding of Jupiter's atmospheric elements and its continuous changing winds will help determine its origin and evolution.

Europa, about the size of our moon, is believed to have a 100 kilometer thick crust of ice, because of its very few craters. It is also postulated that liquid water may be under the ice crust, which may support life. So determining the actual material of Europa is of great scientific interest.

The moon with tidal heating today is Io. No impact craters can be seen on the surface of Io, as they all have been covered over by continuing volcanic activity. Understanding what causes the Io volcanos and the composition of the ejected material is of great interest, and will help determine the origin and evolution of Jupiter. There are large black features thought to be lakes of liquid sulphur, with surface temperatures $\sim 40^{\circ}\text{C}$, that are much warmer than the rest of Io's surfaces (-160°).

The geological history and composition of all Jupiter moons would help determine Jupiter's origin and how it has evolved.

NEW VISION SOLAR SYSTEM MISSION STUDY

JUPITER SCIENCE

SCIENCE GOALS

TECHNIQUE

INSTRUMENT

Understanding The Atmosphere of Jupiter

Occultation

Radio Science Between Bus and S/C

Understanding The Magnetic Field of Jupiter

Direct Measurement

Magnetometer Fields and Particles

Understanding Jupiter Auroral Phenomenon

Imaging

UV, Near UV, Thermal IR

Understanding The Gravity Field of Jupiter

Doppler Ranging

Radio Science

Understanding Process In Neutral Atmosphere and Ionosphere

Spectroscopy

Imaging Spectrometer

Understanding of Elemental Composition of Galilean Satellites

Impact Flash From Impactor
Close Observation
Lander / Penetrator

Spectrometer
Gamma Spectrometer
Mass Spectrometer

Understanding of Io Torus Composition

Direct Measurement of Torus

Mass Spectrometer

Understanding of Io Flux Tube

Direct Measurement of Flux Tube

Magnetometer
Fields and Particles

Understanding of Io Neutral Atmosphere

Direct Measurement

Mass Spectrometer
Fluorescence Spec

Determine if Europa Has An Ocean Under Ice

Remote Observation of
Exact Shape of Europa

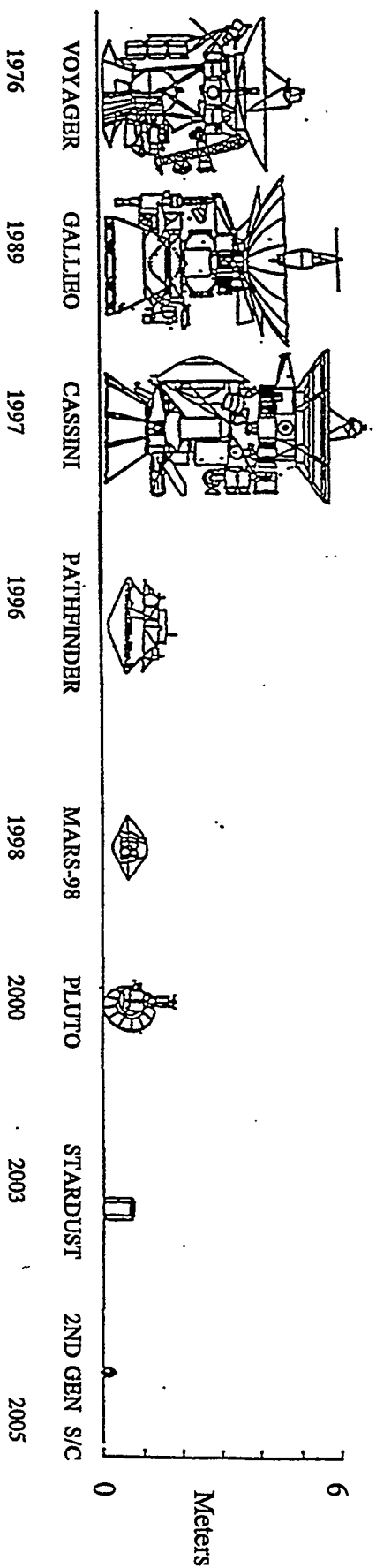
Doppler Laser Altimeter

NEW VISION SOLAR SYSTEM MISSION STUDY JUPITER SCIENCE

The most significant scientific goals of Jupiter and two of its most interesting moons are listed on the facing page. The techniques and instruments to obtain the scientific data are presented also.

NEW VISION SOLAR SYSTEM MISSION STUDY SECOND GENERATION MICROSPACECRAFT

TRENDS IN PLANETARY SPACECRAFT



NEW VISION SOLAR SYSTEM MISSION STUDY

SECOND GENERATION MICROSPACECRAFT

TRENDS IN PLANETARY SPACECRAFT

The early trends in spacecraft mass, volume, and scientific capability was to grow larger from Mariner/Mars to Voyager to Galileo to Cassini because of the larger launch vehicle capability. However the large launch vehicles are very expensive, and when the NASA budget began to shrink, the planetary spacecraft mass and volume was reduced while keeping the same scientific capability. Many of the planetary spacecraft subsystems now can be miniaturized with the new technology for computers, software, instruments, etc., and require less power. Therefore, the trend since Cassini is smaller and smaller to a predicted second generation microspacecraft of outer planet flybys and orbiters of 10 to 20 kg mass and approximately a 30 cm cube. This trend is shown pictorially on the facing page.

NEW VISION SOLAR SYSTEM MISSION STUDY SECOND GENERATION MICROSPACECRAFT

SPACECRAFT APPLICATION

MISSION
GLASSES

NEAR
EARTH
OBJECT
FLYBY

RENDEZVOUS
AND
LANDING

OUTER
SOLAR
SYSTEM
FLYBY

SPACE PHYSICS
AND
ASTROPHYSICS

EARTH
OBSERVATION

CORE BUILDING BLOCKS

INFORMATION
PROCESSING
& CONTROL

POWER
ELECTRONICS

OPTICS &
FOCAL
PLANE

TELECOM
ELECTRONICS

OTHER COMMON ELEMENTS

VALVES

FLUOVERS

IFRU

SECONDARY
BATTERIES

PRESSURE
SENSORS

MISSION UNIQUE ELEMENTS

INSTRUMENTS

SOLAR
ARRAYS

ANTENNAS

PRIMARY
BATTERIES

PROPULSION
STAGE

NEEDED TECHNOLOGIES

NEW VISION SOLAR SYSTEM MISSION STUDY SECOND GENERATION MICROSPACECRAFT

PURPOSE

Help focus technology development on enabling the kind of spacecraft NASA would like to be able to begin flying within the first decade of the new millennium (by 2010).

OBJECTIVES

Greatly reduce flight system, launch, and operations cost, improve mission benefit-to-cost ratios, enable frequent/simultaneous flight of numerous spacecraft, and provide a catalyst for U.S. innovation and leadership in technology.

APPROACH

Utilize innovative system design and new technology to minimize the following cost drivers: communications, power, nuclear material, size and mass, and uplink.

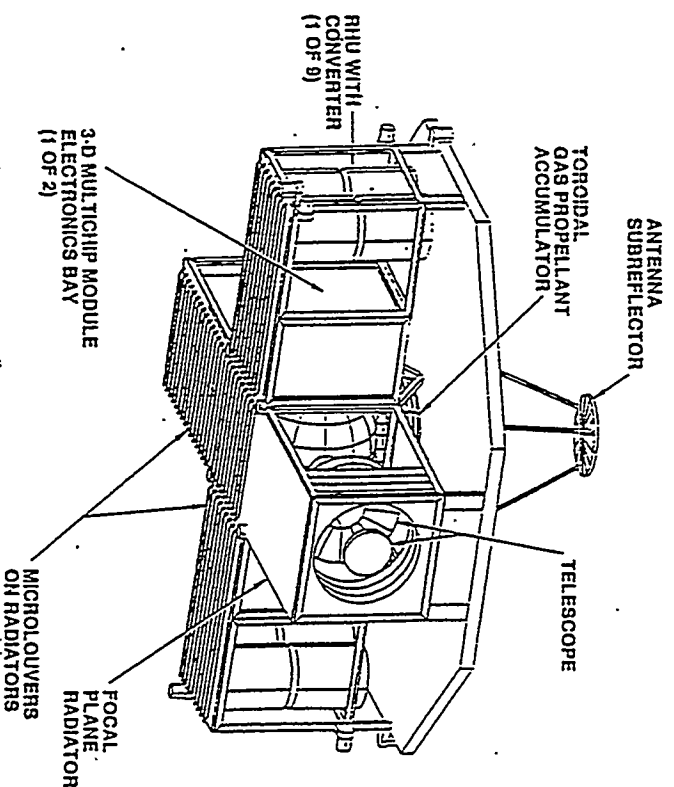
Develop flexible (multi-mission), high-capability building blocks with manufactured-in high quality/reliability and low recurring costs, eliminate extensive redundancy to lower the cost.

NEW VISION SOLAR SYSTEM MISSION STUDY **SECOND GENERATION MICROSPACECRAFT** **OUTER SOLAR SYSTEM FLYBY SPACECRAFT** **AND ASSOCIATED TECHNOLOGY NEEDS**

SCIENCE FOCUS: IMAGING AND IMAGING SPECTROSCOPY

POWER: 0.1-15 W LAUNCH SIZE: 46 CM DIA. X 30 CM LAUNCH MASS: 8.4 KG

REQUIRED UPLINK: NONE



OTHER COMMON KEY TECHNOLOGIES

SPECIAL MISSION SPECIFIC TECHNOLOGIES

RHU-Based, Thermoelectric Power Source Precision Clock & Timer Low-Mass MGA Supplemental On-Board Data Analysis Optional Microinstruments

NEW VISION SOLAR SYSTEM MISSION STUDY

SECOND GENERATION MICROSPACECRAFT

OUTTER SOLAR SYSTEM FLYBY SPACECRAFT

The spacecraft autonomously provides imaging and imaging spectroscopy of objects in the outer solar system for spacecraft solar ranges of 3 to 39 AU and Earth ranges up to 38 AU. Estimated spacecraft wet mass, launch configuration size, and load power are respectively 8.4 kg, 46-cm diameter x 30-cm height, and 0.1 to 15 W (depending on operating state). In this mass regime, missions to the outer solar system with relatively short trip times appear possible using small launch vehicles with appropriate upper stages. This spacecraft spends most of its time in cruise in a "hibernation" state in which only a clock/timer is operating and electrical power is being stored. Also, since communication rates are low and operating periods are limited, more on-board data analysis is utilized, particularly for long-range outer planet targets.

NEW VISION SOLAR SYSTEM MISSION STUDY

SECOND GENERATION MICROSPACECRAFT

OUTTER SOLAR SYSTEM FLYBY SPACECRAFT

WET MASS	8 to 10 kg
LAUNCH CONFIGURATION	46cm dia. x 30 cm height
LOAD POWER	1 to 15 watts

OUTTER SOLAR SYSTEM ORBITER SPACECRAFT

WET MASS	15 to 25 kg
LAUNCH CONFIGURATION	20 x 35 x 30 cm³
LOAD POWER	10 to 40 watts

OUTTER SOLAR SYSTEM LANDER SPACECRAFT

WET MASS	40 to 80 kg
LAUNCH CONFIGURATION	65 cm dia. x 40 cm height
LOAD POWER	1 to 20 watts

NEW VISION SOLAR SYSTEM MISSION STUDY

SECOND GENERATION MICROSPACECRAFT

The microspacecraft designed, developed, built, and used for near-Earth missions can be used for outer planets, except for the power subsystem, when the space reactor Bimodal spacecraft is used to collect data from the microspacecrafts and send the data to Earth. Therefore the outer solar system flyby, orbiter, and lander microspacecraft can have small communication antennas and small radioisotope power subsystems. The three microspacecraft could use 20 watts, 2 kg radioisotope thermal photovoltaic power subsystems. If 40 watts of power is required, two 20 watt power subsystems with a mass of 4 kg could be used.

NEW VISION SOLAR SYSTEM MISSION STUDY

SECOND GENERATION MICROSPACECRAFT TECHNOLOGIES BEING DEVELOPED

MICRO-ELECTRONICS

SMALL X, Ka TRANSPONDERS

LOW-MASS PHASED ARRAY ANTENNA

LOW COST COMPOSITE STRUCTURES

LOW-MASS INTEGRATED SPACECRAFT

LOW-MASS THERMAL CONTROL

PENETRATOR/LANDER SYSTEMS

IMPROVED RADIOISOTOPE POWER SYSTEMS

ADVANCED RECHARGEABLE BATTERIES

LOW-MASS INTEGRATED INSTRUMENTS

MICRO *IN SITU* INSTRUMENTS

IMPACT FLASH SPECTROMETRY

SPACECRAFT AUTONOMY

NEW VISION SOLAR SYSTEM MISSION STUDY SECOND GENERATION MICROSPACECRAFT TECHNOLOGIES BEING DEVELOPED

The technologies now being developed and/or improved for the second generation microspacecraft is listed on the facing chart. This list includes spacecraft instrumentation and operation technologies which are being pursued to ultimately reduce the cost of obtaining more detailed planetary science.

The low-mass integrated spacecraft and low mass integrated instruments may be combined in the future into an integrated sciencecraft to reduce cost and mass. Most of these technologies will be flight demonstrated in the New Millennium Program by the year 2000.

NEW VISION SOLAR SYSTEM MISSION STUDY INSTRUMENT FOR MICROSPACECRAFT MISSIONS

1993 PLUTO FAST FLYBY CHALLENGE:

To show that a science payload consisting of a visible camera, IR spectrometer, UV spectrometer and uplink radio occultation experiment satisfying the core Pluto science objectives is achievable with the following mass, power and cost constraints.

Mass: 7 kg

Power: 6 watts

Pointing Stability: 10 microrads/sec

Cost: \$30M for two science payloads

RESULTS:

Pluto integrated camera spectrometer was designed and a prototype built, delivered and demonstrated in September 1994 to support a 1995 NASA Announcement of Opportunity with the following parameters.

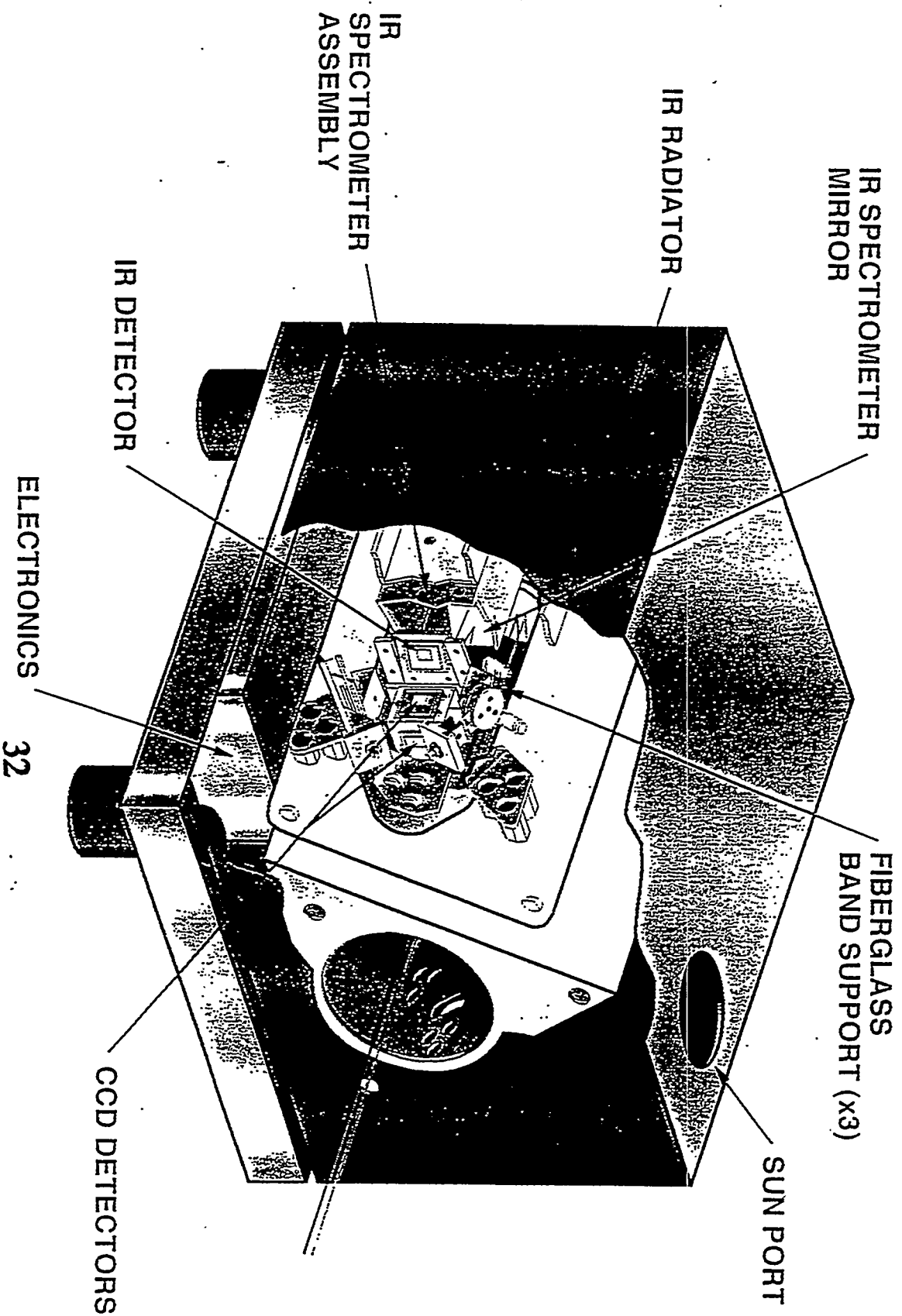
Mass: 6.8 kg

Power: 5.9 watts

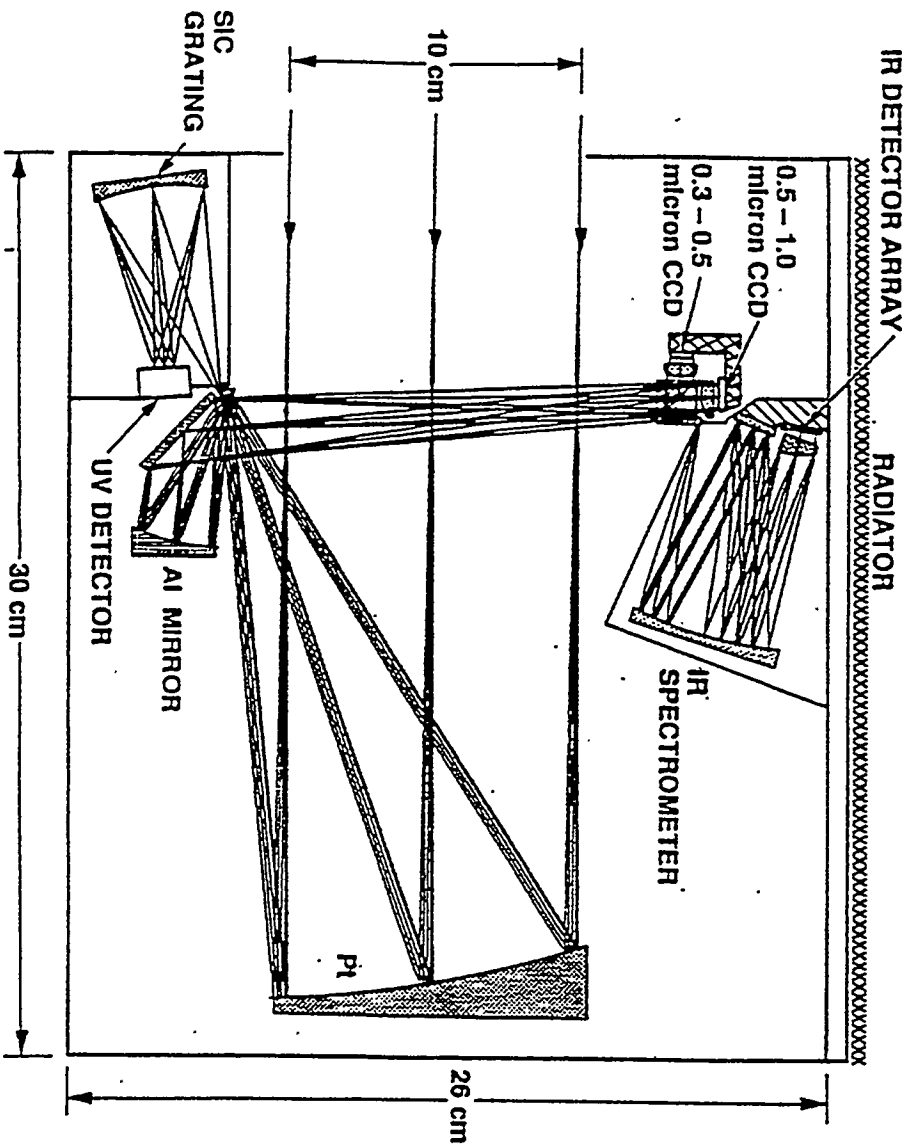
Pointing Stability: Better than 10 microrads/sec

Cost: < \$1M for the total development

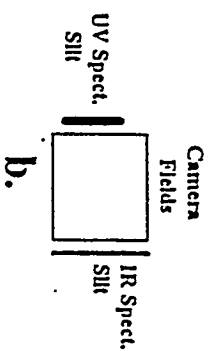
NEW VISION SOLAR SYSTEM MISSION STUDY
INSTRUMENT FOR MICROSPACECRAFT MISSIONS
PLUTO INTEGRATED CAMERA-SPECTROMETER (PICS)



NEW VISION SOLAR SYSTEM MISSION STUDY INSTRUMENT FOR MICROSPACECRAFT MISSIONS Pluto Integrated Camera-Spectrometer (PICS) Instrument



a.



b.

- a.) Optical Raytrace of Off-axis Gregorian Multi-wavelength Telescope
- b.) Schematic layout of the Camera and Spectrometers' Field of View

NEW VISION SOLAR SYSTEM MISSION STUDY INSTRUMENT FOR MICROSPACECRAFT MISSIONS

PLUTO INTEGRATED CAMERA-SPECTROMETER (PICS)

MASS AND POWER PER ELEMENT

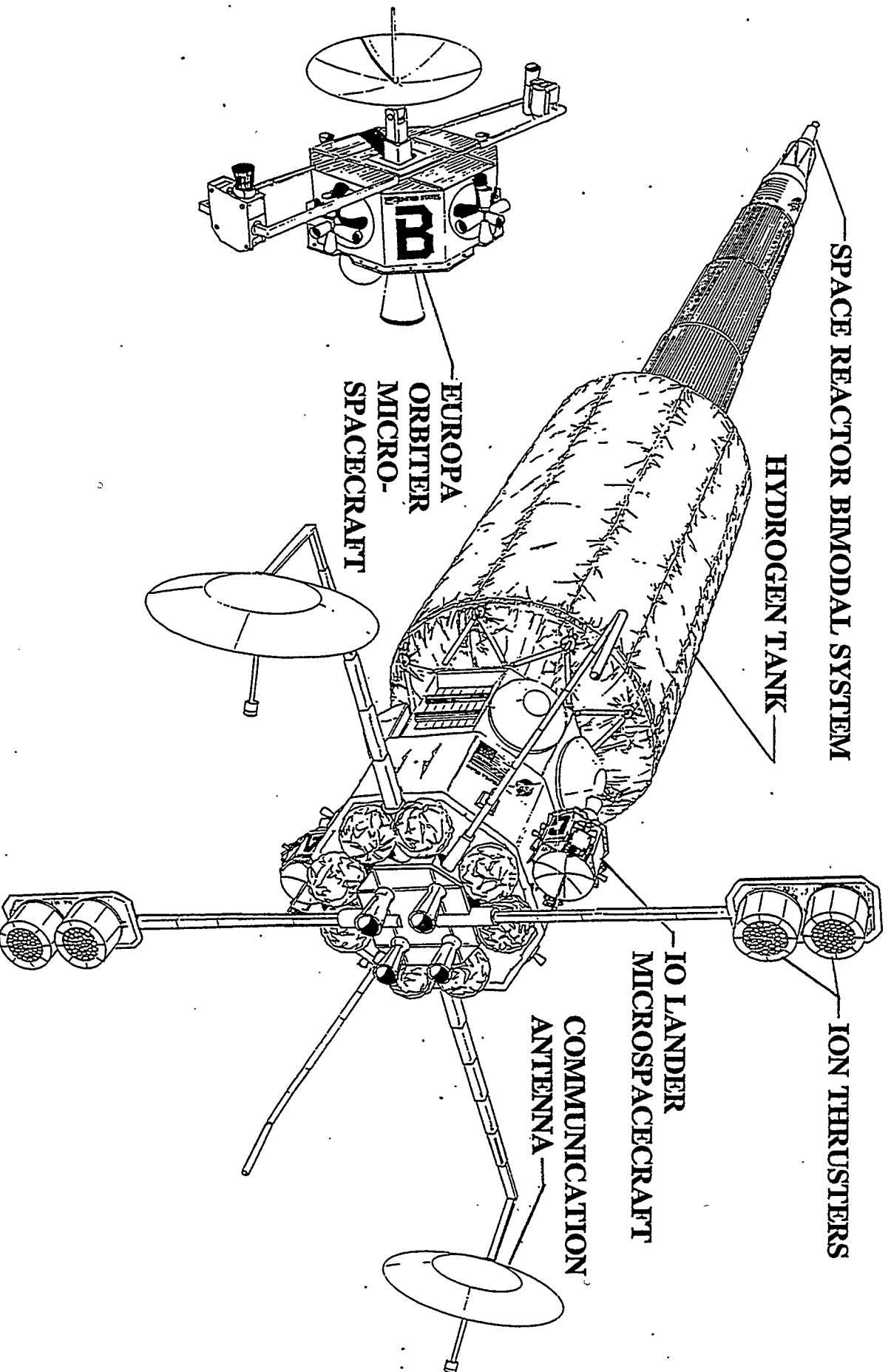
Element	Mass (kg)	Power (watts)
Telescope	2.3	-
Cover & Mechanism	0.5	-
Integrated Radiator/Detectors	0.5	-
Radiation Shielding	0.5	-
Integrated Electronics	0.4	3.9
Power Supply	0.5	1.5
Cabling	0.4	-
Sub-total Contingency	5.1 (33%) 1.7	5.4 (10%) 0.5
TOTAL	6.8	5.9

NEW VISION SOLAR SYSTEM MISSION STUDY INSTRUMENT FOR MICROSPACECRAFT MISSIONS

Pluto Integrated Camera-Spectrometer (PICS) Instrument

The mass and power breakdown shown on the facing page was the original allocated mass and power for the prototype instrument. The prototype instrument actual measured mass is 6 kg, and the power required is 5.5 watts. With some improvements it is estimated that a Flight System PICS can be designed and built with a mass of 4 kg and a required power of 5 watts.

Bimodal S/C Deploying MicroS/C Orbiter at Saturn



Bimodal S/C Deploying MicroS/C at Saturn

The SRB microspacecraft augmented mission to Saturn begins with the bimodal spacecraft launched by at Atlas 2AS to LEO. The bimodal spacecraft then ejects itself from LEO onto a 2EGA trajectory which returns it to Earth with a C3 of about $80 \text{ km}^2/\text{s}^2$. This is not quite enough to reach Saturn, so after the Earth encounter about 1 km/s of ΔV (2 km/s with gravity losses included) must be generated by electric propulsion to accelerate the spacecraft to a C3 of about $100 \text{ km}^2/\text{s}^2$, which allows it to reach Saturn in 3.9 years (5.9 years total flight time). Alternatively, if the mission is launched in a year for which a Jupiter gravity assist is available (3 years in a row out of every 12), the 2EGA trajectory becomes sufficient to reach Saturn via Jupiter without NEP acceleration on the outbound leg. In this case Jupiter is encountered about 4.5 years into the mission, at which time if a SRB spacecraft is operating in orbit around Jupiter a few microspacecraft can be released to reinforce the investigations there, after which the main spacecraft flies on to encounter Saturn about 7 years after launch. In either case, electric propulsion is used to reduce the hyperbolic velocity on approach to about 3.9 km/s (see Fig. 15) after which the spacecraft captures into an $2 \text{ RS} \times 100 \text{ day}$ elliptical orbit using its high thrust bimodal propulsion system with NH_3 propellant. The total electric ΔV prior to capture is 1.5 km/s (0.5 year burn time) for the 2EJGA mission, and 4.5 km/s (1.5 year burn time) for the 2EGA mission. Such short NEP burn times make this mission highly feasible for near-term NEP technology.

After capturing, the SRB S/C releases a micro-orbiter (see facing page), leaving it in the initial moderately inclined, (Saturn's equator is inclined 27° to the ecliptic) highly elliptical orbit from which it can survey the whole Saturn system from above.

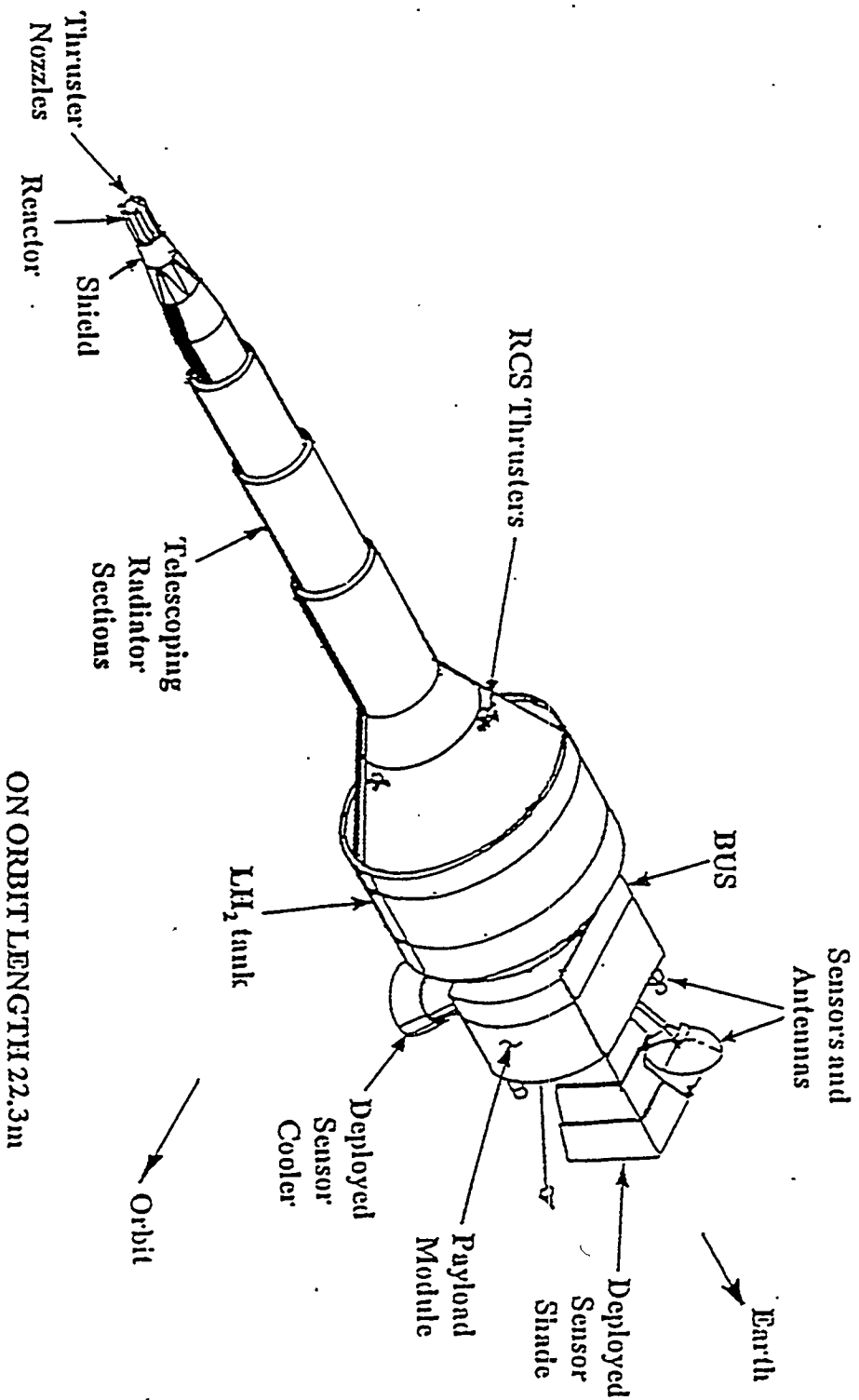
Mass of Bimodal NEP Spacecraft

	<u>5 kWe</u>	<u>10 kWe</u>
TWTAs (1 kW rf X-band units, 7 kg each)	21	35 kg
TWTA Power Converters (2 kWe, 9 kg each)	27	45
TWTA Low temperature radiators	4	8
Other Telecom (1,2 3-m dishes, gimbal etc.)	18	35
Auxiliary Solar Power (2,3 square meters)	10	15
Batteries (600,1200 W-hr NiMH batteries)	12	24
Attitude Determination and Control	18	18
C&DH Payload ICAPS (redundant units)	10	10
Thermal protection	10	10
ACS Propulsion system	18	18
Bus structure	30	30
Adaptive Structure (boom and cannister)	70	90
Science Instruments	50	100
S/C Contingency (20%)	60	88
S/C total	358	526
Reactor System (10 kWe, 250 lbf, 850 s)	1350	1500
Electric Propulsion System (Xe-ion 18 kg/kW)	90	180
Total Dry Mass	1798	2206 kg

Mass of Bimodal NEP Spacecraft

The spacecraft was conceptually designed to transport at least 100 kg of science instruments or equivalent mass of several microspacecrafts to the outer planets, and provide high communication rates back to Earth. Therefore the 5 kwe spacecraft has one 3 meters fixed communication antenna and the 10 kwe has two 3 meters communication antenna. So the total power of the Space Reactor Bimodal (SRB) system can be used to send science data at high data rates. The mass breakdown of the spacecraft is shown on the facing page.

NEBA-3 Orbital Configuration



NEBA-3 Orbital Configuration

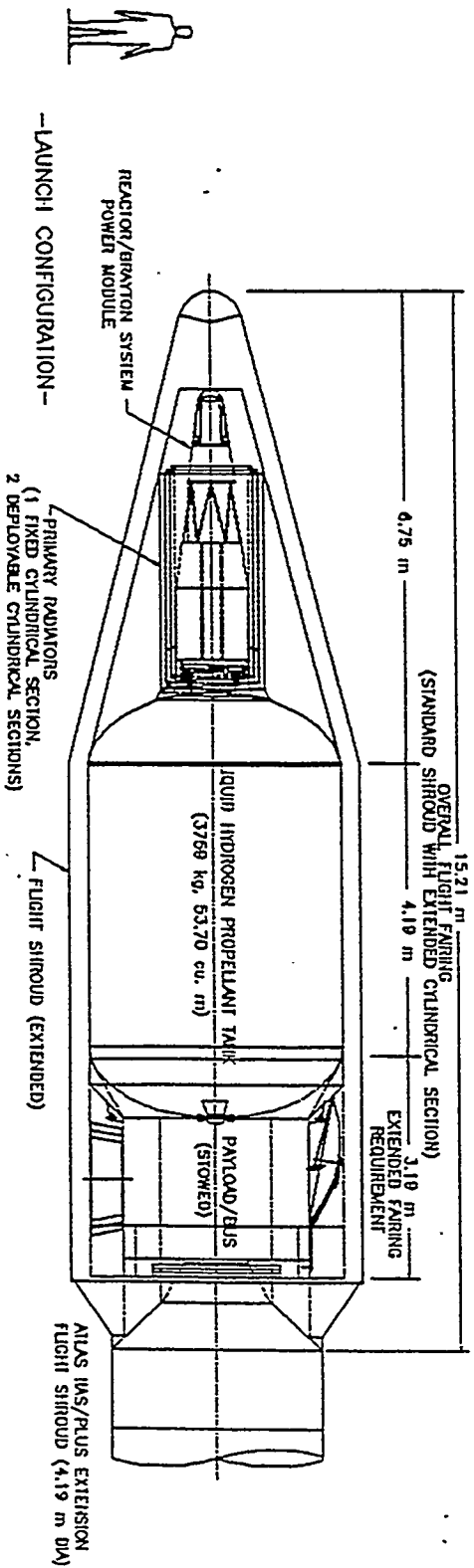
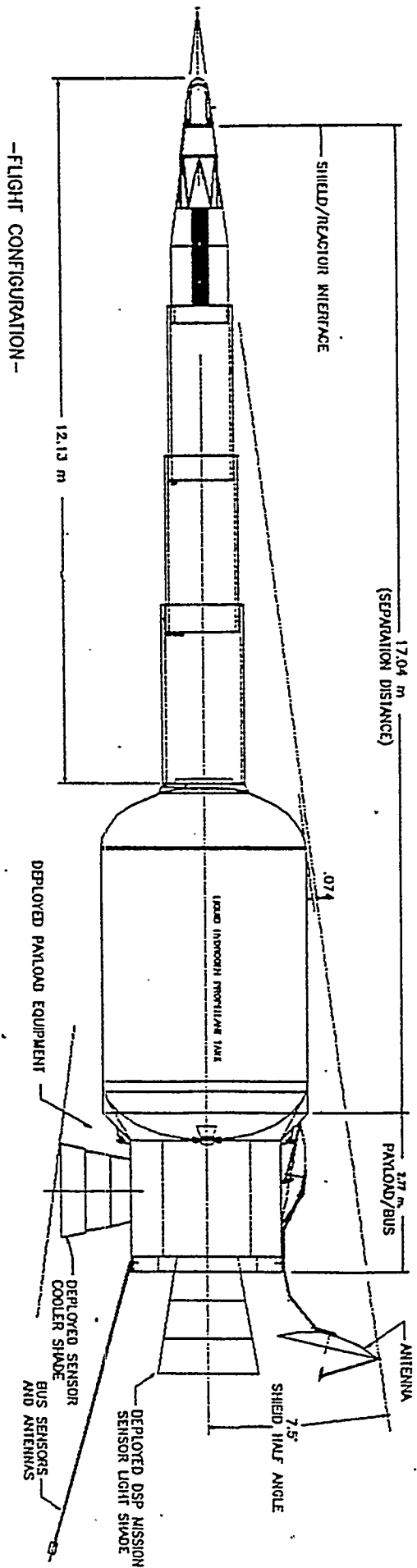
A 10 kW_e SRB flight spacecraft was defined for outer planet missions based on the Air Force Phillips Laboratory Earth Orbital 19 kW_e spacecraft, and is shown on the facing page. A mass estimate for the SRB spacecraft configured for a Jupiter or Saturn orbiter mission using a 2-year Earth gravity assist (EGA) is shown below.

Mass of 10 kW_e SRB Spacecraft

<u>Parameters</u>	<u>Value</u>
Science Instruments (kg)	100
Spacecraft Total (kg)	425
Space Reactor Bimodal Subsystem (kg)	1460
Electric Propulsion Subsystem (Xe-ion 18 kg/kW _e) (kg)	180
NH ₃ Propellant & Tanks (kg)	370
Xenon Propellant & Tanks (kg)	490
<u>Hydrogen Propellant and Tanks (kg)</u>	<u>3325</u>
Total Mass in Low orbit (kg)	6350

An Atlas 2AS has a LEO launch capability of about 8000 kg. Therefore, the above spacecraft would have a launch margin of about 26%, or could transport an additional twelve 20 kg microspacecraft, and still have a 20% launch margin.

NEBA-3 in Atlas 2AS and Deployed



NEBA-3 in Atlas 2AS and Deployed

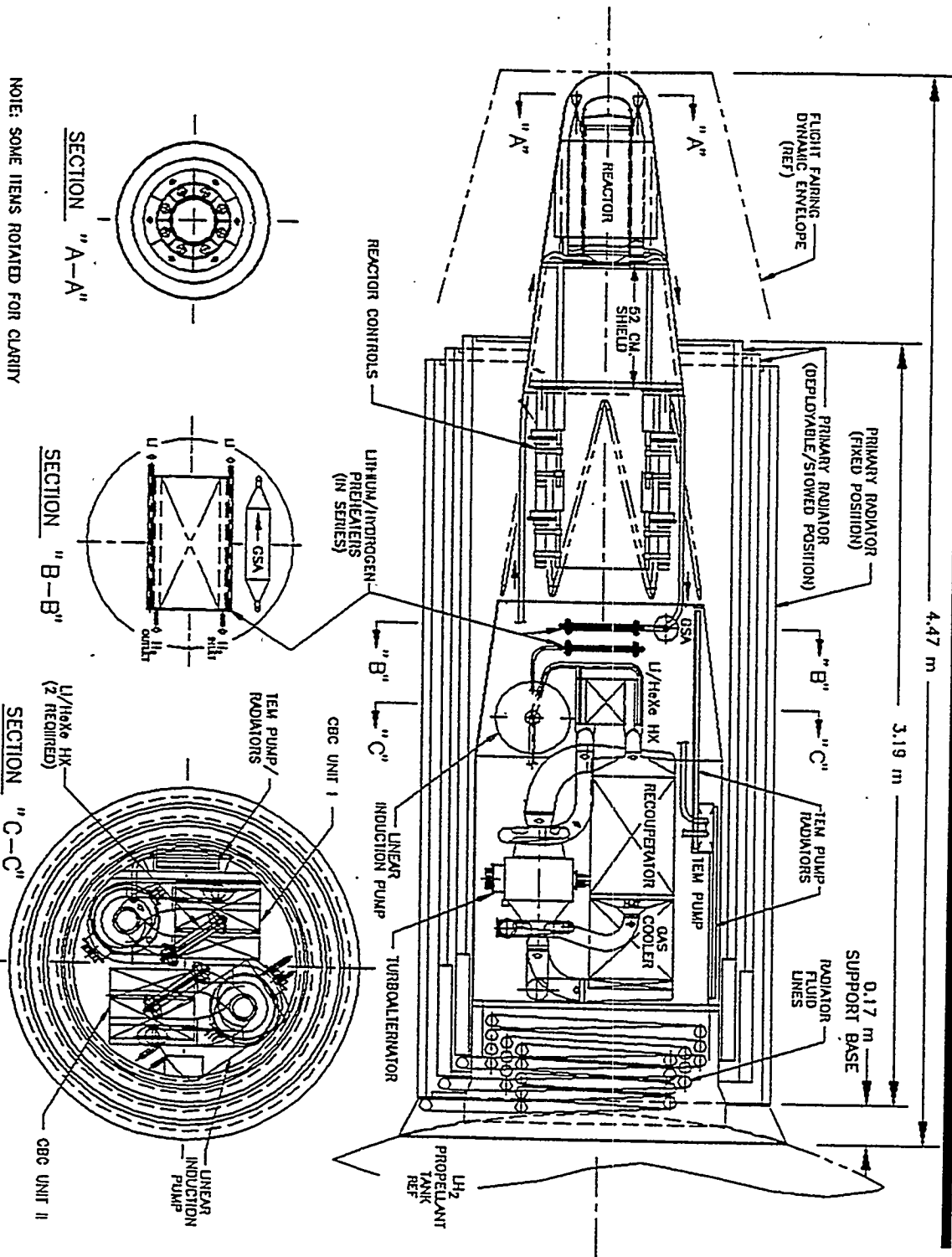
The deployed and stowed configurations of the NEBA-3 spacecraft are shown on the facing page. The drawing illustrates the telescoping radiator concept which has been selected in this conceptual design stage as the baseline configuration. The stowed package is constrained by the diameter of the standard Atlas IIAS shroud, with an added fairing extension to accommodate the required liquid hydrogen propellant for the mission. The hydrogen tank shown is for the case of the moderate thrust system and includes a 5 percent ullage.

The payload module is located behind the hydrogen tank, and when deployed has a separation distance of 17 meters from the base of the reactor. The separation distance is indicated by the 7.5 degree one-half cone angle selected for the reference design and the diameter of the fairing at 4.19 meters.

The heat rejection radiator consists of three cylindrical sections, and when deployed acts as the telescoping boom that separate the reactor module from the rest of the spacecraft. The poor view factor to space of this cylindrical radiator which translates into higher radiator mass, is more than compensated by the elimination of a separation boom and the simplicity in its deployment when compared to the flat plate and conical radiator configurations that were investigated.

The liquid hydrogen required to accomplish the mission is the largest component of the spacecraft from both the mass and volume standpoints. Two approaches to package the hydrogen tank in the Atlas IIAS launch vehicle are considered. The internal tank concept stores the tank inside the fairing skin typical of the conventional approach, while the external approach employs the tank surface as part of the fairing skin, thus allowing a wider tank diameter and a shorter tank length. The use of the external tank concept reduces the fairing extension requirement by about 1.3 m.

NEBA-3 CBC Power System Conceptual Layout



NOTE: SOME ITEMS ROTATED FOR CLARITY

NEBA-3 CBC Power System Conceptual Layout

CBC POWER SYSTEM CONCEPTUAL LAYOUT

The layout of the closed Brayton cycle conversion subsystem is illustrated on the facing page for the moderate thrust system which includes a linear induction pump and larger lithium-to-hydrogen preheaters. All the CBC components are located aft of the reactor control drives, which require open view to space for radiative cooling of the drive motors. Each closed Brayton cycle (CBC) unit with its three heat exchangers is packaged such that the pipe lengths for HeXe flow between components are minimal in order to reduce pressure drops and increase conversion efficiently.

The three cylindrical radiator sections are configured in parallel hydraulically with flexible fluid lines stowed between the rotating CBC units and the hydrogen propellant tank, wrapping around the perimeter of the deployment mechanism.

REACTOR

The design of the UN fuel pins of the lithium-cooled space reactor is essentially the same as the design of the SP-100 fuel pins. In fact, the dimensions of the fuel pellets used in the NEBA-3 design evaluations are those of the pellets previously fabricated and stored at Los Alamos National Laboratory. The quantity of highly enriched pellets now in storage would be sufficient to fabricate several reactors of the current NEBA-3 design. Although the small diameter pellets were originally sized for the SP-100 Nuclear Assembly Ground Test Reactor, they are needed for the NEBA-3 design to accommodate the high power densities required during the moderate thrust propulsion mode. The method of suspending and securing the fuel pins in the inner vessel would be the same system as used in the SP-100. The reactor head closure design was taken from a flight design approach different from that of the SP-100 ground test reactor which used ACME threads in addition to welding. The NEBA-3 reactor head closure and weld must be compact and lightweight. The weld must be full penetration, inspectable, and designed to permit reactor head removal at least twice after welding.

Mission Strategy Bimodal Spacecraft

SPACECRAFT DELIVERED TO 400 km CIRCULAR ORBIT BY LAUNCH VEHICLE.

THERMAL THRUSTER (850 s Isp, 250 lbf THRUST) USED WITH H₂ IN SERIES OF PERIGEE KICKS TO RAISE ORBIT TO ESCAPE & PERFORM INTERPLANETARY INJECTION.

- Thrust arcs + -30 degrees of perigee keep LEO-escape gravity losses to less than 5%.
- Full power burn-time to escape on order of 5 to 10 hours.
- Total flight time till Earth escape on order of 5 to 10 days.

TRAJECTORY TO PLANET IS ESSENTIALLY BALLISTIC. NEP BURN TIME IS MINIMIZED.

--NEP used mainly to reduce hyperbolic approach velocity near planet & for mid-course corrections. Also can be used to accelerate 2+ΔV-BGA trajectories after Earth fly-by, allowing Saturn, Uranus, Neptune, Pluto to be reached.

FINAL CAPTURE INTO ELLIPTIC ORBIT IS DONE WITH THERMAL THRUSTER USING NH₃ (450 s Isp, 500 lbf). PERIOD < 100 days. ΔV = 0.5 km/s.

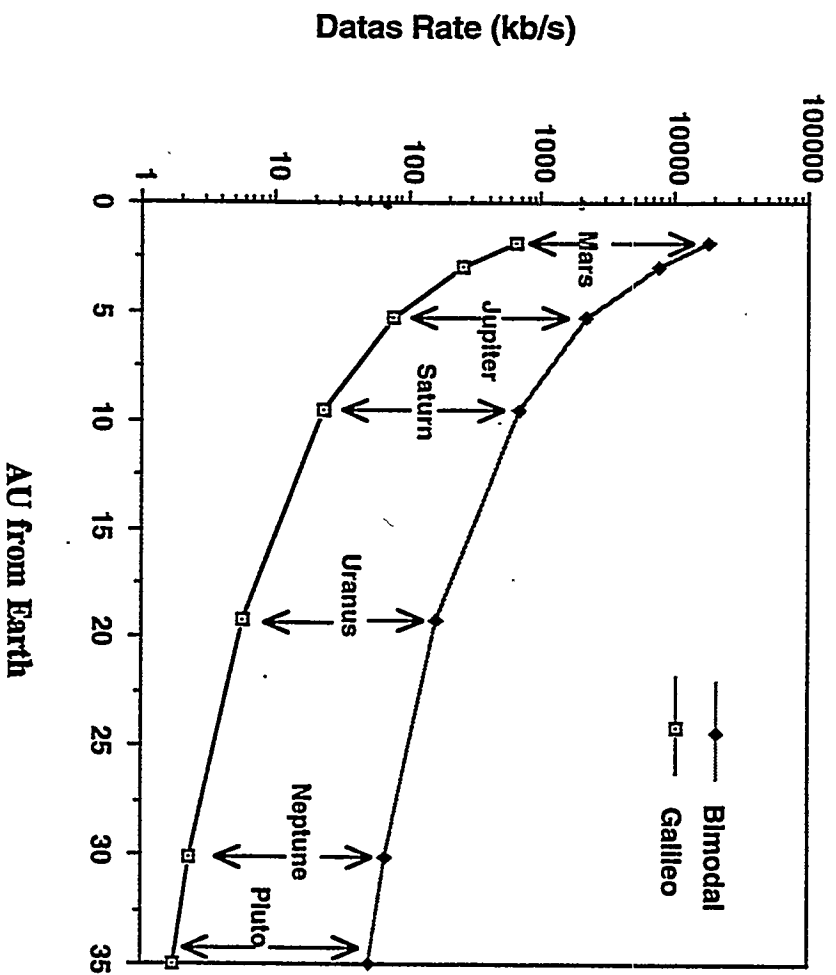
MISSION STRATEGY BIMODAL SPACECRAFT

The bimodal spacecraft has an intermediate thrust to weight (T/W) ratio (about 0.02), one which is very low compared to spacecraft propelled by chemical thrust systems (with $T/W \sim 1$) but very high compared to electric propulsion systems (with $T/W \sim 0.00001$). For this reason it must adopt a LEO escape strategy which differs substantially from either of these two more traditional systems. Specifically, rather than burn directly out of LEO onto an interplanetary trajectory as a chemical system would, or spiral out to Earth escape with a continuous burn lasting many months as an electric propulsion spacecraft would, the bimodal spacecraft escapes from LEO into interplanetary space via a series of approximately 30 minute long perigee kicks. Each of these kicks raises the apogee of the orbit substantially while keeping the perigee about the same. For this reason, so long as the burns are kept in a thrust arc no more than 60 degrees about the actual perigee, little gravity losses are sustained, until the final burn when Earth escape is reached and the thruster must be made to operate continuously until the desired hyperbolic energy, or C_3 , is reached. In the case of the 10 kWe SRB spacecraft, about 13 perigee kicks are required over a period of 3 days to achieve Earth escape, with a total burn time on the thermal thruster of about 6.5 hours spread over that period.

The trajectory to the planet is essentially ballistic, keeping nuclear electric propulsion (NEP) burn time to a minimum. NEP is used to reduce hyperbolic approach velocity near the planet and for mid-course corrections.

The capture into an elliptical orbit about the planet is done by using the reactor thermal thruster with ammonia, NH_3 , propellant at 450 sec Isp and 2225 Newton thrust.

Data Rates of 10 kW_e Bimodal & 300 W Galileo S/C



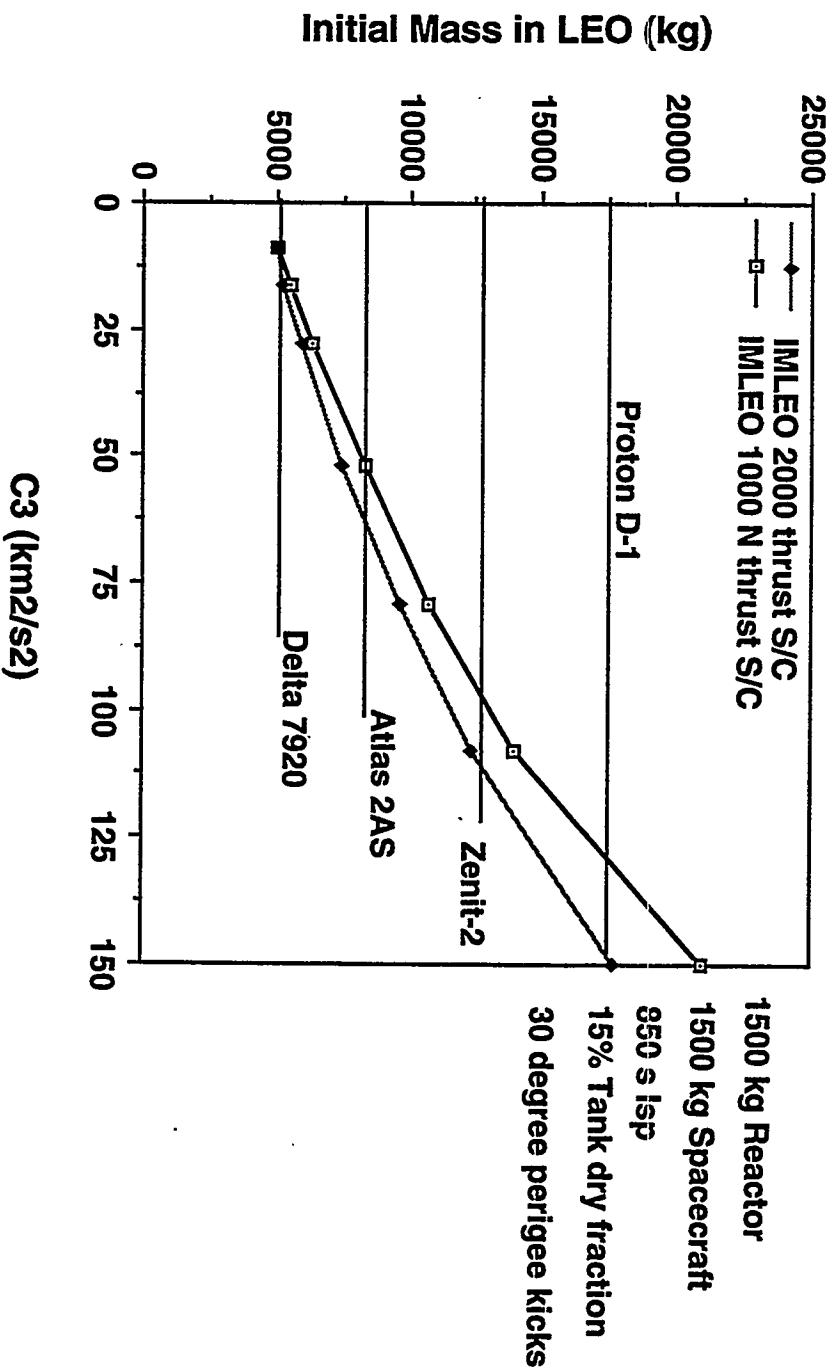
Analysis assumes Galileo S/C with fully functional 5 m dish, Bimodal with two 3 m dishes.

Both S/C are transmitting via X-band to 70 m DSN dish.

Data Rates of 10 kWe Bimodal and 300 W Galileo S/C

The figure on the facing page shows data transmission rates for a probe in orbit around other planets, assuming either a 200 W RTG (40 W rf) for transmitter power with a 5 M X-band dish or our design 10 kWe bimodal spacecraft transmitting with two 3 m X-band dishes to a 70 m DSN antenna. If the cheaper-to-use 34 m dishes are employed instead, data rates for both would be 1/4 those shown. It can be seen that, anywhere in the solar system, the 10 kWe Bimodal spacecraft with two 3-meter dishes can return nearly two order of magnitude more data than a Galileo class spacecraft with a 5 m dish.

Mass of Missions Launched by Bimodal Systems

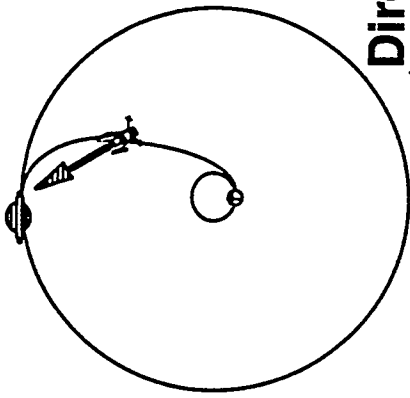


Mass of Missions Launched by Bimodal Systems

It can be seen that if the C3 is less than $52 \text{ km}^2/\text{s}^2$, then the point design spacecraft can be launched (with no margin other than spacecraft contingency) by an Atlas 2AS. It will also be noted, however, that if the launch C3 is $28 \text{ km}^2/\text{s}^2$, then the launch margin for the spacecraft on at Atlas 2AS is about 27% (8000 kg capability vs 6300 kg requirement). This is a significant result since a C3 of $28 \text{ km}^2/\text{s}^2$ is what is required to perform a 2 year Earth Gravity Assist, a maneuver by which a spacecraft can be delivered to Jupiter (and therefore beyond by means of a Jupiter gravity assist) with a launch window occurring every year.

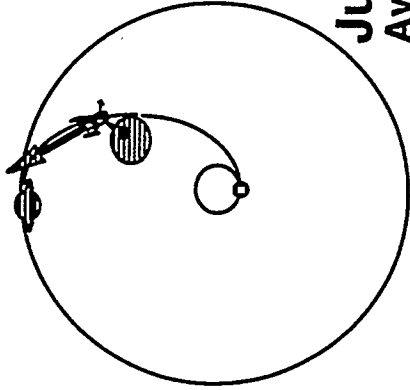
The 27% launch margin made available by adopting such a trajectory is excessive, and part of it can be used to increase the spacecraft's capability. For example, if a more standard launch margin requirement of 15% were adopted, then an initial spacecraft mass in LEO of 6930 kg could be tolerated. This would translate into an increase in dry mass of the spacecraft of 216 kg, which could all be added to the science payload thereby tripling it from 100 kg to 316 kg. Alternatively, the 216 kg could be used to add Xenon propellant to the electrical propulsion system, thereby increasing the post injection ΔV capability of the spacecraft from the point design system's 7 km/s up to about 12 km/s. Such an increased ΔV capability would allow the spacecraft to be delivered into orbit around Neptune or Pluto (in other words to any planet in the solar system). These sensitivities are illustrated in the appendix.

Types of Trajectories Examined



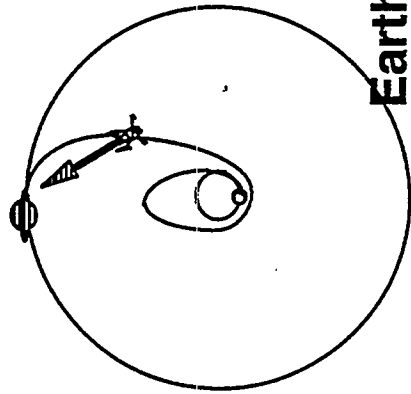
Direct

Available Every Year
Highest Launch Energy



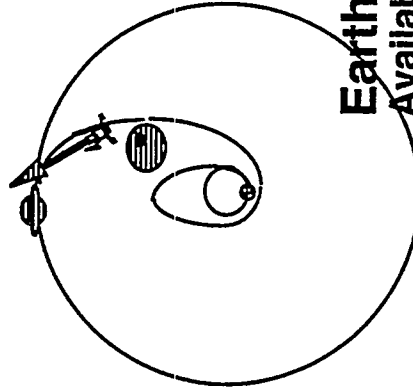
Jupiter Gravity Assist

Available 3 years in every 12
High Launch Energy



Earth Gravity Assist

Available Every Year
Low Launch Energy



Earth-Jupiter Gravity Assist

Available 3 Years in Every 12
Lowest Launch Energy

TYPES OF TRAJECTORIES EXAMINED

Strategies for minimizing the LEO mass of the bimodal spacecraft benefit strongly by keeping its injection C3 to a minimum. Furthermore, since the mass which must be added to achieve higher injection C3 is bulky hydrogen propellant, minimizing C3 may also be important from the point of view of meeting launch packaging requirements. For this reason two unusual types of Earth gravity assist trajectories, termed 1 EGA and 3/2 EGA were investigated.

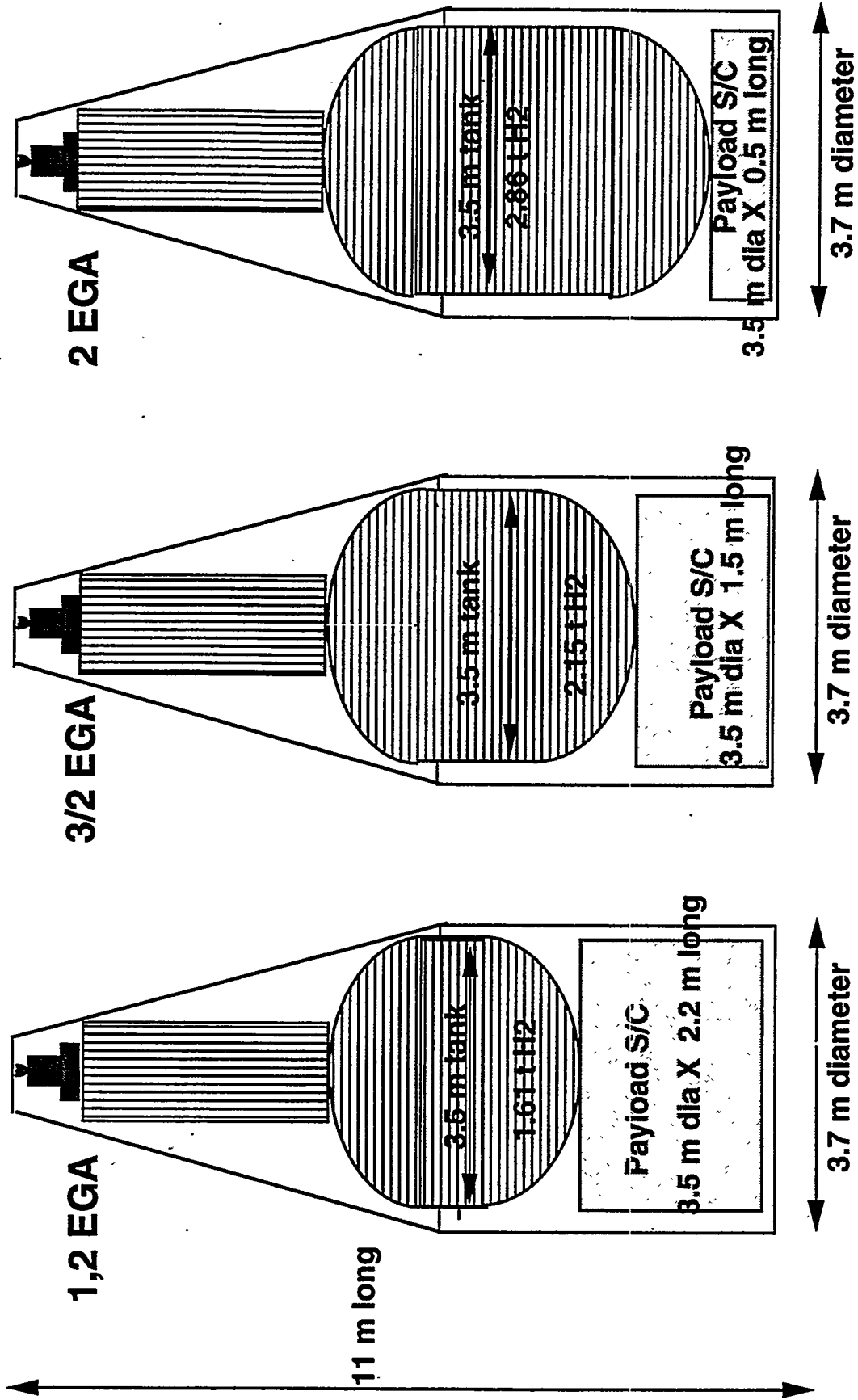
The 1 EGA trajectory leaves Earth with a C3 between 1 and 2 km²/s² and heads out on a trajectory with an aphelion of about 1.1 AU and perihelion about 0.9 AU. At aphelion, a substantial midcourse correction ΔV maneuver of about 1.6 km/s is done, which causes the spacecraft to have a C3 upon Earth return, (1 year after departure) of around 23 km²/s². This is almost enough to send the spacecraft onto a conventional 2 year EGA trajectory, after which the spacecraft will re-encounter Earth with a C3 of about 90 km²/s² and have sufficient energy to reach the outer solar system (a 1,2 EGA). Thus in exchange for adding 1 year to the total mission flight time, the 1 EGA strategy greatly reduces the mission C3.

The 3/2 EGA trajectory idea uses an injection C3 of about 11 km²/s² to send the S/C out on a trajectory with an aphelion of about 1.62 AU and a perihelion close to 1 AU. This trajectory has a period of 1.5 years, so after going twice around the Sun it will re-encounter the Earth 3 years after initial departure. The maximum useful C3 that can be generated upon Earth return by this trajectory is about 45 km²/s², which requires a midcourse ΔV of about 0.6 km/s, which could be taken on either of (or split between) the two orbits prior to Earth re-encounter.

These trajectories have not been examined in the past as a means of facilitating high energy missions to the outer solar system, because with conventional high-thrust chemical propulsion systems they do not make sense (i.e., the required midcourse ΔV is greater than the extra ΔV required to reach the high C3 by burning directly out of LEO).

However is the case of the bimodal system, the ΔV required to reach high C3's directly out of LEO is inflated by gravity losses, while the midcourse ΔV can be disposed of with electric propulsion. Thus the 1 and 3/2 EGA trajectories are uniquely beneficial to bimodal systems.

10 kWe Bimodal NEP Spacecraft Configurations



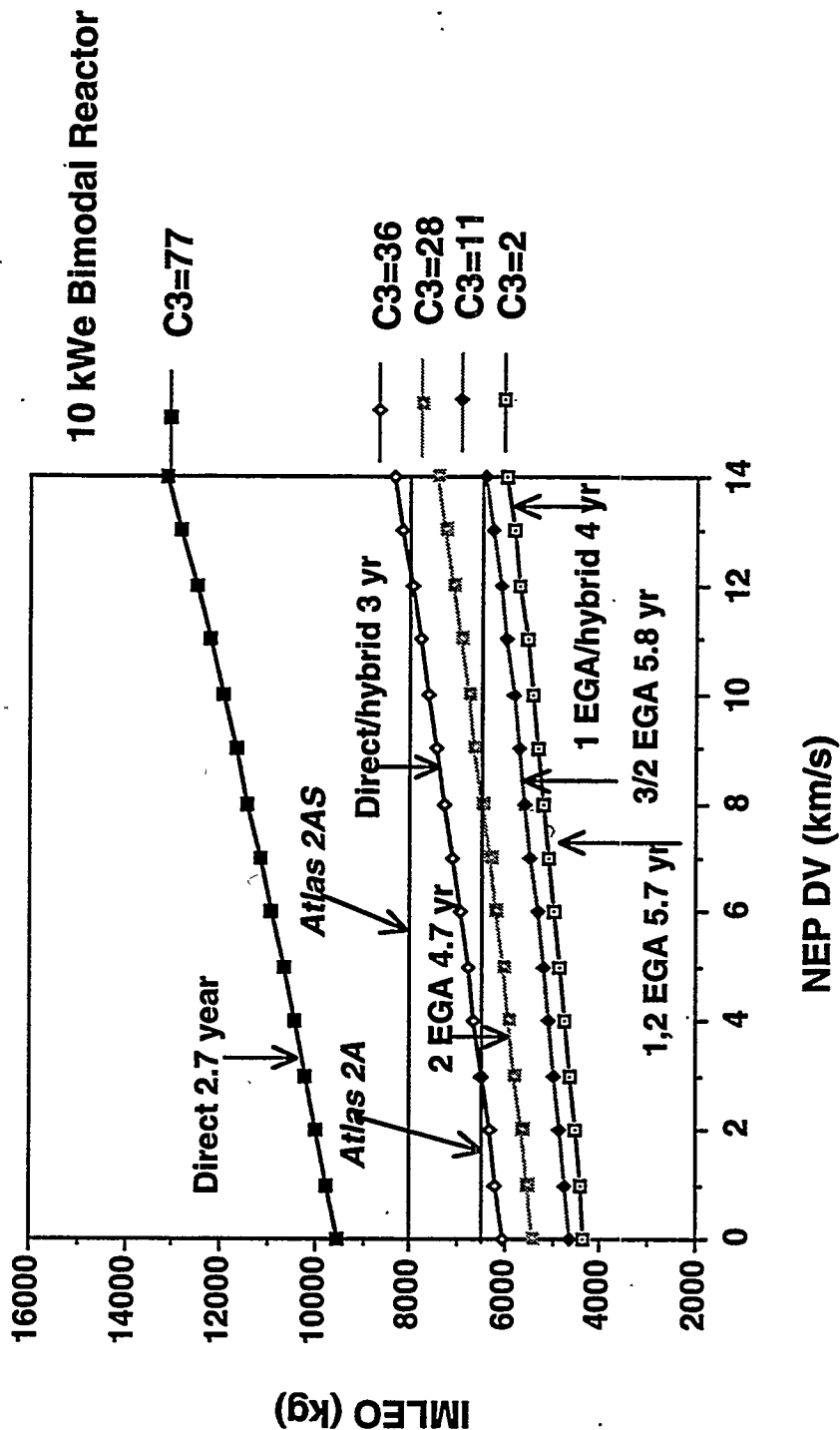
Missions to Jupiter or Saturn
Atlas 2A Launch

Max Sci RZ 8/1/95-1

10 kW Bimodal NEP Spacecraft Configurations

The facing page illustrates the amount of hydrogen propellant required for missions to Jupiter or Saturn using an Atlas 2A launch vehicle and different Earth Gravity assists (EGA) trajectories. The 1 year EGA followed by a 2 year EGA reduces the mass of hydrogen by 1.25 tonnes over the 2 year EGA, and adds 1 year to the total flight time to Jupiter or Saturn. This allows much more volume on the Atlas 2A for the spacecraft payload. A 3 year EGA saves 0.71 tonnes of hydrogen over the 2 year EGA, and only uses Earth for one gravity assist. For more details on advantages and disadvantages of these trajectories, see the Appendix.

Mass & Flight Time of Bimodal Jupiter Missions

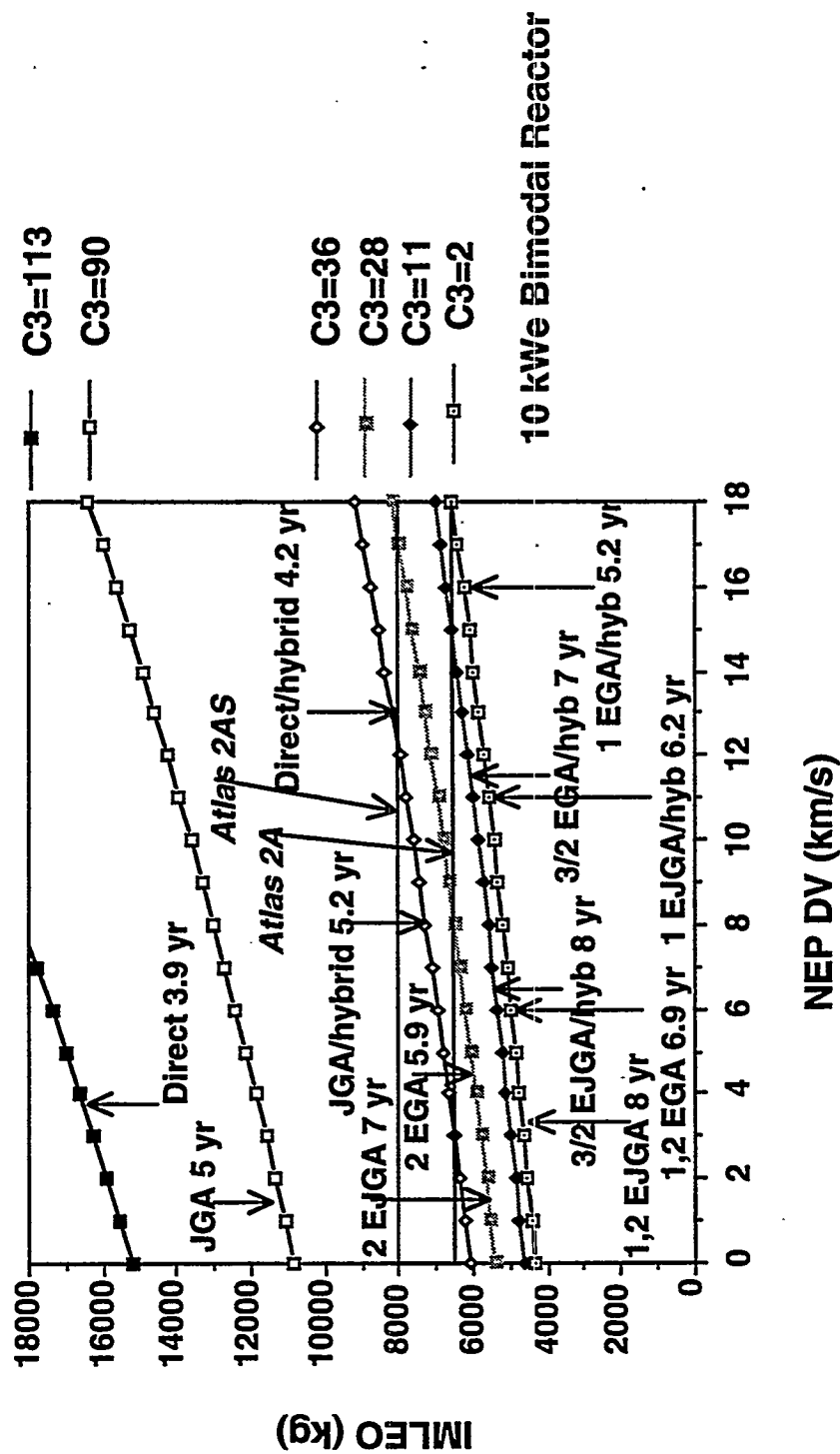


Max Sci RZ 11/30/95-12

Mass & Flight Time of Bimodal Jupiter Mission

A map of options for Jupiter orbiter missions delivered by the Space Reactor Bimodal (SRB) spacecraft is presented in the graph on the facing page. It can be seen that while direct missions can be achieved with flight times of 2.7 years and NEP burns of less than 3.5 km/s, such missions considerably exceed the launch capability of the Atlas 2AS. The standard missions plan of a simple 2-year EGA trajectory is shown ($C3 = 28 \text{ km}^2/\text{s}^2$, $\text{IMLEO}=6300 \text{ kg}$, flight time = 4.7 years, NEP burn = 3.8 years), and can be compared with such options as the 3/2 EGA ($C3=11 \text{ km}^2/\text{s}^2$, $\text{IMLEO}=5500 \text{ kg}$, flight time = 5.8 years, NEP burn = 8.4 km/s) and the 1,2 EGA ($C3=2 \text{ km}^2/\text{s}^2$, $\text{IMLEO}=5100 \text{ kg}$, flight time = 5.7 years, NEP burn = 7.2 km/s). Compared to the 2 EGA baseline, the 1,2 EGA is only 80% as massive. Thus if this trajectory were adopted, the science payload could be increased to about 600 kg and the mission would still have 27% launch margin against the capability of the Atlas 2AS. Such a large science payload is an important capability for Jupiter orbiter missions because ten to twelve microspacecraft can be delivered to Jupiter and its satellites. The number of years cited for each option is the actual time of flight. The NEP burn time is given by correlating the required NEP ΔV with the burn for that ΔV , and is approximately $\Delta V(\text{km/s})/3$. Arrows are placed for ΔV for minimum mission requirement. To include post orbit capture maneuver ΔV in mission plan, move arrow to the right along the specified C3 curve.

Mass & Flight Time of Bimodal Saturn Orbiter Missions



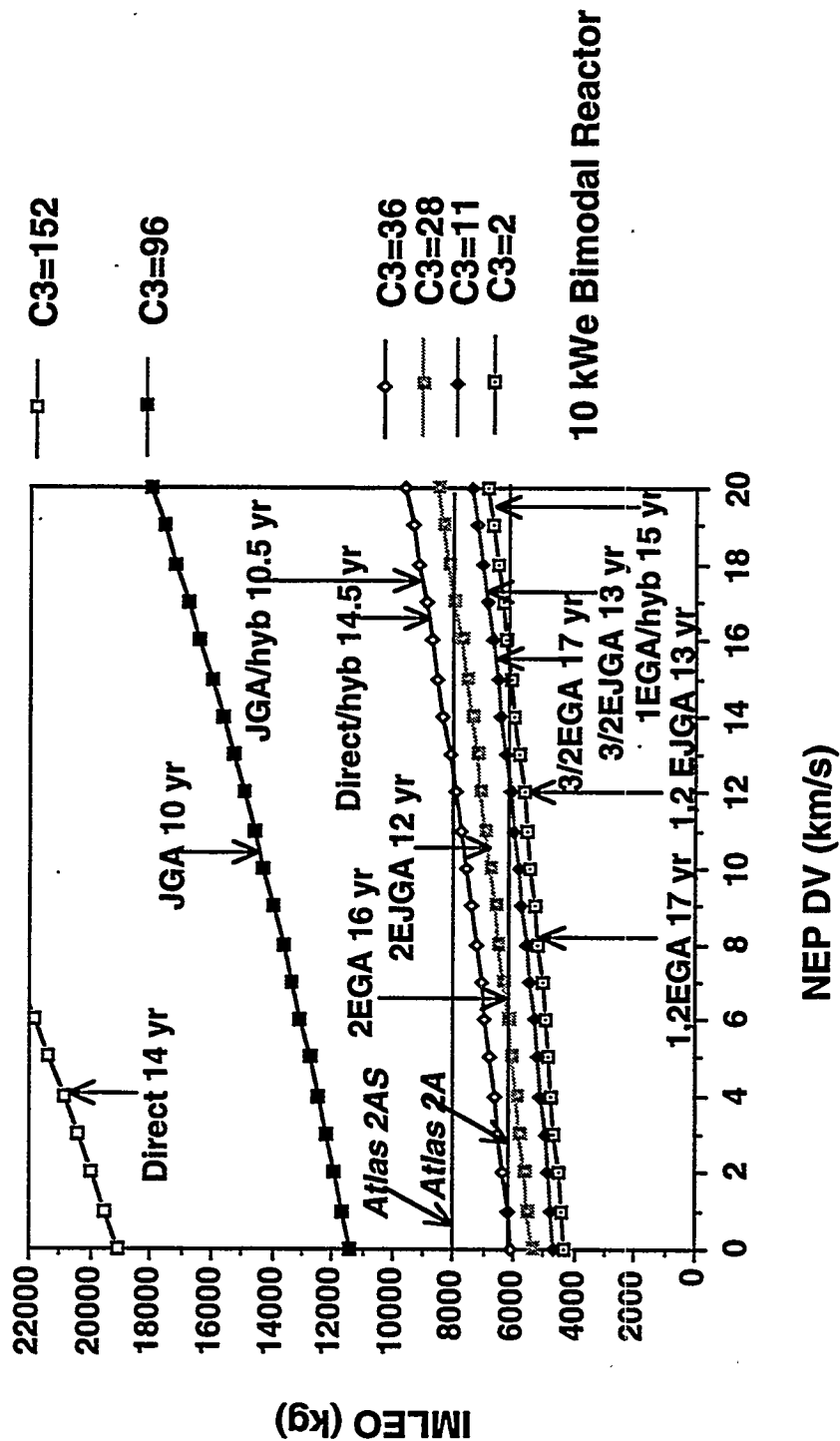
LOCKHEED MARTIN

Max Sci RZ 11/30/95-13

Mass & Flight Time Of Bimodal Saturn Orbiter Missions

This graph depicts similar trajectory space maps for Saturn orbiter missions with the SRB spacecraft. Once again, in the case of the Saturn missions, ten to twelve microspacecraft (indicated by the very large mass margin against the Atlas 2AS when carrying the standard 100 kg science payload) can be delivered by a 1,2 EGA trajectory and a 6.9 year flight time.

Mass & Flight Time of Bimodal Neptune Orbiter Missions



Max Sci RZ 11/30/95-14

Mass & Flight Time of Bimodal Neptune Orbiter Missions

This graph shows similar trajectory space maps for Neptune orbiter missions with the SRB spacecraft. Once again, in the case of the Neptune missions, several (ten to twelve) microspacecraft can be delivered to Neptune orbit with a 13 year flight time if the 1,2 EJGA trajectory is used. This trajectory will have the spacecraft fly through the Jupiter system about 6 years into the mission, allowing microspacecraft to be delivered and emplaced in Jupiter (or Jovian moon) orbit at that time.

10 kWe Bimodal Planetary Orbiter Missions

<u>Planet</u>	<u>C3</u>	<u>Flt Time</u>	<u>Vhyp</u>	<u>NEP DV</u>	<u>Burn time</u>	<u>IMLEO</u>	<u>Launch Vehicle</u>
Mars (dir)	12	0.7 yrs	3.5 km/s	1.5	0.5 yrs	4400 kg	Delta 7920
Jupiter (dir)	77	2.73	5.6	3.31	1.1	10230	Zenit-2
Jupiter(hyb)	36/77	3.0	5.6	8.86	2.95	7280	Atlas 2AS
Jupiter (EGA)	28	4.73	5.6	3.81	1.27	5700	Atlas 2
Saturn (dir)	113	3.9	7.6	3.71	1.24	13500	Proton D-1
Saturn (hyb)	36/113	4.0	7.6	12.97	4.32	7910	Atlas 2AS
Saturn (JGA)	89	5.0	5.0	1.11	0.37	10050	Zenit-2
Saturn (EGA)	28	5.9	7.6	4.21	1.40	5750	Atlas 2
Uranus (dir)	141	9.0	7.6	4.03	1.34	19950	Titan IV
Uranus (JGA)	85	7.0	14.1	10.52	3.5	12600	Zenit-2
Uranus (EGA)	28	10.0	11.7	8.64	2.88	6120	Atlas 2
Neptune (dir)	151	14.0	7.74	4.03	1.32	20900	Titan IV
Neptune(JGA)	96	10.0	14.1	10.3	3.43	12700	Zenit-2
Neptun(EJGA)	28	12.0	14.1	10.8	3.61	6420	Atlas 2
Pluto (dir)	152	15.0	7.28	6.2	2.06	21120	Titan IV
Pluto (JGA)	114	11.0	12.1	11.0	3.67	14590	Proton D-1
Pluto (EJGA)	28	12.0	13.7	13.1	4.37	6840	Atlas 2A

All missions place an orbiter equipped with 100 kg of instruments and a 4 kW rf X- band transmitter into an elliptical orbit with less than 100 day period about planet.



10 kWe Bimodal Planetary Orbiter Missions

The facing page chart is a summary of key parameters using the 10 kWe Space Reactor Bimodal (SRB) system with different launch vehicles and different trajectories to perform orbiter missions of all the outer planets. It shows that by using Earth and Jupiter Gravity Assists, all the outer planet orbiter missions except Pluto, can be accomplished using an Atlas 2, and for Pluto orbiter missions use an Atlas 2A. It also shows that all the outer planets orbiter can be put in place with the Electric Propulsion lifetimes of less than 4 years, except for Pluto which requires an EP lifetime of 4.37 years.

The SRB spacecraft is placed in orbit about the planet and places microspacecraft at key scientific targets for each outer planet. This 10 kWe SRB spacecraft is then available as a 10 kWe communication orbiter for decades.

USES OF MICROSPACECRAFT WITH NUCLEAR CARRIER S/C

USE MICROS TO COLLECT / ASSESS SAMPLES

Return Samples to S/C or to Earth

USE MICROS TO GAIN 3-D MAPPING PERSPECTIVE

Radar, light, lasers used for illumination

USE MICROS TO CARRY INSTRUMENTS THAT CAN'T WORK NEAR REACTOR

GRS, Rad sensitive optics

USE MICROS AS LOCAL PROBES

Floater, gliders, divers for atmosphere dynamic & chemistry
High resolution imaging from ground rovers and aircraft

USE MICROS AS RECEIVERS FOR RADIO SCIENCE

Atmosphere grazing measurements

USE MICROS AS ORBITAL, ATMOSPHERIC, OR GROUND STATIONS LEFT BEHIND

S/C returns for data later

USE MICROS AS HIGH AND LOW VELOCITY PENETRATORS

Micro on surface can act as sonar relay

USE MICROS AS "FLASHBULBS" TO STRONGLY ILLUMINATE TARGET AREAS

Chemical or nuclear explosives

Uses of Microspacecraft With Nuclear Carrier S/C

The primary purpose of this study was to investigate using fleets of microspacecrafts at each outer planet to create scientific information that interest and includes the general public in the exploration of our solar system. A listing of potential scientific uses of microspacecraft is given on the facing page.

The SRB spacecraft would transport these microspacecraft to the planet and place them at scientific targets to obtain the scientific data of interest. Then the SRB would collect, process, and communicate the scientific data to earth. The list portrays micros as individual, independent spacecraft, and has not included using two or more micros simultaneously to obtain science data that cannot be obtained by one spacecraft.

**NEW VISION SOLAR SYSTEM MISSION STUDY
JUPITER MISSION SCIENCE**

ELEVEN MICROSPACECRAFT AND ONE SPACE REACTOR BIMODAL SPACECRAFT

IO SKIMMER

IO LANDER

IO POLAR ORBITER/FLUX TUBE MONITOR

IO TORUS SAMPLER

EUROPA LASER ALTIMETER

GANYMEDE LANDER/PENETRATOR

CALLISTO IMPACTOR WITH FLASH SPECTROMETER

TWO JUPITER ATMOSPHERIC GLIDERS

TWO LOW JUPITER ORBITERS

ONE SPACE REACTOR BIMODAL JUPITER ORBITER SPACECRAFT

NEW VISION SOLAR SYSTEM MISSION STUDY

JUPITER MISSION SCIENCE

Even though the SRB spacecraft and several microspacecrafts can be delivered to all the outer planets, Jupiter was chosen as probably the first outer planet to be explored in detail. Jupiter was chosen so that a more in-depth mission could be described with the limited funds available. Based on individual mission ideas to Jupiter being proposed at JPL, a single mission to Jupiter with eleven microspacecraft and one SRB spacecraft is discussed. The eleven microspacecraft includes four at Io, four at Jupiter, and one each at Europa, Ganymede, and Callisto. The science and purpose of each spacecraft is described in the following charts.

IO SKIMMER STRAWMAN SCIENCE PAYLOAD

ION AND NEUTRAL MASS SPECTROMETERS:

- Simultaneous Ions/Neutrals

X-RAY FLOURESCENCE SPECTROMETER:

- Mercuric Iodide Solid State Detector

IMAGING (VIS, UV, IR):

- Planetary Integrated Camera Spectrometer (PICS)

PLASMA ENERGETICS PACKAGE:

- MAG (0.5 kg, 1 W), EPD, PWS (1.4 + 2.0 kg antenna, 1.6 W)

MINERALOGY:

- IR Spectroscopy

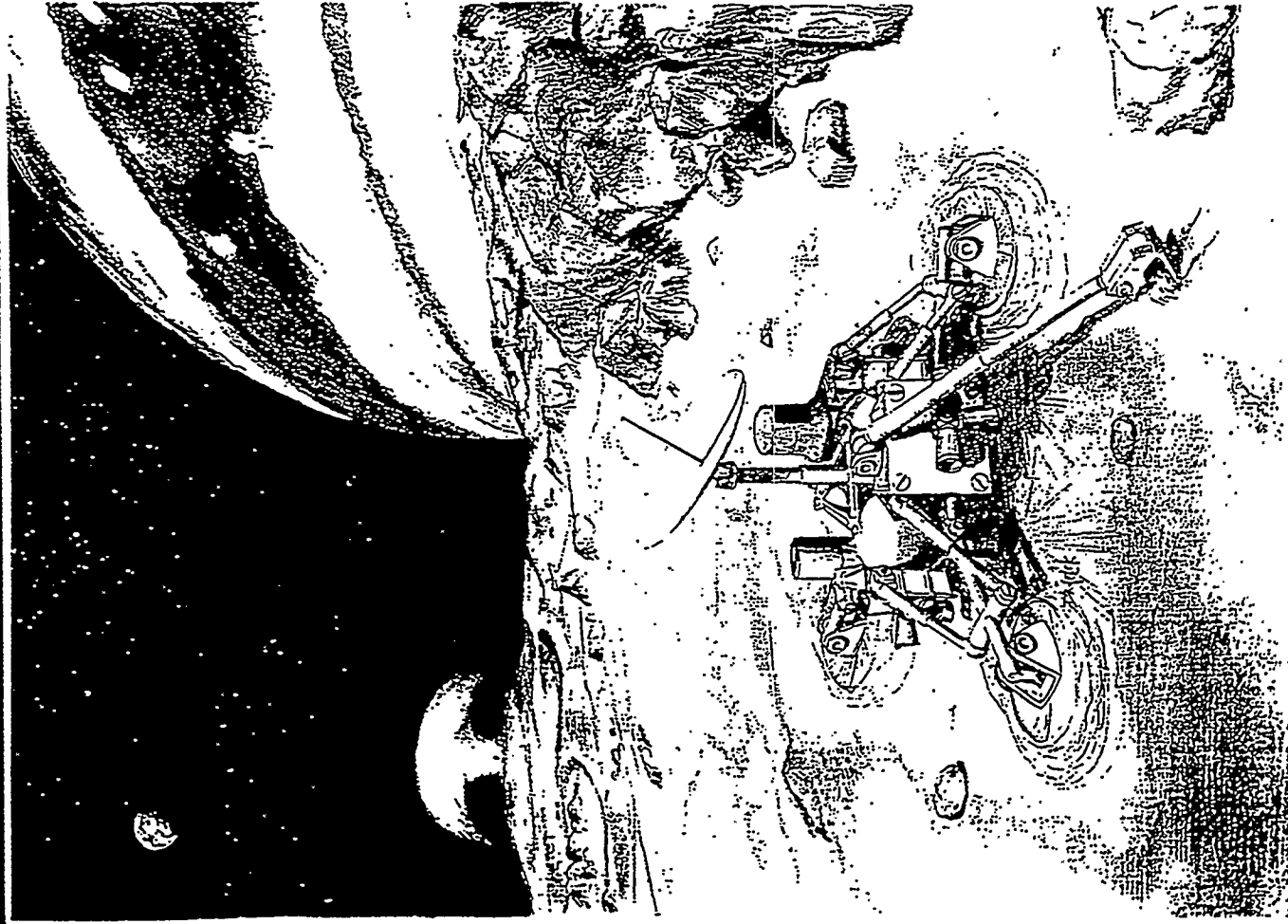
STRAWMAN PAYLOAD CHARACTERISTICS: (includes 20% contingency):

<u>Instrument</u>	<u>Mass (kg)</u>	<u>Power (Watts)</u>
Ion Mass Spectrometer	1.8	1.8
Neutral Mass Spectrometer	3.6	3.6
X-Ray Fluorescence Spectrometer	1.2	1.2
Planetary Integrated Camera Spectrometer	4.0	5.0
Plasma Energetics Package	TBD	TBD
Mineralogy	TBD	TBD
	-----	-----
Totals	> 10.6	> 11.6

Io Skimmer

The Io Skimmer microspacecraft would be placed in an elliptical orbit of Io with the Perigee of the orbit just skimming over the surface of Io. The microspacecraft would determine the surface and atmospheric materials of Io. Visual images of Io would be taken at very close range to obtain high resolution of Io's surface characteristics. An IR spectrometer would be used to determine surface mineralogy of Io. The scientific data would be stored on the microspacecraft and then communicated to the SRB spacecraft when it was near-by. The SRB spacecraft would process and transmit the data from Io to Earth. A strawman science payload is given on the facing page.

Microspacecraft Lander on Io



After deploying from the bimodal S/C, the 10 kg microlander soft lands on Jupiter's volcanically active Moon Io.

Power for the microlander is provided by 10 W of RHU's, converted into 1 W electricity by a thermoelectric device, and stored in a battery for intermittent lander activity.

Data is stored in the lander and periodically relayed to the bimodal S/C for transmission to Earth.

MICROSPACECRAFT LANDER ON IO

The microspacecraft soft lands on Io. It uses a robotic arm to collect Io material, which is examined by the instruments aboard, and then periodically the scientific results are sent to the SRB spacecraft for transfer to Earth. The microlander might be powered by a 10 watt thermal radioisotope heater unit (RHU) which is used as a heat source to produce 1 watt of electrical power to charge a battery. Battery power would be used intermittently to collect and examine the material on the surface of Io.

IO POLAR ORBITER/FLUX TUBE MONITOR

FLUX TUBE FOLLOWS MAGNETIC LINES OF FORCE TO JUPITER'S POLE

IO POLAR ORBITER WOULD INTERSECT FLUX TUBE ONCE PER ORBIT

INSTRUMENTATION

**Magnetometer
Field and Particles
Mass Spectrometer**

IO POLAR ORBITER/FLUX TUBE MONITOR

A microspacecraft would release from the SRB spacecraft at Io. The micro then would use its own propulsion system and propellant to get into a polar orbit around Io.

Io has a flux of material that is connected from the pole of Io to the pole of Jupiter. So an Io polar orbiting microspacecraft will intersect this flux tube every orbit. By using a magnetometer, fields and particle instrument, and a mass spectrometer, the forces, magnetic fields, and type of particles in the flux tube would be determined.

IO TORUS SAMPLER

IO TORUS CIRCLES JUPITER AT THE RADIUS OF AN IN THE PLANE OF IO

IO CONTINUALLY REPLENISHES THE TORUS

LONG-TERM MEASUREMENT OF TORUS COMPOSITION WOULD PROVIDE INFORMATION ABOUT IO GLOBAL DYNAMICS

INSTRUMENTS

**Fields and Particles
Mass Spectrometer**

IO TORUS SAMPLER

Io has volcanos which continually put the volcanic material into a Torus that surrounds Jupiter at the Io orbit. Thus a microspacecraft would be put into the same orbit about Jupiter as Io. The microspacecraft would use a field and particles instrument and a mass spectrometer to make long-term measurements and determine the material in the Torus. This information would help explain Io's global dynamic behavior.

EUROPA LASER ALTIMETER

**EITHER A FLYBY MISSION OR ORBITER WOULD PROVIDE ACCURATE KNOWLEDGE OF
EUROPA GEOMETRY**

**SCIENTISTS THINK THAT THERE IS ABOUT A 50% PROBABILITY THAT EUROPA HAS AN
OCEAN UNDER THE SURFACE ICE**

**ACCURATE GEOMETRY MEASUREMENTS WOULD RESOLVE BETWEEN TWO MODELS OF
EUROPA AND PREDICT THE INTERNAL COMPOSITION**

**THE POSSIBILITY OF THE EXISTENCE OF A LARGE OCEAN ON EUROPA HAS FUELED
SPECULATION OF POSSIBLE LIFE INSIDE EUROPA**

EUROPA IS LIKELY TO BE A HIGH PRIORITY BODY AFTER GALILEO

EUROPA LASER ALTIMETER

A microspacecraft would be deployed at Europa and navigate itself into orbit about Europa. The microspacecraft would accurately determine the geometry of Europa. With this information, scientists could predict whether or not the internal composition of Europa is liquid water. The possibility a large body of liquid water on Europa creates speculation that there is life in the oceans of Europa.

GANYMEDE LANDER / PENETRATOR / ROVER

LOW VELOCITY PENETRATORS (LESS THAN 300 M/S) ARE WITHIN CURRENT TECHNOLOGY

SURVIVABLE INSTRUMENTATION USES EXISTING SURFACE MOUNT ELECTRONICS AND PLOTTING MEDIUM

PENETRATION DEPTH WOULD BE UP TO A FEW METERS

PENETRATORS WOULD BE BATTERY (OR MICRO RTG) POWERED

PENETRATORS WOULD DETERMINE SUBSURFACE COMPOSITION

ROVERS WOULD DETERMINE SURFACE COMPOSITION

LANDER S/C WOULD COLLECT AND STORE DATA FROM PENETRATORS AND ROVERS

LANDER S/C WOULD TRANSMIT SCIENTIFIC DATA TO THE ORBITAL BIMODAL S/C

GANYMEDE LANDER / PENETRATOR / ROVER

Ganymede is the second of four moons of Jupiter discovered by Galileo in 1610. In size and makeup, Ganymede is a virtual twin of Callisto, but it does not look like Callisto at all. Ganymede's surface has large dark plates separated by lighter regions. Impacts have splashed bright snow out onto the surface. The dark regions are believed to be part of the original crust of Ganymede, suggesting that early in its history Ganymede may have looked like Callisto. However, within the first 500 million years of Ganymede's formation, geological activity replaced about half of the old, dark surface with lighter material.

It is unlikely that this activity was caused solely by radioactive heating, suggesting that tidal heating may have played an important role. Tidal effects are familiar on the Earth because our Moon raises tides in the ocean twice a day as the Earth rotates beneath it. The Earth in turn raises a tide in the Moon's crust, and as the Moon always has one side facing the Earth, the bulge in its surface is stationary.

Ganymede also has one side permanently facing Jupiter. But at some earlier time it may well have had a non-circular orbit so that its surface too, was being pumped up and down. Flexing of its surface may have added enough extra heat to have caused a period of geological activity during which half of the original crust was replaced with a younger terrain with parallel ridges. But this is not the only example of geological activity in the outer solar system. The subsurface composition to depths of a few meters would be determined with penetrators. The surface composition would be determined by the lander spacecraft and rover.

CALLISTO IMPACTOR WITH FLASH SPECTROMETER

IMPACTORS USED TO IMPACT DIFFERENT AREAS OF CALLISTO

IMPACT FLASH SPECTROMETER EXAMINES FLASH FROM EACH MULTIPLE IMPACTORS

IMPACTORS USED TO CREATE EJECTA

EJECTA CAPTURED DURING A HIGH VELOCITY FLYBY

EJECTA CAN BE TURNED INTO A PLASMA FOR ONBOARD ANALYSIS

CALLISTO IMPACTOR WITH FLASH SPECTROMETER

Jupiter's moon Callisto looks much as might have been expected for bodies in the outer solar system where the temperatures are very much colder. Although Callisto is about the size of the planet Mercury, it is half ice and half rock. The rocky material forms a core surrounded by a deep layer of ice that is heavily scarred by impacts of meteorites, asteroids and comets, but little modified by tectonic processes. This is what was expected in the cold outer solar system, but other moons of Jupiter showed nature to be much more complex than expected.

The microspacecraft would be in a very low orbit about Callisto. Impactors would be released to impart Callisto in front of the orbiter, and the flash spectrometer would determine the composition of the ejected material from the impactor. Impact flash spectrometer currently is being tested at JPL with good progress in determining surface elemental composition from emission line spectrometry from high velocity impacts (greater than 1 km/sec). Impactors would be used to create ejecta from Callisto's surface which then is captured by an orbiter spacecraft that comes very close to the surface of Callisto immediately following the impact.

JUPITER CLOSE POLAR ORBITERS

TWO DIFFERENT MISSION TYPES

RADIO SCIENCE / MAGNETOMETER MISSION

Radio Science Celestial Mechanics

High-field Magnetometer

Radio Science Atmospheric Occultation

JUPITER AURORAL OBSERVER MISSION

Multi-spectral Imaging and Spectrometry

UV

Near IR

Thermal IR

Fields & Particles

Magnetometer

Charged Particles

Radio / Plasma Wave

JUPITER CLOSE POLAR ORBITERS

SCIENCE OBJECTIVES: RADIO SCIENCE / MAGNETOMETER

Radio Science Celestial Mechanics would be used to map the interior gravity field and upper tropospheric structure of Jupiter to high degree and order. Latitudinal and longitudinal coverage would be provided by the polar orbiter.

High-field magnetometer would map the magnetic field to a high degree and order. The solar wind and magnetosphere interaction would be mapped to a limited extent. Both latitudinal and longitudinal coverage would be provided. An extended mission would address variations of magnetic fields with time.

Radio science atmospheric occultation would provide ionospheric e^- density structure, stratospheric P-T profiles, buoyancy wave detection, tropospheric P-T profiles, NH_3 abundance profiles, and latitudinal coverage.

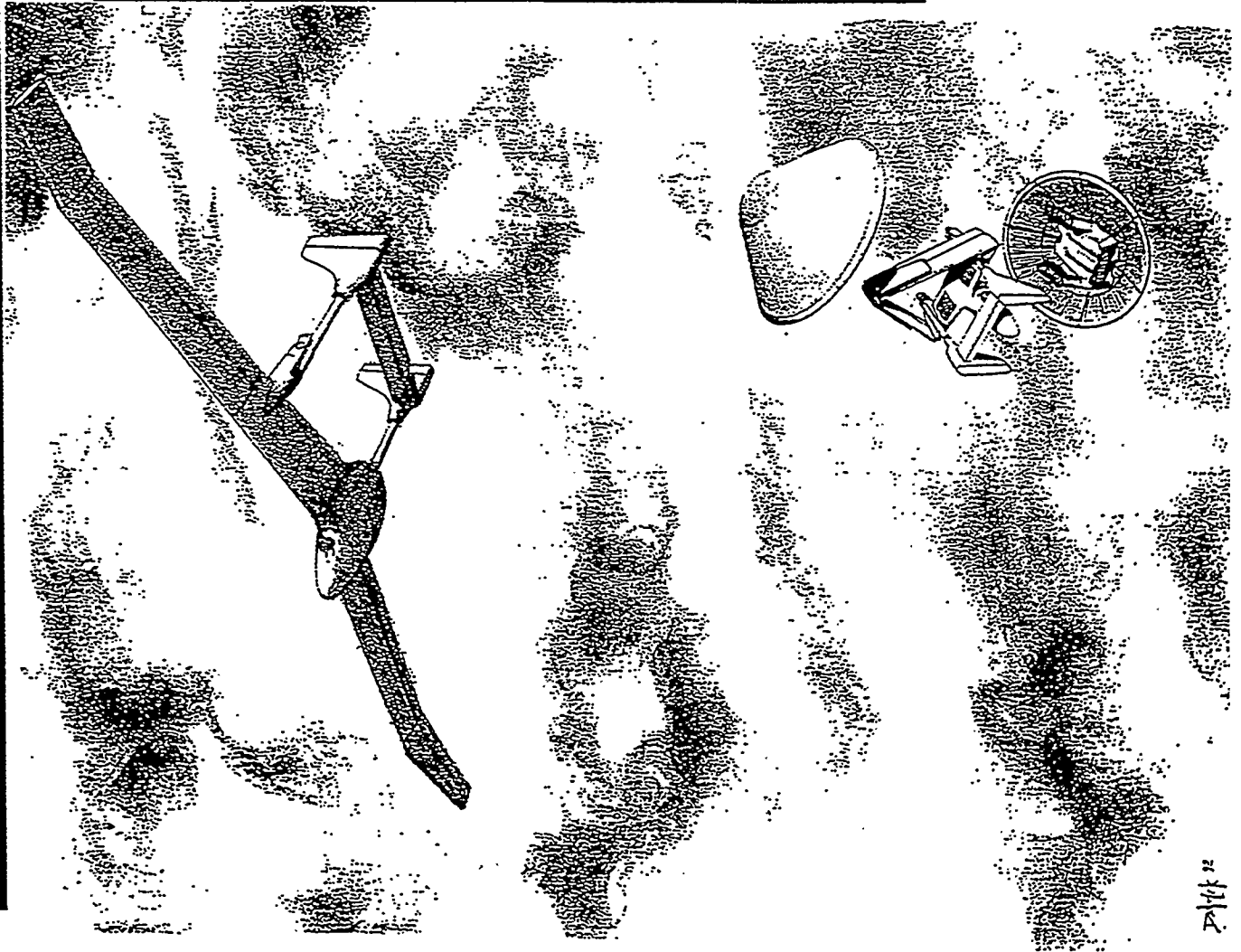
SCIENCE OBJECTIVES: AURORAL OBSERVER

Imaging would show the Aurora geometry and its relation to magnetic field structure and to neutral atmosphere and ionosphere. High-latitude tropospheric structure, circulation, etc would also be imaged.

Spectroscopy (imaging system does imaging spectroscopy) would be used to determine composition of neutral atmosphere and ionosphere, the processes in neutral atmosphere and ionosphere, and for monitoring Io.

Fields and particles instruments would determine the magnetic field structure to high degree and order, the ionic composition of the magnetosphere, the Auroral particle acceleration mechanism, and the solar / magnetosphere interactions.

Microspacecraft Glider Probe at Jupiter



After leaving the bimodal S/C, the 10 kg microS/C glider aero-enters at Jupiter and deploys from its capsule.

With an L/D of 20, the glider can perform a 2000 km traverse during a 100 km dive. When it detects an updraft, it circles like a seagull, allowing it to ascend to the upper atmosphere to transmit its data to a microS/C relay satellite. The relay satellite then stores the data and periodically relays it to the bimodal S/C for transmission to Earth.

MICROSPACECRAFT GLIDER PROBE AT JUPITER

The most unique of the Jupiter system microspacecraft are the atmospheric gliding probes, as shown on the facing page. The purpose of these probes is to assess chemical composition deep within Jupiter's atmosphere.

Gliders would operate between the 5 bar (~ 0 C) and 100 bar (~ 250 C) levels of Jupiter's atmosphere, a depth distance on the order of 100 km. At the 5 bar level the density of the atmosphere is about 0.45 kg/m^3 , assuming an airspeed of 100 m/s and a lift coefficient of 0.5, and taking into account Jupiter's gravity of 2.53 times Earth, the required wing area for a 10 kg aircraft to remain aloft is 0.22 m^2 , which clearly is a manageable wing size. Sailplanes are built with lift to drag (L/D) ratios greater than 20, so using this L/D the total horizontal glide path that the aircraft has available to it during a 100 km descent is about 2000 km. At an airspeed of 100 m/s, this represents a descent speed of 5 m/s and a flight time during descent of 5.6 hours, which is about the same as the 5.6 hour period of a low Jupiter orbiter. More importantly, 2000 km is much longer than the characteristic length scale for local Jupiter atmospheric phenomenon. This means that within 2000 km of horizontal travel the odds are very high that both updrafts and downdrafts will be encountered. Since the speed of such updrafts is likely to be much greater than the glider's 5 m/s descent speed (typical wind speeds on Jupiter are on the order of 100 m/s), a glider which finds such an updraft will be able to ascend by circling in it much as seagulls do on Earth. Using this technique, gliders will be able to ascend and descend in a controller manner, repeatedly sampling the chemistry and other environmental parameters of Jupiter's deep atmosphere. Assuming 1000 km of horizontal travel per descent, about 113 dives would be needed for a given glider to navigate through 90 degrees of latitude.

The glider would be power limited. Therefore it would adopt a strategy of cycling its instruments on for 0.1 seconds once every 100 seconds, taking a chemistry, pressure, temperature, E and B field and acceleration readings. Such a data set may be estimated at 1000 bits (8 bits each for P, T, E, B, acceleration, and each of 120 chemical substances). If such a data set is taken every 100 seconds, the aircraft will be able to store 10 million such sets, or 32 years worth of data in its 1 Gb memory. This is much longer than the probable time for an encounter between the glider and the SRB S/C once it is in its 15 day 5 RJ X 26 RJ orbit. This would allow the gliders to transmit all their data to an orbiter relay without the use of a directional antenna.

MICROFLEET MISSION TO JUPITER

Mass of Microspacecraft Probes for Jupiter System

Jupiter Orbiter	13kg	Io 60 kg	Europa 37 kg	Ganymede 22 kg	Callisto 15 kg
Lander	20kg	156 kg	83 kg	61 kg	39 kg
Science Payload					Mass
SRB S/C Instruments					20 kg
(2) Jupiter Atmospheric Gliders					40
(2) Jupiter Close Polar Orbiters					26
(3) Io Orbiter					180
Io Lander					156
Europa Orbiter					37
Ganymede Lander / Penetrator / Rover					61
Callisto Orbiter / Impactor					15
Total Science Payload					----- 535 kg

MICROFLEET MISSION TO JUPITER

A strawman fleet of microspacecraft for a Space Reactor Bimodal (SRB) spacecraft mission to Jupiter was selected from advanced single scientific mission concepts proposed for Jupiter. The mass of each microspacecraft and its propellant needed to get into orbit or land was estimated. Each microspacecraft with its mass and its scientific target are listed on the facing page.

The total payload for an eleven microspacecraft strawman fleet is 535 kg. Using the 10 kWe SRB and an Atlas 2AS, this mission could be accomplished with an Earth Gravity Assist in 4.7 years and an electric propulsion burn time of less than two years.

NEW VISION SOLAR SYSTEM MISSION STUDY JUPITER MISSION SCIENCE

SPACE REACTOR BIMODAL SPACECRAFT MISSION IN JUPITER ORBIT

DETERMINE LONG TERM JUPITER ATMOSPHERIC CHANGES

RECEIVE, PROCESS AND TRANSMIT DATA FROM ELEVEN MICRO S/C TO EARTH

ONLY USES FULL POWER DURING HIGH RATE DATA TRANSMISSION TO EARTH

AN AVAILABLE POWER SYSTEM FOR DECADES IN ORBIT AT JUPITER

**PROVIDE COMMUNICATIONS AND DATA TRANSMISSION FOR FUTURE
LOW COST MICROSPACECRAFT LAUNCHED TO JUPITER ORBIT.**

NEW VISION SOLAR SYSTEM MISSION STUDY

JUPITER MISSION SCIENCE

For the strawman Jupiter mission, the Space Reactor Bimodal (SRB) Spacecraft would be in an elliptical orbit about Jupiter with instruments and on-board power of 10 kWe to:

1. Determine the long-term changes of the Jupiter atmosphere.
2. Receive scientific data from each of the eleven microspacecraft as it flew by each of their locations.
3. Process the scientific data and transmit the data to Earth.
4. Since the SRB would use only the full 10 kWe of power to transmit data to Earth and to periodically examine the Jupiter atmosphere, its lifetime would be decades. Therefore, the SRB would be a communication resource for future low cost microspacecraft launched into Jupiter orbit for specific scientific data.

MICROFLEET MISSION TO SATURN

SCIENCE PAYLOAD OF SRB MICROFLEET CARRIER SATURN EXPLORATION MISSION

SRB S/C INSTRUMENTS	20 kg
SATURN HIGH INCLINATION ORBITER	13 kg
SATURN HIGHLY ELLIPTICAL EQUATORIAL ORBITER	13 kg
(2) TITAN MICRO-COM ORBITERS	50 kg
(8) TITAN AIRCRAFT	160 kg

TOTAL SCIENCE PAYLOAD	256 kg

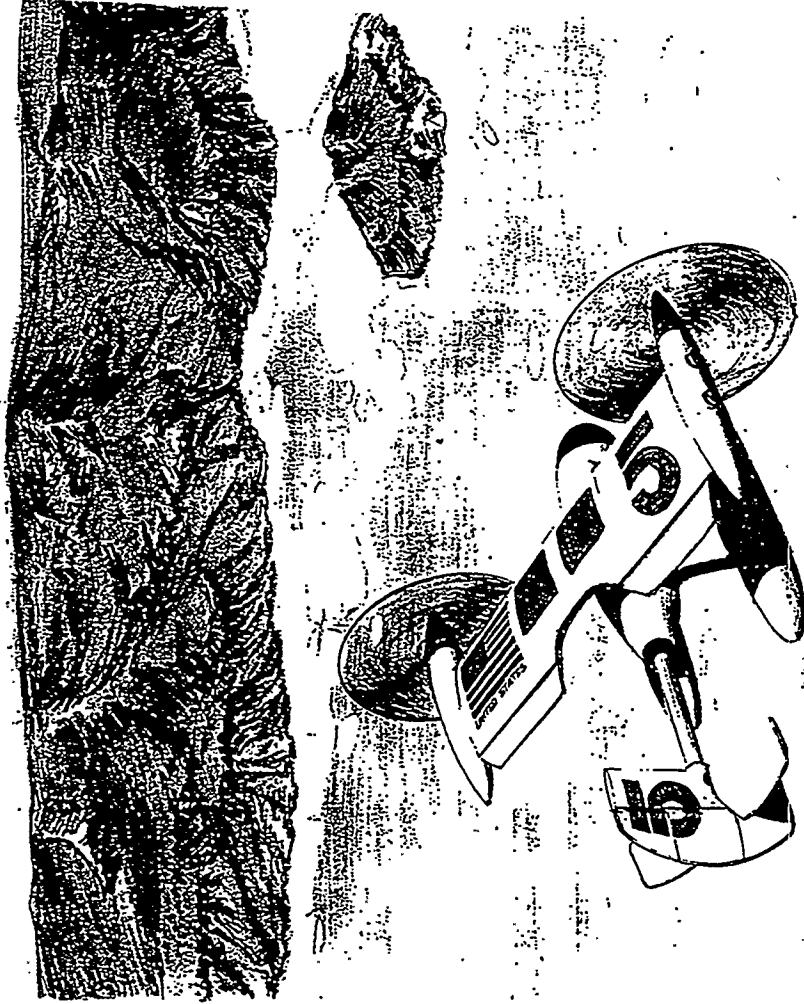
THIS SCIENCE PAYLOAD MASS STILL AFFORDS LARGE LAUNCH MARGIN FOR THE SRB S/C
IF AN ATLAS 2AS IS USED.

MICROFLEET MISSION TO SATURN

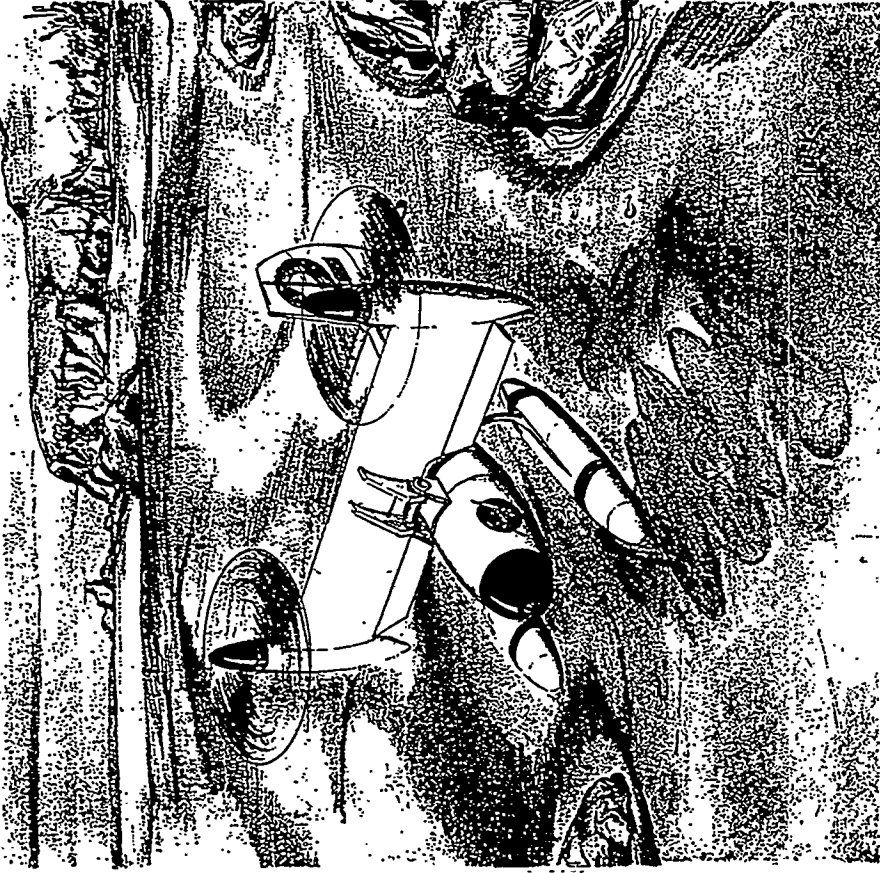
Each planet has its own particular targets of interest. For example at Saturn, the cloud covered moon Titan is almost of as much interest as Saturn itself. This suggests the following mission strategy for Saturn. After a 5 to 9 year flight, the high thrust bimodal propulsion system uses NH_3 to place the spacecraft in a 2RS X 100 day elliptical orbit. After capture, the SRB S/C releases a micro-orbiter, leaving it in the initial moderately inclined, (Saturn's equator is inclined 27 degrees to the ecliptic) highly elliptical Saturn orbit from which it can survey the whole Saturn system.

The SRB S/C then performs an apoapsis burn with its electric propulsion system to raise itself into an equatorial orbit with its periapsis at 3 RS ($\Delta V = 216$ m/s), which is safely outside the ring system of Saturn (except for the ethereal E ring). It then releases a microspacecraft orbiter in this Saturn orbit. This second microspacecraft will be in an orbit that allows it to periodically fly by Mimas, Enceladus, Tethys, Telesto, Calypso, Dione, Helene, Rhea, Titan, and Hyperion (which are all in equatorial orbits), and get fairly close to Iapetus (14.7 degree inclination) as well. The SRB S/C then performs another NEP ΔV of 659 to raise its periapsis from 3 RS to 20 RS which is the distance of Titan from Saturn. A series of gravity assists then are performed to lower the apoapsis from 132 RS to close to 43 RS, which puts the SRB spacecraft into a 32 day orbit around Saturn which is synchronous with Titan (which has a 16 day orbit). At Titan encounter two microspacecraft orbiters are released into elliptical Titan orbits, which later are turned into low Titan polar orbits with perpendicular orbit planes. Then 8 entry probes each carrying a small robotic aircraft are released into Titan and function as described on the next two charts.

Microspacecraft Tilt-Rotor Aircraft on Titan



PAF/K



After leaving the bimodal S/C, the 10 kg micro-Aircraft aero-enters at Titan and deploys from its capsule. Given Titan's 1/7 g and 4 times Earth atmospheric density environment, the A/C can fly at 10 mph with a wing area of 0.5 m². Assuming an L/D of 10, the aircraft requires a thrust of 1.4 N and a power of 7 W to remain airborne. With a 20 Wt RHU, Titan's 90 K environment allows dynamic conversion at 70% efficiency, or 14 We, so the A/C can remain aloft indefinitely. The tilt rotor motor arrangement combined with pontoons enables repeated vertical landing and takeoff anywhere on Titan.

Max Sci RZ 11/30/95-30

MICROSPACECRAFT TILT-ROTOR AIRCRAFT ON TITAN

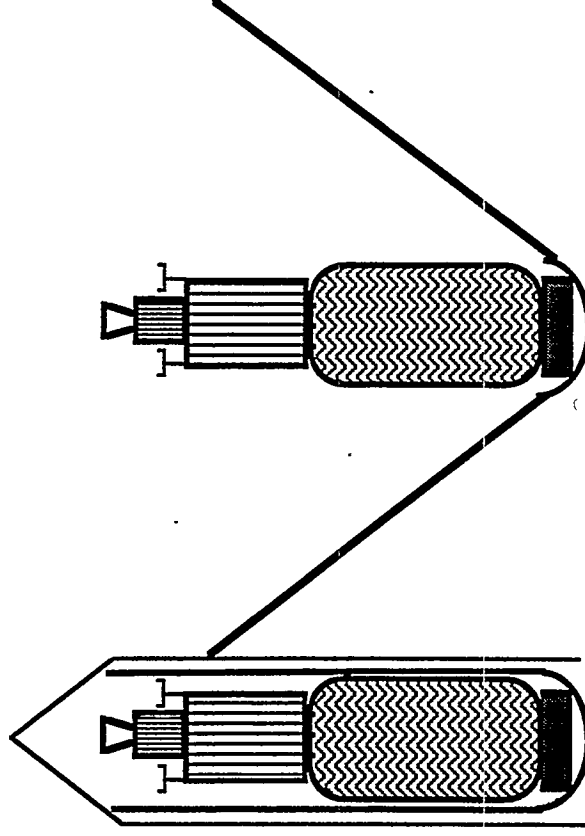
The aircraft for Titan missions each have a mass of 10 kg and a wing area of 0.5 m^2 . Titan has a surface atmospheric density of about 5 kg/m^3 and a gravity $1/7$ that of Earth. Conditions that are very favorable for heavier-than-air aviation. Under these conditions the aircraft will only have to move at an airspeed of 4.8 m/s to stay aloft, and assuming an L/D of 10, will require a thrust of 1.4 N and thus an effective propeller power of 6.8 W . This 6.8 W can be provided by a 20 Wt radioisotope heat source using the 90 K nitrogen atmosphere and dynamic conversion of heat to electricity at efficiencies greater than 70%. The power system supplies 14 We, which allows it to fly continuously as well as operate strobe flashlights to support imaging in the visible light range, as well as continual radar sounding of the ground to reveal subsurface features and assure safe flight. If the plane is designed as a tilt-rotor with pontoons, it can be made capable of soft landing on the ground/ocean, allowing direct sampling of the surface. Ascent from such a condition requires burst power of 70 W, which can be made available by expenditure of a small charge stored in the aircraft's battery. The aircraft will come into contact with the low orbiting microspacecraft about once every other day, at which time they can uplink all their data. The microspacecraft are equipped with 10 Gb memories, allowing each one to store up to 50,000 512 X 512 black and white images, or comparable data. Every 32 days the SRB S/C will fly by, and the orbiters will dump all the collected data for relay to Earth.

Five Year Neptune Orbiter Mission

- Delivery of S/C to Neptune orbit within 5 years can only be done using aerocapture.
- Large entry velocity requires very ambitious aerocapture technology.
- Assume such technology becomes available.
Let aerobrake mass = 20% of decelerated mass.

- Use JGA Trajectory.
Jan. 25, 2006 launch. Launch C3=81 km²/s².
Use NEP to accelerate trajectory to C3=125 km²/s²
Jupiter encounter June 5, 2007. Fly-by distance = 5 RJ.
Neptune arrival Jan. 25, 2011. Entry velocity = 44 km/s.

<u>Mass Breakdown</u>	
10 kWe Bimodal reactor and S/C	2200 kg
Microspacecraft Fleet	100 kg
NH3 Propellant and Tanks (0.2 km/s ΔV)	200 kg
Xenon Propellant and Tank (5 km/s ΔV)	795 kg
Aeroshell	1200 kg
Hydrogen Tank	1515 kg
Hydrogen Propellant	10100 kg
Total Launch Mass	16,110 kg



The mission could be launched by a Proton D-1 (18,000 kg to LEO) with about 12% launch margin.

FIVE YEAR NEPTUNE ORBITER MISSION

A generalized chart showing a selection of good mission options for delivering the SRB spacecraft to Neptune orbit is given on an earlier page of this report. However, in response to recent interest at NASA in a very fast orbiter missions to Neptune, a study of the potential of the SRB system was done to perform a five year mission to Neptune.

The trajectory chosen for this mission analysis was a Jupiter gravity assist with a January 2006 launch. There are only two ways that this mission can be accomplished. NEP could be used to take out a very large chunk of hyperbolic velocity prior to entry, so that high thrust propulsion systems can deal with the remaining ΔV required for orbital capture. Alternatively, aerocapture can be employed. If short flight times to Neptune are to be achieved, both of these strategies become very difficult.

Turning first to the case of propulsive capture, the NEP burn time cannot be greater than half the flight time. So accomplishing an all propulsive Neptune orbiter mission with the SRB spacecraft with flight times of less than 8 years is impossible.

The alternative method of attempting a fast Neptune orbiter mission would be by employing aerocapture. Assuming that aerocapture technologies are developed, SRB spacecraft could use them to accomplish a five year Neptune orbiter mission. The spacecraft would be launched from Earth on a Proton on January 25, 2006. After reaching LEO, the SRB would use its direct thrust capability with hydrogen propellant to engage in a series of orbit-raising perigee kicks and then escape Earth with a C3 relative to Earth of $125 \text{ km}^2/\text{s}^2$. NEP then would be used to accelerate this trajectory to a C3 relative to Earth of $125 \text{ km}^2/\text{s}^2$. The spacecraft would reach Jupiter on June 5, 2007, flying by at a distance of 5 RJ. It would reach Neptune on January 25, 2011, and enter the atmosphere with an entry velocity of 44 km/s. The spacecraft would emerge from Neptune's atmosphere a few minutes later and then perform a small periapsis raising burn to achieve stable orbit. The science mission at Neptune would begin 5 years to the day after Earth departure, with a 10 kWe SRB spacecraft in Neptune orbit and up to ten microspacecraft to be deployed in the Neptune system. A mass breakdown for this mission is presented on the facing page.

NEW VISION SOLAR SYSTEM MISSION STUDY CONCLUSIONS

- 10 KWE SPACE REACTOR BIMODAL (SRB) S/C CAN TRANSPORT TEN OR MORE MICROSPACECRAFT TO THE OUTER PLANETS WITH ELECTRIC PROPULSION BURN TIMES OF 1 TO 3 YEARS.
- THE SCIENCE RETURN FROM THE SRB S/C IS GREATLY ENHANCED BY THE 10 KWE POWER SYSTEM WHICH PROVIDES VERY HIGH DATA TRANSMISSIONS RATES FROM JUPITER, SATURN OR NEPTUNE.
- THE 10 KWE SRB CAN COLLECT, PROCESS AND RAPIDLY TRANSMIT SCIENTIFIC DATA TO EARTH FROM SEVERAL MICROSPACECRAFT AT MAIN BELT ASTEROIDS, JUPITER, SATURN AND NEPTUNE.
- SRB S/C CAN PROVIDE COMMUNICATIONS AND DATA TRANSMISSION FOR DECADES TO SUPPORT FUTURE LOW COST MICROSPACECRAFT LAUNCHED TO THE OUTER PLANETS.
- SRB ENABLES VERY HIGH SCIENCE RETURN OUTER PLANET SOLAR SYSTEM MISSIONS AT LOWER COST THAN IS POSSIBLE WITH ANY OTHER TECHNOLOGY.
- WE RECOMMEND THAT THE SPACE REACTOR BIMODAL (SRB) SYSTEM TECHNOLOGY BE DEVELOPED AND MISSIONS USING SRB BE FURTHER DEFINED.

NEW VISION SOLAR SYSTEM MISSION STUDY

CONCLUSIONS

In summary, the SRB spacecraft launched by an Atlas 2AS is capable of delivering ten or more microspacecraft to orbit around Jupiter, Saturn, or Neptune with electric propulsion burn times of 1 to 3 years. Furthermore, the science return from the delivered instruments is greatly enhanced by the fact that the SRB spacecraft can provide about 2 orders of magnitude higher data transmission rate than a 500 We powered outer solar system spacecraft. By exploiting these capabilities, outer solar system missions could provide scientific data needed to understand the origin and evolution of the solar system. While cost analysis was not addressed in the study, the large reduction in launch cost associated with moving the spacecraft from a Titan IV to an Atlas 2AS, combined with the elimination of ten launch vehicles for each microspacecraft is likely to more than offset the added recurring cost of the bimodal reactor power and propulsion system. The SRB spacecraft would also be a communication resource at each outer planet and provide high data transmission rates for future micro spacecraft launched to the outer planets for decades. Thus the development and use of the bimodal reactor system will allow outer solar system missions that return much more science at significantly lower cost than is possible with any other technology. Therefore, we recommend that the bimodal technology be developed and its potential mission applications be studied further.

NEW VISION SOLAR SYSTEM MISSION STUDY

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