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NASA Contractor Report 202345

An Assessment of Smallsat Technology to Future Exploration Missions

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Final Report
An Assessment of Smallsat Technology to Future Exploration Missions

Task Order No. 3 of Defining Satellite Systems Concepts

NASA Contract No. NAS3-27559

Space Systems/Loral

Building 45

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SPACE SYSTEMS LORAL

Final Report Outline

- Smallsat Survey
- Mars Exploration Missions Summary
- Communications Technologies
- Advanced Subsystems
  - Propulsion
  - Battery
  - Advanced Material
- Ground Operations
- Cost
- Conclusions

Final Report Outline

The final report is prepared in the form of a presentation package with facing page texts. The intent is to facilitate NASA Lewis to pull out pages that are desired for briefings to groups of its choice.

Survey of Small Satellites

- Survey criteria
  - Innovative and unique
  - Launched/to be launched between 1988 - 1998
- Presentation
  - Summary of Innovative Technologies
  - View of particular satellites

Survey of Small Satellites

The review of the spacecraft emphasized those which were innovative and unique and could thus benefit NASA's future exploration missions. In particular spacecraft were considered that significantly reduced mass.

The spacecraft considered were launched no earlier than 1988 or are to be launched before 1998. This window was selected on the grounds that those spacecraft launched earlier than 1988 have antiquated technologies, and those to be launched after 1998 have not been described in sufficient detail in the open literature.

The results of the survey are summarized in the first part of this presentation. A more detailed view of the spacecraft under review is given in second part.

All of the surveys are obtained from the Web except the DS-1 which is from a paper by Lehman, David and Rayman, Marc of JPL, entitled "NASA's First New Millennium Deep Space Technology Validation Flight".

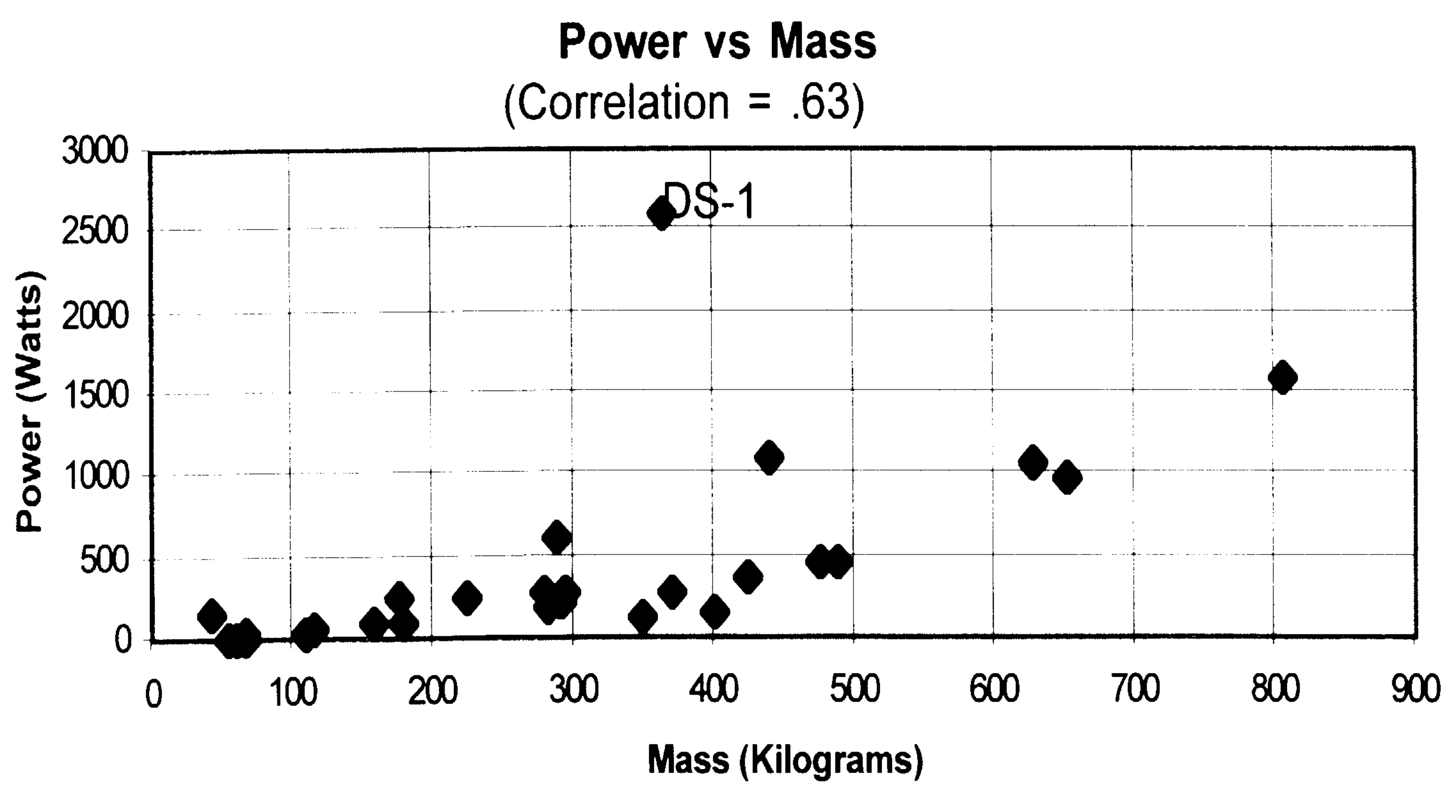
Summary Of Satellite Attributes

- Mass range 50 kg to 800 kg, covering three classes
  - NASA (14)
  - DOD (8)
  - Commercial (6)
- Technological innovations in the areas of
  - power generation
  - command and data handling
  - attitude determination and control
  - structures
  - mechanisms
  - instrumentation

Summary Of Satellite Attributes

For the purposes of the survey a small satellite was defined to be a spacecraft whose mass (including payload) was between 50 kg and 800 kg. Further it was found that it was appropriate to divide the spacecraft into three classes. Class one were those built for NASA or for NASA sponsored projects. These craft were usually special purpose i.e. the bus was designed for a particular payload. The second class were those built for the Defense Department or for defense related purposes (such as those built for DARPA or Los Alamos) and the third class were those space craft built for commercial purposes. This last group was in general (though not uniformly) more massive, more powerful, and of a more general design i.e. it could accommodate more than one kind of payload.

SATELLITE POWER VS MASS



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SATELLITE POWER VS MASS

This show the correlation of power developed by the space craft considered in the survey as a function of the dry mass of the space craft. The correlation is .63.

As would be expected the power increases as the mass increases. Note the exception however, in the point way out from the group. This is the power developed by the Deep Space 1 satellite. It results from a new design of the of the the solar panels.

To achieve the kind of power generation at lower masses exemplified by DS-1 NASA has looked to the new technologies which we will summarize. These innovations have been created by NASA, by DOD contractors, DARPA and commercial satellite companies.

Summary Of Spacecraft Attributes

- Electrical Power
  - S/W control of solar cells
  - Innovative Solar Arrays
    - Thinner
    - Use of Concentrators
  - Solid State Power Controllers
- Command and Data Handling
  - Open architectures
  - 32 bit processors
  - 3d cube memories
  - radiation hardened FPGAs
- Attitude Determination and Control
  - mini star tracker
  - new horizon and solar sensors
  - hemispherical resonator gyro

Summary Of Spacecraft Attributes

By using a software control to turn banks of solar cells on and off mass savings are achieved through the elimination of parts of the thermal control system needed to remove excess heat when more power is generated than is used.

Combining innovations in the use of multi-junction photo voltaic cells, thinner solar arrays, use of concentrators on the solar arrays, and the uses of amorphous silicon cells greater efficiency with less mass is achieved in electrical power production.

Using solid state power controllers in place of switches and fuses in the power system increases reliability and decreases mass.

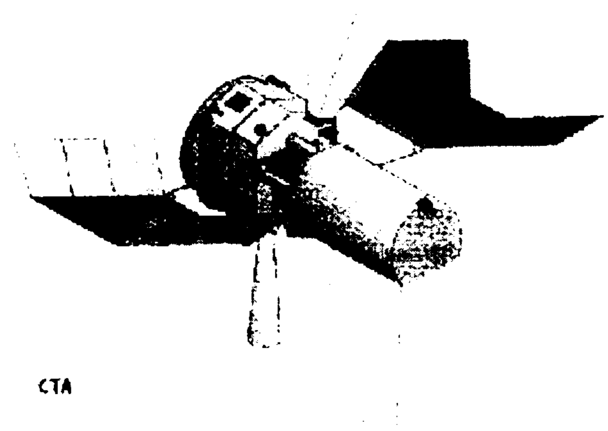
Summary Of Spacecraft Attributes (Cont'd)

- Structures
  - Composite shell structure
  - Highly conducting thermal fibers
- Mechanisms
  - Shape memory metals
- Instruments
  - Digitization
  - Data compression
  - Solid state devices
    - Memory devices

Summary Of Spacecraft Attributes (Cont'd)

A composite shell structure, covered with 1.5 mil Al to form a ground plane, incorporating highly conducting thermal fibers decreases mass and costs while increasing the thermal performance.

NASA Clark Satellite



NASA Clark Satellite

Clark was selected in the NASA SSTI (Small Spacecraft Technology Initiative) program, along with Lewis, to demonstrate advanced spacecraft technologies. Contract start was July 11, 1994.

Launch Planned for July 1996 using LLV

Orbit 475 km, circular, polar, sun synchronous i=97.3 deg

Size 36.5 in. hex by 69.5 in. Mass 278 kg

Design Life 3 years (required life is 1 year)

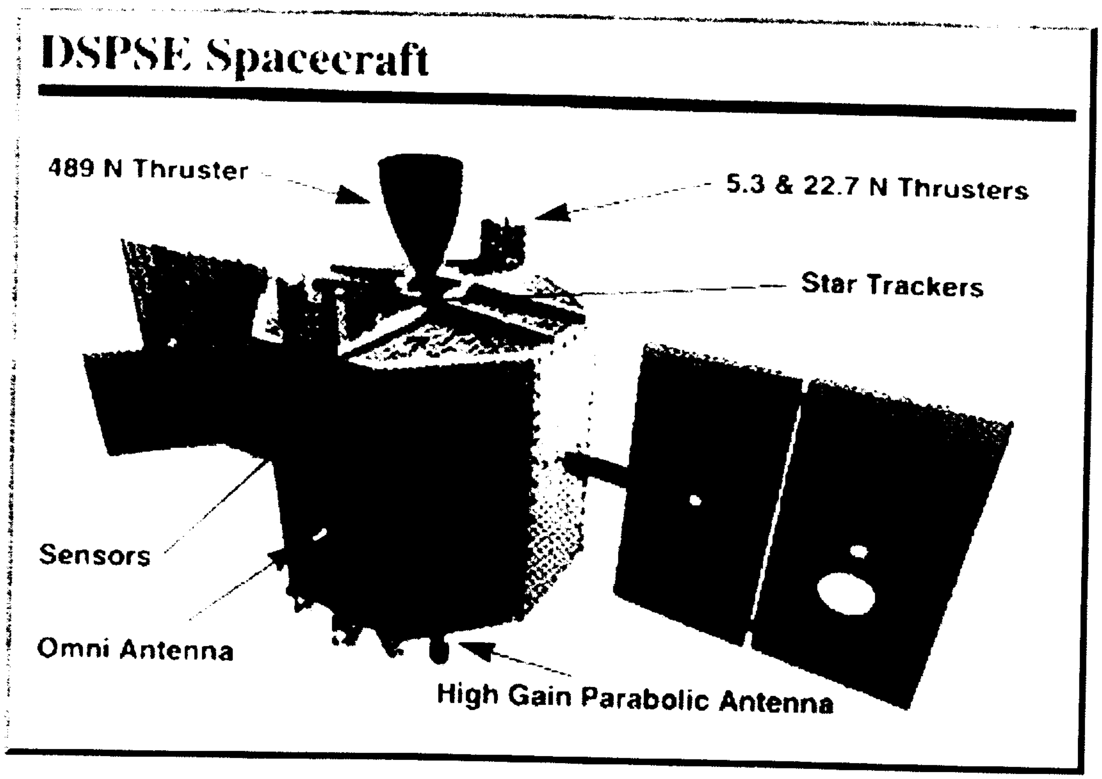
Spacecraft in Series 1

It is a 3-Axis stabilized, zero momentum biased control system with 0.05 deg control and 0.03 deg knowledge. Two solar arrays generate 289W orbit average power. Two 15-Ahr NiH2 CPV batteries. Hydrazine propulsion system with 12.5 kg propellant. 2 Gbit solid state storage. UHF command and telemetry links. 25 Mbps X-band image data downlink. Graphite composite structure.

Payload Description

Worldview Imaging Corp's panchromatic 3-m resolution, 15-m multispectral resolution with cloud editing capability. Micro-Measuring Air Pollution from Satellites (μMAPS). X-Ray Spectrometer (soft and gamma). Atmospheric Tomography (3D pollution mapping).

NASA Clementine Satellite



SPACE SYSTEMS LORAL

NASA Clementine Satellite

Clementine was jointly sponsored by BMDO and NASA as the Deep Space Program Science Experiment (DSPSE). The principal objective was to space qualify lightweight imaging sensors and component technologies for the next generation of DOD spacecraft. After entering lunar orbit, Clementine providing over 1.6 million images of the Moon's surface.

Launch January 1994 on Titan IIG from WTR using a Star 37FM kick motor

Trajectory 1.5 to 7 days in LEO, 27 day transfer orbit to enter Lunar orbit, 2 months in lunar orbit, 4 month transfer to asteroid.

Orbit 5 hour, polar, lunar Size 1.14 m diameter, 1.9 m length

Mass 424 kg Design Life 7 months

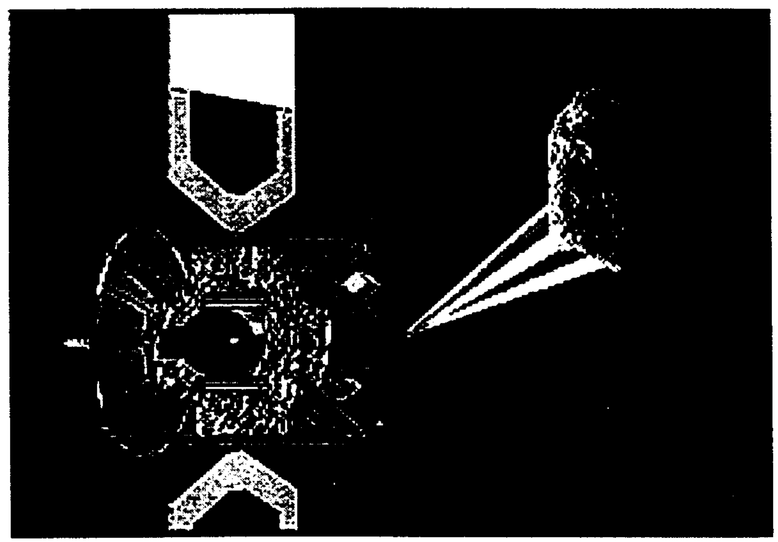
Manufacturers NRL, LLNL(imaging sensors) Spacecraft in Series 1

3-axis stabilized. Dual GaAs solar arrays with 1-axis articulation provide 360 W. Fixed 1.1 m HGA. S-Band downlink to NASA DSN and DOD tracking stations with downlink rate up to 128 kbps. 1.9 Gbit solid state recorder. Bipropellant system with 489 N thruster. Hydrazine system with 10 x 5.3 N and 7 x 22 N thrusters. 32-bit R3000 processor. Lightweight RLG and IFOG (1 deg/hr). NiH2 CPV battery (15 AHr). Lightweight reaction wheels and star tracker cameras. JPEG image compression chip (from Matra Marconi).

Payload Description

Two miniature star tracker cameras. UV/Visible camera. Near-IR camera. Long wave IR camera. High resolution camera. Laser transmitter. Charged particle telescope. Dosimeters (4). Radiation experiment. Orbital meteoroid and debris counting experiment. Total payload is 8 kg and 68 watts.

NASA DS-1 Satellite



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NASA DS-1 Satellite

DS-1 is the first of the new millennium programs spacecraft. It has the double purpose of testing new technology and asteroid exploration. The new technologies to be tested on the mission include a new solar electric propulsion system, the SCARLET II solar concentrator, and a new integrated space physics package. In addition a small deep space transponder/beacon, an autonomous optical navigation system, and a three dimensional memory package are included in the mission.

The spacecraft will fly by asteroid 6053 1993 BW3. This will allow tests of the thrusters and the navigational system as well as the sensor suites. The primary mission of the craft will conclude with an encounter of the comet P/TEmpe 1 in January of the year 2000.

Launch: 1998 on a : LM-LV2 Mass: 365 kg

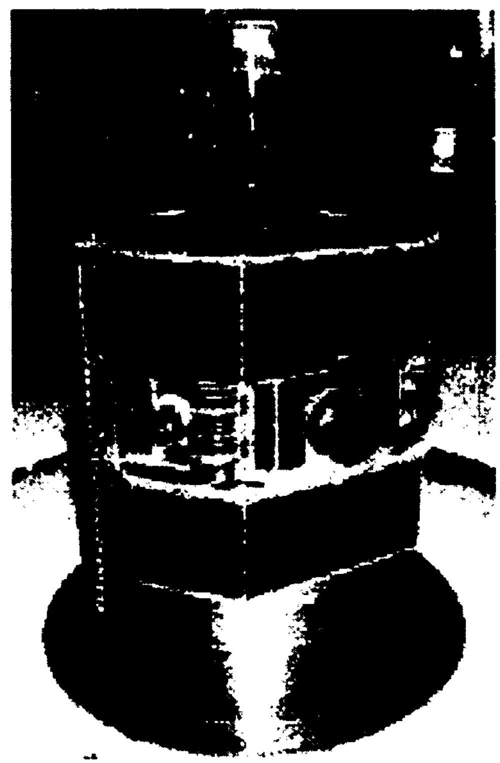
Communication: compact mmic and silicon ASICs small deep space transponder at X and Ka bands combining in one package rcvr, command detector, telemetry modulation, exciters, beacon tone generation, and control functions

Structure: Al space frame based on MSTI spacecraft; S/C Power 2.6 kW

Power Sources : solar concentrator array. Thermal: std insulators, heaters and radiators Configuration: Most components mounted on exterior of bus; attitude control sensors include 5 sun sensor heads distributed to provide 4 pi sr coverage; two inertial reference units, and one wide field of view star tracker

Payload Description: integrated camera/spectrometer, integrated electron/ion energy and mass spectrometers. The total p/l mass is 10 kg.

NASA FAST Satellite



SPACE SYSTEMS LORAL

NASA FAST Satellite

FAST will observe and measure rapidly varying electric and magnetic fields and the flow of electrons and ions above the aurora. FAST will correlate measurements of electrical and magnetic fields from other sensors and simultaneously correlate these forces with their effects on electrons and ions at altitudes of 350 to 4200 km. These observations will be complemented by data from other spacecraft at higher altitudes, which will be observing fields and particles and photographing the aurora from above, thus placing FAST observations in global context. At the same time, auroral observatories and geomagnetic stations on the ground will provide measurements on how the energetic processes FAST observes affect the Earth.

Launched 21 August 1996, using a Pegasus XL
Orbit 350 x 4200 km, 83 deg inclination
Size 1.2 m dia x 1.8 meter cylinder Mass 180 kg
Design Life 1 year Manufacturers GSFC
Spacecraft in Series 1

Spin stabilized (12 rpm) keeping spin axis aligned with orbit-normal. Body mounted GaAs solar cells. Aluminum structure. Dual 80C85 rad hard processors. S-Band transponder for command and telemetry. Passive thermal control.

Payload Description Electric Field Experiment uses 3 orthogonal boom pairs to measure plasma density and electron temperature. Magnetic Field Experiment has 2 magnetometers 180 deg apart deployed on graphite booms. Time-of-Flight Energy Angle Mass Spectrograph (TEAMS) is a mass-resolving spectrometer designed to measure 3-dimension distribution functions of major ion species. Sixteen Electrostatic Analyzers (ESA) are used for electron and ion measurements from 3 eV to 30 KeV.

NASA Lewis Satellite

SPACE SYSTEMS LORAL

NASA Lewis Satellite

Lewis was selected in the NASA SSTI (Small Spacecraft Technology Initiative) program, along with Clark, to demonstrate advanced spacecraft technologies. Contract start was July 11, 1994.

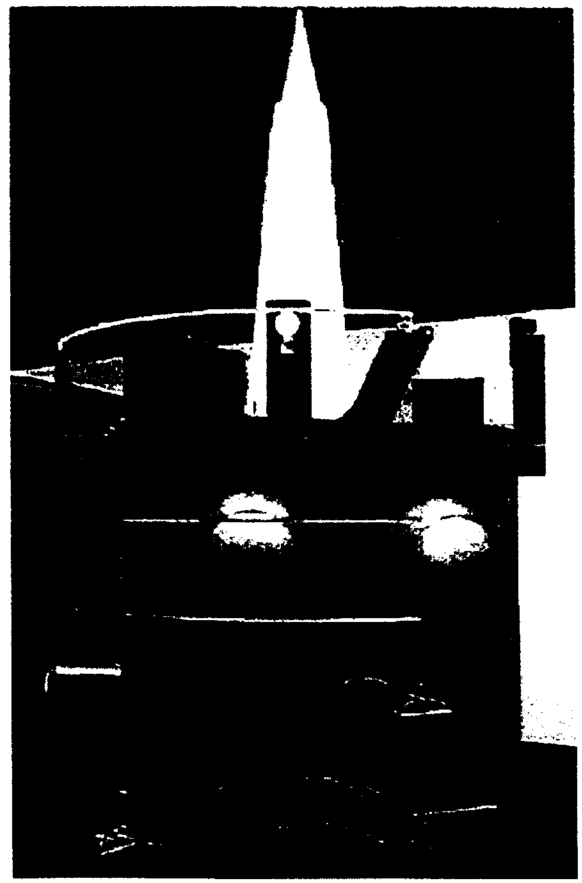
Launch Planned for on or before July 11, 1996 using Pegasus XL
Orbit 523 km, circular, sun synchronous
Size 36.5 in. hex by 69.5 in. Mass 288 kg
Design Life 3 years (required life is 1 year)
Spacecraft in Series 1
Spacecraft Description

3-Axis stabilized, zero momentum biased control system with 0.004 deg knowledge. Single solar array generates 600W power at 1 year. Hydrazine propulsion system with eight 1-lbf thrusters. System reliability is 0.86 at 5 years. Ground stations: primary at TRW, Chantilly VA, receive only at Univ. Alaska, Fairbanks, Stennis Space Center, MS, HBCU network. Data archive by Stennis Space Center, accessible on Internet, T1 line.

Payload Description

Earth imaging Hyperspectral Imager (HSI) with 384 bands (0.4 - 2.5 μm, 30 m pixels, 7.7 km swath, panchromatic: 0.45 - 0.75 μm, 5 m pixels, 13 km swath). Linear Etalon Imaging Spectral Array (LEISA), from NASA GSFC (1.0 - 2.5 μm, 256 channels, 300 m pixels, 77 km swath). Ultraviolet Cosmic Background (UCB), from UC Berkeley (35 - 85 nm). In addition to the above instruments, several spacecraft systems technologies will be flown. A partial list of planned technology demonstrations is: pulse tube cryocooler, solid state recorder, fiber optic data bus, GPS attitude determination, wide FOV star tracker, magnetically suspended reaction wheel, lightweight structure with integrated thermal control, GFRP overwrapped propellant tank.

NASA Lunar Prospector Satellite



SPACE SYSTEMS LORAL

NASA Lunar Prospector Satellite

Lunar Prospector is the third of NASA's Discovery missions and will orbit the moon in June 1997. The mission is led by Lockheed and NASA's Ames Research Center and the entire cost, including launch vehicle, should be less than $63M ($59M in FY92 $). Lunar Prospector is designed to determine the origin, evolution, and current state of resources of the Moon via low-altitude mapping of its surface composition, magnetic fields, gravity fields, and gas release events. The mission will provide a new map that shows in unprecedented detail the chemical composition and the magnetic and gravity fields of the Moon.

Planned for June 1997 using LLV2

Orbit 100 km lunar orbit, period of 118 min.

Size . 1.22 m tall, 1.4 m diameter (excluding booms) Mass 290 kg (including fuel) Design Life One year in lunar orbit

Spacecraft in Series 1

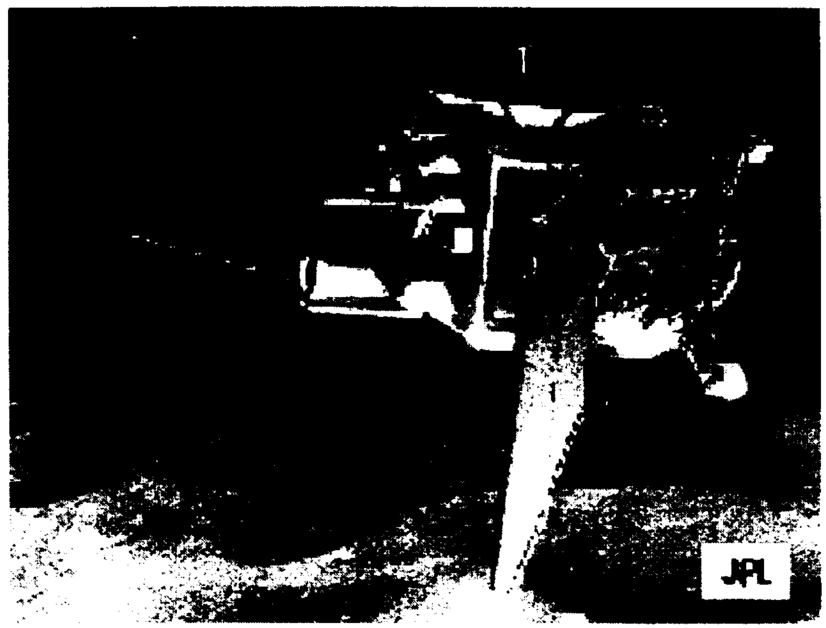
Spacecraft Description

The spacecraft bus is a 126 kg graphite-epoxy drum with three deployed booms carrying instruments. Spin stabilization is used with Sun and limb sensors for attitude determination. Hydrazine propulsion system with six 22-N thrusters provide 1430 m/s of delta-V. Body mounted solar cells provide 206 W. 15 AHr CPV NiH2 batteries. Communications use two S-Band transponders with slotted, phased array medium gain antennas. The spacecraft has no on-board computer. Three 2.5 meter deployable booms for science instruments.

Payload Description

Gamma-ray and neutron spectrometers will provide information on the composition of the Moon's crust. A magnetometer and electron reflectometer will map the lunar magnetic field. An alpha particle spectrometer will identify the location and type of gasses released from the lunar surface. Doppler tracking in S-Band will provide gravity field measurements.

NASA Mars Global Surveyor Satellite



NASA Mars Global Surveyor Satellite

MGS is the first mission of the Mars Surveyor Program. MGS will be a polar orbiting spacecraft at Mars designed to monitor global weather and provide global maps of surface topography and distribution of minerals. After the initial elliptical capture orbit, four months of thruster firings and aerobraking maneuvers will be used to reach the nearly circular mapping orbit. Mapping operations will begin in late January 1998. After collecting data for a full Martian year (approx. 2 Earth years), the spacecraft will be used as a data relay station for future missions for an additional three years.

Launch: Planned for November 1996 from ETR on a Delta II
Trajectory: 10 months, elliptical capture orbit
Orbit: 405 km, polar (i=92.5 deg), sun synchronous (2 pm), 2 hour period
Size 1.2 m x 1.2 m x 1.8 m Mass 651 kg (dry)
Design Life 6 years
Spacecraft in Series 1

Spacecraft Description

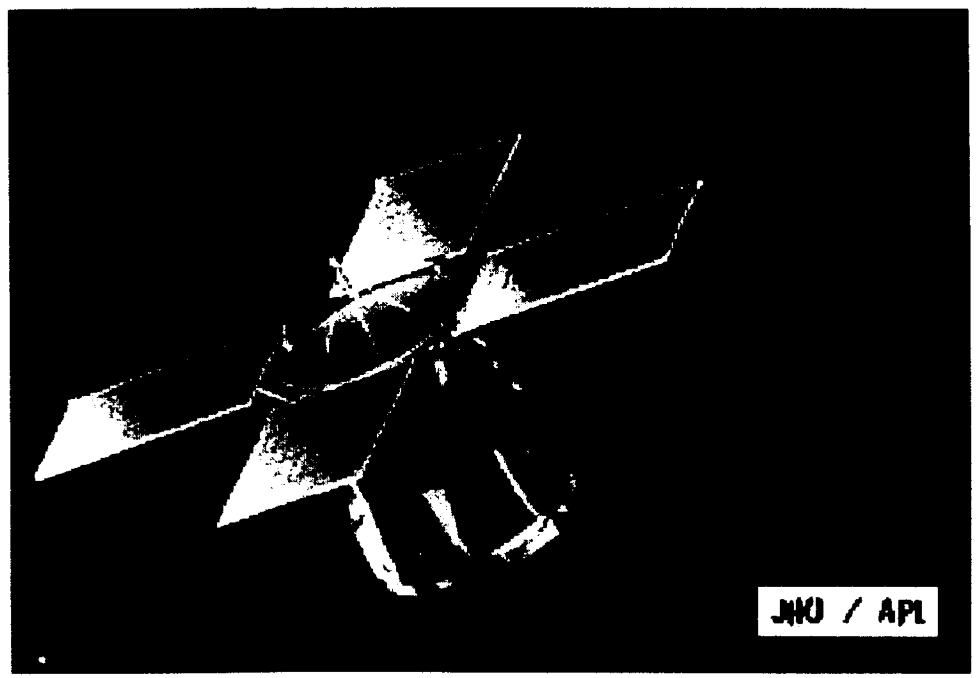
X-band uplink at 500 bps. X-band and Ka-band downlinks at 85.3 kbps max. 1.5m articulated HGA on a 1.5m boom plus 4 LGAs. Dual-mode propulsion system (MMH/NTO) with one 596 N and twelve 4.45 N thrusters. Total propellant load 361 kg. Passive thermal control design. 3-axis stabilized with 10 mrad control and 3 mrad knowledge using reaction wheels, thrusters, sun sensors, horizon sensors, celestial sensors, and IMU. Four articulated 1.7 m x 1.8 m solar panels (2 GaAs, 2 Si) provide 656 watts (980 W max). Two 20 Ahr NiH2 batteries. 1750A-based central computer. Total data storage capacity is 3 gigabits.

Payload Description

Mars Orbiter Laser Altimeter (MOLA) provides local and global maps of Martian topography with 2 to 30 meter vertical resolution. Mars Orbiter Camera (MOC) supports study of climate, surface geology, and surface and atmospheric interactions. Camera system includes wide angle (140 degree) and narrow angle (0.4 degree) optics for producing global coverage (7.5 km/pixel), selective moderate resolution images (280 m/pixel) and very selective high resolution (1.4 m/pixel) images.

Magnetometer and Electron Reflectometer for global study of intrinsic magnetic field. Thermal Emission Spectrometer to map the mineral content of rocks, ice caps, and clouds. Radio Science (Ultrastable Oscillator) to study gravitational field, atmospheric refractive indices, and temperature profiles. Mars Balloon Relay to serve as a data relay for planned future surface stations and atmospheric experiments. Ka-Band Link Experiment (KaBLE) to provide Ka-band signal for atmospheric attenuation studies. Total science payload mass is 78 kg.

NASA NEAR Satellite



NASA NEAR Satellite

The NEAR mission is the first launch in NASA's Discovery Program , and will be the first spacecraft ever to orbit an asteroid. The mission will help answer fundamental questions about the nature and origin of near-Earth objects, as well as provide clues about the how Earth itself was formed. The vehicle will rendezvous with the asteroid 433 Eros in December 1998, and will perform up-close observations for about one year. The primary scientific goals are to measure the asteroid's: (1) bulk properties (size, shape, volume, mass, gravity field, and spin state); (2) surface properties (elemental and mineral composition, geology, morphology, and texture); and (3) internal properties (mass distribution and magnetic field). Prior to its encounter with Eros, NEAR will also fly by the Main Belt asteroid Illya.

Launch February 17, 1996 on Delta 2 from ETR

Trajectory 2 year Delta-V Earth Gravity Assist
Asteroid Illya flyby: August 1996 Earth flyby: January 1998 Asteroid Eros rendezvous: December 1998

Orbit Variable, ranging from 1000 x 1000 km to 35 x 35 km about Eros

Size 1.7 x 1.7 x 2.75 meter Mass 805 kg (including fuel)

Design Life 4 years

Spacecraft in Series 1

Spacecraft Description

3-axis stabilized. 4 deployed, non-articulating fixed solar panels provide 1600 watts at 1 AU. Fixed 1.5 m diameter high gain antenna. X-band communications using via DSN with selectable data rates between 1 and 27 kbps. Passive thermal control. 1 Gbit solid state data storage. Hydrazine propulsion system with one 100 lbf, four 5 lbf, and seven 1 lbf thrusters provide a total delta-V of 1425 m/s. Mission control provided by JHU APL with navigation support from JPL

Payload Description

The instrument payload totals 55 kg and 48 watts.

MultiSpectral Imager (MSI - a refractive telescope with passively cooled Si CCD array (244 x 537). 2.25 x 2.9 deg FOV, 10-16 meter resolution from 100 km altitude, sensitive between 400 and 1100 nm.

X-Ray/Gamma-Ray Spectrometer (XGRS - containing two sensors (an X-ray fluorescence spectrometer and a gamma-ray spectrometer).

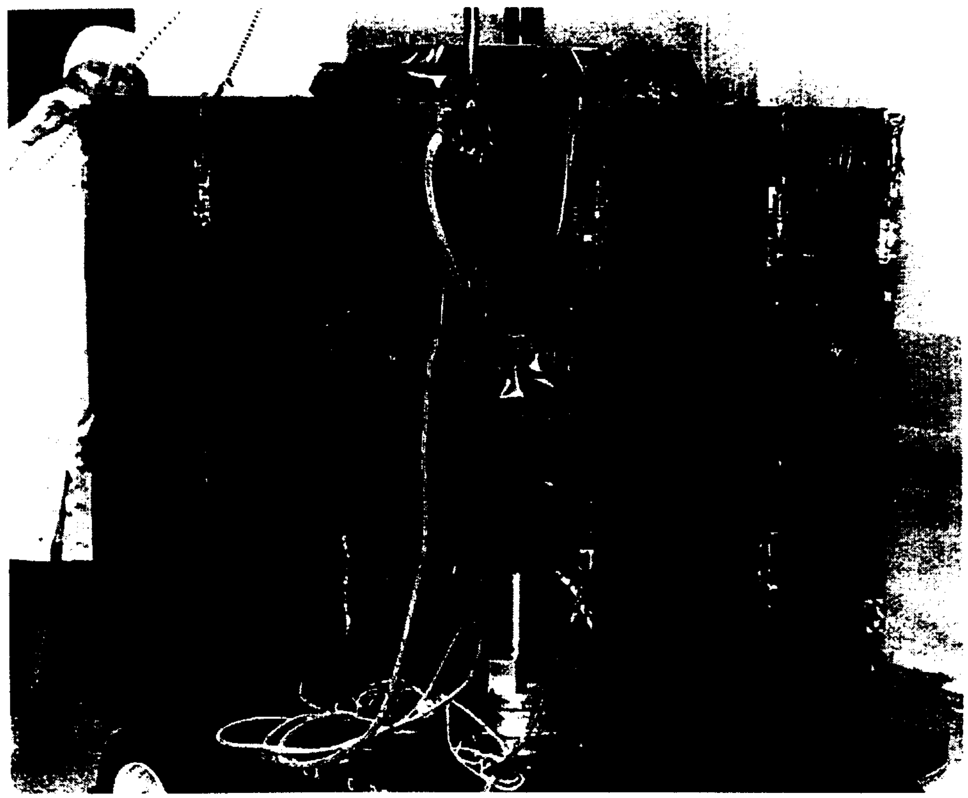
Near-Infrared Spectrograph (NIS) - a spectrometer covering 800-2700 nm.

Magnetometer - a three-axis fluxgate sensor .

NEAR Laser Rangefinder (NLR) - an altimeter that uses a solid-state pulsed laser to measure the distance between the spacecraft and the surface of the asteroid. Nd-YAG laser operating at 1.064 mm wavelength, 6 meter resolution, 50 km range.

Radio Science - uses the satellite's telemetry system to map Eros' gravity field.

NASA SAMPEX Satellite



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NASA SAMPEX Satellite

SAMPEX is the first of NASA's Small Explorer (SMEX missions. SAMPEX is designed to study the energy, composition and charge states of four classes of charged particles that originate beyond the Earth: (1) galactic cosmic rays (from supernova explosions in our Galaxy); (2) anomalous cosmic rays (from the interstellar gas surrounding our solar system); (3) solar energetic particles (from explosions in the solar atmosphere); and (4) magnetospheric electrons (particles from the solar wind trapped by the Earth's magnetic field).

Launch July 3, 1992 on Scout

Orbit 520 x 670 km, 82 deg inclination Size 1.5 m tall x 0.86 m dia. (stowed) Mass 158 kg Design Life 1 year, 3 year goal

Spacecraft in Series 1

Spacecraft Description

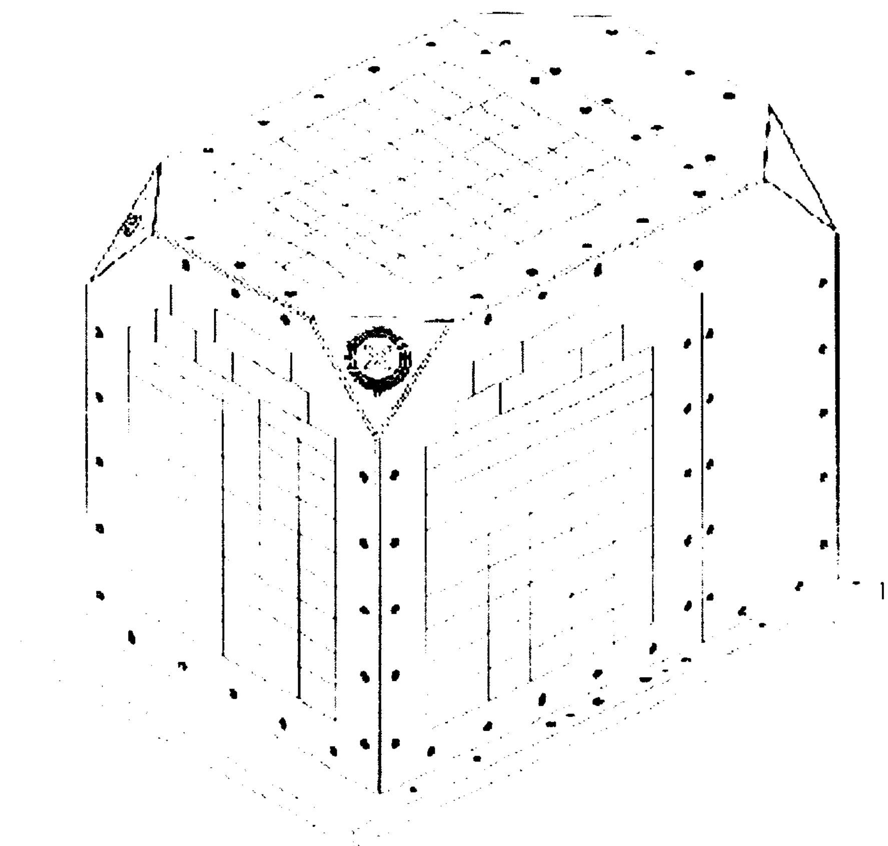
Single string architecture. Momentum-bias control system with 1 momentum wheel and 3 torque rods for vehicle pointing. Attitude knowledge better than 2 deg (3 sigma).

Attitude sensing via sun sensors, star sensors, magnetometer. 2 deployable, articulated solar panels with GaAs cells provide 102 W orbit average power (no eclipse). One 9 A-hr NiCd battery. Aluminum structure. Passive thermal control. Two omni antennas and near-Earth 5W S-band transponder. Solid-state memory. Fiber optic data bus (Mil-Std 1773).

Payload Description

SAMPEX carries 4 instruments. HILT (Heavy Ion Large Area Proportional Counter Telescope) measures heavy ions from 8 to 220 MeV/nucleon for oxygen. LEICA (Low Energy Ion Composition Analyzer) measures 0.5 to 5 MeV for solar and magnetospheric ions. LEICA and HILT were originally designed and constructed as Get-Away-Specials for shuttle flight. MAST (Mass Spectrometer Telescope) measures isotropic composition of elements from Li to Ni in the range of 10 MeV to several hundred MeV. PET (Proton/Electron Telescope) complements MAST by measuring the energy spectra and relative composition of protons (18 to 250 MeV) and helium nuclei (18 to 350 MeV/nuclei) of solar, interplanetary, and galactic origins, and the energy spectra of solar flare and precipitating electrons from 0.4 to 30 MeV.

NASA Surfsat-1 Satellite



SPACE SYSTEMS LORAL

NASA Surfsat-1 Satellite

(Summer Undergraduate Research Fellowship Satellite #1

SURFSAT-1 is a small satellite built by undergraduate college students and the Jet Propulsion Laboratory to support experiments by NASA's Deep Space Network. The satellite is designed to mimic signals from planetary spacecraft, and radiates milliwatt level RF signals in X-, Ku-, and Ka-band. These signals support research and development experiments supporting future implementation of Ka-band communications, tests of new 11m ground stations built to support the Space Very Long Baseline Interferometry project (SVLBI), and training of ground station personnel. From conception through launch, the spacecraft cost ~$3 million, including design, fabrication, test, and launch integration. The mission's goal is to provide one year of continuous operation, although the lack of any consumables aboard the spacecraft may permit several additional years of operation.

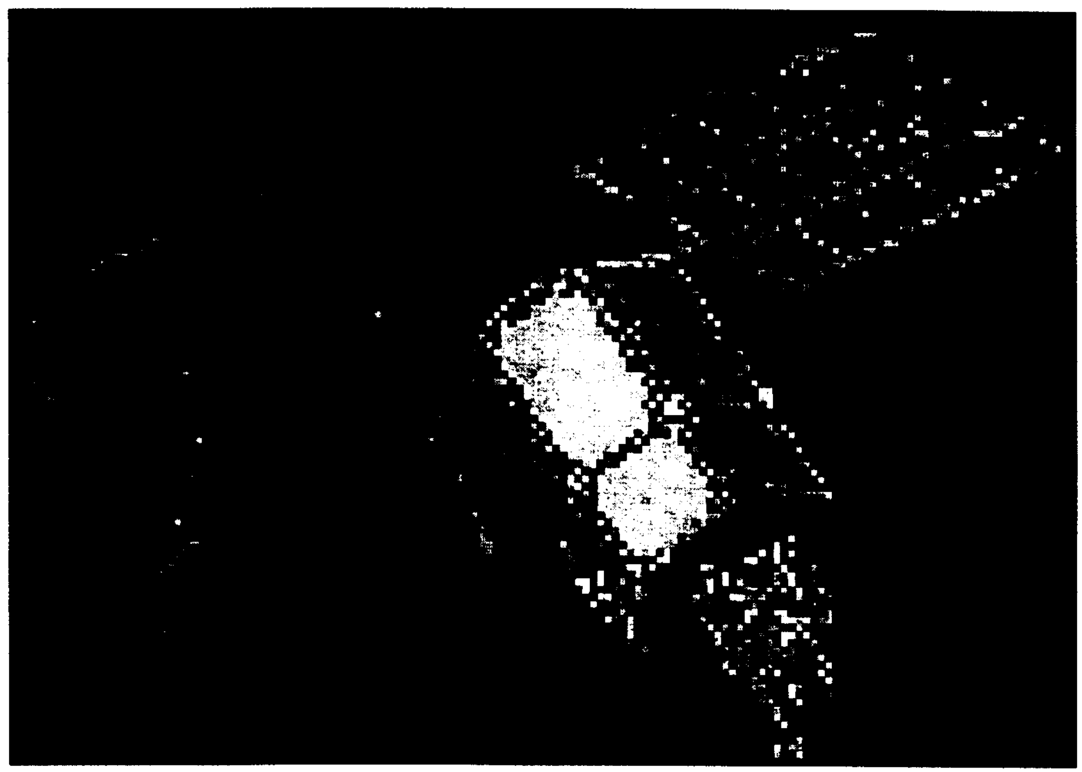
Launch November 4, 1995 on Delta 2 from WSMC. Launched as secondary payload with RADARSAT
Orbit 1000 x 1400 km, sun-synchronous polar orbit
Size 2 boxes, each 30 cm x 30 cm x 41 cm Mass 55 kg
Design Life 1 year
Spacecraft in Series 1
Spacecraft Description

The spacecraft consists of two aluminum boxes that remain permanently bolted to the launch vehicle's 2nd stage rocket. The SUSFSAT-1/Delta 2 system is gravity gradient stabilized. Power is provided by body-mounted solar panels providing ~15W orbit average power. The boxes are electrically connected by a interconnecting wire harness. 3 omni antennas (2 on the "primary" box, 1 on the "secondary" box) are used for uplink and downlink. 3 independent command detector units are used to control operational state. The vehicle uses passive thermal control. There is no propulsion system, batteries or attitude control / determination.

Payload Description

One transponder a carried in each box. The "primary" transponder operates at X-band (8 Ghz) and Ka-band (32 Ghz). The "secondary" transponder operates at Ku-band (15 Ghz). Both transponders radiate milliwatt level signals through the omni antennas.

NASA TOMS-EP Satellite



SPACE SYSTEMS LORAL

NASA TOMS-EP Satellite

Total Ozone Mapping Spectrometer - Earth Probe

TOMS-EP is the third mission in NASA's TOMS program, which provides long-term observations of the global distribution of the Earth's ozone layer and measurements of sulfur-dioxide released in volcanic eruptions. Previous versions of the spacecraft's TOMS instrument were launched aboard the Nimbus-7 (1978) and Soviet Meteor-3 (1991) satellites. These instruments provided detailed maps of the world's global atmospheric ozone distribution, and observed the Antarctic "ozone hole" which forms September through November of each year. TOMS-EP will add to the data set collected by these missions, and will be followed by other TOMS instruments carried aboard ADEOS (1996) and Russian Meteor-3M satellite (2000).

Launch July 2, 1996 on Pegasus XL

Orbit 500 km, circular, sun-synchronous, incl. = 97.36 deg. Originally planned for 955 km, circular, sun synchronous

Size 1.1 m diameter x 1.73 m tall (stowed), solar panels 3.9 m across when deployed Mass 295 kg (at launch)

Design Life 2 years

Spacecraft in Series 1

Spacecraft Description

Based on the TRW Eagle family of light weight spacecraft. 3-Axis stabilized with pointing control to 0.5 deg (3-sigma) in pitch/roll, 1 deg in yaw and knowledge to 0.25 deg (3-sigma). Two solar arrays provide up to 275W. Hydrazine propulsion system (54 kg fuel) with eight thrusters (0.4 and 1 lbf) is used to raise the orbit and desaturate momentum wheels. Communications via DSN, GSTDN. Downlink rate: 50 and 202 kbps. Uplink rate: 2 kbps. 16 Mb solid state storage. Designed for 24 hour autonomous operation.

Payload Description

The TOMS instrument has a mass of 35 kg and 25 watts average power. Resolution is 47 km at nadir and 62 km average. Swath width is 2750 km.

TOMS/EP measures total ozone by observing both incoming solar energy and back scattered ultraviolet (UV) radiation at six wavelengths. "Back scattered" radiation is solar radiation that has penetrated to the Earth's lower atmosphere and is then scattered by air molecules and clouds back through the stratosphere to the satellite sensors.

TOMS makes 35 measurements every 8 seconds, each covering 30 to 125 miles (50 to 200 kilometers) wide on the ground, strung along a line perpendicular to the motion of the satellite.

NASA TRACE Satellite

SPACE SYSTEMS LORAL

NASA TRACE Satellite

Mission Description: This part of NASA's SMEX program. TRACE will observe the Sun to study the connection between its magnetic fields and the heating of the solar corona.
Launch: Sept. 1997
Orbit: 700 km sun synchronous Mass: 225 kg
Design Life: 1 year

Spacecraft Description:

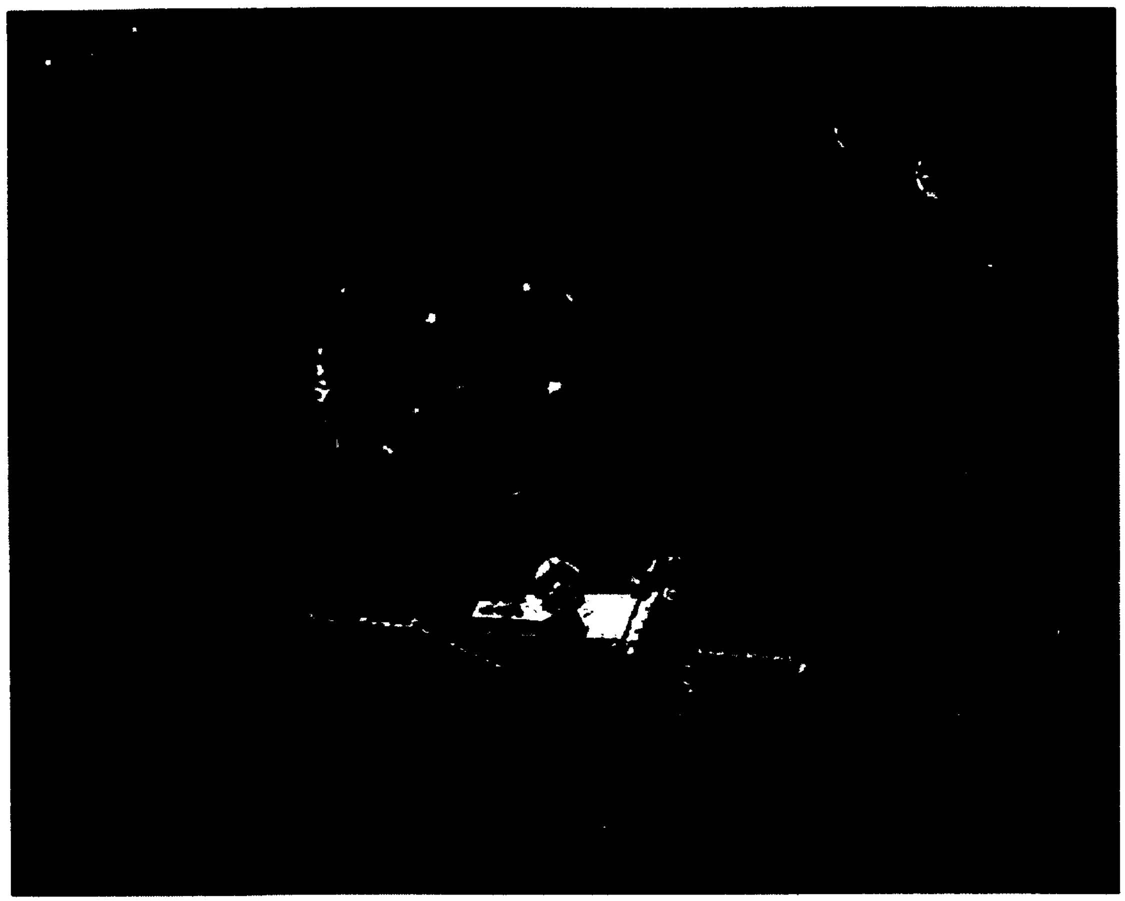
Power Sources four deployed GaAs solar arrays, 9A-Hr "super" Ni-Cd battery. Attitude Control 3 axis stabilized; 3 axis magnetometer six coarse sun sensors, bright object sensor, digital sun sensor, 3 gyros, guide telescope, 3 torque rods, 4 reaction wheels It also carries an S band transponder with -5w transmitter capability.

It can provide up to 110 Mbytes of data storage. The spacecraft can deliver 250 w of power.

Payload Description:

The payload mass is 51 kg and it consumes 33w.

NASA Ulysses Satellite



NASA Ulysses Satellite

Ulysses is a joint NASA / ESA mission designed to study the polar regions of the Sun. Ulysses' primary scientific objectives are to investigate, as a function of solar latitude, (1) the properties of the solar corona, (2) the solar wind, (3) the structure of the Sun-wind interaction, (4) the heliospheric magnetic field, (5) solar and non-solar cosmic rays, (6) solar radio bursts and plasma waves, (7) solar x-rays, and (8) interstellar / interplanetary neutral gas and dust. The spacecraft was placed onto a trajectory that intercepted Jupiter and used the gas-giant's gravity to leave the ecliptic plane.

Launch October 6, 1990 from Shuttle using IUS and PAM-S upper stages.

Trajectory Jupiter gravity assist and plane change (JGA) with Jupiter closest approach on February 8, 1992

Orbit Heliocentric, 5 x 1.3 AU, polar

Size 3 m x 3 m x 2 m Mass 370 kg Spacecraft in Series 1

Spacecraft Description

Spin stabilized at 5 rpm (Earth pointing). X-, S- band communications using 1.65 m HGA. RTG used for power generation, provides 285 W (BOL). Hydrazine propulsion for trajectory correction maneuvers. Redundant tape recorders store 46 Mb each. 3 deployable booms for science instruments.

Payload Description

Ulysses carries 9 instruments in addition to using the radio system for scientific investigations. Total payload mass is 55 kg.

Magnetometer (VHM/FGM - designed to measure changes in the interplanetary magnetic field at different heliographic latitudes,

Solar Wind Plasma Experiment (SWOOPS - studies protons,

electrons and heavy ions in the solar wind

Solar Wind Ion Composition Instrument (SWICS - studies the elemental and ionic-charge composition, and the mean temperatures and mean speeds of all solar-wind ions from hydrogen to iron,

Unified Radio and Plasma Wave Instrument (URAP - designed to determine the direction and polarization of distant radio sources, as well as radio bursts from charged particles in the solar wind,

Energetic Particle Instrument (EPAC - measures intensities and energies of interplanetary ions

Heliosphere Instrument for Spectral, Composition and Anisotropy at Low Energies (HISCALE - designed to measure elemental abundance's of interplanetary ions and electrons,

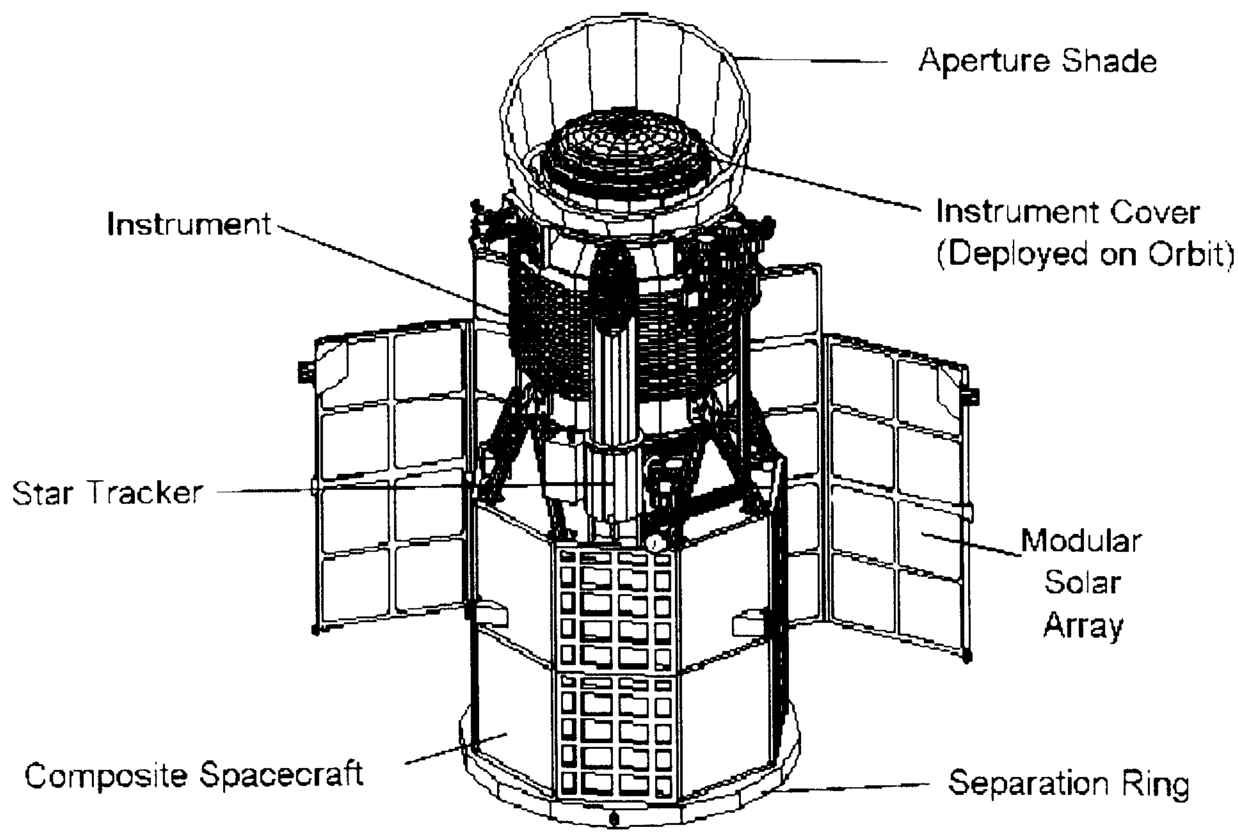
Cosmic Ray and Solar Particle Instrument (COSPIN

Solar X-Ray and Cosmic Gamma-Ray Burst Instrument (GRB - measures electrons in solar flares and determines the direction of gamma-ray bursts from distant galaxies,

Cosmic Dust Experiment (CDE - will provide direct observations of particulate matter and its interaction with solar radiation

Additionally, the spacecraft's radio subsystem will be used to measure density, turbulence and velocity of the plasma in the Sun's corona, as well as search for the presence of passing gravity waves.

NASA WIRE Satellite



SPACE SYSTEMS LORAL

NASA WIRE Satellite

Wide Field Infrared Explorer

WIRE was one of two missions selected in 1994 as part of NASA's Small Explorer Program (SMEX). The spacecraft will survey the celestial sky in the infrared bands and will build on the results of the IRAS mission. The specific objectives of the WIRE mission are to: (1) determine what fraction of the luminosity of the universe at a red shift of 0.5 and beyond is due to starburst galaxies; (2) determine whether luminous protogalaxies are common at red shifts less than 3; (3) amass a catalog larger than the IRAS Point Source Catalog; and (4) make a survey more than 500 times fainter than the

IRAS Faint Source Survey at 12 and 25 µm.

Launch Planned for October 1998 on Pegasus

Orbit 400 km, circular, polar, sun-synchronous

Mass 282 kg (incl. 82 kg instrument) Design Life 4 months

Spacecraft in Series 1

Spacecraft Description

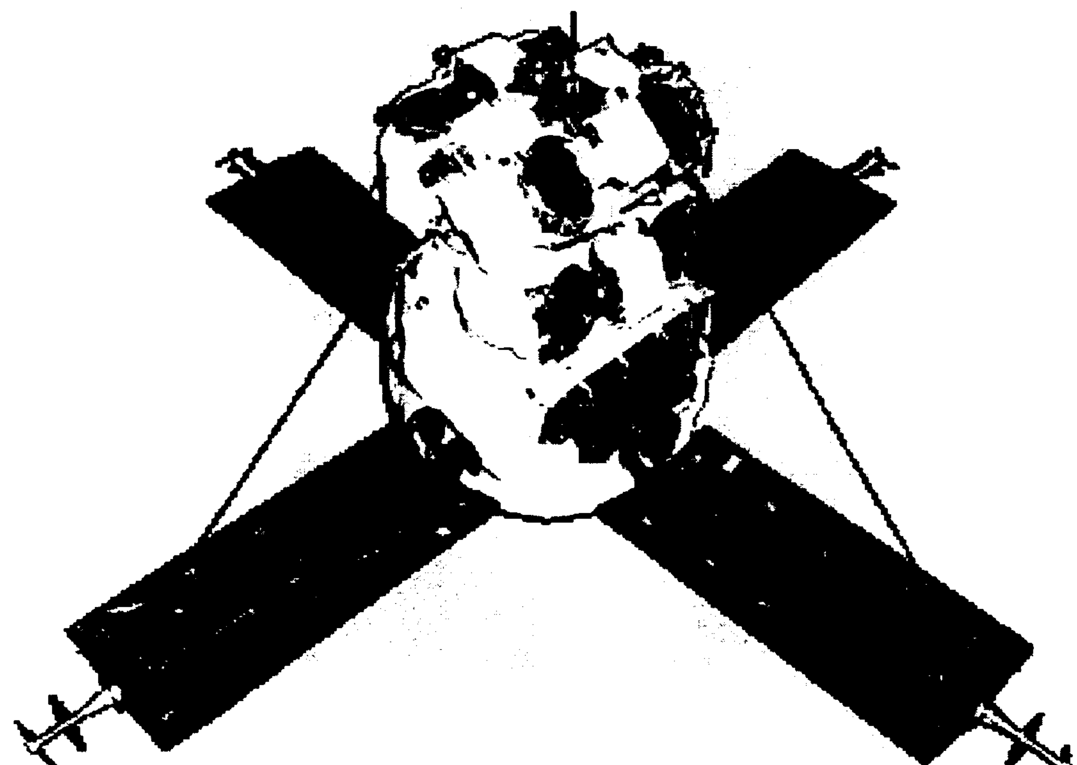
Small Explorer bus. 3-axis stabilized, momentum biased control system with 2 arc-minutes pointing. Uses reaction wheels, gyros, star tracker, torque rods, sun-sensors. Non-articulated solar array provides 170 W orbit average power using GaAs cells. One 9 AHr Super NiCd battery. 80386 processor with 88 MBytes memory. Uplink at 2 kbps, downlink at 18.75, 900, and 1800 kbps using 5 W S-Band transponder.

Payload Description

A 30 cm aperture Cassegrain telescope, diffraction limited at 25 &#181 m, with no moving parts or crumb optics. Two 128 x 128 Si:As BIB focal plane arrays. Optics are cooled to less than 19 Kelvin and detectors are cooled to less than 7.5 Kelvin using 3 kg solid hydrogen. Instrument power is 35 watts, with average data rate of 9 kbps. Each exposure lasts 32 - 128 seconds.

SPACE SYSTEMS LORAL

DoD Alexis Satellite



SPACE SYSTEMS LORAL

DoD Alexis Satellite

ALEXIS is a small spacecraft built for the Los Alamos National Laboratory (LANL). Its mission is to provide high resolution maps of low-energy astronomical X-Ray sources. The mission is also intended to demonstrate the feasibility of quickly building low cost sensors for arms treaty verification. The spacecraft's cost is reported to be approximately $17 million. An anomaly caused one of the satellite's four solar panels to break away from the vehicle during launch, causing a temporary loss of the spacecraft. Ground controllers were eventually able to control the spacecraft and begin on-orbit operations.

Launch April 25, 1993 on Pegasus
Orbit 750 km circular, 65 deg inclination
Size Cylinder 1 m high, 60 cm diameter Mass 115
Design Life 1 year, goal of 3 years
Spacecraft in Series 1

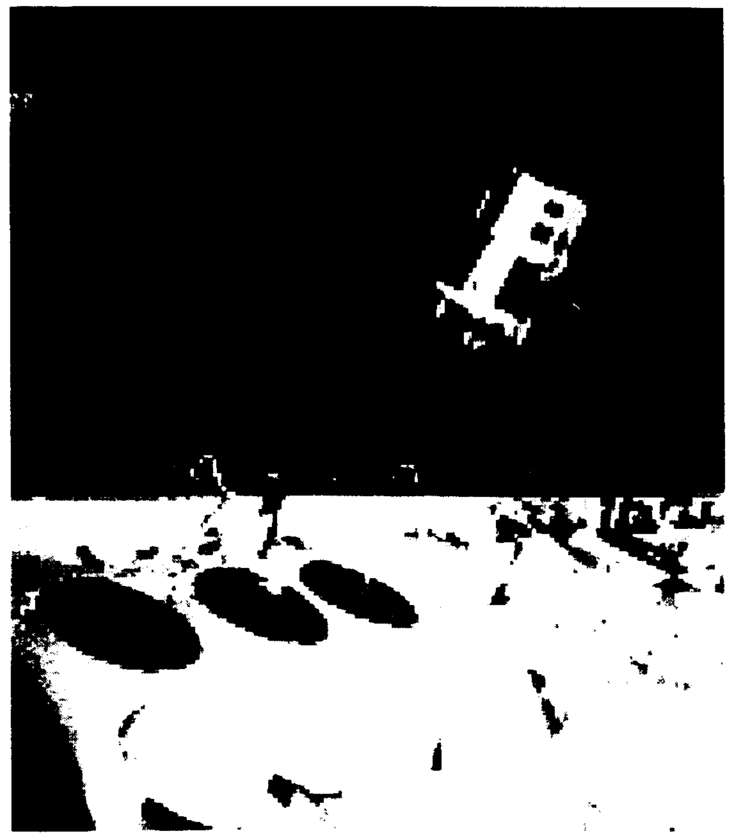
Spacecraft Description

Four deployed solar panels provide 50 watts average power. Four NiCd batteries. Redundant 80C86 processors. S-Band communications with downlink rate 750 kbps and uplink rate of 9.6 kbps. Mass memory of 100 Mbytes EDAC solid state storage. The bus points the payload in the anti-sun direction and rotates at 2 rpm about that sun-line. Sun sensors and limb sensor provide attitude knowledge of 0.1 deg. Attitude is controlled using magnetic torque coils. Single ground station installed at LANL has 6 ft antenna.

Payload Description

An ultra soft X-ray monitor which consists of 6 compact normal-incidence telescopes tuned to narrow bands centered on 66, 71, and 93 eV.

DoD CRO Satellite



SPACE SYSTEMS LORAL

DoD CRO Satellite

Chemical Release Observation

The CRO satellites were part of an SDIO program designed to test the ability of space-based, ground-based, and airborne sensors to track incoming ICBMs. The experiment was designed to determine how the intentional release of rocket propellants from an incoming ICBM would mask the missile's signature. Managed by the Los Alamos National Laboratory, the 3 CRO satellites were deployed from Get-Away-Special (GAS) canisters carried by STS-39. After release, the satellites' chemical payloads were released by ground command, and were subsequently tracked by a number of sensors, including sensors carried aboard STS-39. These sensors were used to study the chemicals' optical, infrared, and RF characteristics. All three satellites performed as designed and decayed approximately one week after deployment.

Launch April 28, 1991 aboard STS-39. Deployed from GAS canisters May 2, 3 Decayed on May 12, 13, 14

Orbit 260 km, 57.0 deg inclination Mass Approx. 82 kg each

Design Life 1 week

Spacecraft in Series 3

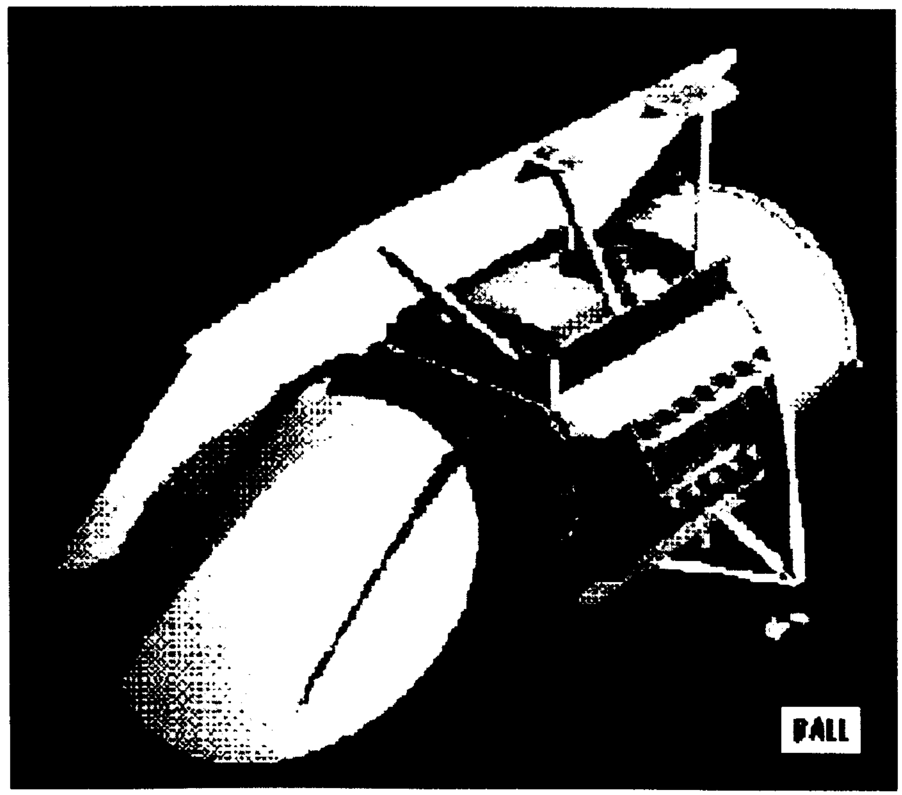
Spacecraft Description

Command and data handling system used an 1805 microprocessor interfacing to a UHF receiver and an S-band transmitter. Stabilization provided by atmospheric drag on a large corner reflector on the end of a deployable boom

Payload Description

CRO-C released 6.8 kg of nitrogen tetroxide, CRO-B released 23.6 kg of UDMH, and CRO-A released monomethyl hydrazine.

DoD GFO Satellite



DoD GFO Satellite

GEOSAT Follow On

GFO is a follow-on to the successful GEOSAT program which flew between 1985 and 1990. GFO will provide real-time ocean topography data to 65 Navy users at sea and on shore. This data on wave heights, currents and fronts will also be archived and made available to scientific and commercial users through NOAA. The GFO contract was awarded in September 1992 and is worth $46M for the first satellite and $115M if options for two additional spacecraft are exercised. These costs include launch, ground software and 60 days operation. Contract includes incentive fees based on long-term on-orbit performance.

Launch Planned for September 1996 on Taurus

Orbit 880 km circular, 108 deg inclination, 17 day repeat

Size 3 m long

Mass 347 kg

Design Life 8 years

Spacecraft in Series 1, with options for 2 additional S/C

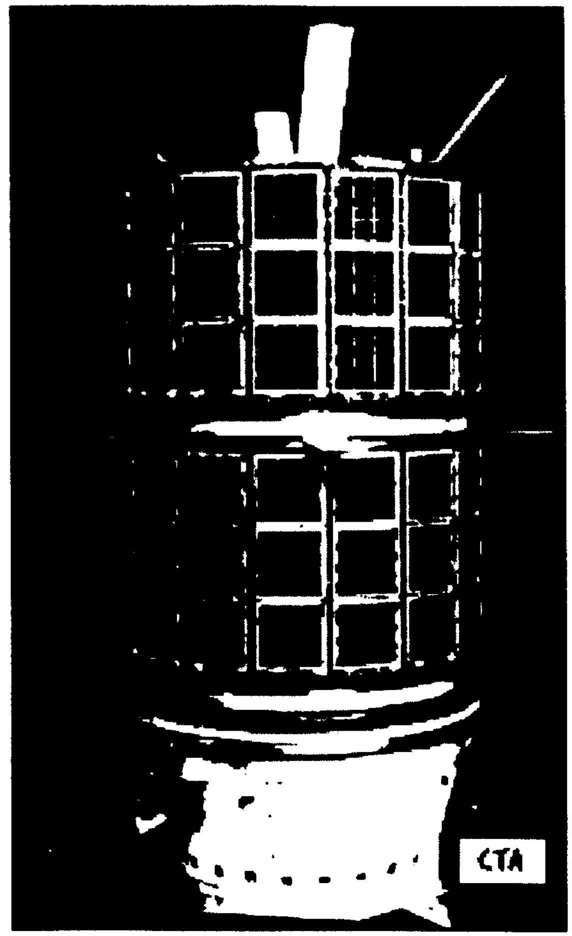
Spacecraft Description

3-axis stabilized. Single solar array with 1-axis articulation providing 126 W. Hydrazine propulsion system for orbit maintenance. Redundant GPS receivers provide precision orbit determination with an rms. accuracy (radial component) of 10 cm. A Doppler beacon transmitter is also carried to support orbit determination.

Payload Description

A radar altimeter is the primary payload. This 13.5 GHz radar provides 3.5 cm height measurement precision. The other payload is a dual frequency (22 and 37 GHz) water vapor passive radiometer with a path correction accuracy of 1.9 cm rms. Total payload is 47 kg and 121 watts.<p>

DoD MACSAT Satellite



SPACESYSTEMS LORAL

DoD MACSAT Satellite

Multiple Access Communications Satellites)

The two MACSAT spacecraft are third generation DSI digital communications satellites designed to demonstrate tactical UHF voice, data, fax and video store and forward capabilities for the U.S. military. The gravity gradient boom on one spacecraft appears to have failed to deploy. The other spacecraft was used during Operation Desert Storm for message relay to and from military troops in the Gulf region.

Launch May 9, 1990 tandem launch on Scout

Orbit 613 x 739 km, 90 deg inclination

Size 61 cm diameter, 35.6 cm high

Mass 68 kg

Spacecraft in Series 2

Spacecraft Description

Gravity gradient stabilized (approx. 5 degrees control) using a 9+ meter boom and 2.3 kg tip mass. Damping achieved via hysteresis rods. Z-coil used to invert the spacecraft should it stabilize upside down. Dual digital processors provide redundancy and 2.4 to 16 megabytes of data storage. 16 sided cylindrical structure. Body mounted solar cells provide 10 to 17 watts of orbit average power. Redundant NiCd batteries provide 150 Whr capacity.

Payload Description

Each satellite contains two digitally tunable 10 watt transmitters, a 65 watt high power auxiliary receiver for spacecraft command and hardware reconfiguration, and two antenna systems. This equipment was used to conduct store and forward communications demonstrations.

SPACE SYSTEMS LORAL

DoD MSTI-3 Satellite

MSTI-3 mounted to a test stand at Edwards AFB.

SPACE SYSTEMS LORAL

DoD MSTI-3 Satellite

Miniature Sensor Technology Integration : Collecting data on the Earth's atmosphere and terrestrial environment; demonstrate advanced technology for critical sensors, components, on-board processing technologies and architectures for future development of advanced space systems

Launch: 15 May 1996 on a standard Pegasus

Orbit: 361 X 296 km elliptical orbit inclined at 97 degrees

Size: 123 cm high

Mass: 211 kg

Manufacturer:Spectrum ASTRO

Design Life: 1 year

Spacecraft in Series 3

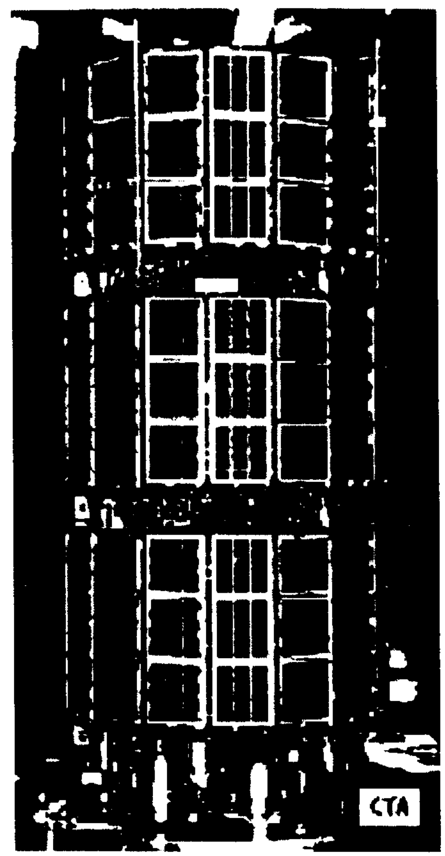
Spacecraft Description

GPS: supplies 100 m Ephemeris. 3-Axis Control via Reaction Wheels and a Star Tracker 0.010 (radians?).Hydrazine Propulsion (49 lbs) Power supplied by a 250w GaAs/Ge Solar Array and 10 A-hr NiH2 Battery .It has a 175OA and uses a R-3000 Processor to do Onboard Track File Generation

Payload Description:

Miniature sensors for 14 band short and mid range IR and 150 hyperspectral band

DoD Stacksat (POGS, TEX, and SCE) Satellite



SPACE SYSTEMS LORAL

DoD Stacksat (POGS, TEX, and SCE) Satellite

The U.S. military's STACKSAT mission involved the launch of three similar spacecraft, POGS, TEX and SCE. The POGS (Polar Orbiting Geomagnetic Survey) satellite was designed to measure the Earth's magnetic field vector as a function of position. Data from the experiment will be used to improve Earth navigation systems, and is stored in an experimental solid state recorder. TEX (Transceiver EXperiment) carries a variable power transmitter used to study ionospheric effects on RF transmissions. Data from the experiment will be used to determine minimum spacecraft transmitter power levels for transmission to ground receivers. SCE (Selective Communications Experiment) carries a variable frequency transmitter to study ionospheric effects at various RF frequencies, and is also designed to demonstrate message store and forward techniques. Six low cost ground stations were designed, built and located around the world to operate these spacecraft.

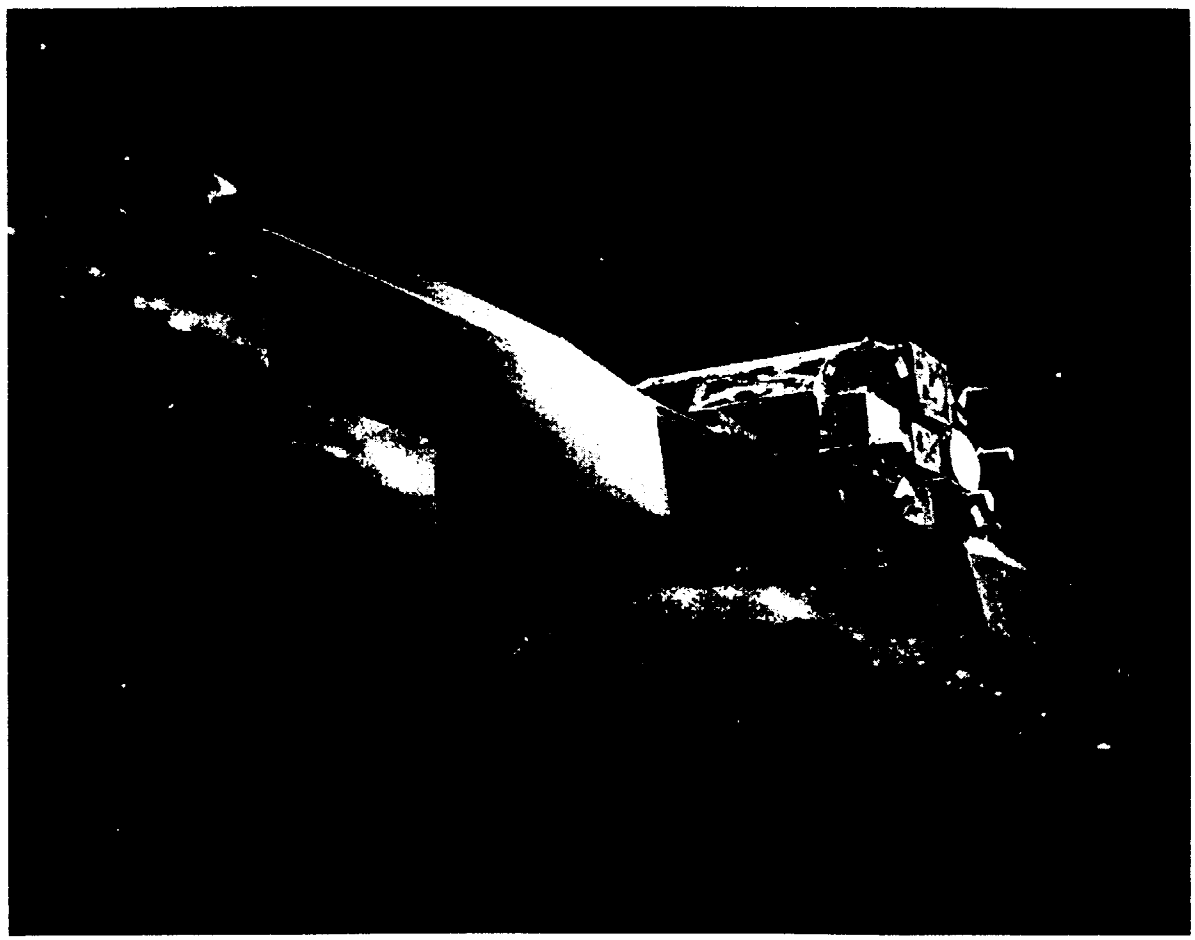
Launch April 11, 1990 by Atlas E from WTR

Orbit 741 km, incl. = 90 deg. Mass 55-68 kg

Design Life 3 years Spacecraft in Series 3

The spacecraft is gravity gradient stabilized. Body mounted solar arrays provide ~15W orbit average power. POGS carries a magnetometer to accurately map the Earth's magnetic field for the Defense Mapping Agency. Data is stored in a new Fairchild solid state recorder. TEX and SCE each carry specially designed transmitters to study ionospheric effects on radio signals. Attempt to obtain more information on the variable power amplifier by contacting the person listed on the Web page was unsuccessful due to the military nature of the satellite.

DoD STEP-0 Satellite



DoD STEP-0 Satellite

Purpose: to test technology for autonomous, survivable satellites	Launch :May 1994 on a Taurus

Size 1.12 x 1.78 m

Mass 488 kg

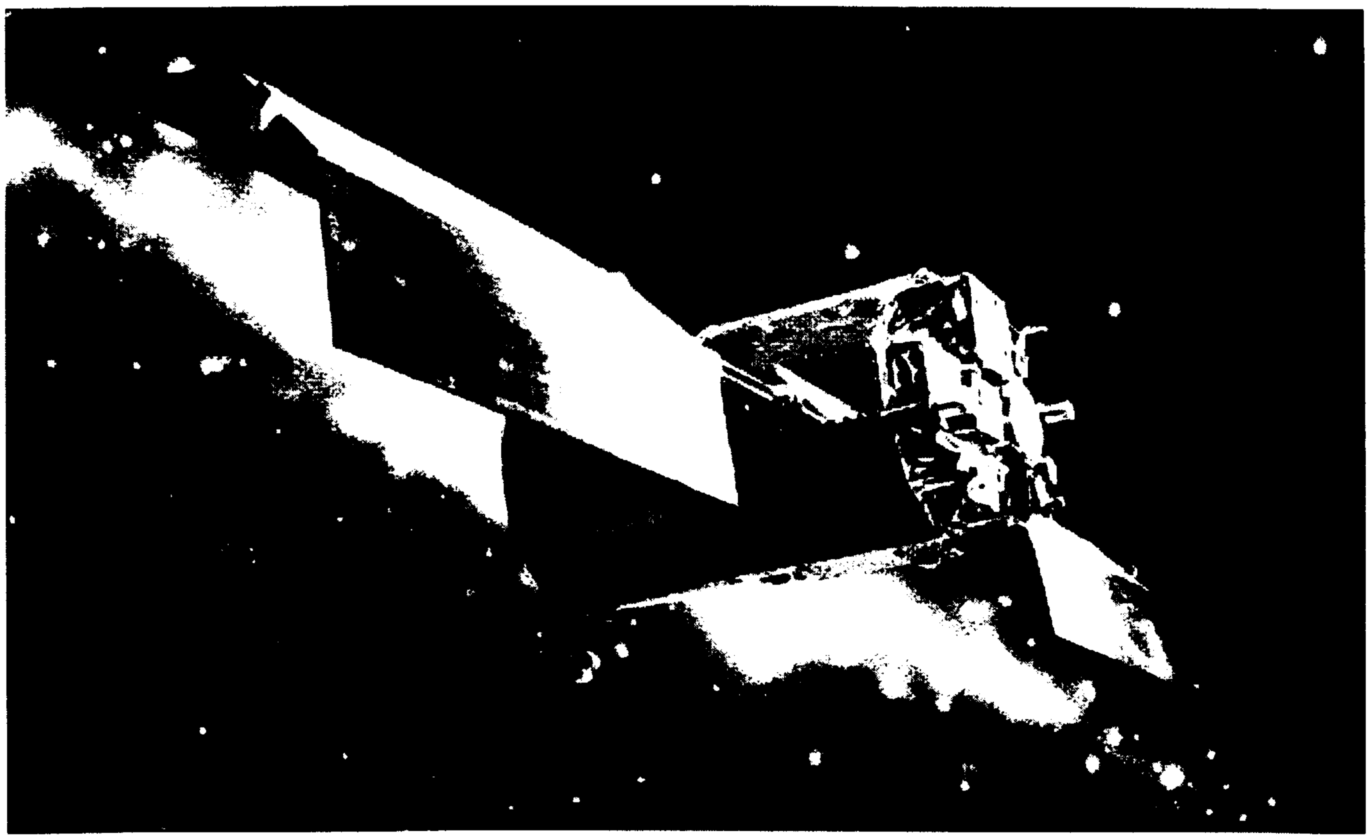
Design Life: 18 months

Spacecraft Description: S, C and X-band communication links Power 450W Power Sources two deployed wing solar panels Configuration:, 3-axis stabilized

Payload Description: Amongst the 10 instruments carried are two laser sensors, Dual cone sensors, Radar Illumination Verification System, and an advanced lightweight GPS sensor

SPACE SYSTEMS LORAL

DoD TAOS Satellite



SPACE SYSTEMS LORAL

DoD TAOS Satellite

Technology for Autonomous Operational Survivability

TAOS is a technology demonstration satellite whose purpose is to demonstrate

autonomous space navigation systems to reduce satellite ground support needs. Lowering the needs for ground support of future spacecraft would 1) increase the survivability of satellites during wartime conditions and 2) reduce satellite operations costs. TAOS also incorporates several new satellite bus components designed to improve reliability and maintainability of future spacecraft while reducing life-cycle costs. TAOS is the first mission flown under the USAF's STEP (Space Test Experiments Platform) program, and is designated STEP Mission 0.

Launch March 13, 1994 on Taurus (first Taurus launch)

Orbit 538 km, incl. = 105 deg. Size 0.95 m dia. x 1.73 m tall (stowed)

Mass 502 kg Design Life 1 year

Spacecraft in Series 1

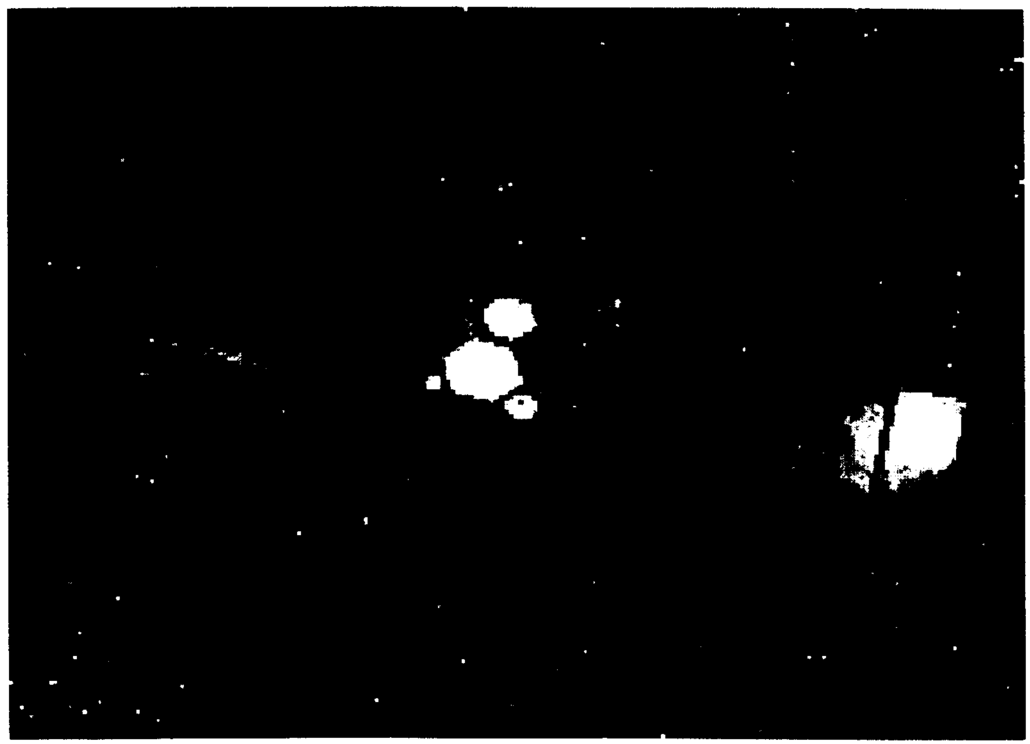
Spacecraft Description

3-Axis stabilized. Two solar arrays with 1-axis articulation. Hydrazine propulsion system used for orbit maintenance and to desaturate momentum wheels. S-Band communications link with SGLS RTS stations. MIL-STD 1553B data bus used for intra-satellite communications. Two MIL-STD 175OA computers used for spacecraft control and payload operation.

Payload Description

10 experiments to investigate technologies applicable to autonomous space operations. Microcosm Autonomous Navigation System (MANS) used two spinning sensors to provide Earth/Sun/Moon position measurements to determine satellite position and attitude. The system had a goal of 100 meters in position and 0.03 degrees in attitude. A six-channel Global Positioning System (GPS) receiver was used to calculate orbital position to within several meters. The 1553B data bus and MIL-STD 175OA computers were carried as technology demonstrations. The satellite also carried a radar sensor and two laser sensors.

Commercial Globalstar Satellite



SPACE SYSTEMS LORAL

Commercial Globalstar Satellite

The planned Globalstar system will provide global personal communication links. It consists of 48 LEO spacecraft. plus spares. To achieve low cost, reduce technological risk and accelerate deployment of the Globalstar system, Globalstar's system architecture uses small satellites incorporating a well-established repeater design that acts essentially as a simple "bent pipe," relaying signals received directly to the ground. All the system's call processing and switching operations are on the ground, where they are accessible for maintenance and can benefit from continuing technological advances. In addition, the CDMA digital technology selected by Globalstar promotes efficient use of satellite resources.

First Launch: 1997 Uses Delta, Zenit-2, or LongMarch

Orbit: LEO, 1414 km circular orbits with different inclinations

Mass: 440 kg

Design Life: 7.5 years

Spacecraft in Series: 48

Spacecraft Description:

The structure is rigid Al frame with body composed of Al honeycomb panel The 4 panel Silicon solar array generates 1.1 kW Power Sources . The system uses a single 14 cell Ni H2 battery. There is an automated attitude control system which uses 4 for 3 momentum wheels for yaw steering. A combination of GPS receivers and earth sensors provide attitude data.: The system also contains a completely automated thermal control subsystem.

Payload Description: Bent pipe transponders using the following frequencies

1610-1626.5 MHz (user-to-satellite); 2483.5-2500 MHz (satellite-to-user)

5091-5250 MHz (gateway-to-satellite); 6875-7055 MHz (satellite-to-gateway)

Commercial Iridium Satellite

SPACE SYSTEMS LORAL

Commercial Iridium Satellite

The Iridium system is a planned commercial communications network comprised of a constellation of 66 LEO spacecraft. The system will use L-Band to provide global communications services through portable handsets. A total of 125 spacecraft will be built by Lockheed for more than $700M. Commercial service are planned to begin in 1998. Total cost of the project is estimated at $3.45 billion and subscriber fees are estimated at $3 per minute. The system will employ 15-20 ground stations with a master control complex in Landsdowne, VA, a backup in Italy, and a third engineering center in Chandler, AZ.

First Launch in 1996. Launches to use Delta II, Proton, and Long March vehicles

Orbit LEO, 780 km circular, 75 deg inclination, 6 planes of 11 satellites each

Size 13 m length, 4 m width Mass 689 kg Design Life 8 years

Spacecraft Description

3-axis stabilized. Hydrazine propulsion system. Two solar panels with 1-axis articulation.

Payload Description

The system employs L-Band using FDMA/TDMA to provide voice at 4.8 kbps and data at 2400 bps with 16 dB margin. Each satellite has 48 spot beams for Earth coverage and uses Ka-Band for crosslinks and ground commanding.

Commercial ORBCOMM Satellite

SPACE SYSTEMS LORAL

Commercial ORBCOMM Satellite

Orbcomm is a commercial venture to provide global messaging services using a constellation of 26 low-Earth orbiting satellites. The planned system is designed to handle up to 5 million messages from users utilizing small, portable terminals to transmit and receive messages directly to the satellites. The first two satellites of the constellation experienced communications problems after launch, but were recovered and placed into operational status. The nominal 26 satellite constellation will be deployed by 1997, with the potential for an additional 8 satellite plane and two more polar orbiters depending on demands for increased coverage. The vehicles will be controlled from a single control center located in Dulles, Virginia. The cost per satellite has been estimated at $1.2 million. A small forerunner vehicle, Orbcomm-X, was launched in 1991 as a feasibility demonstration. This vehicle had a different design than the operational vehicles and will not be included in the operational system.

Launch F 1, 2: April 3, 1995 on Pegasus Future satellites will be launched 8 at a time on Pegasus

Orbit Nominal constellation 2 Polar (F 1, 2): 785 km circular 24 Inclined: 3 planes with 8 equidistantly spaced satellites in each plane, 780 km circular, 45 deg inclination Augmented constellation 2 more Polar + additional 8 satellite plane

Size Bus: 1.05 m diameter x 0.17 m thick

Mass 43 kg

Design Life 4 years

Spacecraft in Series 26 + 10 more if needed for additional coverage

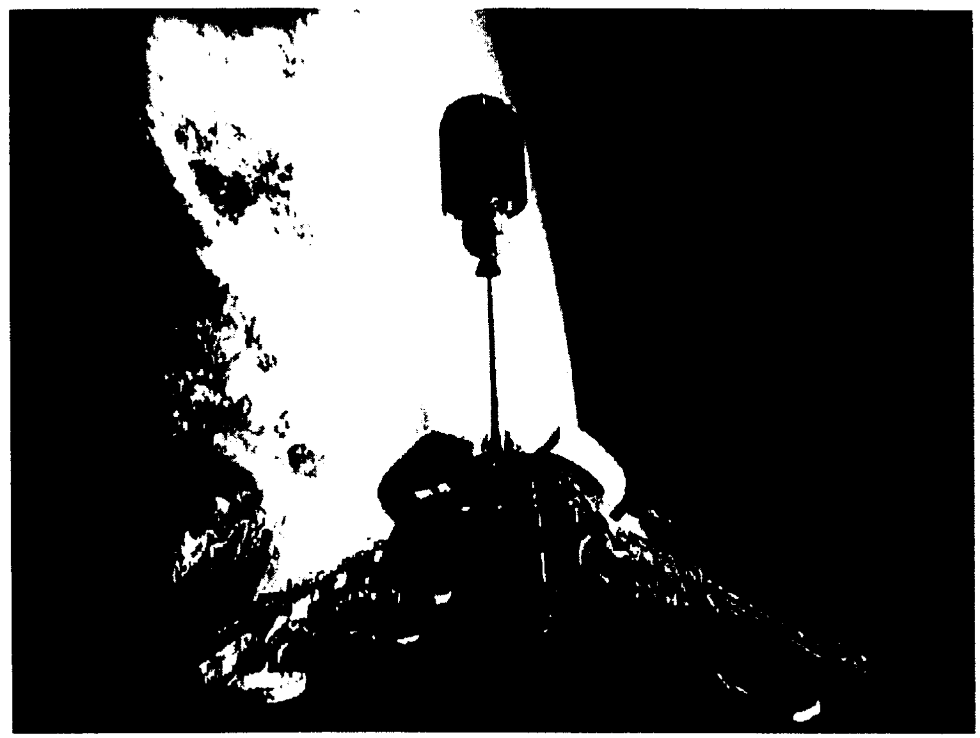
Spacecraft Description

Circular disk shaped spacecraft. Circular panels hinge from each side after launch to expose solar cells. These panels articulate in 1-axis to track the sun and provide 160W. Deployed spacecraft measures 3.6 m feet from end to end with 2.3 m span across the circular disks. VHF telemetry at 57.6 kbps. On-board GPS navigation and timing system. 14 volt power system. Gravity gradient stabilization provides 5 degrees control with magnetic torquers for damping. Cold gas (nitrogen) propulsion system.

Payload Description

Each spacecraft carries 17 data processors and seven antennas. Designed to handle 50,000 messages per hour. Long boom is a 2.6 meter VHF/UHF gateway antenna. Receive: 2400 bps at 148 - 149.9 MHz. Transmit: 4800 bps at 137 - 138 MHz and 400.05 - 400.15 MHz. The system uses X.400 (CCITT 1988) addressing. Message size is 6 to 250 bytes typical (no maximum).

Commercial Palapa B Satellite



SPACE SYSTEMS LORAL

Commercial Palapa B Satellite

Palapa B satellites were four times as powerful and twice the size of their predecessors, the Palapa A series. While the A series was designed for domestic/regional communications within Indonesia, the new system also served the Philippines, Thailand, Malaysia and Singapore.

Launch All from ETR B1: June 18, 1983 on STS-7 B2: February 4, 1984 on STS-10 B2P: March 20, 1987 on Delta 2 B2R: April 13, 1990 by Delta 2 B4: May 5, 1992 on Delta 2

Orbit All in GEO B1: 118 deg E B2P: 113 deg E B2R: 108 deg E B4: 118 deg E

Size 6.83 m high, 2.16 m diameter; Mass 628 kg; Design Life 8 years

Spacecraft in Series 4

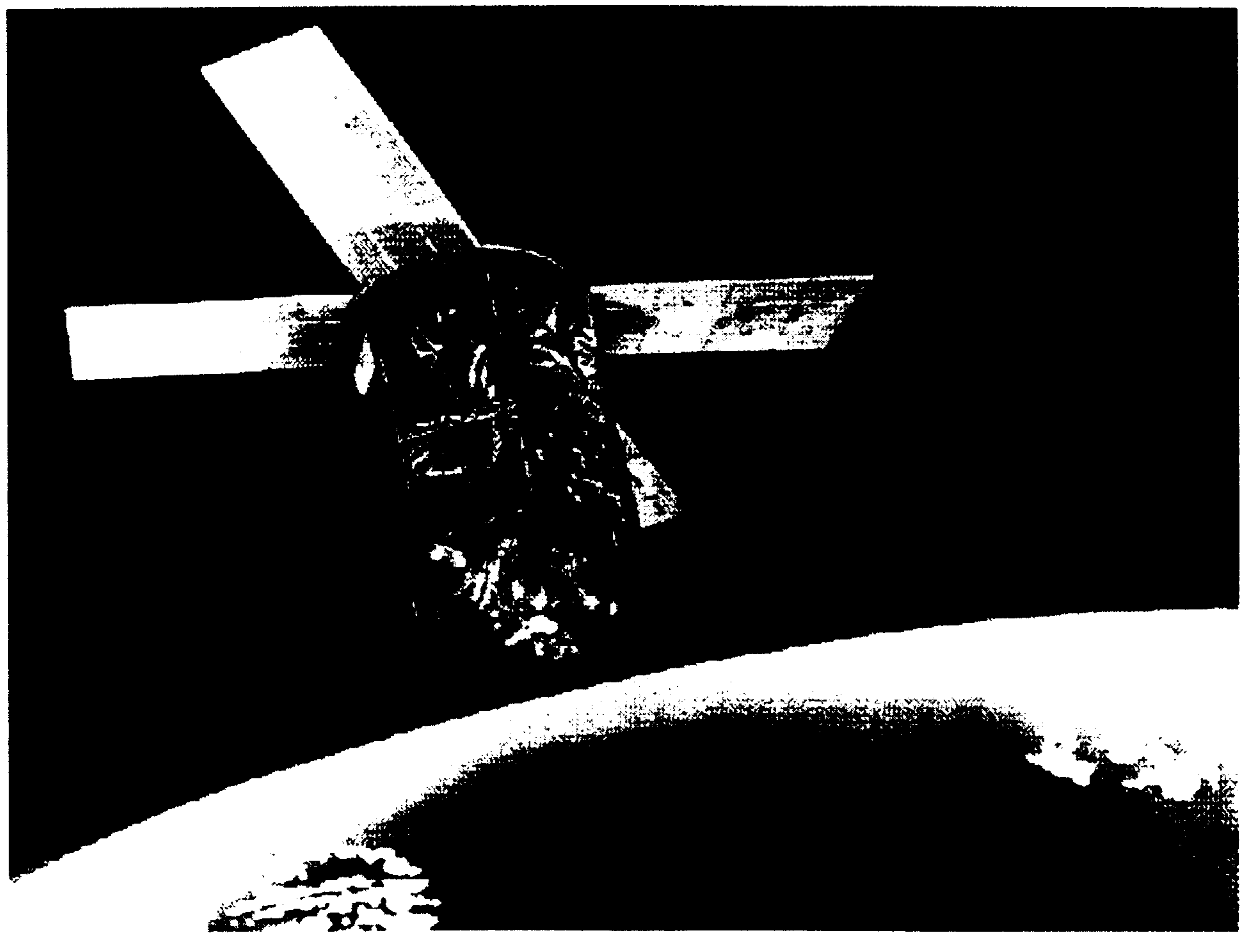
Spacecraft Description

Based on Hughes HS-376 design. Cylindrical structure. Spin stabilized. Hydrazine propulsion system for attitude control, orbit maintenance. Body mounted solar cells provide 1060 W BOL. Despun antenna platform.

Payload Description

Each carried 24 C-band transponders (+6 spares).

Commercial Seastar Satellite



SPACE SYSTEMS LORAL

Commercial Seastar Satellite

The SeaStar satellite carries the SeaWiFS instrument which is designed to monitor the color of the world's oceans. Various ocean colors indicate the presence of different types and quantities of marine phytoplankton, which play a role in the exchange of critical elements and gases between the atmosphere and oceans. The satellite will monitor subtle changes in the ocean's color to assess changes in marine phytoplankton levels, and will provide data to better understand how these changes affect the global environmental and the oceans' role in the global carbon cycle and other biogeochemical cycles. Complete coverage of the Earth's oceans will occur every two days. The NASA sponsored mission was contracted as a "data buy" from a Orbital Sciences Corporation, who will build, launch, and operate the satellite, and then sell data from the satellite to NASA. NASA will retain all rights to data for research purposes, while will OSC retain all rights for commercial and operational purposes. The mission is a follow on to the Coastal Zone Color Scanner (CZCS).

Launch Planned for 1996-97 on Pegasus XL from WTR

Orbit 705 km, circular, sun-synchronous (noon-midnight)

Mass 400 kg (incl. 85 kg payload) Design Life 5 years

Spacecraft in Series 1

Spacecraft Description

Nadir pointing. 3-axis stabilized to 0.5 deg with 0.08 deg knowledge using 2 momentum wheels, torque rods. Attitude determination via redundant sun sensors, horizon sensors, and magnetometers. Hydrazine propulsion using four 1-lbf thrusters is used for orbit raising and orbit maintenance. Nitrogen propulsion system provides stabilization during launch. Downlink using L-Band at 665.4 kbps, and S-Band at 2 Mbps. Uplink using S-Band at 19.2 kbps. Redundant GPS receivers for orbit determination. Four deployed solar panels with zenith-facing cells and two body-mounted side-facing solar panels produce 165 watts orbit-average after 5 years. 160 MBytes solid state recorder.

Payload Description

Only one instrument is carried: the Sea-viewing Wide Field-of-view Sensor (SeaWiFS). Consisting of an optical scanning telescope and an electronics module, SeaWiFS will image the Earth's oceans in 8 frequency bands.

The scanning telescope rotates at six revolutions per second in the cross-track direction to provide scan coverage with a spatial resolution of 1.13 km

Commercial Teledesic Satellite

SPACE SYSTEMS LORAL

Commercial Teledesic Satellite

The Teledesic system will provide global communication links via a constellation of 840 LEO spacecraft. The system will provide "fiber-optic like" links to customers around the world. The system will act as a network operator and will support communications ranging from high-quality voice channels to broadband channels supporting video conferencing, interactive multimedia and real-time two-way digital data flow. The system will use L-band to send and receive signals from users. Each satellite acts as a node in a large-scale packet-switching network. Service is planned to begin in 2001. Total cost of the project is estimated at $9 billion.

Launch: Planned for 1999-2000? Multiple launches per vehicle (e.g. 8). Designed to be compatible with 20+ different launchers

Orbit: LEO, 700 km circular, near polar (98.2 deg) sun synchronous, in 21 orbit planes with 40 S/C (plus 4 spares) in each plane

Mass: 700 kg Design Life: 10 years

Spacecraft in Series: 840 + 84 spare

Spacecraft Description

3-axis stabilized. Large solar panel is articulated to remain sun pointing. Designed to be compatible with over 20 different launch vehicles to permit launch option flexibility.

Payload Description

The antenna footprint for each satellite is about 700 km. Large deployed phased array antenna.

Mars Exploration Background Survey

- Synopsis of Programs
  - Discovery
  - Surveyor
  - New Millennium
- Overview of Missions to Mars

Mars Exploration Background Survey

The purpose of this overview of the existing and planned NASA Mars exploration programs is to provide the background for the study of advanced technologies for the Mars Relay Satellite.

This presentation will provide a brief synopsis of the three programs which are directly relevant to applying new technologies to Mars exploration;

Discovery Program

Mars Surveyor Program

New Millennium Program

We will then present an overview of the current and planned missions to Mars as currently understood.

Discovery Program Goals

- Perform high-quality scientific investigations
- Pursue innovative ways of doing business
- Encourage the use and transfer of new technologies
- Enhance general public awareness of, and appreciation for, solar system exploration

Discovery Program Goals

The general goal of the Discovery Program is the support of the Solar System Exploration Division's program of planetary science. It has four specific goals.

By conducting a series of focused, cost-effective missions to answer critical questions in planetary science it will help maintain the excellence of the United States' solar system exploration program. By launching Discovery missions on a frequent, regular basis a mechanism will be provided by which the most pressing questions in planetary science may be addressed in a timely fashion.

The innovative ways of doing business involve encouragement teaming arrangements among industry, university, and Government partners. Competitively selected teams will have responsibility and authority to accomplish the mission with limited NASA oversight limited.

The Discovery Program will identify and support development and application of candidate new technologies for its missions and will encourage the transfer of those technologies to the broader scientific community and other markets. The teaming of industry, university, and Government is meant to foster an environment conducive to technology transfer occurring in parallel with technology development.

The will identify appropriate techniques to capture and hold public interest in its and all of NASA's exploration programs

SPACE SYSTEMS LORAL

Discovery Program Missions

- NEAR: rendezvous with an asteroid
- Pathfinder - place a rover on Mars
- Lunar Prospector - detailed maps of lunar composition and gravity
- Stardust - sample cometary dust

Discovery Program Missions

NEAR

Launched on Feb. 17,1996. - description given in previous presentation on small satellites

Pathfinder

Launched Dec. 1996; will be described later in the presentation

Lunar Prospector

To be launched in October of 1997; description given in previous presentation on small satellites

Stardust

To be launched in Jan. 1999. It will take samples of a comet's tail and return them to earth. The S/C is built of light weight composites, uses single axis gimbals, and body mounted science experiments. Virtually all the components will have been flown and/or flight tested on other missions.

Surveyor Program

- Objective
  - Robotic survey of Mars
- Missions
  - Mars Global Surveyor
  - Mars Surveyor '98 Orbiter
  - Mars Surveyor '98 Lander
  - Mars '01 Orbiter , MARS '01 Lander
  - Mars '03 Orbiter , MARS '03 Lander

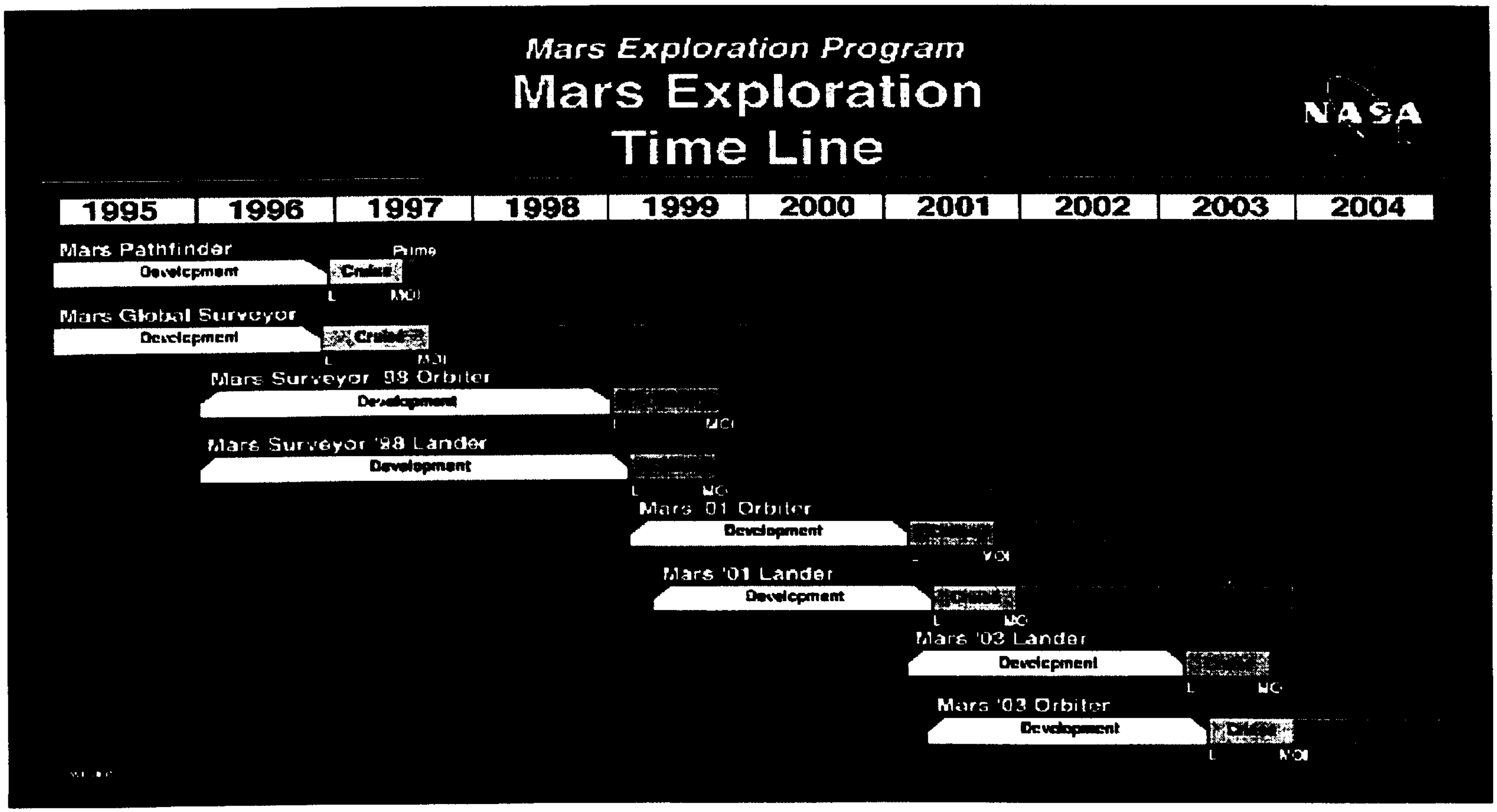
SPACE SYSTEMS LORAL

Surveyor Program

1996's Mars Global Surveyor is the first of a new, decade-long program of robotic exploration of Mars, - the Mars Surveyor Program. This will be an aggressive series of orbiters and landers to be launched every 26 months, as Mars moves into alignment with Earth.

The program will cost about $100 million per year. International participation, collaboration and coordination will be encouraged in all missions of the program. The Mars Surveyor landers planned in future years -- 1998, 2001, 2003 and 2005 -- will capitalize on the experience of the Mars Pathfinder lander. Small orbiters launched in the 1998 and 2003 opportunities will draw on the experience of Mars Global Surveyor and carry other scientific instruments into orbit to complete Mars Global Surveyor's science mission.

Mars Exploration



SPACE SYSTEMS LORAL

Mars Exploration

This is the proposed time line for the Mars missions. The launches are scheduled to exploit the orbital alignments of Mars and Earth.

Since the program will have launches spread out over a number of years it should be able to take some technological risks. Further the planned muti-year launches should allow for response to changing scientific and technical needs.

New Millennium Program

- Objectives
  - Developing and validating revolutionary technologies
  - Reducing development times and lifecycle costs
  - Evaluating highly capable and agile spacecraft
  - Promoting nationwide partnering and coordination
- Missions
  - DS-1
  - Martian Microprobes

New Millennium Program

The New Millennium Program is intended to demonstrate the technologies that will become the mainstays of deep space exploration for the first part of the 21st century. Themajor program goals are:

- Developing and validating revolutionary technologies
- Reducing development times and lifecycle costs
- Evaluating highly capable and agile spacecraft
- Promoting nationwide partnering and coordination

The first of the New Millennium missions will be DS-1 which will demonstrate several new technologies. This was described in the small satellite survey part of the presentation.

The second New Millennium mission is to place two micro probes aboard the 1998 Mars Surveyor Lander. The microprobes will have mass of about 2kg. They will be mounted on the spacecraft's cruise ring and will reach the surface in a single aero shell. They will separate into two parts the first part burying itself in the soil the second remaining near the surface. The two parts will be joined by a cable and the upper part will communicate with the Deep Space Network.

Technologies proposed for demonstration include a light weight, single-stage entry aeroshell, a miniature, programmable telecommunication subsystem, power microelectronics with mixed digital/analog integrated circuits, an ultra low-temperature lithium battery, a microcontroller and flexible interconnects for system cabling.

Mars Exploration

Mars Missions Overview

Mars Exploration

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Mars Global Surveyor (MGS)

- Purpose
- Spacecraft
- Instrument package
- Mission description

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MGS Purpose

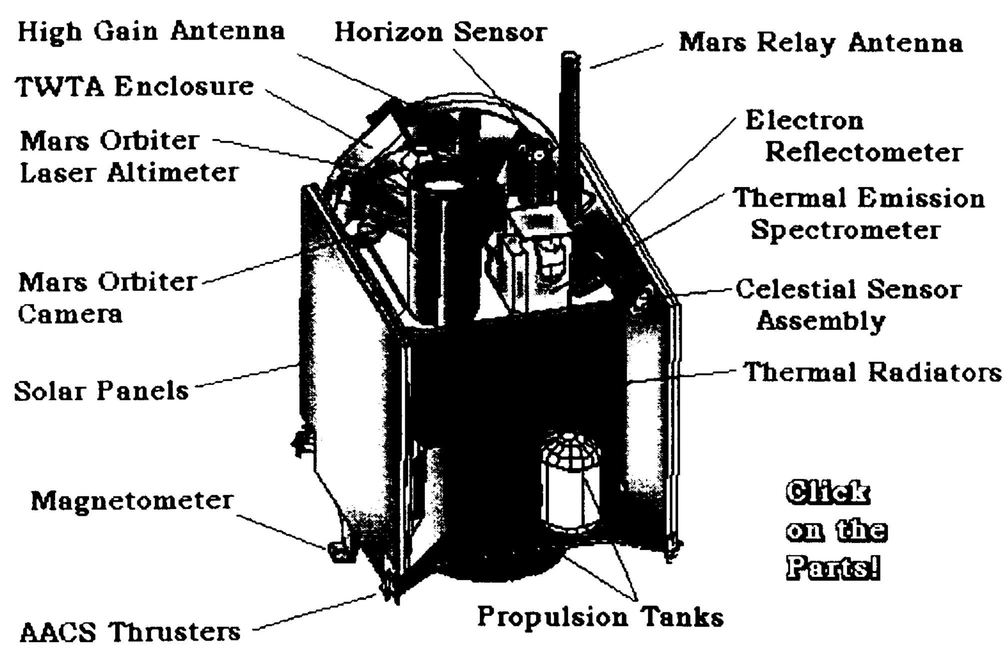
- Low cost replacement for Mars Observer
  - High resolution images of martian surface
  - Gravity studies
  - Surface and atmospheric composition studies
  - Evaluation of Martian magnetic field
- Provide data relay

MGS Purpose

The Mars Global Surveyor is designed to orbit Mars over a two year period and collect data on the surface morphology, topography, composition, gravity, atmospheric dynamics, and magnetic field. This data will be used to investigate the surface processes, geology, distribution of material, internal properties, evolution of the magnetic field, and the weather and climate of Mars.

At the conclusion of the two year period, the orbiter will act as a relay until January of 2003 in support of the other missions of the Mars Surveyor program.

MGS Spacecraft - Launch Configuration



Click on the Parts!

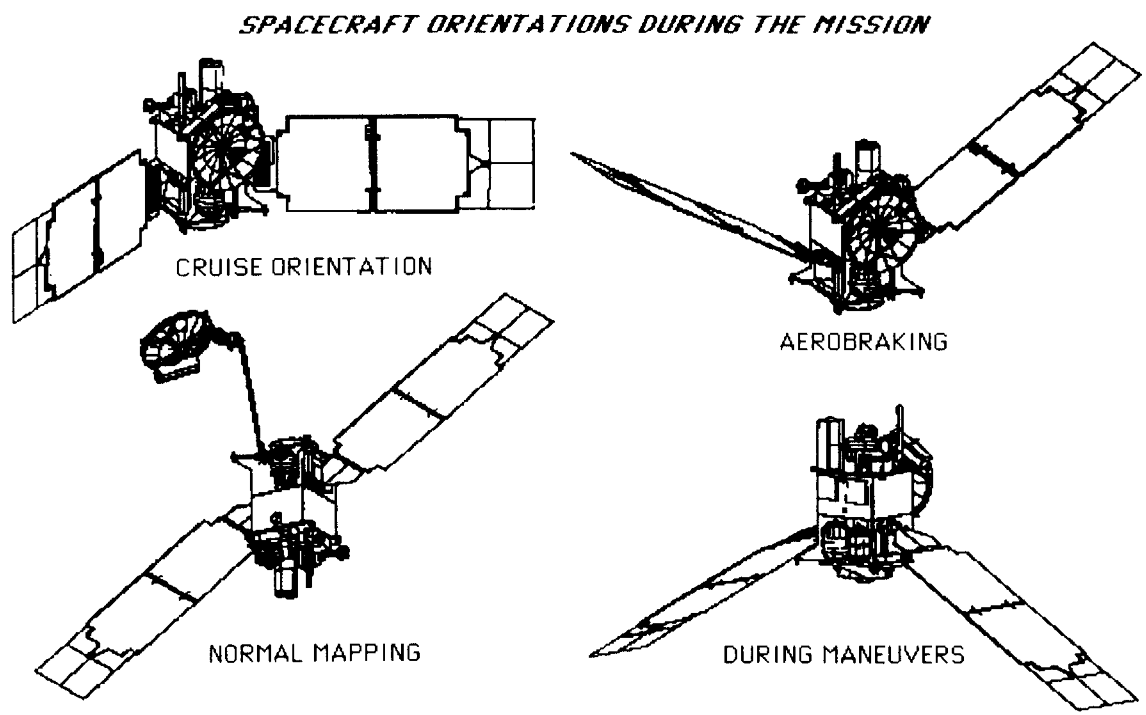
LAUNCH CONFIGURATION

SPACE SYSTEMS LORAL

MGS Spacecraft - Launch Configuration

The spacecraft itself is a rectangular box approximately 1.17 x 1.17 x 1.7 meters in size, made up of two parts, the equipment module and the propulsion module. All instruments except the magnetometer are stored on the nadir equipment deck, on one of the 1.17 x 1.17 meter surfaces. This is the top of the equipment module, which is 0.735 m high. The main thruster and propulsion tanks are on the opposite side from the instruments, on the propulsion module, which is approximately 1 meter high. Two solar panels, each 6 square meters in area, extend out from opposite sides of the craft. A 1.5 meter diameter parabolic high gain dish antenna is mounted on an adjacent side, and attached to a 2 meter boom, which is extended for mapping operations so the antenna is held away from the body of the spacecraft

MGS Spacecraft - Flight Configuration



SPACE SYSTEMS LORAL

MGS Spacecraft - Flight Configuration

This slide shows the different orientations that the spacecraft will assume during the mission.

The spacecraft will be three-axis stabilized with no scan platform. The main 596 N thruster will use hydrazine and N_2O_4 propellant. Control will be through 12 4.45 N hydrazine thrusters, mounted in four groups of three (two aft facing and one roll control thruster). The initial propellant load will be 216.5 kg of hydrazine and 144 kg of N_2O_4. Four solar array panels (2 GaAs, 2 SI) will provide 667 W of power to the Global Surveyor at aphelion. Energy will be stored in two 20 Amp-hr nickel hydrogen batteries, and supplied at 28 V DC. Temperature control will be primarily passive with multilayer insulation, thermal radiators, and louvers, augmented by electrical heaters. Communications will be achieved via the deep space network using the high gain antenna and two low gain antennas, one mounted on the high gain antenna and one on the equipment module. Uplink will be in the X-band, downlink in the X and Ka bands. Minimum downlink rate will be 21.33 kbps, 2 kbps engineering data downlink, and 10 bps emergency downlink.

MGS - Instruments

- Mars Orbital Camera
- Thermal Emission Spectrometer
- Mars Orbital Laser Altimeter
- Radio Science Investigations
- Magnetic Field Investigations

MGS - Instruments

The Mars Orbital Camera will take high resolution images, on the order of a meter or so, of surface features. It will also take lower resolution images of the entire planet over time to enable research into the temporal changes in the atmosphere and on the surface.

The Thermal Emission Spectrometer is a Michelson interferometer that will measure the infrared spectrum of energy emitted by a target. This information will be used to study the composition of rock, soil, ice, atmospheric dust, and clouds.

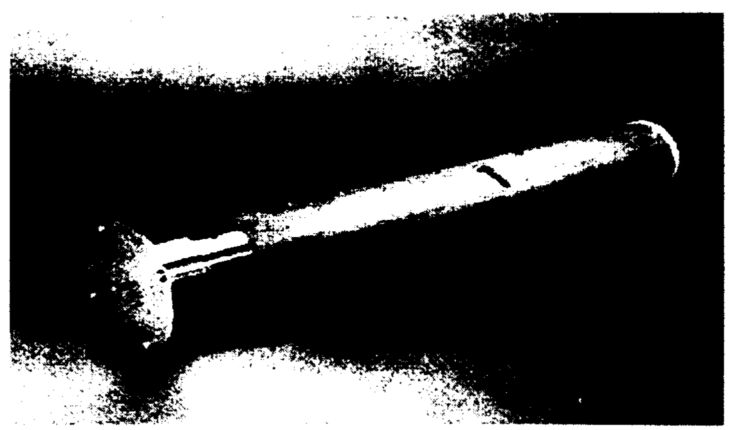
Mars Orbital Laser Altimeter will measure the time it takes for a transmitted laser beam to reach the surface, reflect, and return. This time will give the distance, and hence the height of the surface. Combining these measurements will result in a topographic map of Mars

Radio Science Investigations: Measurements of the Doppler shift of radio signals sent back to Earth will allow precise determination of changes in the orbit, which will allow for a model of the Mars gravity field. As the spacecraft passes over the poles on each orbit, radio signals pass through the Martian atmosphere on their way to Earth. The way in which the atmosphere affects these signals allows determination of its physical properties

Magnetic Fields Investigation (A magnetometer will be used to determine whether Mars has a magnetic field, and the strength and orientation of the field if one exists. An electron reflectometer will measure remnant crustal magnetization.

MGS - Mars Relay Instrument

- Mass: 10.50 kg
- Average Power: 12.50 W
- Average bit rate: 128 kbps



SPACE SYSTEMS LORAL

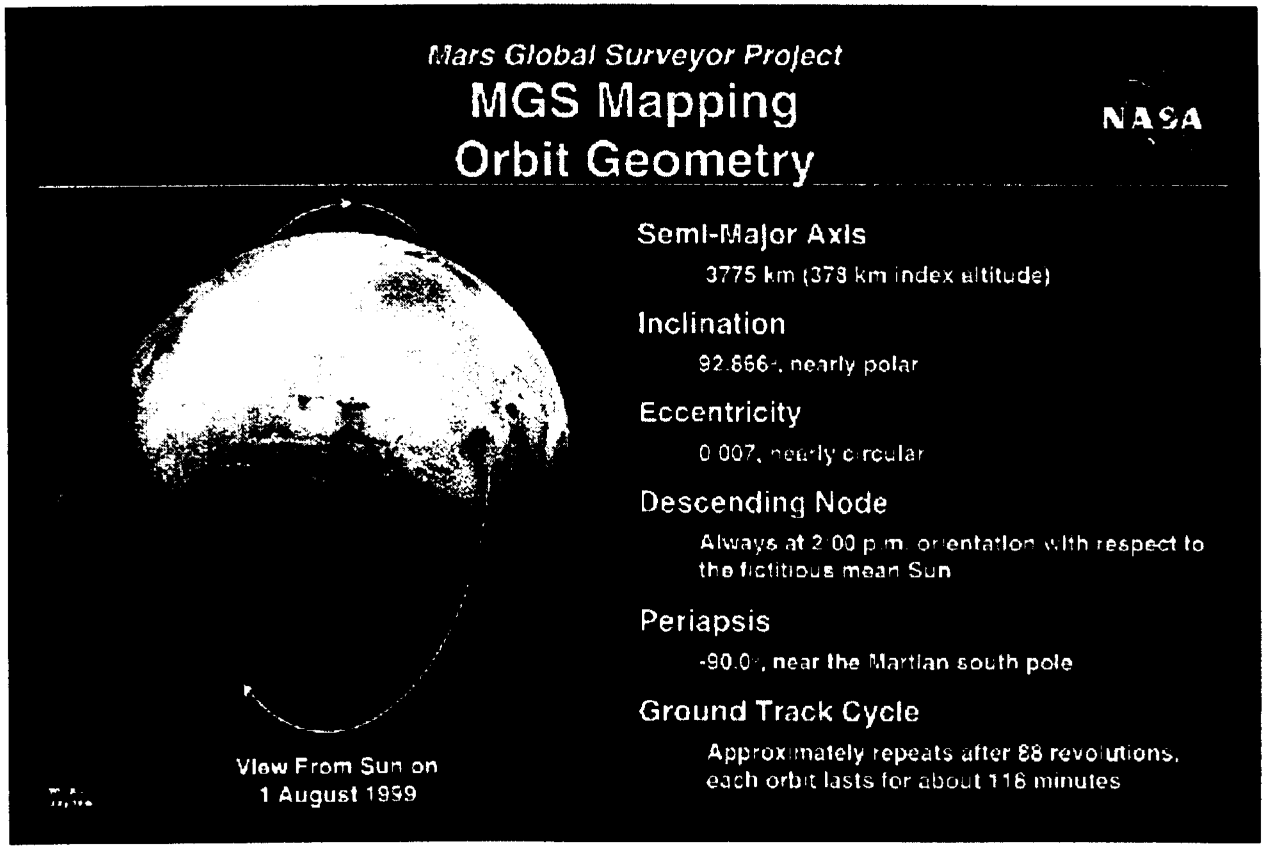
MGS - Mars Relay Instrument

The Mars Global Surveyor Mars Relay is a communications link between the Earth or other orbiting spacecraft and various Mars surface lander and rover missions being planned by the U.S., Europeans, and Russians. It will allow high speed reliable communications to and from these surface instruments, without requiring them to have the capability of transmitting directly to Earth.

The Mars Relay consists of a 86 cm high, 2.5 kg quadrifilar helix fiberglass antenna mast mounted on the spacecraft and an electronics box and coaxial cable. The MR can communicate with three different landers in the same beamwidth without jamming. The MR will be commanded by uplinked sequence commands (to the orbiter) to select one lander or two alternating landers. The system receives data from the surface missions and routes them for storage to the large on-board storage memory buffer of the Mars Observer Camera (MOC). The stored transmissions are then routed to Earth via the MOC.

The transmission frequency to the Mars ground stations is 437.1 MHz at 1.3 W. The receive frequencies (from the stations) are 401.5 and 405.6 MHz. The antenna field of view ranges from limb to limb. The receive data rate is selected on the orbiter by sequenced commands. Permissible data rates of 8 and 128 kb/s before convolutional encoding are 8003 or 128038 b/s if 401.5275 MHz is used, and 8085 b/s or 129345 b/s if 405.6250 Mhz is used.

MGS Mission Description



SPACE SYSTEMS LORAL

MGS Mission Description

After launch and a 10 month cruise phase, the Mars Global Surveyor will be inserted into an elliptical capture orbit in September, 1997. Over the next four months, aerobraking maneuvers and thrusters will be used to change the orbit to the final mapping orbit, a 117.65 minute polar orbit with an index altitude of 378 km, an inclination of 92.9 degrees, and an eccentricity of 0.0072. The orbit will be sun-sychronous, so data will be collected with the sun at the same mid-afternoon azimuth for each pass. The spacecraft will collect data from this mapping orbit over the course of one martian year, from January, 1998 until December, 1999. After this time, the orbiter will act as a relay until January of 2003 in support of the other missions.

Mars Pathfinder

- Purpose
- Spacecraft/Lander
- MicroRover
- Instruments
- Mission

Mars Pathfinder

We shall briefly discuss the objectives of the Mars Pathfinder mission, then provide a description of the spacecraft, micro rover, and scientific instruments.

We will conclude with a mission description.

Mars Pathfinder - Purpose

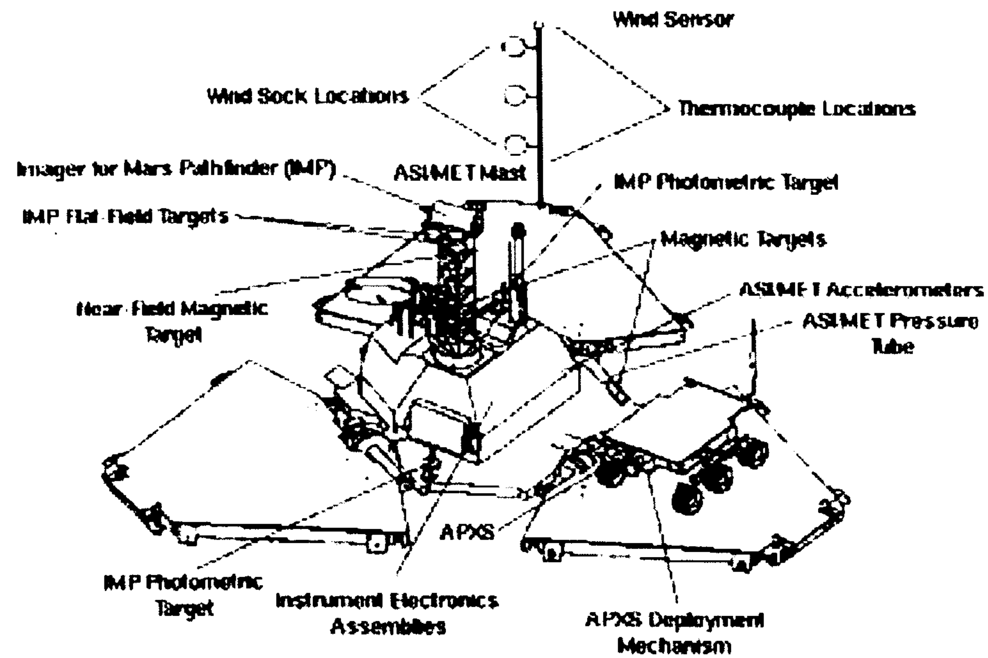
- Demonstrate feasibility of free ranging rover
- Investigate structure of Martian atmosphere
- Investigate composition of Martian rocks
- Investigate Martian surface geology

Mars Pathfinder - Purpose

It will land a single vehicle with a microrover (Sojourner) and several instruments on the surface of Mars in 1997. Sojourner's mobility provides the capability of "ground truthing" a landing area over hundreds of square meters on Mars. Pathfinder will be investigating the surface of Mars with three additional science instruments

These instruments will allow investigations of the geology and surface morphology at sub-meter to a hundred meters scale, the geochemistry and petrology of soils and rocks, the magnetic and mechanical properties of the soil as well as the magnetic properties of the dust, a variety of atmospheric investigations and rotational and orbital dynamics of Mars. Landing downstream from the mouth of a giant catastrophic outflow channel offers the potential for identifying and analyzing a wide variety of crustal materials, from the ancient heavily cratered terrain to intermediate-aged ridged plains to reworked channel deposits. Examination of the different surface materials will allow first-order scientific investigations of the early differentiation and evolution of the crust, the development of weathering products and the early environments and conditions that have existed on Mars.

Mars Pathfinder - Spacecraft/Lander



SPACE SYSTEMS LORAL

Mars Pathfinder - Spacecraft/Lander

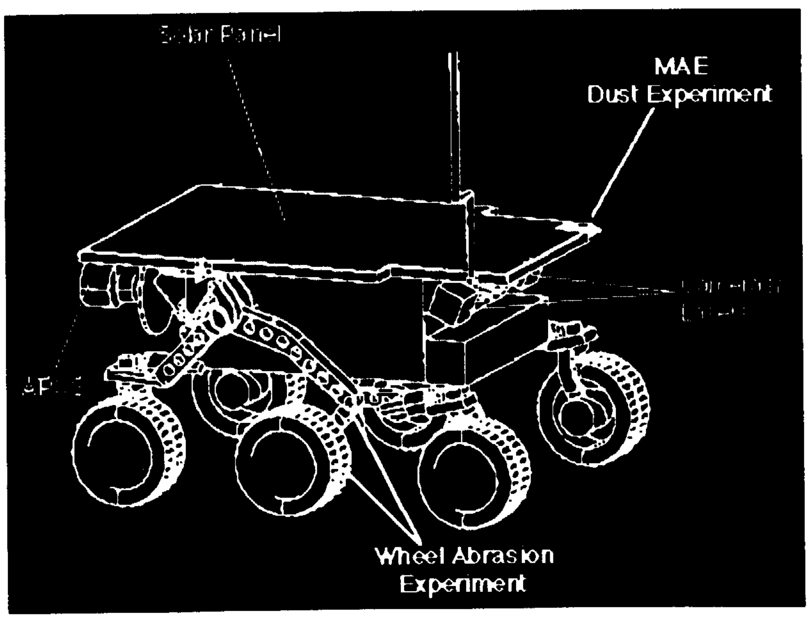
The Pathfinder spacecraft resembles a tetrahedron, a small pyramid standing about 0.9 meters (3 feet) tall with three triangular-shaped sides and a base. At launch the spacecraft will weigh about 890 kilograms including its cruise stage, aeroshell and backshell, solar panels, propulsion stage, medium- and- high-gain antennas and 100 kilograms of cruise propellant. The cruise stage measures 2.65 meters in diameter and stands 1.5 meters tall.

The entry configuration is essentially the spacecraft minus the cruise stage and propellant. When Pathfinder is configured for entry into the Martian atmosphere, it main components will be the aeroshell, folded lander and rover, parachute, airbag system and small rocket engines. Combined, the spacecraft will weigh approximately 570 kilograms at entry.

Command And Data Handling is performed by an Integrated Attitude And Information Management System . The lander has a R6000 Computer with VME Bus it performs 22 Millions Of Instructions Per Second It has a 128 Mbyte mass memory The system is solar powered during cruise stage and lander stag.

Surface Operations Telemetry Rate Via the High Gain X-band Antenna is 6 Kb/s to the 70-m Deep Space Network Surface Operations Command Rate Via the HGA X-Band is 250 b/s

Mars Pathfinder - MicroRover



Mars Pathfinder - MicroRover

Total Mass16 KG

Mobile Mass 11.5 Kg including

Lander-Mounted Rover Equipment Mass:4.5 Kg including ultra-high frequency (UHF) modem and support structure.

Autonomous Navigation: On board, using laser striping for obstacle detection

Mobility System:Six-wheel, rocker bogie suspension

Command And Telemetry:UHF link with lander

Payload ; aft and fore cameras, APXS, APXS deployment mechanism

Solar panels, peak power 16 W-hours: primary battery, 50 W-hours

Thermal Control:Three radioisotope header units (RHUs)

Computer Characteristics<:80C85, 0.1 MIPS; 0.5 Mbyte RAM mass storage; 0.5 Kg, 1.5 W

Surface Ops Time:10 A.M. to 2 P.M. each martian day

Mars Pathfinder - Instruments

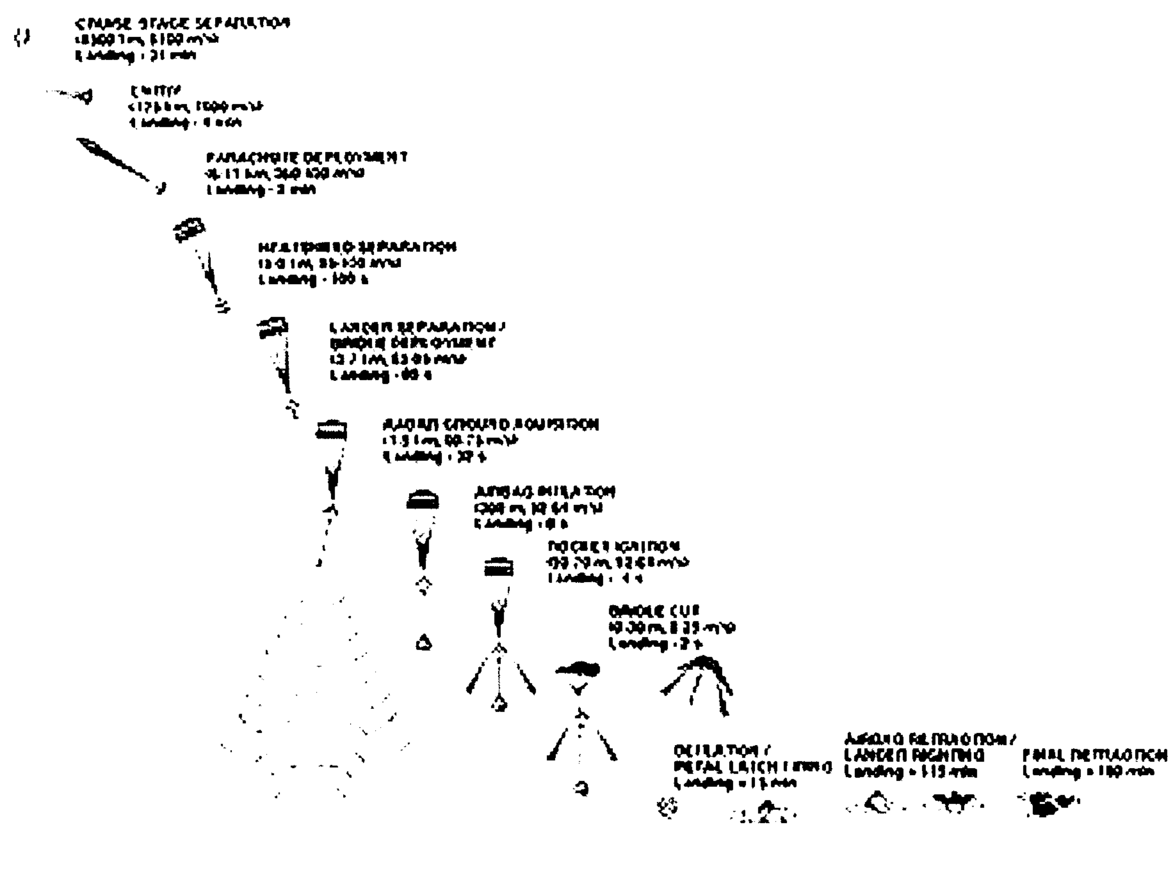
- Imager for Pathfinder
- Alpha/Proton X-ray spectrometer
- Atmospheric structure instrument/meteorology package

Mars Pathfinder - Instruments

Imager For Mars Pathfinder will reveal martian geologic processes and surface-atmosphere interactions similar to what was observed at the Viking landing sites. Observations of the general landscape, surface slopes and thedistribution of rocks will be obtained by panoramic stereo images at various times of the day. Any changes in the scene over the lifetime of the mission might be attributed to the actions of frost, dust or sand deposition, erosion or other surface-atmosphere interactions.

Alpha-Proton X-Ray Spectrometer (and the visible to near-infrared spectral filters on the imp will determine the dominant elements that make up the rocks and other surface materials of the landing site. A better understanding of these materials will address questions concerning the composition of the martian crust, as well as secondary weathering products (such as different types of soils). Atmospheric Structure Instrument/Meteorology experiment will be able to determine the temperature and density of the atmosphere during Entry, Descent and Landing (EDL). In addition, three-axis accelerometers will be used to measure atmospheric pressure during this period. Once on the surface, meteorological measurements such as pressure, temperature, wind speed and atmospheric opacity will be obtained on a daily basis. Thermocouples mounted on a meter-high mast will examine temperature profiles with height. wind direction and speed will be measured by a wind sensor mounted at the top of the mast, as well as three wind socks interspersed at different heights on the mast.

Mars Pathfinder - Mission



Mars Pathfinder - Mission

After launch, the spacecraft requires 6 to 7 months to reach Mars, depending upon the exact launch date. During this phase, a series of, in order to fine tune the flight path. Tracking, telemetry, and commandoperations with the spacecraft are conducted using the giant dish antennas of the NASA/JPL Deep Space Network

Upon arrival at Mars on July 4, 1997, the spacecraft will enter the Martian atmosphere, and then deploy the parachute, rocket braking system, and air bag system for a soft, upright landing. At this point the primary data-taking phase begins, and continues for 30 Martian days or "sols" (24.6 hours).

Mars ‘98 - Orbiter

SPACE SYSTEMS LORAL

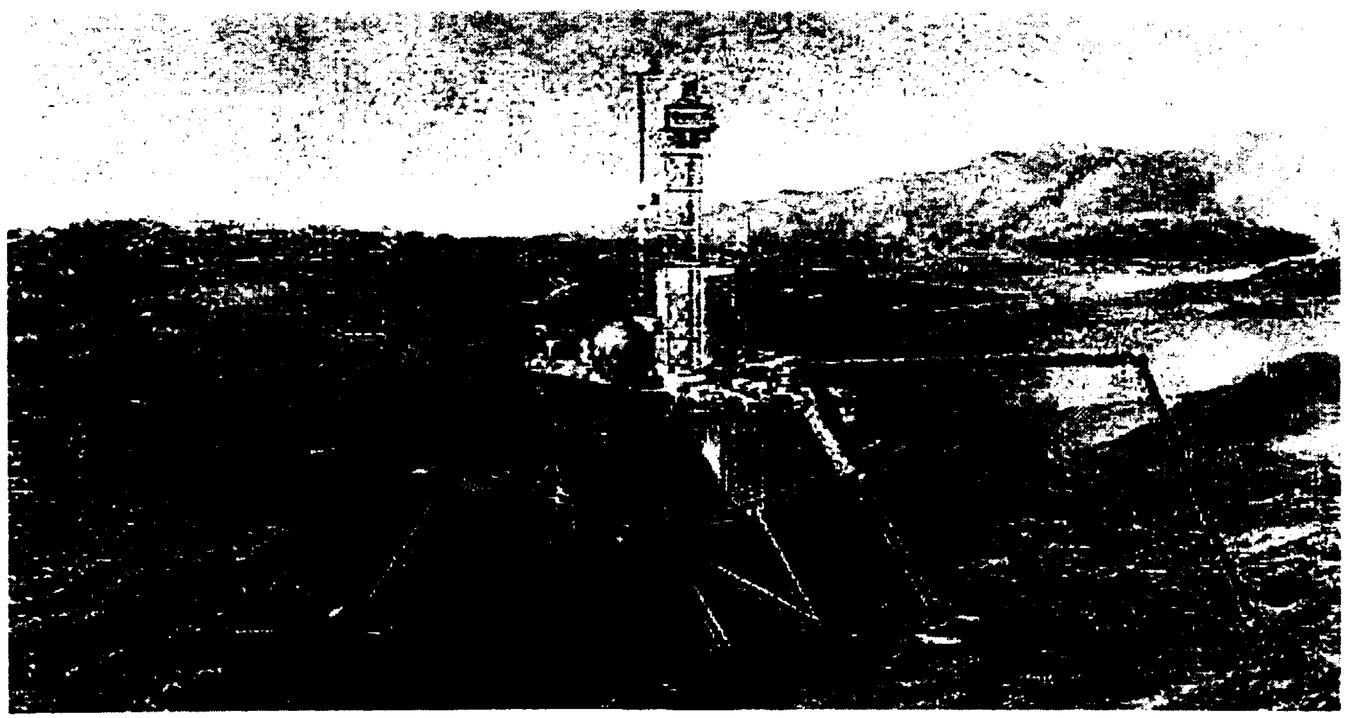
Mars ‘98 - Orbiter

The Orbiter has as its primary science objectives to: 1) monitor the daily

weather and atmospheric conditions; 2) record changes on the martian surface due to wind and other atmospheric effects, 3) determine temperature profiles of the atmosphere; and 4) monitor the water vapor and dust content of the atmosphere. The orbiter will use two instruments to carry out these investigations. The Mars Surveyor '98 Orbiter Color Imager (MARCI) will acquire daily atmospheric weather images and high resolution surface images and the Pressure Modulator Infrared Radiometer (PMIRR) will allow measurement of the atmospheric temperature, water vapor abundance, and dust concentration. The orbiter will also serve as a data relay satellite for the Mars '98 Lander and other future NASA and international lander missions to Mars.

The 1998 orbiter will be just one-half the weight of Mars Global Surveyor

Mars '98- Lander



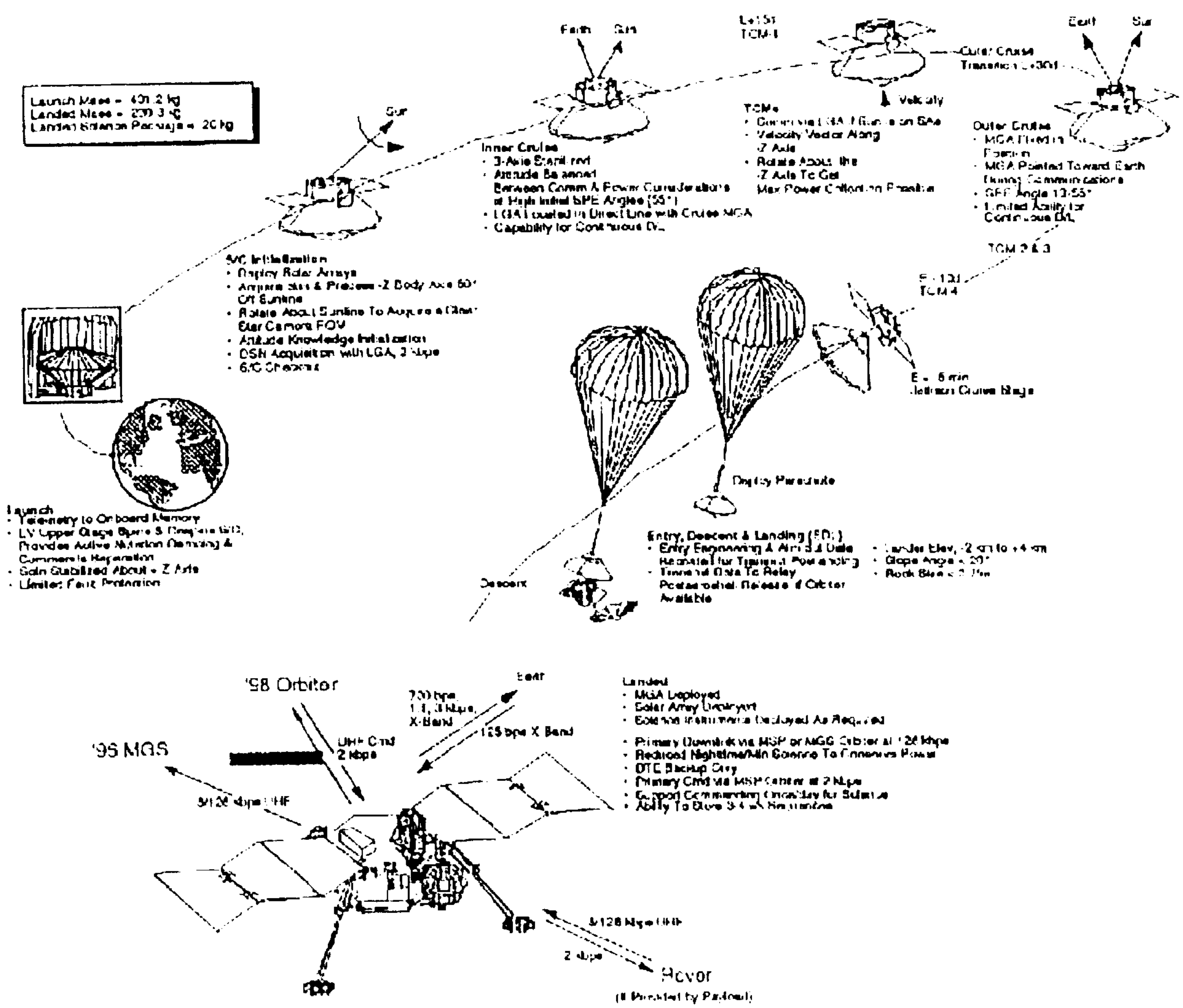
SPACE SYSTEMS LORAL

Mars '98- Lander

The Lander has as its primary science objectives to: 1) record local meteorological conditions near the martian south pole, 2) analyze samples of the polar deposits for volatiles, particularly water and carbon dioxide, 3) dig trenches and image the interior, 4) image the regional and immediate landing site surroundings. These goals will be accomplished using a number of scientific instruments, including a Mars Volatiles and Climate Surveyor (MVACS) instrument package which is comprised of a robotic arm and attached camera, mast-mounted surface stereo imager and meteorology package, and a gas analyzer. In addition, a Descent Imager will capture regional views from parachute deployment at about 8 km altitude down to the landing. The Russian Space Agency will also provide a laser ranger (LIDAR) package for the lander, which will be used to measure dust and haze in the Martian atmosphere.

The lander consists of a base with three landing legs and an exposed upper deck to hold the instruments. Two solar power arrays extend from the deck parallel to the ground on opposite sides. The imaging and meteorology masts extend upward from the deck. A 2 meter long robotic arm is attached to one side of the deck, and a small parabolic dish antenna is on the opposite side. A thermally regulated payload electronics enclosure houses the temperature sensitive components.

Mars Exploration



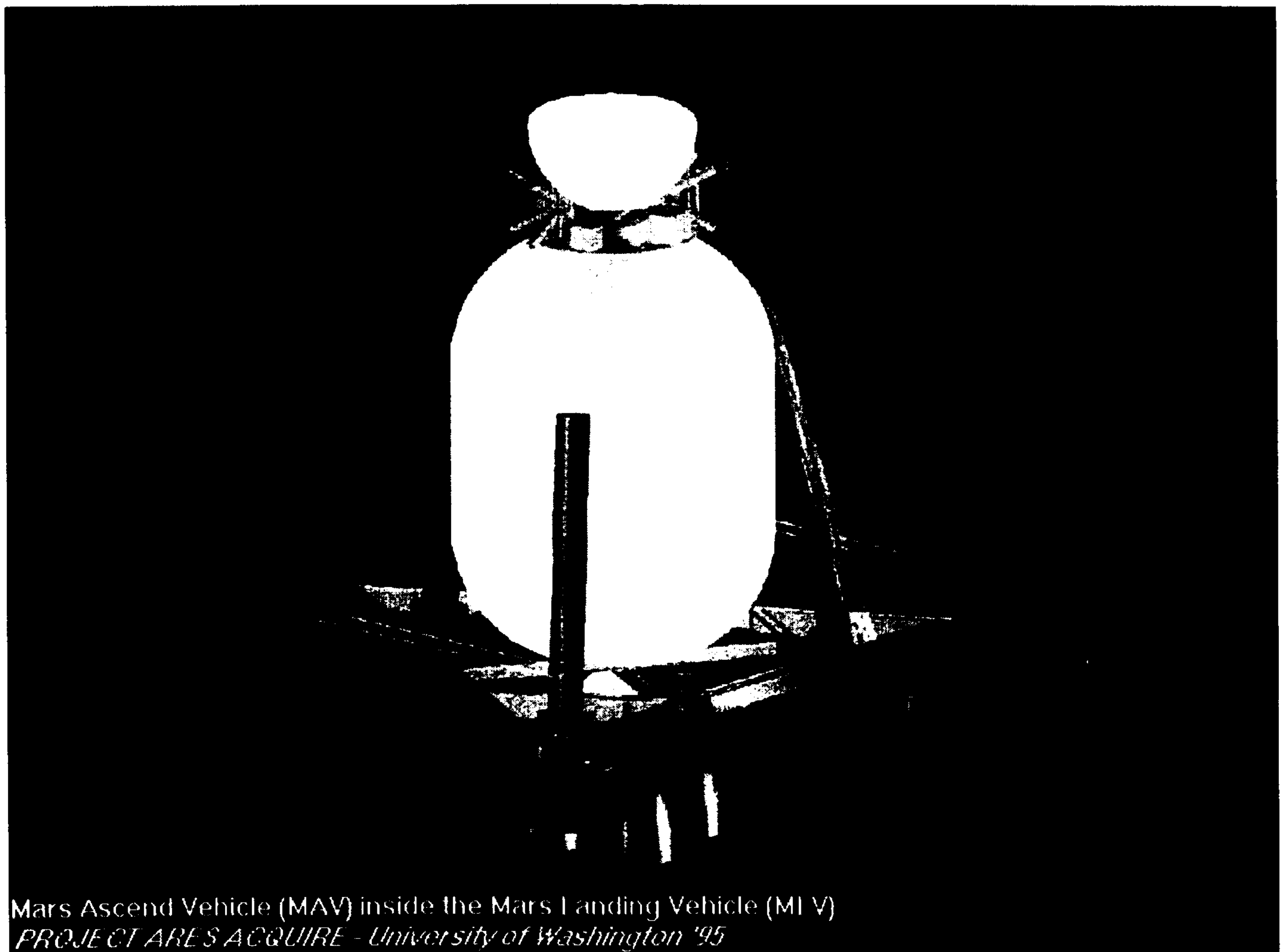
Spacecraft Activities by Mission Phase

SPACESYS LORAL

Mars Exploration

After an 11 month cruise, the lander will make a direct entry into Mars' atmosphere in December, 1999. After using an active descent propulsion system and deploying a parachute at approximately 8 km altitude it will soft-land at about 71 S latitude, 210 W longitude. This spot represents the most northerly extent of the south polar layered deposits. These deposits appear to be made of alternating layers of clean and dust-laden ice, and may represent a long-term record of the climate, as well as an important volatile reservoir. The lander will touch down during the late southern spring season and begin its primary mission which is scheduled to last until 28 February 2000. The secondary mission will go to 31 May 2000. There are also plans for a later spring mission from 01 July 2001 to 30 September 2001.

Mars Sample Return



SPACESYSTEMS LORAL

Mars Sample Return

Project Ares Acquire is an unmanned Mars sample return mission that utilizes propellant manufactured in situ from the Martian atmosphere for the return trip. In order to meet the 200 million dollar budgetary constraint for this mission, Project Ares Acquire will rely primarily on existing or near term technology and hardware for the construction of its components

Ares Acquire launches during the March 2001 launch opportunity on top of a Delta II 7925 launch vehicle into low Earth orbit (LEO). The spacecraft consists of a Mars Landing Vehicle (MLV) with a Mars Transfer Cruise Stage (MTCS), a Mars Ascent Vehicle (MAV), and a Sample Return Capsule (SRC) with an Earth Transfer Cruise Stage (ETCS).

The sample acquisition portion of the surface operations will be completed within ten days, after landing on Mars and the sample acquisition device (planetary rover) will begin an extended autonomous mission. Once all sample cylinders are stowed in the sample containment canister and the containment canister is locked in the SRC, the propellant production from the Martian atmosphere is started to ensure enough propellant is produced during the rest of the approximately 570 day stay on Mars to give the MAV a DV of 6100 m/s (6300 m/s for the TERP MAV) for the return trip to Earth.

Background

- Mars Global Surveyor (MGS) Program
- Relation between MGS and MRS

Background

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Background - Mars Global Surveyor

- Life on Mars resulted in Mars Global Surveyor Planning facing budget reduction
- JPL tasked to facilitate "sample return" from Mars from the Life on Mars potential
- Sample return expected to cost even higher than the original MGS budget
- Three internal discussion program plans, in cooperation with the Russian Mars Robots, were prepared: regular, accelerated and aggressive
- Impact on Sample Return in view of the recent Russian Mars Exploration satellite failure

Background - Mars Global Surveyor

The possibility of life on Mars apparently has pushed NASA planners to direct their attention to returning some samples from Mars in shortened time frame and resulted in a lot of funding uncertainty on the Mars Global Surveyor Program.

On the one hand, they are directed to reduce the budget that, according to Mars Program Office at JPL, was insufficient in the first place, and to submit a plan for an early sample return. On the other hand, the budget necessary for sample return is expected to be higher than even the pre-reduction MGS program. Additionally, the emphasis on sample return has also focused on an "international" aspect in that the program is to cooperate with the Russian Mars Robots.

With the recent failure of the Russian Mars effort and the resultant loss of the Mars Robots, its impact on the future of the program remains to be sorted out.

Background - MRS in MGS

- MGS always intended to carry its own relay package
- Mars Relay Satellite (MRS) was never part of the MGS program plan despite program office awareness of MRS
- Mars Program Office not involved with any funding of the MRS effort - though MRS technology, if required, is assumed
- Only in the Sample Return "planning papers" was the MRS explicitly included
- As of October, 1996, the "planning papers" are still in internal discussion stage

Background - MRS in MGS

While the MRS was designed to support Mars exploration in general, the MGS program plans have never explicitly include an MRS concept. Instead, they always expect to carry their own relay package. Hence, even as MGS planners are aware of the MRS effort, there was apparently not much coordination between the two.

The possibility of "life on Mars", on the other hand, changed that in two respects, one positive and the other negative. The positive is that with a push toward sample return, funding is reduced from the MGS so that JPL's replanning effort did include MRS as a means to provide data relay. The negative is that "sample return" plans calls for even more funding than the original MGS budget and JPL is by no means certain that it will be funded.

In any event, as of October 1996, the sample return project plans are simply preliminary, committed on paper for the purpose of generating internal (to JPL) discussion only.

Mars Exploration Status

- Four opportunities beyond the 96 launch of Global Surveyor
  - Mars Surveyor 98 Orbiter and two “piggy back” payloads for DS-2
  - 2001 Orbiter and Highland Lander
  - 2003 Lander(s) Network and Orbiter
  - 2005 Lander(s) (possibly precursor to Sample Return) and Orbiter
- 2003 and 2005 opportunities dependent on some form of cooperation with the Europeans negotiation for which has fallen through so that both opportunities are became highly uncertain
- “Life on Mars” resulted in a shorten schedule for sample return

Mars Exploration Status

Visit to JPL at the end of September provided some updated status on the Mars Exploration Program the responsibility for which falls on The Mars Program Office. Tentatively identified were two launches each for four "opportunities" beyond the November 96 launch of the Mars Global Surveyor. These are: Mars Surveyor 98 Orbiter and another one with Polar Lander, and two "piggy back" payloads for DS-2, a 2001 Orbiter and Highland Lander, 2003 Lander(s) Network and Orbiter, a 2005 Lander(s) (possibly precursor to Sample Return) and Orbiter. However, the 03 and 05 opportunities were dependent on some form of cooperation with the Europeans negotiation for which has fallen through so that both 03 and 05 opportunities are became highly uncertain. To complicate matters further, the publicity of possible life on Mars resulted in a desire to return some physical sample to earth in a shorter time frame.

Mars Exploration Status (Cont'd)

- An alternate plan has been developed
  - An Orbiter and U.S. Cruise State plus a Russian Marsokhod and Descent Module in 2001
  - A Lander that might become the test for the 05 opportunity in 2001, and sample collection and return to earth
- The alternate plan is uncertain at this point as three other plans, referred to as paced, accelerated, and aggressive, are being investigated for an earlier sample return date
- Funding at this time is a major issue
- Role of MRS is evolving

Mars Exploration Status (Cont'd)

Concurrent with the opportunities, funding reduction also results in an alternate plan being developed that includes an Orbiter and U.S. Cruise State plus a Russian Marsokhod and Descent Module in 2001, a Lander that might become the test for the 05 opportunity in 2001, and sample collection and return to earth. As of the end of September, however, even the alternate plan is rather uncertain due to the development, for the purpose of discussion within JPL, of three other plans, referred to as paced, accelerated, and aggressive, all of which serve the purpose of pushing for a earlier sample return date and were. Unfortunately, the fixed budget given them originally may be further reduced and even if it were not, additional funding in significant amount will be required for the sample return goal.

One interesting outcome of the push for early sample return and the uncertainty in funding is the clarification of the role of MRS. Since the original plan always include an Orbiter which always carries its own relay package, MRS was never officially included. Pushing for an early sample return, on the other hand, has necessitated the cutting back of Orbiter launches for budgetary reason so that the MRS is now included in all three of the pace, accelerated and aggressive plans, but is not expected to materialize until after 2005 to 2006.

Mars Exploration Status (Cont'd)

- The Russians launch of a French Mars Balloon Relay (MBR)
  - Planned for 94 but is now delayed to possibly 1996
- MBR is a (substantially) one way UHF communication
  - Supports data transfer from the lander(s) to the MBR
  - Limited number of tones from the MBR to the lander(s) only for the purpose of "waking" up the lander(s).
- Two way UHF package currently being developed by Cincinnati Electronics
  - To be launched with the 96 Orbiter
  - Mars Program Office selected FSK modulation whose optimality is not certain

Mars Exploration Status (Cont'd)

In terms of communication support, the Russians were planning to help launch a French Mars Balloon Relay (MBR) in 94 but is now delayed to possibly 1996. The MBR is a (substantially) one way UHF communication package for data to be transmitted from the lander(s) to the MBR, with a limited number of tones that can be sent from the MBR to the lander(s) for the purpose, and only for the purpose, of "waking" up the lander(s).

A two way UHF communication support package is currently being developed by Cincinnati Electronics using FSK modulation scheme to be launched with the 96 Orbiter. The FSK format was specified in the RFP while the modulation scheme is not universally supported within JPL. The 96 Orbiter will also carry the French designed MBR package to provide a communication capability that is compatible with the penetrators.

Communication Support Status

- The Telecommunication Laboratory at JPL currently designing an ASIC chip for a transceiver at 400 Mhz
  - April of 1997 completion expected and compatible with the MBR
  - Supports 8 kbps (BPSK) uplink from Landers a limited number of tones for downlink for "wake up" the
  - Expecting a 98 launch that is expected to also carry the MBR package
- Another effort addresses a UHF package to fit in the landers, rovers, or the penetrators to communication with a relay package on board (not necessarily the MRS)
  - Focused for the next six months on requirement identification for a different missions

Communication Support Status

The Telecommunication Laboratory at JPL is currently designing an ASIC chip for a transceiver at about 400 Mhz. It is expected to be completed in April of 1997. The uplink from the Landers, with BPSK modulation, is expected to support 8 kbps in BPSK modulation. The downlink, with a limited number of tones supporting about 10 bps, serves only the purpose of "waking up" the landers. It is expected to be compatible with the MBR package that is supposed to go on the 96 launch. The new chip is supposed to go on the 98 launch that is expected to also carry the MBR package.

There is another effort under way that addresses a UHF package expected to fit in the landers, rovers, or the penetrators to communication with a relay package on board (not necessarily the MRS). For the next six months or so, however, the effort will be focused on requirement identification for different missions.

New Millennium and DS-1

- The New Millennium Program office is concerned with new technologies in 13 areas
  - Goal is to serve “future customers”, i.e., users
  - “Small satellites” are expected to be in the 400 kg range allocating about 225 kg to “solar electric propulsion”.
- DS-1 (Deep Space), to be launched in July of 1998 will have a small DS transponder in both X and Ka band, a micro-miniaturized package with an SSPA for demo purposes

New Millennium and DS-1

The goal of the New Millennium Program office is concerned with new technologies in 13 area such as miniaturized spacecraft electronics, autonomy remote agent for smart spacecraft, Mars microprobe, in-situ instruments, layered computers, etc. Their goal, however, is to serve "future customers", i.e., users. The mass of their "small satellites", however, is expected to be in the 400 kg range and thus considerably heavier than the "small satellite" of the MRS class. About 225 kg of the New Millennium satellite is allocated to "solar electric propulsion".

The DS-1 (Deep Space), to be launched in July of 1998, will have a small DS transponder in both X and Ka band, a microminiaturized package with an SSPA for demonstration purposes.

Latest NASA Programs Status

- Effort to obtain a final NASA Program Status before the final report met with aarying degrees of success
- Programs and Persons contacted or contact attempted:
  - Clark: Barry Meredith & Dr. Harry Benz (NASA Langley)
  - Lewis: Jim Sarina (TRW)
  - Clementine: Alan Hope (NRL)
  - FAST: Tim Watson, & J. Catena (GSFC)
  - Lunar Prospector: Alan Binder (LMMSC)
  - MGS: Wayne Lee (JPL)

Latest NASA Programs Status

To provide a current update of the NASA programs, we tried to reach a number of perple associated with the various programs. The following provides the most current program update as of March 17, 1997.

Lewis - Launch delay till 12/96 then to 4/17/97 and slipped again due to LLV review

Clementine - Anomaly caused satellite to spin up to 80 rpm after reaching moon orbit, followed by a power loss on 6/4/94 or 6/5/94. Reorientation of solar array was not successful though eventually it did leave the moon's gravity pull and now in an orbit similar to the Earth's. The year after anomaly, some information was downloaded using the 70 m DSN antenna. Continued contact once a year is now no longer possible.

FAST - On orbit and doing great. Hope to use another year and a half.

Lunar Prospector - Satellite construction completed in 2/97, now into thermal vac testing, etc. in preparation of 9/24/97 launch by L-MLV2 (LV's first). Total mass expected to be 305 kg., with a 1.2 m spinner and 1.4 m antenna. Instrument on a 2 m boom off the side of the spacecraft.

MGS - In excellent condition and expect to reach Mars on 9/11/97. Mars Observer stopped communicating 3 days prior to Mars orbit insertion. Suspect cause is a leak in the fuel tank bringing into contact the two chemicals outside the thruster combustion chamber.

Latest NASA Programs Status (Cont'd)

- Other programs whose current status are obtained include:
  - NEAR: Tom Coughlin & Bob Farquhar (APL)
  - SAMPEX: Glen Mason (U of Md) & Jim Willet (NASA Hq)
  - Surfsat-1: Al Berman (JPL)
  - TOMS-EP: Donald Margolies (GSFC)
  - TRACE: (Roberto Aleman, Jerry Daelemans, Joe Burt & Haydee Maldonada (GSFC)
  - Ulysses: Willis Weeks (JPL)
  - WIRE: Dave Henderson (JPL), Orlando Figueroa & Gryan Fafaul (GSFC)

Latest NASA Programs Status (Cont'd)

NEAR - Normal operation, set a new solar cell record at 2.19 AU (previous about 1.7 AU). Preparing to fly by the asteroid Mathilda on 6/27/97, an added mission prior to launch, made possible by less than expected fuel consumption (good launch from Atlas). Main mission with asteroid Eros expected on 1/10/99 at 5m/s, and ends on its landing on Eros on 2/6/00 due to funding.

SAMPEX - Launched 6/12/92, still in excellent shape, exceeded all expectation of lifetime. Completed primary mission (during solar flare minimum) and on extended mission of observing solar flare maximum. Full Cost Accounting will be implemented at the beginning of FY 99. Funding review by Senior Review Board in 6/97 is expected.

TRACE - Spacecraft integration a shade ahead of schedule: all bus components, communication equipment, and the instrument. Solar array not yet on. Ready for 3 weeks system test and then to environmental test. Working toward a 10/15/97 launch by Pegasus XL from Vandenberg despite a contracted launch of 12/15/97. Launch will likely slip if held to contracted date due to the extremely busy manifest for the Pegasus XL.

WIRE - In development with most of the electronics completed. Scheduled for 12/15/98 launch by Pegasus XL from Vandenberg. Integration expected to start on 9/3/97 in Goddard with the first subsystem expected to arrive on 9/28/97, and the JPL instrument on 1/15/98.

Communications Technologies

- Antenna arrays and digital beamforming
- X-band high power SSPA transmitters
- Forward error correction codes
- On-board storage devices

Communications Technologies

In a typical power-limited communications system, the main components which impact the channel throughput are the antennas, power amplifiers, and forward error correction coding (means of ensuring data integrity). Generally, the most simple and efficient modulation scheme is assumed, usually BPSK in most situations, so that modulation trades are merely confirming this choice. Of course, if the throughput is increased, and if the communications channel is not available during periods of data generation and collection, then data storage technologies become important also.

The survey of smallsats uncovered several advanced technologies which are, or could be, applicable to future NASA exploration missions. These fall in the following areas:

- Antenna arrays and digital beamforming
- X-band high power SSPA transmitters
- Forward error correction codes
- On-board storage devices

Each one will be discussed briefly.

MRS Communication

- UHF for Lander-to-MRS
- X-band for MRS-to-Earth

MRS Communication

Communications between the Mars landers and the MRS will be at about 400 MHz using a UHF transceiver. This allows simple antennas and low power on both the landers and MRS with no need to mechanically steer. If a more directional antenna on the MRS is considered, it is natural to consider a higher frequency band. However, given that the lander antenna beam must be very broad, regardless of frequency, the MRS antenna gain must then increase more rapidly than the increase in space loss associated with higher frequencies. Suppose that X band were used. The increase in space loss from UHF (400 MHz) to X band (8 GHz) is 26 dB. To achieve this, and more so that the switch in frequency bands is worthwhile, the beamwidth of the x-band antenna would need to be less than 6° (assuming a UHF antenna beamwidth of 120°). [$26 = 20 \log(120/\phi)$, where ϕ is the X-band beamwidth.] Such a beamwidth requires an X-band antenna diameter of about 1.5 feet. Because there are several landers that require simultaneous communication with the MRS, up to six such antennas would be required on the MRS. This is not feasible. The use of UHF is confirmed. The Telecommunication Laboratory is currently designing an ASIC chip for a UHF transceiver. It is expected to be completed in April.

Communication between the MRS and Earth is at X band and will make use of the X-band Small Deep-Space Transponder (SDST) which is in development. The effort is based on the technology established by the Pluto Fast Flyby Advanced Technology Insertion Activity. It involves two major technology developments: a digital X-band receiver using advanced GaAs MMICs and a miniature high-frequency, high-density electronic packaging technology.

UHF MRS (In-Situ) Transceiver

- Two versions under development
  - Mars Balloon: simple tones to the Landers/Rovers for activation
  - Full two way communications package: that can actually transmit more complicated commands to the Lander/Rovers, requirement definition in progress at JPL

UHF MRS (In-Situ) Transceiver

There are two UHF communications packages being developed, a Mars Balloon Relay (MBR) and a two-way communications subsystem. A MBR is, practically speaking, a one-way UHF communication package to be placed on MRS for receiving data transmitted from the landers. It also can transmit a limited number of tones to the landers for the sole purpose of waking up the landers.

A two-way communication package is in development by Cincinnati Electronics. It is to be carried on the Mars Orbiter. It will use FSK modulation, as required in the associated RFP. The use of BPSK has been investigated and is shown as the preferred modulation type in the MRS study led by STel.

The data rate on the uplink, with the lander EIRP fixed by its need for a broad beam (low gain) antenna and its very limited power generation capability, is determined essentially by the MRS UHF antenna gain. An antenna array placed as a belt around the MRS spacecraft is mentioned in the Alenia Spazio documentation. An enhanced version of this array may be possible to increase the gain and, thereby, the mission throughput (see chart__ herein).

No other UHF developments or commercially applicable initiatives were identified. Note that the current satellite-based cellular systems make use of higher frequencies, in both L band and S band.

X-Band Transmitter - SDST

- Small Deep Space Transponder under development by Motorola for JPL and used in the MRS design
  - Mass $\leq$ 2.5 kg
  - DC Power $\leq$ 10 W (for X/X) or $\leq$ 13.6 W DC (for X/Ka)
  - specification is still under negotiation

X-Band Transmitter - SDST

For Mars missions, mass, power, and volume are precious commodities and it is crucial that a truly miniaturized, high efficiency telecommunication system be available for these missions. The Mars Exploration Office has defined such a development as their number one technology need.

A program to develop a miniature X-band transponder in on-going to both reduce the size and mass of the existing transponder and improve upon its performance. The development builds upon technology base of the Pluto Fast Flyby Advanced Technology Insertion Activity and has two major technology components: 1) a digital X-band receiver using advanced GaAs MMIC techniques and 2) a miniature high-frequency, high-density electronic package. The synergistic goal of these two components is:

- Narrow bandwidth phase lock loop (<20 Hz) with a carrier tracking threshold of -158 dBm
- Wide dynamic range (>70 dB)
- Command detector functionality (for multi-user compatibility)
- Total mass <2.5 kg [this compares, for example, to 6.3 kg for the comparable Cassini system]

This work will bring these two components to a prototype readiness level which will allow the Mars Exploration Office to decide whether or not to proceed with procurement of flight hardware for the 98 Lander mission.

SPACE SYSTEMS LORAL

X-Band Transmitter for Cassini

- Being developed by APL with a 5 W SSPA, first use of solid state X-Band amplifier for deep space mission
- Two convolutional codes
  - Rate 1/2, K=7 for launch and during cruise
  - Rate 1/6, K=15 for the entire asteroid rendezvous phase
- Concatenated with an RS 8 bit (255,233) block code
- Eight downlink data rates: 9.9 bps, 39.4 bps, 1.1 kbps, 2.9 kbps, 4.4 kbps, 8.8 kbps, 17.6 kbps, and 26.5 kbps
  - Higher two rate sufficient for MRS where ≥14.2 kbps required

X-Band Transmitter for Cassini

The availability of solid state power amplifiers (SSPAs) at X band in sufficiently high power for the Mars missions offers an alternative to traveling wave tube amplifiers (TWTAs). The perception is that the reliability of the SSPAs is greater than that of their TWTA counterparts. In return for this probable increase in reliability, the SSPAs would be less than half as efficient. A typical TWTA efficiency, at saturation, will be around 50% to 60%. SSPA efficiencies are between 20% to 30%. Specific SSPAs that are available now are mentioned in chart ___. For still higher power, only TWTAs can suffice.

There is almost no public domain literature on in-orbit reliability performance of X-band TWTAs. Although a moderate number of 20, 40, and 50 W X-band TWTAs are flying, they are primarily dedicated to the MILSATCOM 7.25-7.75 GHz programs (DSCS 2 & 3, Skynet-4, NATO-4, or the "national flag" MILSATCOM compatible programs such as Brazilsat, Hispasat, Telecom-2, and Superbird-2). Operational information on these strategic assets is not in the public domain.

There are also a small number of scientific payloads carrying approximately 40 W TWTA at the 8.4-GHz DSN downlink frequency, such as Landsat, various deep space missions, etc. To our knowledge, no TWTA failures have been reported for these programs.

The most accessible data on in-orbit reliability of TWTAs and SSPAs is the 1991 ESTEC study periodically updated by Bob Strauss. In his July '93 update, Bob compiled the data from over 2000 TWTAs and SSPAs. The report concluded that the established in-orbit reliability performance yielded a TWTA FIT (failures in 10^9) of 680 and an SSPA FIT rate of 790. The author comments that the study was necessarily restricted to the earlier generations of lower power C and Ku band hardware. Since that time, TWT and SSPA products have been produced at even higher power and frequency. In the case of the TWTAs, a number of design improvements have been implemented that have more than offset the reliability burden associated with higher power and frequencies.

X-Band on Mars Global Surveyor

- Receiver portion designed to work with 34-m DSS at maximum range
  - Consistent with MRS design requirements
- Transmit frequency in the 8.4 to 8.45 GHz range
  - Consistent with the 8.425 GHz design for MRS
- 21.33 kbps minimum downlink
  - Sufficient for MRS minimum required data rate
- 25 W TWTA (mounted on HGA to minimize losses and reduce power requirements)

X-Band on Mars Global Surveyor

The transmitter for downlink (to Earth) data transfer from the Mars Global Surveyor is an X-band TWTA. It has a power output of 25 W. A top-level comparison of this amplifier with the SSPA being built by EMS Technologies, Inc. for the Mars Surveyor '98 mission found no significant differences except in their efficiencies and size. The SSPA efficiency is about 30%, that of the TWTA is __%. The SSPA is an integrated amplifier/power supply and occupies a slightly smaller space than the TWTA and its separate power supply. Estimated reliabilities of the two amplifiers gives a slight edge to the SSPA. Nevertheless, there have been no deep space mission failures which are attributed to a spacecraft TWTA.

Array Antenna/Digital Beam Forming

- Array antenna compatible with digital beam forming
- Digital beam forming enables multiple, high-gain beams
  - Each beam pointed to specific lander; maintain high gain
  - High gain provides increased data rate on Lander-MRS link
- Digital beam former mass, power estimates : 10 kg, 100W
  - 5 to 7 simultaneous beams
  - 60-kHz channelization
  - 0.5 MHz total bandwidth
  - ≤ 50 array elements
  - Mass of added array elements not included
- Question of available space for up to 50 array elements (1993 design has 12-element "belt" antenna)

Array Antenna/Digital Beam Forming

This technology is powerful and can be implemented now for the relatively low bandwidth signals which will be transmitted from the Landers to the MRS. To implement this, the existing belt antenna on the MRS must be augmented with several more antenna elements (twice as many for 3 dB increase in gain). The availability of the needed real estate to allow such augmentation is yet unknown. However, even if sufficient real estate can be made available, a preliminary design must first be completed to determine whether the added mass (for the additional antenna elements and the digital beamformer) and power (of the digital beamformer) could be accommodated and justified by the increased throughput.

Naturally, the transmitter throughput may have to be increased as well.

X-Band High Power SSPA

Amplifier Parameter	EMS	MELCO
– Center Frequency:	8.4 GHz	8.1 - 8.4 GHz
– Output Power:	15 W	40 W
– Total Efficiency: (incl. power supply)	29%	27%
– Bandwidth:	300 MHz	100 MHz
– Mass:	1.7 kg	4.1 kg

SPACESYSTEMS LORAL

X-Band High Power SSPA

Deep space missions have generally relied on TWTAs for their on-board X-band transmitters. This chart indicates that SSPAs are now or will shortly become available to serve this function. These two amplifier types differ mainly in their efficiencies and, perhaps, their reliabilities. The axiom that SSPAs were more desirable for space (due to the absence of a wear out mechanism such as the electron source) and not as fragile (no traveling wave helix) has been shown otherwise with the accumulated on-orbit years of TWTAs, especially for the desired life times of the deep space missions.

At least two high-power X-band SSPAs are available for future missions. The first, with 15 W output power and produced by EMS Technologies, Inc., is slated for the Mars '98 Surveyor mission. The CDR on this contract was held on December 4, 1996. Incidentally, output powers up to 30W with no change in efficiency and little change in packaging/footprint may be obtained. The outline for the 15-W version is 8.1 x 5.0 x 2.9 inches. That of the 30-W version would be 10.1 x 7.0 x 2.9 inches. Estimated mass of the 30-W version is 2.2 kg. Even higher powers are achievable by adding more individual FET devices in parallel -- 40 W is thought to be no problem.

The second, produced by Mitsubishi Electric Corporation (MELCO), has nearly the same efficiency and a 40 W output power. The amplifier measures 15.3 x 6.9 x 4.1 inches.

In obtaining the latest update with MGS, Mr. Wayne Lee of JPL was not able to identify any trade between an SSPA vs a TWTA for X-band downlink communications. It was pointed out, though, because of the quick response desired to replace the Mars Observer, as many of the Mars Observer parts as possible were used. The TWTA was used on the Mars Observer. Wayne also stated that the MGS downlink subsystem is more capable than that of the Mars Surveyor '98. It has a higher data rate and it can be continuously operable. The Mars Surveyor '98 has thermal constraints that arise from its operational profile of 5 hours on and 5 hours off.

FEC Turbo Code

- Discovered by Berrou and Glavieux; reported in 1993
- Berrou and Glavieux: performance of the turbo codes “for very large interleaving sizes ... appears to be close to the theoretical limit predicted by Shannon”
- Dariush Divsalar of JPL, and others, confirm coding gain
  - Codes within 0.8 dB of Shannon limit for given throughput and modulation, at BER of 10^{-6}
  - Interleaver length of 16,000 bits
- Applicability of “turbo codes” for deep space communications
  - Easier to implement and 1.6 dB better than Galileo spacecraft concatenated code [(15, 1/4) convolutional code, 10 bit RS code]

FEC Turbo Code

At MILCOM 96, a tutorial session was given on turbo codes by Dariush Divsalar of JPL. One of the subject is the turbo code application for deep space communications with a promise of achieving, with the use of a 16000-bit interleaver, "a very large coding gain" - within 0.8 dB of the Shannon limit for a given throughput and modulation, at an BER = 10^{-6}. Berrou and Glavieux also made the statement that "for sufficiently large interleaving sizes, the correcting performance of turbo-codes, investigated by simulation, appears to be close to the theoretical limit predicted by Shannon."

The JPL turbo code is a 1.2 dB improvement over a concatenated code (a low rate convolutional code (15, 1/6) with a 10 bit Reed-solomon code) described in the Preliminary Conference Publication, of the NASA-sponsored Advanced Modulation and Coding Technology Conference held on June 21-22, 1989. Note that the concatenated code was, itself, a 2.1 dB improvement over the then current standard, that was used on the Voyager Spacecraft. [See Daniel J. Costello, Jr. (principal investigator), Error Control Techniques for Satellite and Space Communications (NASA Grant Number NAG5-557), Annual Status Report, December 1995, pp. 51, 52 for the following: "It can be expected that when the performance requirements are high enough, the Turbo Code will outscore convolutional codes in decoding complexity."]

The promise is that when performance requirements are sufficiently high, turbo decoders are more easily implemented than are convolutional codes with Viterbi decoding (Error Control Techniques for Satellite and Space Communications, Annual Status Report, Daniel J. Costello, Jr., principal investigator (University of Notre Dame), NASA-CR-199729, December 1995, p. 52).

On-Board Digital Storage Capacity

- Increased Lander-MRS data rate may be accompanied by increased on-board storage capacity requirements
- MRS on-board storage required (at 1009 km altitude) is just under 128 Mbits
- Mass allocated for solid state recorder (SSR) of MRS is 19 kg
  - Current land/air ruggedized SSRs weigh < 1 kg
  - Storage card itself (without read/write/control electronics) ≤ 0.1 kg

On-Board Digital Storage Capacity

According to the "Mission Analysis Aspects" of the Mars Relay Satellite Definition Study, (p. 43 to 62), presented by Alenia Spazio in July 1993, the Mass Storage required for the MRS at 1009 km altitude is just under 30 Mbits. (Work done for NASA JPL by STel indicates that 128 Mbits are required) The associated On-board Data Handling (OBDH) system, including the solid-state recorder (SSR) was allocated 19 kg (ibid, p. 2.2.1). It is not known what the SSR weighs, by itself. However, SSRs with capacities to 640 Mbits are available in ruggedized version suitable for use in rotary wing aircraft and in tracked land vehicles. One particular unit, made by Targa (San Ramon, CA 94583), weighs only 460 g. It is powered from a 5-vdc supply at 600 ma. Its dimensions are 1.1 x 3.3 x 6.5 inches. The reliability is specified as greater than 50,000 hours (or MTBF > 4.7 years). These units are not radiation hardened.

Advanced Subsystems

- Propulsion Subsystems
- Advanced Battery Subsystem
- Advanced Material
- Miniaturization

SPACE SYSTEMS LORAL

Advanced Subsystems

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Propulsion Subsystems

- Two types of propulsion subsystems reviewed
  - PPT suitable for really small satellites with limited power (100 W range) such as the one designed by Stanford Telecommunications Inc./ Orbital Science Corporation under a JPL subcontract
  - SPT suitable for satellites with higher power (at least 500 W) such as those by larger satellite vendors such Lockheed Martin and Space System/Loral

Propulsion Subsystems

Two types of advance propulsion subsystems were reviewed: the Pulsed Plasma Thruster Subsystem and the Stationary Plasma Thruster Subsystem.

Both propulsion systems were originally developed in the United States in the 1960, the PPT by Lincoln Laboratories and the SPT by NASA Lewis Research Center. PPT is particularly suitable for extremely small, precise impulse bits for drag make up or attitude control. Six flights sets of various versions of the original PPT have flown without failure. SPT was eventually abandoned in favor of the development of the Kaufman type ion thruster due to efficiency considerations. Various research institutes in Russia built upon the initial NASA work and were able to eventually solve the thruster efficiency problems that NASA had experienced. SPT is more suitable for moderately sized satellites providing electrical powers of 500 W and up and satellites with more demanding requirements such as orbit raising and NS station keeping.

Propulsion Subsystems - PPT

- Pulsed Plasma Thruster (PPT) appears to be one attractive alternative for small satellites
  - Extremely small minimum impulse bits
  - High specific impulse
  - Low power requirements
  - Low level of system complexity
  - Flight proven reliability

Propulsion Subsystems - PPT

Due to the system complexities of typical satellite propulsion systems, only one type of propulsion technology appears to be attractive today for application to very small spacecraft, pulsed plasma thrusters or PPT's. This particular technology has various unique capabilities attractive to small spacecraft propulsion systems and attitude control (i.e. the elimination of momentum wheels) for large geosynchronous spacecraft. These performance attributes include: extremely small minimum impulse bits, high specific impulse, low power requirements, low levels of system complexity, and flight proven reliability.

Pulsed Plasma Thruster - Background

- Developed by Lincoln Lab to provide extremely small, precise impulse bits for drag make-up or attitude control
- Six flight sets have flown without failure
- Most significant program: Navy NOVA/TIP spacecraft
  - Provided a true inertial orbit for navigation purposes
  - Use of PPT to correct all measured perturbation forces
- Recent interest in small, simple propulsion system resulted in a NASA contract to Olin Aerospace to develop next generation PPTs

Pulsed Plasma Thruster - Background

The Pulsed Plasma Thruster has an extensive test and flight history dating back to the 1960's. The original concept was developed by the Lincoln Laboratories to provide extremely small, precise impulse bits for drag make up or attitude control. Six flights sets of various versions of the original PPT have flown without failure, the most significant program being the Navy NOVA/TIP spacecraft. These spacecraft provided a true inertial orbit for navigation purposes by using PPT's to correct all measured perturbation forces. Recent interest in small, simple propulsion systems for large constellations of small LEO spacecraft along with the potential for replacing momentum wheels in geostationary spacecraft have created a renewed in interest in PPT technology. This interest has resulted in NASA awarding a contract to Olin Aerospace to develop the technology for the next generation of PPT's.

Pulsed Plasma Thruster - Operation

- Primary components:
  - Teflon propellant bar
  - Discharge chamber
  - Large storage capacitor
  - Ignition spark plug - triggers release of energy stored in capacitor
- As current flows between two electrodes, the Teflon propellant bar is ablated and ionized
- Thrust produced when ionized particles accelerated by the Lorentz force created by the interaction of the discharge current and the induced magnetic field
- Thrust level controlled by altering discharge pulse rate

Pulsed Plasma Thruster - Operation

The PPT consists of the following primary components: a Teflon propellant bar, a discharge chamber, a large energy storage capacitor, and an ignition spark plug. Operation occurs when the spark plug fires and releases the energy stored in the capacitor. The face of the Teflon propellant bar is then ablated and ionized as current flows between the two discharge electrodes. Thrust is produced when the ionized particles are accelerated by the Lorentz force created by the interaction of the discharge current and its induced magnetic field. The discharge dissipates after the capacitor energy is depleted. Thrust levels are controlled by altering the rate of discharge pulses.

Pulsed Plasma Thruster - Performance

- Based on data from LES 8/9 at 1 Hz pulse rate

Parameter	Value
Thrust (μN)	300
Specific Impulse	1000
Power (W)	25
Total Impulse (Ms)	10,000
Total Pulses	18,500,000
Capacitor Energy (J)	20
Total Mass (kg)	16.5

Pulsed Plasma Thruster - Performance

As a baseline, Olin is using performance numbers from existing PPT data, specifically the thrusters developed for LES 8/9 at a 1 Hz pulse rate.

Significant improvements are expected to be made during the development program, primarily the result of advances in capacitor technology over the last 20 years.

Pulsed Plasma Thruster - Current Status & Future Projection

Two NASA programs with Olin Aerospace for PPT system:

- Joint AF Phillips Lab, NASA Lewis, & Olin Aerospace effort
  - Provide orbit raising for 1999 launch of the Mightysat II.1 S/C
- New Millennium EO-1 mission
  - Replace function of one momentum wheel in an experimental capacity

Pulsed Plasma Thruster - Current Status & Future Projection

The PPT system is being developed by NASA under a contract with Olin Aerospace for two separate programs. The first program, a joint Air Force Phillips Laboratory, NASA Lewis, and Olin Aerospace effort, is to provide orbit raising for the scheduled 1999 launch of the Mightysat II.1 spacecraft. The second is for the New Millennium EO-1 mission, where the PPT will be employed in an experimental capacity to replace the function of one momentum wheel.

Propulsion Subsystems - SPT

- Offers another alternative for small satellite propulsion
- A range of capabilities in terms of thrust and power requirements
- Even the low thrust version is capable of accomplishing various spacecraft orbital adjustment maneuvers

Propulsion Subsystems - SPT

SPT offers another propulsion alternative. While PPT is more appropriate for very small satellites with 100 W or so of power, SPT is more appropriate for medium size satellites with demanding mission requirementswith a moderate power subsystem.

Stationary Plasma Thruster - Background

- Concept originally developed at NASA-Lewis in the 1960s
- Built upon by the Russians who were able to overcome the thruster efficiency problem
- SPT-50's and SPT-60's were flown on three Meteor spacecraft in the early 1970's after extensive tests - numerical designation refers to the thruster discharge chamber diameter in millimeters
- The low power SPT's were eventually replaced by the higher power ($\approx$ 700 W) higher thrust ($\approx$ 40 mN) SPT-70's on the Kosmos and Luch spacecraft series flown in the 1980's and 1990's

Stationary Plasma Thruster - Background

Stationary Plasma Thrusters (SPT's), have an extensive test and flight history dating back to the 1960's. The original concept was developed in the United States at the NASA Lewis Research Center in the early 1960's; however it was eventually abandoned in favor of the development of the Kaufman type ion thruster. In Russia, various research institutes built upon the initial NASA work and were able to eventually solve the thruster efficiency problems that NASA had experienced. After an extensive test program, SPT-50's and SPT-60's were flown on three Meteor spacecraft in the early 1970's. The SPT numerical designation refers to the thruster discharge chamber diameter in millimeters. These thrusters were of relatively low power ($\approx$ 400 W) and low thrust ($\approx$ 20 mN); but were still able to accomplish various spacecraft orbital adjustment maneuvers. Eventually, these low power SPT's were replaced by the higher power ($\approx$ 700 W) higher thrust ($\approx$ 40 mN) SPT-70's on the Kosmos and Luch spacecraft series flown in the 1980's and 1990's. In turn, the SPT-70 was replaced by the larger SPT-100 (1350 W power and 80 mN thrust) on the Gals and Express spacecraft series in the 1990's. Although it has been more than a decade since the last lower power SPT-50's and SPT-60's have flown, interest in such low power thrusters have been renewed by recent interest in smaller lower powered spacecraft.

Stationary Plasma Thruster - Operation

- SPT consists of an annular anode/gas distributor and annular discharge insulator with external magnet coils and a redundant pair of cathodes
- Xenon gas is supplied to the discharge chamber and the selected cathode in the proportions 13:1
- Discharge current flows through the inner and outer magnet coils, thereby creating a radial magnetic field that impedes electron movement from the cathode to the anode, and causes electron cyclotron motion around the magnetic field lines
- The electrons diffuse to the anode primarily as a result of collisions with xenon molecules in the discharge chamber, and these collisions, in turn, create xenon ions
- Ions created in this region are accelerated by this potential, creating thrust

Stationary Plasma Thruster - Operation

The SPT consists of an annular anode/gas distributor and annular discharge insulator with external magnet coils and a redundant pair of cathodes. Xenon gas is supplied to the discharge chamber and the selected cathode in the proportions 13:1. Discharge current between the cathode and anode flows through the inner and outer magnet coils, thereby creating a radial magnetic field that impedes electron movement from the cathode to the anode, and causes electron cyclotron motion around the magnetic field lines. The electrons diffuse to the anode primarily as a result of collisions with xenon molecules in the discharge chamber, and these collisions, in turn, create xenon ions. The magnetic field and discharge chamber geometry are designed so that nearly all the impedance to electron flow and nearly all the ion creation occur in a thin (10-mm) annular region near the thruster exit plane. This results in nearly the full 300-V potential difference between the anode and cathode being imparted axially across this thin section. Ions created in this region are accelerated by this potential, creating thrust. Beam neutralization is provided by emission of sufficient additional electrons by the single operating cathode.

Stationary Plasma Thruster - Performance

Thruster	Thrust (mN)	Specific Impulse (s)	Power (W)	Total Impulse (thousand Ns)
SPT-50	20	1300	350	150
SPT-70	40	1400	680	500
SPT-100	80	1500	1350	2000

SPACE SYSTEMS LORAL

Stationary Plasma Thruster - Performance

The lower power SPT-50 requires the least power of the three thruster types shown above, but with reduced thrust, specific impulse and total impulse. All three thruster types are capable of throttling over a relatively large power range ($\approx \pm 30\%$ of the nominal power level) with a corresponding linear change in thrust. This throttleability, and the attractive performance parameters as shown above, make the SPT-50 an attractive thruster for small spacecraft propulsion

Propulsion Subsystem Summary

- The cost of PPT is considerably lower, about $100,000 per unit with about three nozzles but is suitable only for satellites that are not too demanding in propulsion requirements
- The cost of SPT is rather high, not unusual in the million dollar range per system
  - Its main advantage is the reduction in weight typically allows a reduction in the class of launch vehicle required for launch cost savings
  - Clearly a system consideration must be conducted before its cost advantage can be ascertained
  - Suitable for demanding missions especially when NS station keeping and orbit raising are involved

Propulsion Subsystem Summary

The selection of one or the two of the two types of propulsion subsystems depends on size, especially the power level, of satellite in question and is subject to system level trade, including launch cost and mission requirements. Its relevance to the smallsat consideration here will be discussed at the final section.

Power Subsystem - Bipolar batteries

- Bipolar battery offers a new promising technology
  - Delivered as individual modules or as an assembly of modules
  - Voltage at end of charge: Approximately 100V (one module)
  - Average discharge voltage: approximately 72V
- Individual modules
  - approximately 6" diameter, 15 inches tall, & 10 kg mass, provide 650 to 700 watts at 100% Depth Of Discharge (DOD)
- At 80% of DOD, maximum usable power about 560 watts
- Roughly 20% more Ah (ampere-hours) are returned to the battery during charge than are removed during the previous discharge.

Power Subsystem - Bipolar batteries

Bipolar batteries can be delivered as individual "modules," or as an assembly of modules. While the final design is still evolving, most likely, one module will have approximately 100V at end of charge, and an average discharge voltage of roughly 72V. A module will be approximately 6" in diameter, and will be 15 inches tall and will weight about 10 kg. Each module will provide roughly 650 to 700 watts at 100% DOD (depth of discharge). Normally a maximum of 80% of the total capacity is allowed, so the usable power is roughly 560 watts. The battery will dissipate roughly 125 to 150 watts of heat during a normal GEO, 80% discharge in a 72 minute eclipse. Roughly 20% more Ah (ampere-hours) are returned to the battery during charge than are removed during the previous discharge. If more than one bipolar battery module is required for a power system, then the modules can be "attached" to the power bus in parallel. However, each module will require a separate PCU (power control unit) between it and the bus in order to control the charge and discharge current of each module.

Power Subsystem - Lithium-Ion batteries

- Lithium-Ion batteries is another alternative
  - Energy density comparable to Bipolar battery
  - About 80 Wh/kg
- Much lower heat dissipation
  - About 1/2 to 1/3

Power Subsystem - Lithium-Ion batteries

SS/L is also looking at Lithium-Ion battery designs, but no power, mass, or volume information is available today. The energy density of a lithium-ion battery should be comparable to a bipolar battery (about 80 Wh/kg), although the heat dissipation will be about 1/3 to 1/2 of that for bipolar batteries. It is a technology that is worth tracking for the MRS project.

Advanced Composite Material

- The primary driver for the use of advanced composite materials is for weight reduction to accommodate ever increasing payloads
- A large majority of these produced by SS/L are graphite reinforced polymers for:
  - Antenna reflectors
  - Primary structures
  - Solar array substrates
  - Communication payload support structures and panels
- Two types of composite materials application examples will be discussed, their overall cost impact on smallsat presented in the final conclusion section

Advanced Composite Material

Space Systems/Loral has had more than 20 years of experience as a leader in the research, development, and application of organic matrix composite materials to spacecraft structures.

More than 100,000 composite components have been produced by SS/L and are currently in orbit. the large majority of these components are graphite reinforced polymers in the form of antenna reflectors, primary structures, solar array substrates, and communication payload support structures and panels.

Early spacecraft used minor quantities of advanced composites until the 1980s when spacecraft required structural weight reductions to accommodate ever increasing payloads. As an example, the current Intelset VII satellite system has more than 50% of the structural weight of advanced composite materials. Future systems are expected to provide even higher percentages of their structural weight in advanced composites because of increasing satellite launch costs and longer in-orbit performance.

Two examples of the use of advanced composite materials will be presented.

Advanced Composite Material (Cont'd)

- K1100 - a graphite fiber, a light weight, highly thermal conductivity graphite appropriate for space application, in development and application stages at SS/L
  - Composite Scan mirror for an environmental satellite at SS/L
  - Battery Sleeves
- M60J
  - As primary support structure for the Multi-Angle Imaging Spectroradiater (MISR) of JPL

SPACE SYSTEMS LORAL

Advanced Composite Material (Cont'd)

Space Systems/Loral has one of the best material laboratories in the satellite business and has been the leader in the research, development, and application of organic matrix composite materials to spacecraft structures. SS/L has produced more than 100,000 composite components currently in orbit: the large majority of these components were graphite reinforced polymers in the form of antenna reflectors, primary structures, solar array substrates, and communication payload support structures and panels.

Two types of advanced materials particularly worth addressing for the present purpose are designated as K1100 and the M60J. The cost of these materials is generally quite a bit higher than the aluminum that they replaced. As in so many other things throughout this survey, they cannot be considered in isolation, at least insofar as cost is concerned, but must be treated as part of the system trade that takes into account the launch cost that is typically estimated at about $30,000/kg.

K1100 Application - Composite Scan Mirror

- Developed to replace the current beryllium scan mirror used in the imager and sounder of an environmental satellite
- High thermal conductivity fibers for reduction of thermal gradients and for extreme stiffness properties
- Replacing aluminum battery sleeves with composite sleeves for considerable mass savings with no loss in thermal performance

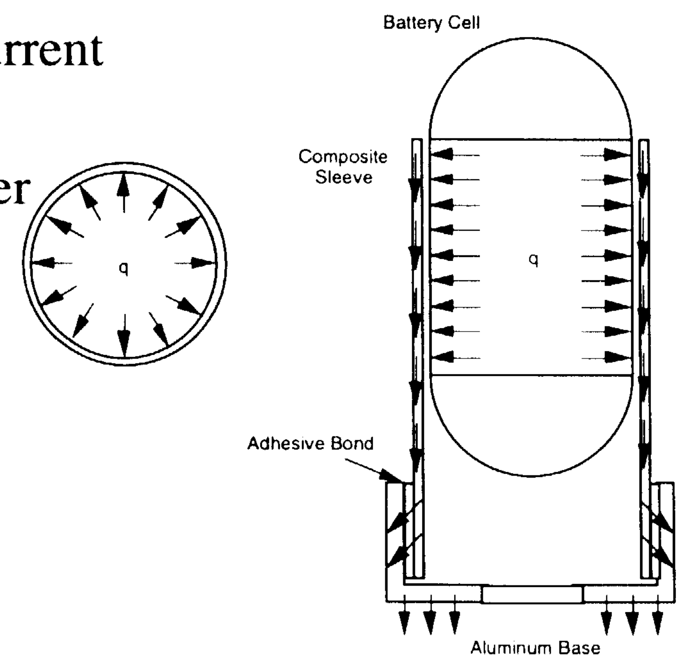
K1100 Application - Composite Scan Mirror

The scan mirror operates in the visible region of the spectrum in a space environment and has strict performance requirements. Specifically, the mirror must not distort more than two wavelengths of visible light peak-to-valley (the maximum elevation difference between any two location on the mirror) when subject to an operating temperature range of almost 100° C. In addition, the mirror must not sag more than half a wavelength of light under its own weight.

High thermal conductivity fibers were chosen to help reduce thermal gradients within the mirror, and for their extreme stiffness properties. Aluminum battery sleeves position the pressurized cell batteries in an array and conduct excess heat away to a radiator. Structurally, the composite sleeves must still satisfy launch load and cell retention requirements. The composite battery sleeves can replace aluminum parts at a considerable mass savings with no loss in thermal performance.

K1100 Application - Battery Sleeve

- Battery sleeves whose main purpose is to carry heat away from a battery cell to a baseplate during operation, while providing adequate support during launch and cell pressurization operations, as shown on the right
  - An obvious choice to replace the current aluminum structure
  - Achieves 50% weight reduction over aluminum design



K1100 Application - Battery Sleeve

A battery sleeve is a simple cylinder whose primary purpose is to carry heat away from a battery cell to a baseplate during operations, wile providing adequate support during launch and cell pressurization operations. Existing designs used aluminum for the sleeve. K1100 graphite fiber was an obvious choice to improve on the sleeve thermal conductivity while significantly reducing mass.

M60J Application - MISR Support Structure

- Development of the Primary Support Structure for the JPL instrument Multi-Angle Imaging Spectroradiometer "MISR"
  - Design, fabrication and proof test accomplished in 1994 by SS/L
  - Reduced the mass from 30 kg which exceeded requirement to 18 Kg (with some design change as well) that meets requirements
  - Survived dynamic and thermal environmental testing in 1995
  - Now part of the MISR fight instrument
- Cost of this material is about twice as high as that of the K1100: about $2000/lb.

M60J Application - MISR Support Structure

The multi-Angle Imaging Spectroradiometer (MISR) is an instrument developed the Jet Propulsion Laboratory (JPL), for the Earth Orbiting Spacecraft (EOS) to be launched in 1997. The detailed design, fabrication and proof test of the composite structure was accomplished in 1994 by SSL under contract to JPL.

The use of a composite structure was selected after early metallic truss and plate concepts did not meet the mass and stiffness requirements. The design approach uses flat graphite-cyanate panels connected with clips and lognerons to simplify the construction of a single article. This approach also simplified the incorporation of the removable access panels that are required on three sides of the structure. The contractual requirement was for a Primary Support Structure for the MISR Engineering Model capable of being upgraded to a flight spare.

The use of graphite-cyanate composite flat panels for the MISR Primary Support Structure met MISR Project requirements and expectations. The mass was about 10% under the MISR allocation, and the structure survived the sine burst proof tests and the instrument environmental tests. Alignment requirements were met, and the resonant frequencies exceeded the Spacecraft 35 Hz requirements by about 10 percent.

Ground Operations

Ground Operations

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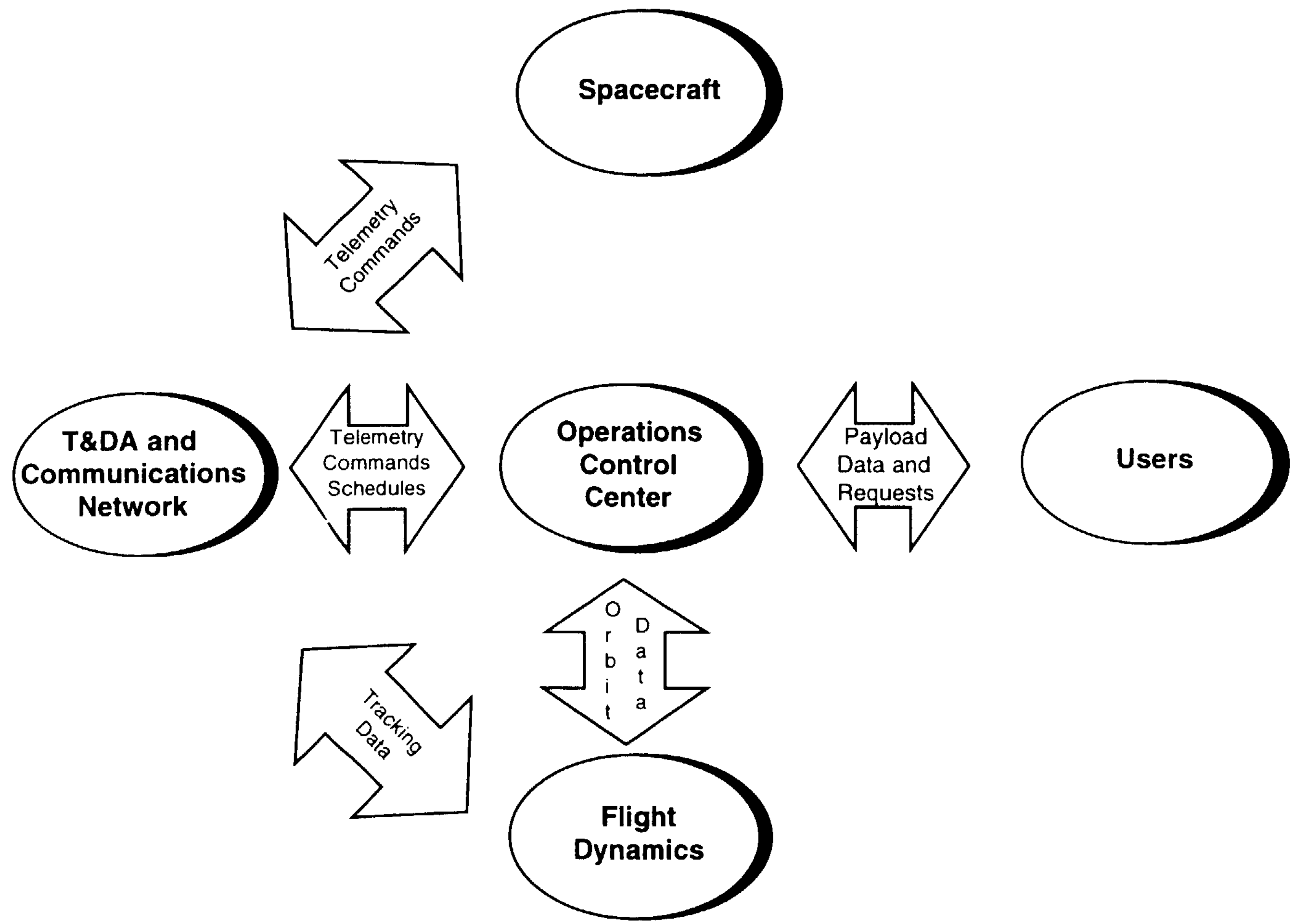
Overview of Mission Operation Environment



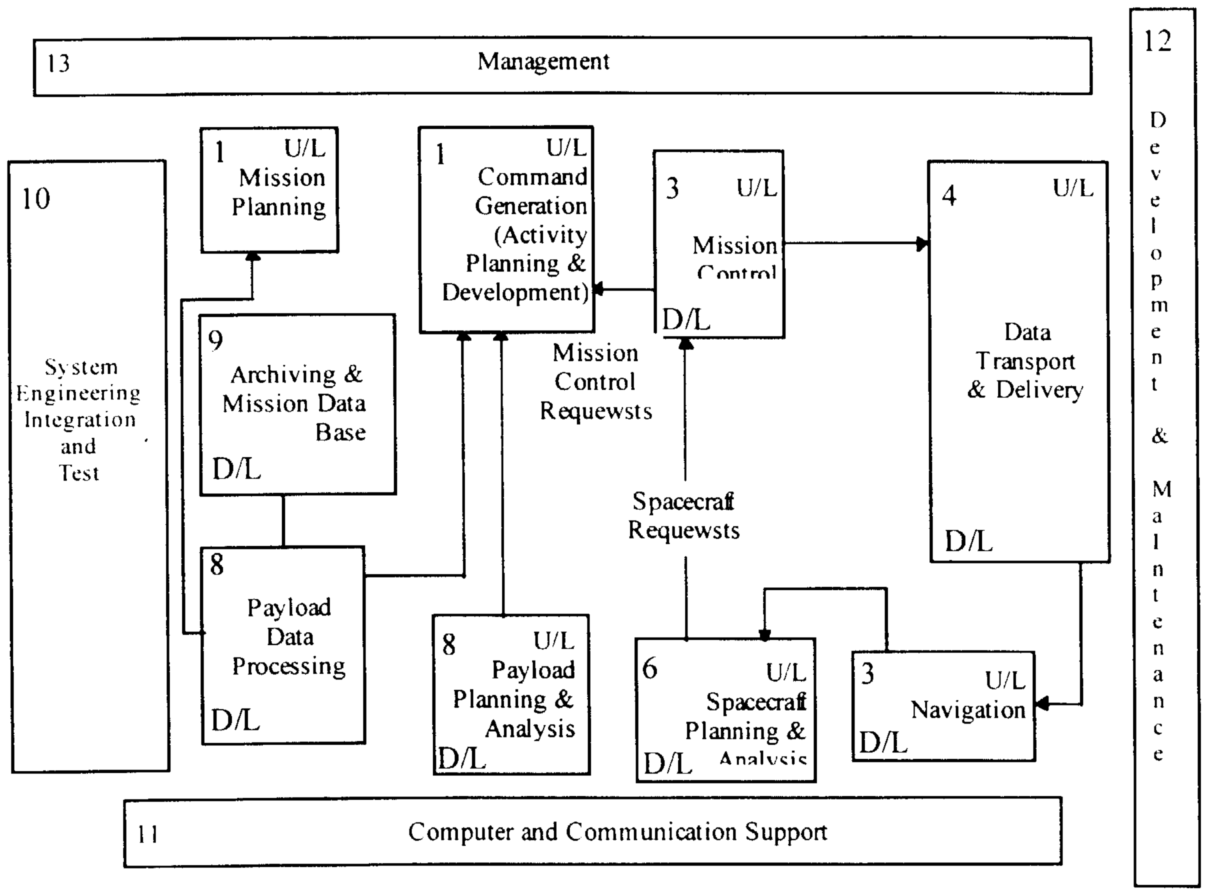
SPACESYSTEMS LORAL

Overview of Mission Operation Environment

An OCC is one of several components of a space system. Other components in the system include the spacecraft, the Tracking and Data Acquisition (T&DA) Network [i.e., NASA Communications (Nascom) switching center], orbit and attitude computation facilities (i.e., Flight Dynamics), and the user community. This viewgraph presents a simplified view of a space system. The OCC controls the spacecraft, which collects payload data. The T&DA Network provides facilities for collecting and transporting spacecraft data. Flight Dynamics provides orbital data essential for planning, scheduling, and maintaining the spacecraft's position. The user community provides the reason for the system; they receive the satellite's payload data and process it into information.

Mars Mission Operation System - MOS

- The thirteen functions in a Mars Mission Operations System



Mars Mission Operation System - MOS

Depicted in the viewgraph is the relationship among the thirteen functions in a Mars Mission Operation System. Most of the top ten functions listed are performed at the OCC. A detailed description of these functions is available in the Proceedings of the Workshop on Reducing the Cost of Space Operations, May 3-4, Arlington, Virginia and from the World Wide Web edition of the Guidebook at URL address:

http://stugyro.stanford.edu/RELATIVITY/GUIDEBOOK/guidebook.html.

SPACE SYSTEMS LORAL

Current Operation Control Center (OCC)

- Rigid operational scenario
- Monitoring and Command constraint checking typically perfumed manually with little to no support for correlating telemetry and related flight dynamics information with anomalous conditions
- Manpower intensive and costly DSN
- Wasted communication bandwidth
- Multivendor environment for computer and communication equipment without an open, i.e., standard, interface
- Manually generated plans and schedules, and if automated, with separate custom software and computers without inadequate netting

SPACE SYSTEMS LORAL

Current Operation Control Center (OCC)

OCC software is typically developed to support a relatively rigid operations scenario developed prior to launch. Manual checking of telemetry and commands by operations staff is required for monitoring and checking command constraint conditions. Little or no support is provided for correlating telemetry data and related flight dynamics information with anomalous conditions. When anomalies arise, operations staff are often presented with masses of difficult to digest information. Even routine trend monitoring is handicapped by excessive information in a poorly presented form. The communications complex supporting the Deep Space Network (DSN) of antennas is manpower intensive and costly to operate. However, reducing manpower may increase risk to the mission. Information that can be processed onboard is processed on the ground, wasting communication bandwidth and computing resources. There are also interoperability issues with computers and communication equipment from different vendors, making it expensive to upgrade the system Some plans and schedules are generated manually. When they are automated per customer requests, the process of generating the commands is performed with separate software and may run on a separate computer.

The reasons for the existence of such poor operational environments at traditional control centers and suggestions for resolving the problems are presented in the main body of this report. Because of additional constraints placed on operators of spacecraft for Mars missions, innovative approaches must be employed to support full spacecraft autonomy.

Improve OCC Efficiency with Onboard Autonomy

- Migrate manpower-intensive activities toward onboard automation
- Typical to consider are:
  - Attitude and pointing control, health management
  - Shifting details of planning, scheduling, and commanding on board - use of COTS high-level command languages
  - Payload information processing through compression/decompression, information extraction, autonomous decisions on data transmission and filtering, for bandwidth efficiency
  - Use of commercial communication protocols, e.g., TCP/IP, SCPS, CCSDS - Deep Space One project of New Millennium with prototype FTP protocol
  - Employ ground-based toolsets, such as expert systems and Graphical User Interfaces (GUI)

Improve OCC Efficiency with Onboard Autonomy

Typical manpower-intensive that can be migrated onboard the satellite are:

a. Performing system health management onboard, i.e., Fault Detection, Isolation, and Recovery including telemetry monitoring, event detection, anomaly resolution, conflict detection, and resolution and trend analysis..

b. Shifting details of planning, scheduling, and commanding by executing schedules and command sequence macros onboard by the use of Commercial-Off-the-Shelf (COTS) products with a high-level command language common to spacecraft and ground systems are recommended..

c. Saving bandwidth by performing payload information processing onboard through data compression/decompression and filtering , information extraction, autonomous decisions.

d. Using established commercial communication protocols such as TCP/IP, SCPS, and CCSDS for reliable file transfer, file handling, transport, security, and the network layer, and for data format at the communication link and physical layers.

e. Employing ground-based toolsets, such as expert systems and Graphical User Interfaces (GUI) using display-by-exception techniques.

[Doyle 94] provides a robust telemetry monitoring tool with multiple monitoring approaches.

OCC Requirement Drivers

- Requirement Evolution
  - Increasingly complex instruments
  - Extended lifetime
  - Increased geographic distribution of principal investigators
  - Shorter system response time
  - International communication connectivity

OCC Requirement Drivers

NASA's Long Range Plan lists the following requirements drivers:

a. The need to support missions with a growing number of increasingly complex instruments.

b. The need to support missions with extended lifetimes.

c. The need to support the increased geographic distribution of investigators.

d. The need to support interactive science operations requiring shorter system response times.

e. The need to support increased international cooperation with international communications connectivity.

OCC Requirement Impacts

- High Impact ●
- Medium Impact ◉
- Low Impact ○

	Spacecraft/Instrument Evolution	Extended Mission Lifetimes	Distributed Investigators	Faster Response Times	International Cooperation
Mission Planning and Scheduling	●	◉	◉	●	◉
Command Management	◉	●	◉	◉	○
Operations	○	●	◉	◉	○
Test and Simulation	○	◉	●	●	○
Flight Dynamics	●	◉	◉	◉	○
Network	○	○	●	○	●

SPACE SYSTEMS LORAL

OCC Requirement Impacts

The increase in satellite and instrument complexity makes planning and scheduling satellite activities and ground-based communications resources more difficult. If onboard resources are scarce, it will be difficult to allocate these resources in a way that optimally meets mission objectives. Increased mission longevity will have a significant impact from a system life-cycle perspective as system robustness becomes a key requirement. Because of the geographically dispersed community of end-users, planning and scheduling will be more complex, increasing the contention for such shared resources as data products and processing services. Verifying the scientific agenda will become increasingly difficult. The scientific community's desire for interactively controlling experiments (i.e., telescience) will require changes to many Mission Operation Directorate (MOD) functions. Supporting interactive science operations will require increasingly advanced planning and scheduling tools. This is especially critical for the MRS if scientists are expected to directly control the landers and rovers. Finally, there will be new requirements for interfacing with international data relay satellite systems that will impact on the OCC not only on the interface and interoperability issues but as the international programs fail or succeed. One needs to remember only the Relay Balloon and the plan with the Russian Rovers .

OCC - Economic & Enabling Technologies

- Design, implementation, and future operations impacts:
  - Cost
    - Decreasing cost for high performance computer resources
    - Decreasing cost (per bit) of data handling by VLSI technologies
    - Decreasing cost for data transport, processing, and storage
  - Enabling Technologies
    - Increasing telecommunication networking capability
    - Reduction of implementation schedule due to computer aided environment
    - Increasing automation at OCC by expert systems & neural networks

OCC - Economic & Enabling Technologies

Advancements in computer technology and price/performance improvements will help MOD respond to the challenges of the 1990's and beyond. Notice the low-end workstations are available that are 40 times faster and cost less than similar computers 10 years ago.

Improvements in data handling techniques and technologies, in mass storage devices, and in database management techniques will reduce data handling costs and make it possible to create large repositories of scientific data at reduced cost per bit with increased reliability.

Cost improvement is not, however, does not a depend so much on new technologies as on demands for the technology in the market.

Improvements in networking and communications capabilities, enhancement in the operational environment by the addition of computer-aided tools will also contribute to the general increase in OCC productivity. Finally, AI and other expert system should be tracked for further performance improvements.

Cost Considerations

Cost Considerations

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Cost Overview

- Quantification of potential cost savings from advanced technologies such as miniaturization is speculative at best
  - Market demands that drive production quantity being the cost driver
- Cost of advanced material such as K1100 will increase so the issue is not so much cost but cost effectiveness that fits the requirements
- Meaningful cost investigation possible only for Ground Operations from automation of both space and ground elements
  - Potential Cost Considerations
  - Potential Cost Savings

Cost Overview

While emerging technologies and miniaturization can be expected to contribute to cost savings for the construction and operation of a Mars Relay Satellite, applying any emerging technology as they become available can, on the other hand, be expected to increase the acquisition cost of a system. As a rule, while a new technology may offer potential cost savings, it is the market demand of that technology that allows the realization of the potential savings. A quick look at the cost history of any new technology, such as a VCR in the consumer product field or the personal computers and solid state random access memories in the high technology field, to name just a few, will readily convince us of that. It will be highly speculative to discuss the cost impact of some of the emerging technologies discussed in this survey, such as the miniaturization effort of the UHF and X-band package now taking place at JPL as they are still in the requirement definition phase. Others such as the propulsion and power subsystems are still in the R&D stage and it is premature to discuss cost issues in any detail with any confidence. In any event, the cost impact will depend on a system level consideration and the mission requirements. The only exception to the above statements is the Ground Operations for which some detailed quantitative data can be presented. Even then, the cost savings are measured only in relative terms.

Propulsion Subsystem

- Cost for Pulsed Plasma Thruster is quite low, it runs for about $100,000 per unit with multiple nozzles
- Cost for Stationary Plasma Thruster could easily run into six or seven figures
- Direct comparison is, however, relatively meaningless because it depends on system trade consideration
  - PPT applicable only with small satellites with limited power (100 W or so) with only routine mission requirements such as E-W house keeping
  - SPT is more suitable for larger satellites with at least several hundred watts of power and demanding mission requirements

Propulsion Subsystem

Cost for PPT is measured in the $100,000 range providing several nozzles. As pointed out previously, it is particularly suitable for non-demanding missions for satellites that provides about 100W or so of power.

The cost of an SPT subsystem could be considerable as it easily runs into six or seven figures. It also is applicable only to satellites with demanding mission requirements such as orbit raising and NS station keeping. Based on a rule of thumb $30,000/kg of launch cost, the advantage of SPT lies in the fact that its use may result in a change of launch vehicle class. The trade here is basically with launch cost.

Power Subsystem - Bipolar Batteries

- Cost (material alone) per module is typically about $14,632, to this must add
  - 36 hours of direct labor for assembly, installation and activation
  - 72 hours of labor plus additional material cost of about $5,640 for battery bus, heater, etc.
- As an example, the cost of this subsystem for a satellite with 32 modules will be about $473,870 plus 1,224 hours of direct labor

Power Subsystem - Bipolar Batteries

Bipolar batteries are still in the IR&D stage at Space Systems/Loral and therefore, cost figures provided herein are not only proprietary but must be viewed as tentative and only as an order of magnitude indication.

While we presented a 32 module system, the cost, except for some fixed cost such as the power bus, is basically linear with respect to the number of modules in the system. Let M be the number of modules in the power subsystem, the cost C and direct labor hours L can be estimated based on the following formula:

$$C = \$(5640 + 14632M)$$

$$L = 72 + 36M \text{ hours}$$

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Composite Material

- The cost of composite materials surveyed are considerably higher than aluminum that they may replace
  - About $1000 per pound for K1100 and $2,000 per pound for M60J
- In one example of an X-band antenna reflector, there is an estimate 30% saving in mass with a 30% increase in cost
- The application of composite materials is a matter of system level trade
  - In commercial satellites it is dry mass vs. propellant mass, i.e., on orbit life
  - It is rated typically at $30,000 to $40,000 per kg for satellites in Geostationary Earth Orbits (less for Satellites in Low Earth Orbits)
  - Similar considerations may continue for Mars Mission satellites

Composite Material

The application of composite material by itself does not reduce system acquisition cost even as it reduces the mass of the satellite. Its cost benefit can be seen only when traded against launch cost which, as a rule of thumb, is rated at about $50,000/ kg for satellites in Geostationary Earth Orbits and about $7,800/kg for satellites in Low Earth Orbits.

In the example of the MISR, a mass savings of 10 kg was achieved. This translates to a launch cost savings in the neighborhood of $500,000.

OCC Cost Considerations

	FUNCTION	*% AVG. ANNUAL MOS COST		**% AVG. ANNUAL LABOR COST (FTE)	
		LOW %	HIGH %	LOW %	HIGH %
1.	Mission Planning	1	3	2	4
2.	Sequence Development	4	9	7	13
3.	Mission Control	6	9	10	11
4.	Data Transport & Delivery	4	9	5	11
5.	Position Location Planning and Analysis	3	6	4	7
6.	Spacecraft Planning and Analysis	13	20	18	25
7.	Payload Planning and Analysis	3	19	3	14
8.	Payload Data Processing	7	25	6	13
9.	Archiving and Mission Database	0	3	0	3
10.	Systems Engineering, Integration, and Test	3	6	3	7
11.	Computer and Communications Support	3	9	2	5
12.	Development and Maintenance	5	8	5	8
13.	Management	9	11	11	14

* Percentage of annual cost of mission operations by each function.

** Percentage of annual average full-time equivalents by each function.

(Table from Gael Squibb, JPL , 3rd International Symposium on Space Mission Operations and Ground Data Systems)

OCC Cost Considerations

Gael Squibb, in his Functional View of Mission Operations, AIAA Report from the Workshop on Reducing the Cost of Space Operation, has developed some cost considerations, as shown in the viewgraph presented here. It was divided into system cost and labor cost (measured in terms of Full Time Equivalent heads). It identifies a realistic estimate of the average cost, within some range as a percentage of the total cost, for each of the thirteen functions that must be performed in a Mission Operation System.

Potential OCC Cost Savings

FUNCTION	*% AVG. ANNUAL MOS SAVINGS		**%AVG. ANNUAL LABOR SAVINGS (FTE)	
	LOW %	HIGH%	LOW %	HIGH %
1. Mission Planning	1	2	1	3
2. Sequence Development	2	5	2	7
3. Mission Control	2	5	4	6
4. Data Transport & Delivery	0	3	5	9
5. Position Location Planning and Analysis	3	6	2	4
6. Spacecraft Planning and Analysis	5	10	5	20
7. Payload Planning and Analysis	3	10	1	10
8. Payload Data Processing	0	7	1	2
9. Archiving and Mission Database	0	2	0	2
10. Systems Engineering, Integration, and Test	1	2	0	2
11. Computer and Communications Support	0	2	1	2
12. Development and Maintenance	0	1	0	1
13. Management	3	4	5	7

* Percentage of annual cost of mission operations by each function.

** Percentage of annual average full-time equivalents by each function.

(Table from Gael Squibb, JPL, 3rd International Symposium on Space Mission Operations and Ground Data Systems)

SPACE SYSTEMS LORAL

Potential OCC Cost Savings

By automating both onboard and/or ground system, the present viewgraph presents the potential cost savings. Again, it is separated into system cost and labor cost, relative to the total cost, in the same percentage format as in the previous viewgraph.

OCC Cost Summary

- Automation of both space and ground segments is most promising in terms of cost savings for the OCC
- Most significant cost consideration, in increasing order of cost effectiveness, is
  - Spacecraft planning and analysis
  - Payload planning and analysis
  - Payload data processing

OCC Cost Summary

Based on the generic cost model, it is clear that overall cost savings can be expected by automating both the space and the ground segments. More than that, however, the previous two viewgraphs suggest automating some of the thirteen functions can be more cost effective than automating. Spacecraft Planning and Analysis function is the first priority since both the system cost the potential cost savings are high, in percentage terms. Payload Planning and Analysis function is the second as it has similar cost and savings characteristics.

It should be pointed out, however, that the previous two viewgraphs represent the cost and potential cost savings for a generic Mission Operation System and is not customized to Mars Missions, let alone the Mars Relay Satellite operation. Nonetheless, the range of percentages are believed to be realistic. Further analysis will be required to to develop a realistic cost saving strategy for MRS.

Conclusions

- Most of the small satellites by both DoD and NASA weighs over 100 kg, some even more than 400 kg
  - Majority between 100 to 400 kg peaking in the range between 100 to 400 kg
- Only three small satellites weigh less than 100 kg
  - Surfsat-1 (Summer Undergraduate Research Fellowship Satellite series 1) by NASA
  - Orbcomm that provides messaging service (4800 bps)
  - DoD Stacksat
- Large commercial satellite vendors such as Space Systems/ Loral, Lockheed Martin, TRW, do not seem to build any satellites that weighs less than 400 kg, let alone in the 50 kg range, such as the MRS studied by Stanford Telecommunication Inc. and OSC

Conclusions (Cont'd)

- Generally, cost data cannot be obtained as it is proprietary information
  - The GlobalStar™ bus cost seems, at this level, compatible with the estimate provided by the MRS report
  - Adaptability of the GlobalStar™ bus for Mars mission of course requires a separate investigation
- Miniaturization efforts at JPL for the UHF and X-band relay were in the requirement definition phase, at least as of last year
  - Potential system cost savings depend not so much on new technology such as miniaturization but on the market demand on the technology

Conclusions (Cont'd)

- Two propulsion subsystems requiring different levels of power capabilities were included in the survey
  - PPT is more appropriate for the small satellite (such as the MRS) with about 100 W of power and only routine mission requirement
  - SPT more appropriate for the larger satellites (such as the GlobalStar™ bus) with about 500 W of power and more demanding mission requirements
- The use of composite materials must be part of a system cost trade (including launch cost) since the reduction in satellite mass is associated with a a higher subsystem cost

Conclusions (Cont'd)

- Bipolar batteries offer an alternative to existing power subsystem though its cost estimate must be viewed with caution as it is still in R&D stage
- Automating the "Spacecraft Planning and Analysis" and the "Payload Planning and Analysis" functions will be the most effective way to achieve cost savings

Conclusions (Cont'd)

- The present work found only several autonomous satellites:
  - An Air Force satellite program titled Technology for Autonomous Operational Survivability, its payloads are manifested on a Space Test Experiment Platform (STEP) spacecraft bus
  - STEP-0 (Satellite Technology Education Program)
  - GlobalStar^{TM} bus is not in operation yet but was specifically designed for autonomous operation for cost savings
- STEP satellite has two automated navigation systems:
  - A six-channel Global Positioning System receiver, and
  - Two spinning sensors that provide Earth/Sun/ Moon position measurements as inputs to the autonomous navigation software
- For Mars Mission, the latter is clearly more appropriate

Conclusions (Cont'd)

- Potential cost savings from the Ground Operations investigation strongly suggests MRS PO should track satellite autonomy technology especially in the absence of empirical data to date
- Preliminary cost comparison between the GlobalStar™ bus and the MRS cost estimate strongly suggests the cost effectiveness of a Mars Relay Satellite may not necessarily lie in a one of a kind micro-satellites but the use of commercial production busses
- The push for sample return and the recent loss of the Russian Rovers may result in some uncertainty and the replanning of many Mars Exploration programs, in the long run, it will enhance the requirements to have in place a number of MRS
  - Closer coordination between the Mars Exploration Program Office & MRS will be required

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13. ABSTRACT ***(Maximum 200 words)***

This reports the results of a general study for NASA Lewis in relation to the use of small satellites for a Mars Relay Satellite (MRS) that supports communications between Mars and Earth: commands to, and telemetry from, Mars Landers and Rover. The scope of the study encompasses a survey of small satellites, those that are lower than 800 kg in mass, by NASA, DoD, and Commercial companies. Additionally, surveys in advanced technologies in the area of composite materials, propulsion subsystems, battery subsystems, communications components and subsystems, and ground operations are also provided. A summary of NASA Mars Programs and their status as relevant to MRS is also included. Attempts to draw detailed cost conclusion is generally not possible due to its proprietary nature. In any event, cost is driven by market demands rather than new technologies. A preliminary comparison with the cost estimate of the S-Tel/OSC report did suggest the possibility cost savings for the MRS by the use of production busses. On the other hand, cost savings in normalized terms from the use of automated ground systems was obtained with some degree of details.

14. SUBJECT TERMS
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