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THE ORBITAL DEBRIS DETECTOR CONSORTIUM: SUPPLIERS OF INSTRUMENTS FOR *IN-SITU* MEASUREMENTS OF SMALL PARTICLES IN THE SPACE ENVIRONMENT

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SUMMARY

The Industry and University participants listed above have joined together to form the IMPA:Ct consortium (In-situ Monitors of the Particulate Ambient: Circumterrestrial) which offers a broad range of flight qualified instruments for monitoring the small particle (0.1 micron to 10 cm) environment in space. Instruments are available in 12 months or less at costs ranging from 0.5 to 1.5 million dollars (US) for the total program. Detector technologies represented by these groups are: impact-induced capacitor-discharge (MOS, metal-oxide-silicon), cratering or penetration of electroactive thin film (polyvinylidene fluoride (PVDF), impact-plasma detection, acoustic detection, ccd tracking of optical scatter of sunlight, and photodiode detection of optical scatter of laser light. The operational characteristics, general spacecraft interface and resource requirements (mass/power/ telemetry), cost and delivery schedules, and points of contact for 7 different instruments are presented.

INTRODUCTION

The IMPA:Ct consortium was formed in response to customer information needs. All consortium members offer flight-qualified instruments. (New members are actively sought, just contact any consortium member if your group has, or is developing, a flight qualified system.) Our intention is to present information on existing technologies in a useful, customer-friendly format that will encourage further contact with consortium members for more detailed information. Customers should find this paper a useful resource for evaluating existing technologies that can meet their mission-specific requirements. The seven flight-qualified instruments described in the following pages are available in relatively short time periods (<12 months) and for modest total mission costs (U.S.$ 0.5 to 1.5 million). Instrument configurations and capabilities range from ultra-thin, surface mountable capacitor-discharge sensors that detect submicron and larger particle impacts, to briefcase size optical scatter instruments that scan large volumes of space for particles in the mm to 10 cm size range. Most of the participating groups are also involved with research and development programs aimed at upgraded and/or hybrid technologies.

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Data on the small particle environment produced by consortium instruments can be displayed in a variety of formats, including multi-dimensional phase-spaces relevant to specific missions. The data can also be used to predict the near-term future particle environment that spacecraft will experience, including short-term, high-flux time periods associated with debris ring and wake trail crossings. [LDEF data showed that nearly 2/3 of the estimated 7 million microparticle impacts on the ram, north and south sides of the spacecraft occurred in less than 1% of the mission time at predictable intervals associated with debris-ring crossings (refs. 1, 2). Data from particle monitors on board Space Shuttle flights have detected a 100 fold increase in the small particle flux associated with the crossing of a wake trail behind an old, spent Soviet (liquid-fueled) upper stage (ref. 3).] Information on the particle environment can be used by operators to minimize damage to sensitive surfaces or institute operational alerts. This situation is entirely analogous to the first use of information from meteorological instruments by aircraft operators. Indeed, a near term goal of present-day military spacecraft operators is the development and deployment of small, lightweight, non-intrusive sensors for monitoring a host of space environment conditions that affect the operations and degradation of their systems.

A consortium affiliate, POD Associates, Inc. of Albuquerque, NM [Contact: Mr. Dale Atkinson (505) 243-2287, FAX (505)243-4677] offers additional pre- and post-mission analytic support for sensor systems and data that includes: incorporation of spacecraft subsystem damage criteria and mission success and/or degradation (probability) criteria into environment predictions; comparison of new data sets with standard and current environment subsystem damage models; and incorporation of new data sets into improved particle environment models.

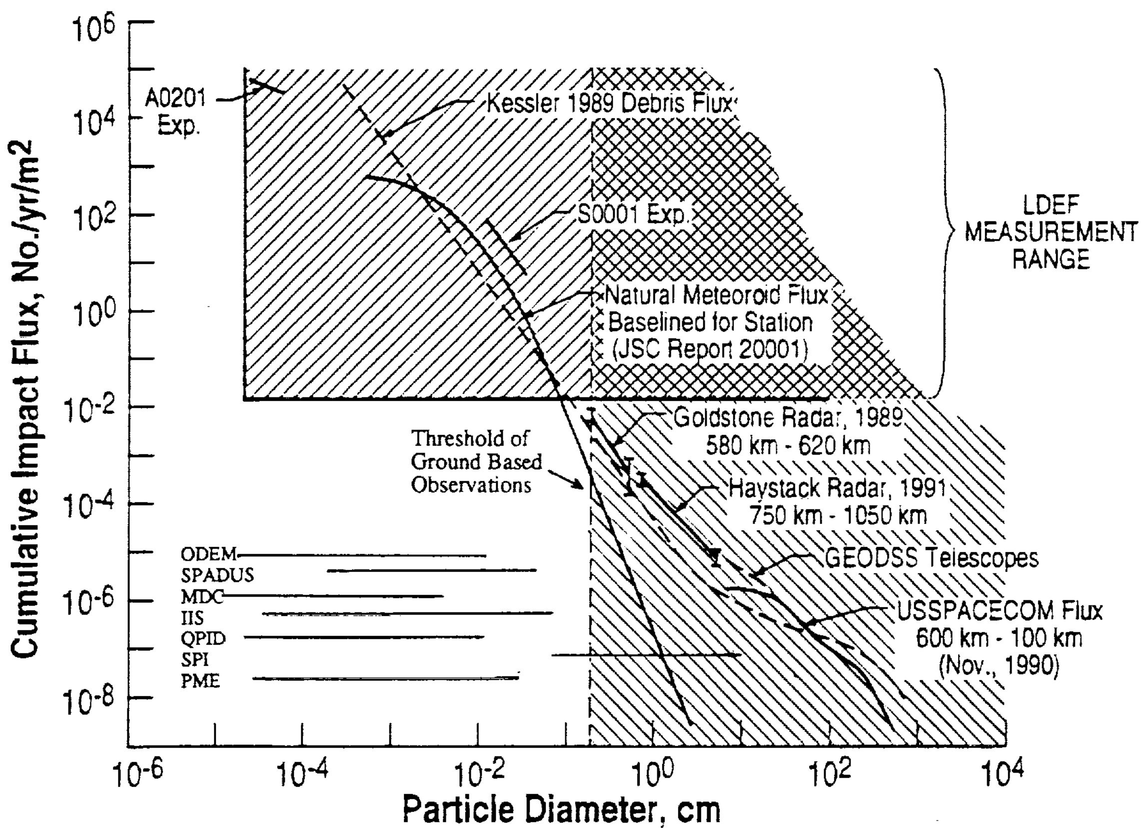


Figure 1. Present knowledge of the circumterrestrial meteoroid and debris environment based on LDEF and ground-based radar and optical data. Particle size detection ranges for all 7 of the IMPA:Ct consortium instruments are shown just above the x axis in this plot. All instruments are capable of detecting the maximum annual flux of particles in their respective detection ranges. Note the broad ranges and high degree of overlap of these instruments.

SENSOR TECHNOLOGY: *MOS (metal-oxide-silicon) capacitor-discharge*

Instrument Generic Name:	ODEM-MOS, (Orbital Debris Environment Monitor)
Detection Method:	impact on electrically active surface induces discharge
Particle Size Detection Range:	0.1 -100 μm (10^{-15} to 10^{-6} g), no size discrimination except for variable threshold sensitivities
Field of View:	180°, dependent on mounting location
Consortium Contact:	Jim Wortman, North Carolina State University Phone: (919) 515-5255 Fax: (919) 515-3027

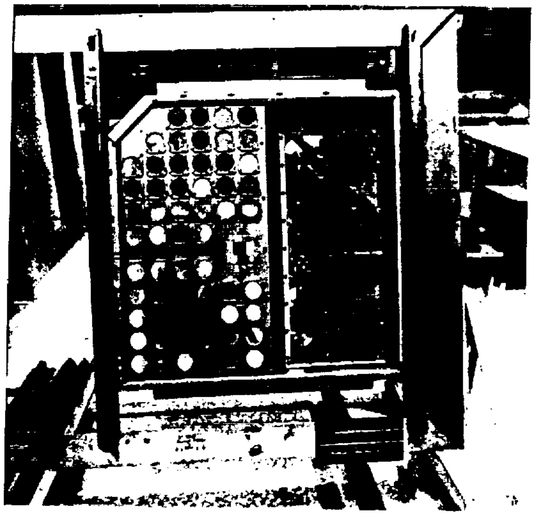


Figure 2. One of 6 MOS sensor arrays that flew on board LDEF for 5.8 years, from April 1984 to Jan. 1990 as the Interplanetary Dust Experiment.

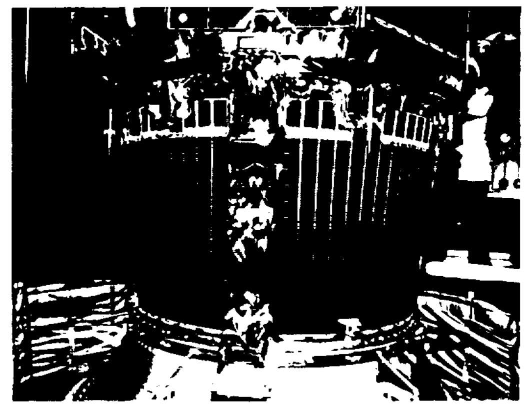


Figure 3. The Orbiting Meteoroid and Debris Counter on the Clementine-1 Interstage Adapter has 48 current-generation MOS sensors on board (Jan 1994- Apr 95).

Nominal Instrument Specifications

dimensions	Sensors: 38 x 76 x 0.05 mm, Electronics: 10 x 20 x 2 cm
active sensing area	3000 cm^2, nominal, variable up to the limit of available spacecraft mounting area including solar panel back sides
mass	<1 kg, includes sensors, electronics and wiring harness
spacecraft electrical interface	compatible with all buses
mounting requirements	sensors are bonded to rigid surfaces using solar cell mounting techniques: electronic controller board can be mounted in an existing box, or can have its own box (add 0.5 Kg mass)
power requirements	2 watts
telemetry requirements	no special requirements, data rate <250 kbit/day

Nominal cost estimates (U.S. $1000)

delivered hardware (only) costs	400-600
pre-launch mission support costs	TBD, dependent on program/documentation requirements; typical range is 250-400.
software development costs	TBD, dependent on phase space requirements; typical range is 25-100.
post-launch mission support (data interpretation and report) costs	TBD, dependent on requirements; typical range is 100-150/year.

Nominal delivery schedule

hardware (only) delivered	3-6 months
mission specific software development time	1-3 months (concurrent)

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

The MOS capacitor-type impact detectors, originally designed and fabricated by J. J. Wortman (co-inventor) at North Carolina State University, have flown successfully on the Explorer 46 Meteoroid Technology Satellite (MTS) in 1977, and on LDEF in 1984-1990 as the Interplanetary Dust Experiment, which discovered the extreme anisotropic nature of orbital debris and tracked several debris clouds and rings (Fig. 4)[1,4-7]. An MOS-based instrument called the Orbiting Meteoroid and Debris Counter (OMDC) is currently operating on the BMDO Clementine-1 interstage adapter (Fig. 3)[8].

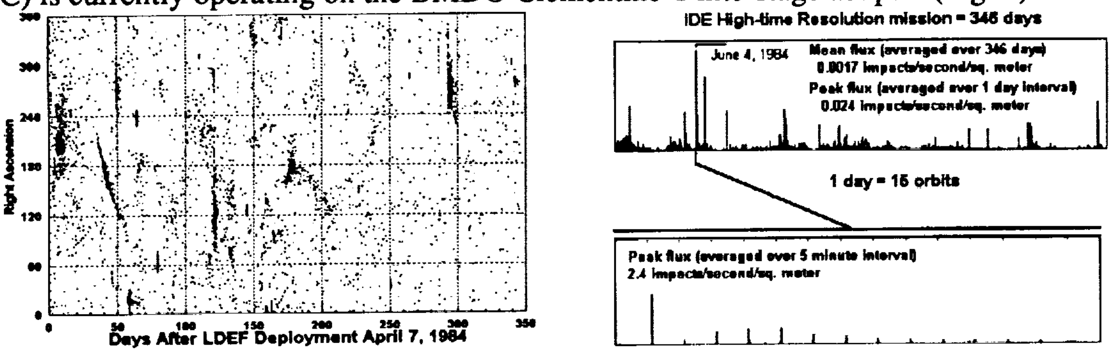


Figure 4. Impact flux data from IDE detectors on the ram side of LDEF showing examples of displays and the orbital periodicity of debris ring intercepts. Evidence indicates a continuous source for these rings.

The MOS detector (or sensor) is based on the fact that a charged parallel plate capacitor will partially discharge when struck by an energetic enough particle. The impact is detected by measuring the voltage transient due to the discharge. Transient power consumption due to impacts is on the order of milliwatts. A schematic drawing of the detector (Fig. 5) shows that the capacitor consists of a thin top metal electrode layer, a thin dielectric layer of silicon dioxide, and a bottom substrate, or electrode, of p-type silicon. They are fabricated utilizing standard silicon integrated circuit manufacturing processes and photo lithography. Threshold sensitivity of the detectors is dependent on the top electrode and dielectric layer thicknesses. Dimensions and shape can be varied from 2.5 cm squares to 10 cm diameter rounds. Sensors are typically bonded directly to rigid external spacecraft surfaces using solar-cell mounting techniques. Extensive ground simulation tests and orbital empirical calibration studies[9-11] have shown that the MOS detector sensitivity can be expressed as a function of particle mass in grams, velocity in m/s and impact angle (ß) in degrees from normal. These data indicate that the MOS detectors respond to submicron Fe particles with a lower size limit of 0.1-0.2 µm (~10^{-14} to 10^{-15}g) for impact velocities ≥10 km/s (Fig. 6).

$$\mathbf{S=[(mass)^{0.33}(velocity)(\cos\beta)^{1.5}][e^{(-0.90T)}] - 0.13}$$

S>0, discharge; S<0, no discharge

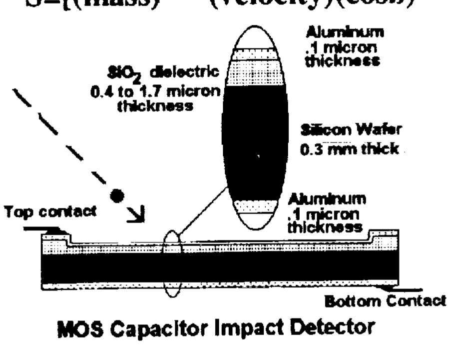


Figure 5. The current generation MOS sensor.

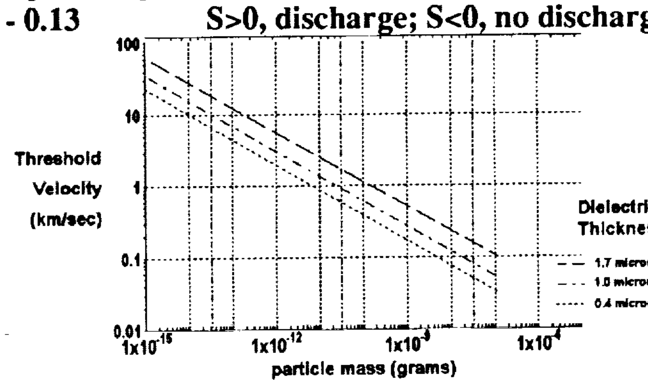


Figure 6. MOS detector threshold response curves.

New Developments

(1) Combined MOS/PVDF array for detection of submicron particles and measurement of size/velocity/trajectory of particles >10 µm (under development: NASA/LaRC, U. Chicago, NCSU, ISST, Visidyne).
(2) Hybrid MOS/Acoustic sensor array for satellite health monitoring and impact detection. Senses all induced vibrations on a spacecraft and, using acoustic spectroscopy, provides information on spacecraft subsystem performance and impactor sizes, velocities, angles, and locations. (C.G. Simon, inventor)

SENSOR TECHNOLOGY: *PVDF Electroactive Thin Film*

Instrument Generic Name:	SPADUS, Space Dust Monitor
Detection Method:	impact on or penetration of electrically active thin films
Particle Size Detection Range:	2-200 μm (10^{-12} to 10^{-5} g); measures size, velocity (±5%), trajectory (±7°) of particles >10 μm with 2 parallel arrays
Field of view:	180° for flux, 120° for trajectory
Consortium Contact:	Tony Tuzzolino, University of Chicago Phone: (312) 702-7798 Fax: (312) 702-6645

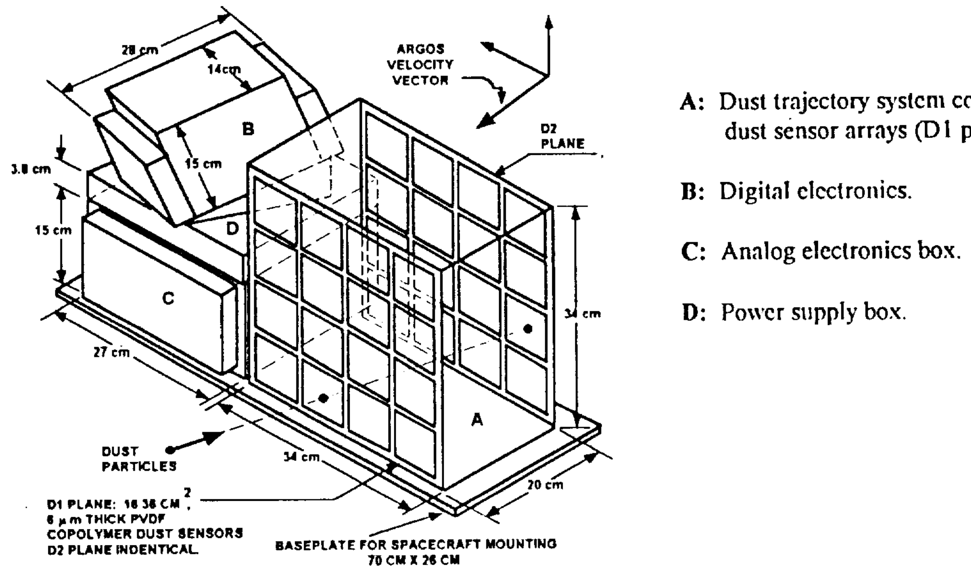


Figure 7. The Space Dust Instrument (SPADUS) for flight on the ARGOS spacecraft (launch, Jan. 1996).

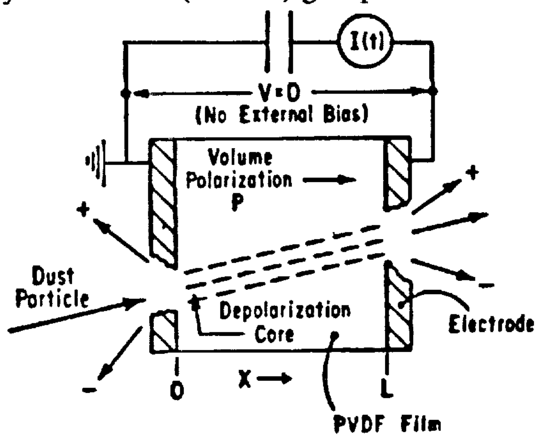
Nominal Instrument Specifications	
dimensions	Sensor Module: 34 x 34 x 20 cm Electronics Module: 30 x 27 x 20 cm
active sensing area	1200 cm², nominal, variable up to the limit of available spacecraft mounting area
mass	8 kg, includes sensors, electronics and wiring harness
spacecraft electrical interface	compatible with all buses
mounting requirements	external rigid mount required for box-shaped sensor array Electronic controller box can be mounted external or internal
power requirements	6.5 watts
telemetry requirements	no special requirements, data rate 1-10 Mbit/day

Nominal cost estimates (U.S. $1000)	
delivered hardware (only) costs	500-750
pre-launch mission support costs	TBD, dependent on program/documentation requirements; typical range is 250-500.
software development costs	TBD, dependent on phase space requirements, typical range is 50-150
post-launch mission support (data interpretation and report) costs	TBD, dependent on requirements, typical range is 100-150/year

Nominal delivery schedule	
hardware (only) delivered	9-12 months
mission specific software development time	3-6 months (concurrent)

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

PVDF dust sensors consist of 2-28 μm thick polarized (dipole-aligned) PVDF (or PVDF copolymer) foils mounted in circular or square frames. Vapor deposited aluminum films over the mounted foils serve as electrodes. When an impacting particle removes material due to cratering or penetration, a signal is generated which depends on the particle mass and velocity (Figs. 8, 9). PVDF sensors flew on the VEGA 1 and 2 spacecraft which encountered comet Halley in 1986 where they made the first direct measurements of the dust structure of a cometary coma (refs. 12-15). The current SPADUS instrument uses 5 μm thick copolymer sensors (36 cm^2) grouped in two 16-sensor arrays spaced 20 cm apart (Fig. 7, ref. 16).



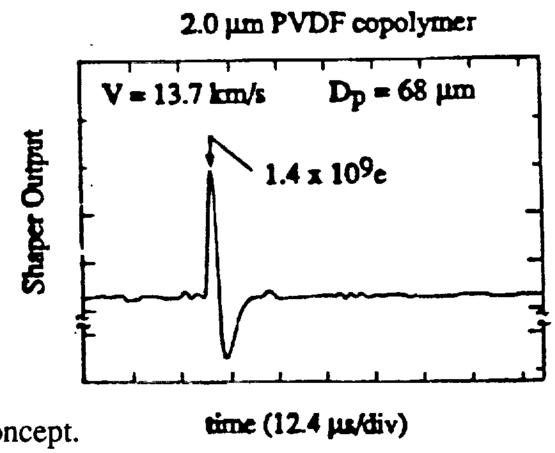


Figure 8. The polyvinylidene fluoride (PVDF) dust sensor concept.

For each impact , SPADUS data include:

- Impact time on D1 for correlation with spacecraft clock to obtain spacecraft attitude and orbital position,
- Identification of D1 sensor and D2 sensor impacted to determine particle trajectory (±7%),
- Time of flight between D1 and D2 (0.25 μs resolution) to determine particle velocity,
- Pulse-height-analysis (PHA) of D1 and D2 signal amplitudes (32 channel PHA for each of the 32 sensors plus signal waveform storage with 2000 time points at 256 channel PHA/point) to determine particle mass.

These data permit discrimination between orbital debris and natural particles. The time-velocity-trajectory capabilities of SPADUS also permit identification of orbital debris cloud and meteor stream encounters.

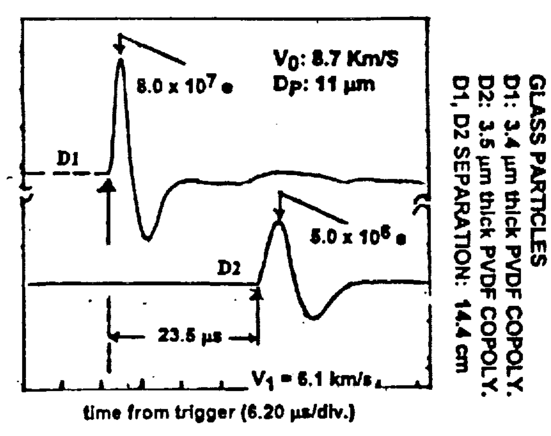


Figure 9. PVDF copolymer response curve.

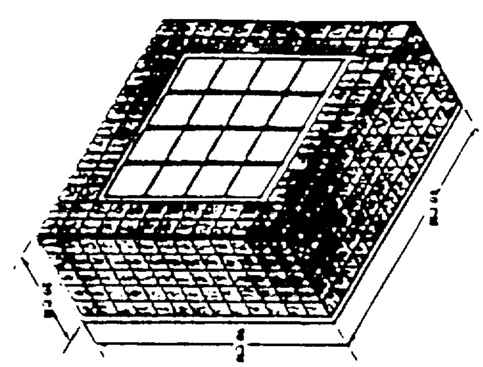


Figure 10. Combined MOS/PVDF instrument (IMOD).

New Developments

The University of Chicago has teamed with NASA/LaRC, the University of North Carolina, Visidyne, Inc. and ISST, Inc. to construct a combined MOS/PVDF instrument (Fig. 10) for detection of submicron to mm size particles and measurement of size/velocity/trajectory of particles >10 μm in size. This combined instrument is known as IMOD (*In-situ* Monitor of Orbital Debris).

SENSOR TECHNOLOGY: *Impact-Plasma Detection*

Instrument Generic Name:	MDC, Munich Dust Counter
Detection Method:	measures electron and ion charges and times of arrival to electrodes from impact-plasmas generated from particles striking gold surfaces
Particle Size Detection Range:	0.1 to 50 μm (10^{-15} to 10^{-7} g), measures mass (±430%), velocity (±150%), and trajectory (±70%)
Field of view:	140°
Consortium Contact:	Ralf Münzenmeyer, Technical University of Munich Phone:(49) 89-2105-2578 Fax:(49) 89-2105-2468

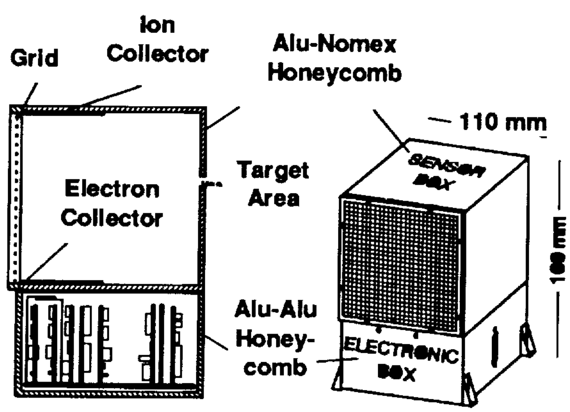


Figure 11. Outline of MDC mechanical design.

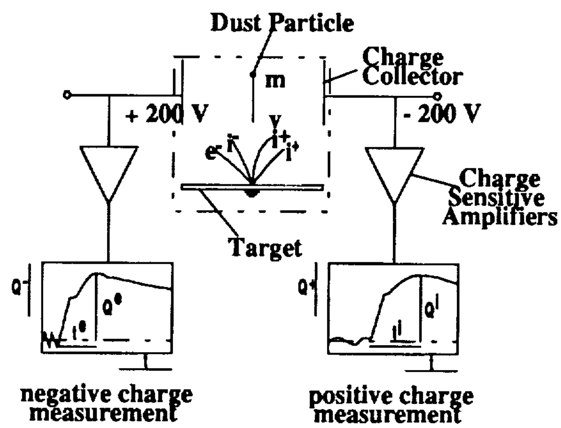


Figure 12. MDC measurement principle.

Nominal Instrument Specifications

dimensions	11 x 11 x 16 cm
active sensing area	100 cm^2
mass	0.6 kg, includes sensor, electronics and internal wiring
spacecraft electrical interface	compatible with all buses
mounting requirements	external or internal rigid mount required for box-shaped combination sensor and electronic controller
power requirements	2 watts
telemetry requirements	no special requirements, data rate 1-10 kbit/day

Nominal cost estimates (U.S. $1000)

delivered hardware (only) costs	TBD
pre-launch mission support costs	TBD
software development costs	TBD
post-launch mission support (data interpretation and report) costs	TBD

Nominal delivery schedule

hardware (only) delivered	9-12 months
mission specific software development time	3-6 months (concurrent)

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

The Munich Dust Counter consists of two honeycomb boxes that are typically mounted together (Fig. 11). The lower box houses all electronics and is fixed to the satellite platform. The upper box serves as the particle detector, or "sensor", and has a 100 cm^2 electrically grounded open steel grid at the front. The five gold plated inner sides, including electron and ion collector areas shown in Fig. 11, serve as target areas. The MDC measures the electrical charges generated by high velocity particle impacts into the gold surfaces. The amplitude and rise time of each charge signal are used to determine the mass and velocity of the particle using the empirical relationships:

$$t = (c_g)(v^n) \qquad \pm Q/m = (c_r v^{\beta})$$

where m is particle mass, v is velocity, Q is the maximum charge, t is the signal rise time, and c_r, c_g, n and ß are calibration constants (refs. 17, 18, Figs. 13 and 14).

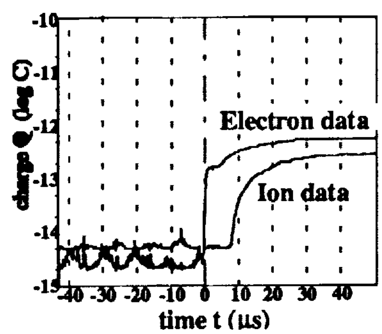


Figure 13. A typical impact induced plasma signal detected by the MDC.

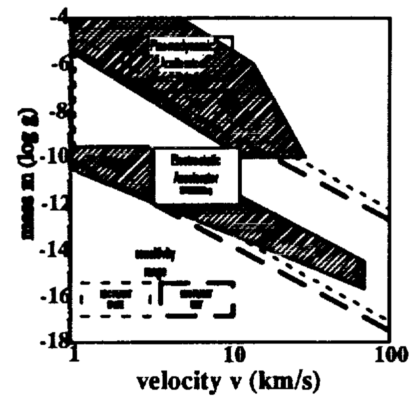


Figure 14. Sensitivity range of the MDC and particle accelerators used in calibration.

The MDC plasma dust sensor first flew on the Japanese MUSES-A (HITEN) Spacecraft in 1990-1993 (refs. 19, 20) where it measured the flux of natural particles in cislunar space and beyond. This mission provided valuable data on the mass, velocity and distribution of cosmic dust particles and ß-meteoroids in the Earth-Moon system. Swarms, groups and random particles were detected and enormous variations in the instantaneous fluxes and impact rates were observed. Measurements of dust flux at the Lagrangian points L4 and L5 showed no indications of the presence of dust clouds.

A second MDC is currently flying in low Earth orbit (400 km, 58°, circular) on board the BREM-SAT, (Fig. 15), a microsatellite dedicated to dust detection. BREM-SAT was designed and constructed at the Center for Applied Space Technology and Microgravity (ZARM) in Bremen, Germany. It was launched from a Space Shuttle Getaway Special canister in February 1994 with a nominal mission duration of 2-6 months (ref. 20). Calculations using the ESABASE Meteoroid / Debris software predict that 360 meteoroids and 600 debris particles >0.5 µm in size will strike the BREM-SAT MDC sensor per year.

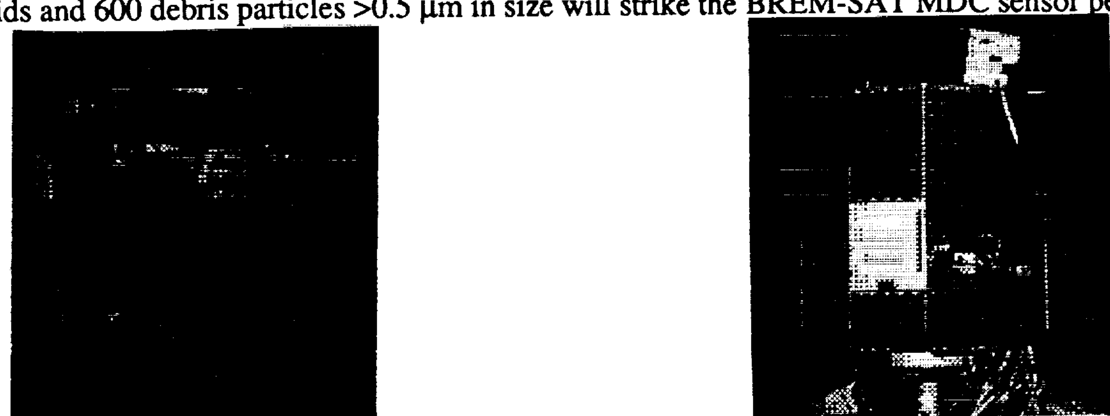


Figure 15. The BREM-SAT microsatellite for meteoroid and debris detection in low Earth Orbit.

SENSOR TECHNOLOGY: *Acoustic Impact Detection*

Instrument Generic Name:	IIS, Impact Impulse Sensor
Detection Method:	detects impact induced phonon displacements; derives impactor size and velocity of particles from momentum
Particle Size Detection Range:	0.5 μm to 1 mm (10^{-13} to 10^{-3} g), measures momentum to ±20%
Field of view:	160°
Consortium Contact:	Bill Tanner, Baylor University Phone: 817-755-3404 Fax: 817-755-3409

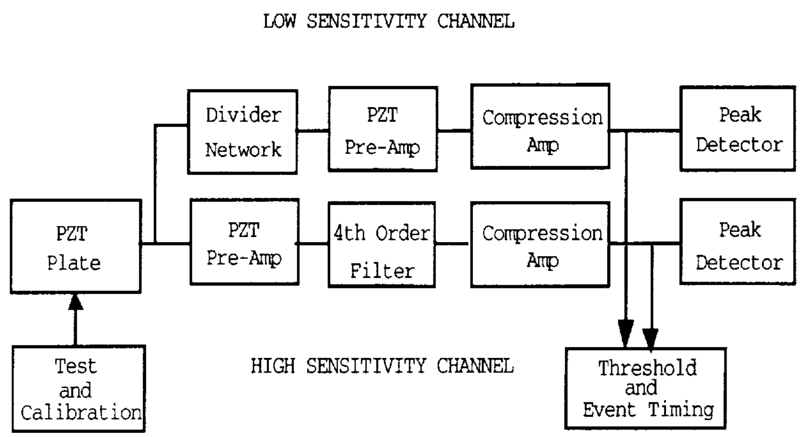


Figure 16. Impact Impulse Sensor functional block diagram. PZT = lead-zirconate-titanate sensor.

Nominal Instrument Specifications

dimensions	Sensors: 2 x 10 x 10 cm Controller: 2 x 14 x 16 cm
active sensing area	100 cm^2/sensor, up to 8 sensors (800 cm^2) per controller
mass	5 kg, includes 8 sensors with housings, controller and wiring harness
spacecraft electrical interface	compatible with all buses, (1553 standard)
mounting requirements	sensors - external rigid mount controller box - internal or external rigid mount
power requirements	10 watts (8 sensor array)
telemetry requirements	no special requirements, data rate 1-8000 bps

Nominal cost estimates (U.S. $1000)

delivered hardware (only) costs	300-400
pre-launch mission support costs	TBD, dependent on program/documentation requirements; typical cost is 300
software development costs	TBD, dependent on phase space requirements; typical cost is 50
post-launch mission support (data interpretation and report) costs	TBD, dependent on requirements; typical cost is 75/year.

Nominal delivery schedule

hardware (only) delivered	6-8 months
mission specific software development time	2-4 months (concurrent)

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

First proposed in the pre-satellite era, acoustic detectors later comprised the main thrust of early sensor development in the U.S. The piezoelectric sensor has high mass sensitivity at medium velocities and outperforms other impact related sensors. The reliability of early piezoelectric sensors was questioned due to the electronic controller limitations of the era. Current high-speed analog to digital (A/D) converters and CPU's have eliminated these earlier difficulties. A piezoelectric acoustic sensor system called DIDSY (The Dust Impact Detection SYstem) was included as one of ten major components of the Giotto Comet Halley scientific payload (ref. 21). Several piezo sensors have also flown on Shuttle missions (ref. 22).

Lead zirconate titanate (PZT) piezoelectric crystal sensors have a threshold momentum sensitivity of 5 x 10^{-11} Ns, which implies that an ~10 picogram grain (density = 0 8 g/cm^3) with a diameter of ~2.9 μm and a velocity of 5 km/s will be detected. The PZT responds only to the normal component of the impulse delivered to the impact surface regardless of the obliquity of the impact.

The Impact Impulse Sensor consists of a PZT crystal bonded to a thin plate. Dust particles striking the plate give rise to acoustic waves which propagate radially from the point of impact, inducing transient charge separations in the PZT crystal. A charge-sensitive amplifier is used to detect these waves and the first few hundred μs of the analog signal are routed through a high-speed A/D converter and stored for processing (Fig 16.). The maximum PZT response is proportional to the particle momentum, and the slope and rise-time of the first acoustic wave is related to the size and velocity of the particle (ref. 23). The dynamic range of an IIS covers six orders of magnitude (Figs. 17, 18).

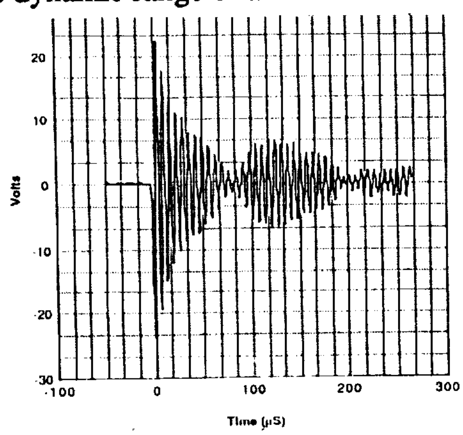


Figure 17. A typical impact-induced acoustic wave detected by the IIS.

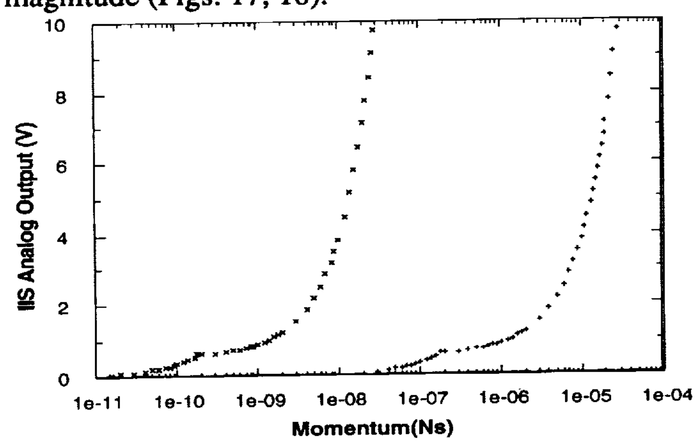


Figure 18. Peak amplitude versus momentum relationship for high and low sensitivity IIS .

New Developments

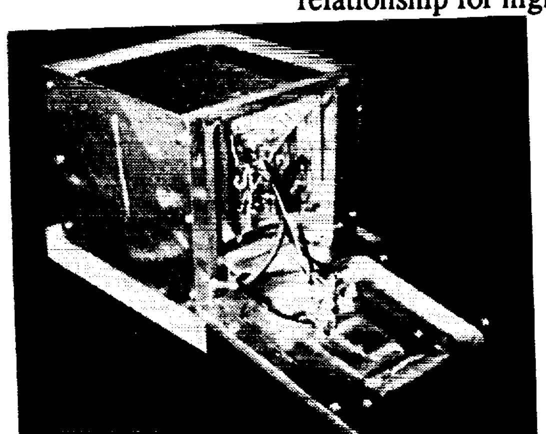
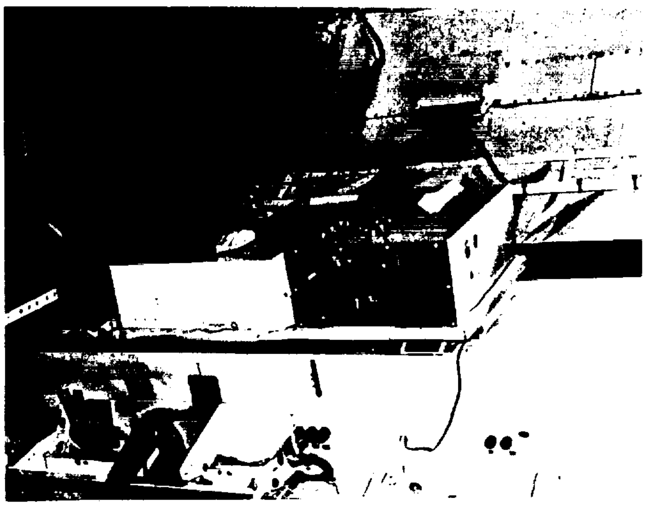


Figure 19. The Cometary Dust Environment Monitor experiment (CoDEM), selected for flight on the CRAF/Cassini mission, uses piezoelectric and plasma-charge sensor arrays to measure mass, velocity and trajectory of particles. Contact Bill Tanner for more information on this highly capable instrument.

SENSOR TECHNOLOGY: ***Acoustic Impact Detection***

Instrument Generic Name:	QPID, Quartz Particle Impact Detector
Detection Method:	impact on electrically active surface
Particle Size Detection Range:	0.4 μm to 100 μm (10^{-12} to 10^{-3} g), measures momenta ranging from 6 x 10^{-13} to 6 x 10^{-8} kg-m/s
Field of view:	150°
Consortium Contact:	Carl Maag, T&M Engineering, Inc. Phone: (818) 852-9772 Fax: (818) 335-9968



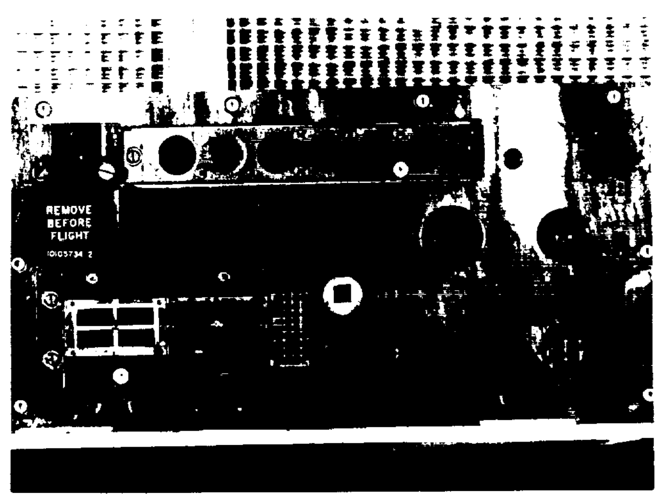


Figure 20. QPID detector mounted in STS-44/IOCM. Figure 21. Close-up view of QPID on STS-44.

Nominal Instrument Specifications

dimensions	Detector: 20 x 9 x 10 cm, Electronics: 12 x 10 x 10 cm
active sensing area	64 cm^2 (8 cm^2/sensor, 8 sensors/detector)
mass	1.5 kg, includes detector package, electronics and wiring harness
spacecraft electrical interface	compatible with all buses
mounting requirements	external or internal rigid mount
power requirements	3 watts
telemetry requirements	no special requirements, data rate 100-200 kbit/day

Nominal cost estimates (U.S. $1000)

delivered hardware (only) costs	200-250
pre-launch mission support costs	TBD, dependent on program/documentation requirements; typical range is 100-150
software development costs	TBD, dependent on phase space requirements; typical range is 50-150
post-launch mission support (data interpretation and report) costs	TBD, dependent on requirements; typical range is 100-150/year.

Nominal delivery schedule

hardware (only) delivered	4-6 months
mission specific software development time	2-4 months (concurrent)

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

The utility of the Quartz Particle Impact Detector (QPID) is based on the piezoelectric response of quartz when struck by a dust particle. The advantages of the QPID include direct counting (as with all impact detectors) and the avoidance of some difficulties associated with accumulation sensors such as collection efficiency, spallation of detector material, release of previously collected material and saturation. The QPID measures the amplitude of the impact induced oscillation of a quartz crystal. Timing the decay of the "ringing" against the crystal frequency eliminates the need for fast pulse electronics.

The fundamental component of the QPID is a Y-cut quartz crystal detector. When impacted, the crystal rings at its base frequency with an amplitude dependent on the momentum transfer. The damping of the crystal and its mounting cause the ringing to decay. By counting the cycles above a fixed amplitude threshold with a scalar, the momentum of the impacting particle may be determined (after proper calibration). The impacts are counted by a separate scalar. When the momentum scalar is replaced by a multichannel analyzer, the momentum spectrum may be obtained directly.

This detector is capable of measuring momenta ranging from 6×10^{-13} to 6×10^{-8} kg-m/s. The high momentum channel also includes counts with momenta greater than the upper limit. A typical system can process at least 100 impacts per second. This limit is set by the duration of the ring for the largest momentum discreetly measured. In terms of Divine's model for comet Kopff (ref. 24) at 100 km distance sunward of the comet at perihelion, a single detector could count dust particles as small as 3×10^{-15} kg (at 295 m/s) and as large as 1×10^{-9} kg (at 73 m/s).

Figure 22. QPID detectors mounted on Galileo.

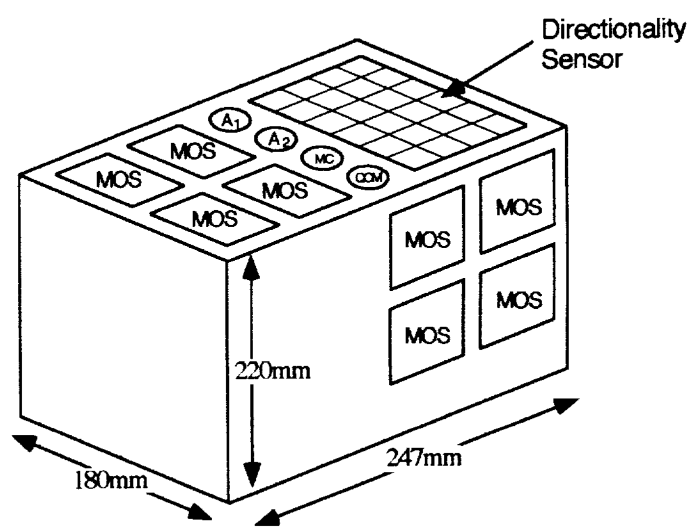


Figure 23. The COMRADE instrument for MIR.

QPID is a flight qualified detector. It has flown on two Shuttle missions for low earth orbit (LEO) debris detection. It aided in determining the scope of the debris cloud encountered during the STS-44 collision avoidance maneuver (ref. 3). In addition to these flights, an earlier version flew on the Galileo mission. The sensors were located on the body and on the RTG outriggers.

New Developments

An instrument for the detection of cometary dust has been accepted for integration onto the Priorda module of MIR. The instrument is known as COMRADE (Collection of Micrometeoroids, Residue and Debris Experiment). The instrument contains both active and passive sensors (Fig. 23).

SENSOR TECHNOLOGY: *CCD Tracking of Scattered Sunlight*

Instrument Generic Name:	SPI, Space Particle Imager
Detection Method:	CCD detection of sunlight scattered from particles
Particle Size Detection Range:	0.1 to 10 cm, measures size and orbital parameters
Field of view:	30° conical FOV, 4 sensors can be grouped to give 60° FOV
Consortium Contact:	R. A. (Bob) Skrivanek, Visidyne, Inc. Phone: (617) 273-2820 Fax: (617) 272-1068

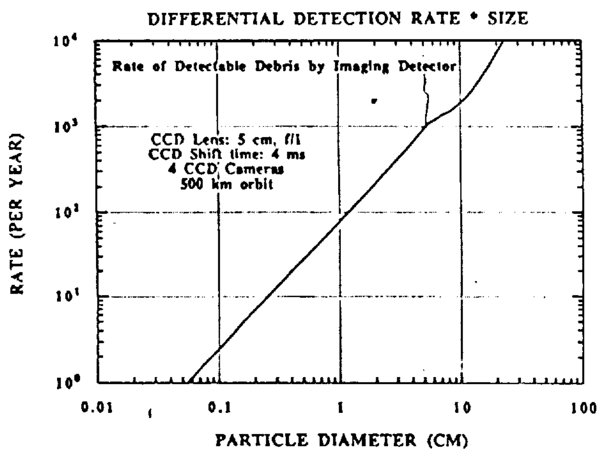


Figure 24. SPI detection rate.

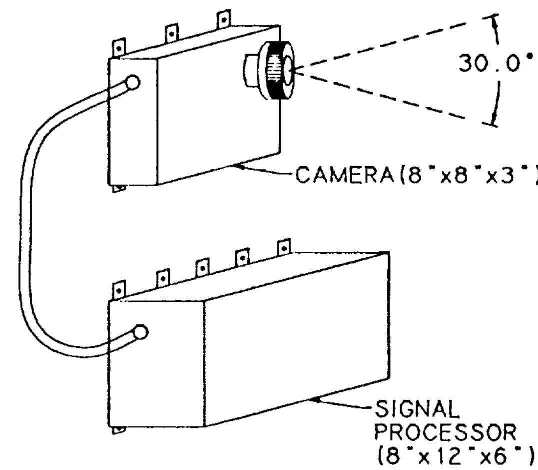


Figure 25. SPI debris detector schematic.

Nominal Instrument Specifications

dimensions	Sensor Module: 16 x 21 x 31 cm Electronics Module: 16 x 21 x 31 cm
active sensing volume	sensing volume is dependent on particle size; larger particles can be seen at very great distances (Fig. 24)
mass	16 kg, includes sensor, electronics and wiring harness
spacecraft electrical interface	compatible with all buses
mounting requirements	external, forward looking (into RAM) mount for sensor module; external or internal mount for electronics box
power requirements	35 watts
telemetry requirements	1.5×10^7 bits per orbit, 40 kbps nominal/100 kbps maximum.

Nominal cost estimates (U.S. $1000)

delivered hardware (only) costs	500
special quality assurance reqmnts.	TBD
special spacecraft interface reqmnts	TBD
pre-launch mission support costs	TBD, dependent upon mission requirements, designated spacecraft, schedule and required documentation; typical range is 250-500.
software development costs	TBD, dependent on requirements; typical range is 100-200
post-launch mission support (data interpretation and report) costs	TBD, dependent on requirements; typical cost is 200/year.

Nominal delivery schedule

hardware (only) delivered	12 months
software development	3-6 months (concurrent)

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

The Visidyne Space Particle Imager has been designed to detect particles in the 0.1 to 10 cm size range, a size range that has the potential for causing serious impact associated problems for any spacecraft and for which there is very little data concerning fluxes. The Visidyne instrument uses a charge-coupled-device (CCD) focal plane array detector that monitors the conical volume in the RAM direction of the spacecraft looking for the sunlight scattered from debris or meteor particles as they pass through the active volume.

The flight instrument consists of two modules, a sensor module that houses the lens, CCD focal plane and pre-processing electronics; and the electronics module that holds the microprocessor. The sensor module must be externally mounted such that the lens has an unimpeded view of the RAM direction. The electronics module can be mounted anywhere within or on the spacecraft.

The instrument measures the sunlight scattered from a particle as it passes through the large field of view. The typical time (t) for a particle to pass through the field of view is given by:

$$\Delta t = [np/fl]r/v$$

where (np)/fl is the angular width of the field of view (n is the number of pixels across the CCD, each of physical size p, and fl is the focal length of the imaging lens), and r is the distance of the particle from the image as it crosses through the field of view with a speed of v. If time tags are placed along the track, crucial distance information can be obtained. The angular velocity, which can be determined from time tags along the track, is given by $\Omega = v/r$, where v is the particle speed in the spacecraft reference frame. If the trajectory angle of the particle in the spacecraft frame of reference is known, then a very good estimate of v can be made. The distance r is then obtained from the measured value of Ω and inferred value of v.

Using a sensor with camera focal length of 5 cm and a 5 cm diameter lens, and using particle albedos as proposed by Henize (ref. 25), in a 500 km orbit, assuming reasonable system parameters such as signal strength and background noise, and using fluxes as predicted by the NASA standard model, this sensor will measure between 8 and 10 particles/year in the 0.5 to 1.0 cm size range, between 25 and 50 particles/year in the 1 to 2 cm size range and between 250 and 500 particles/year in the 5 to 10 cm size range.

The microprocessor in the electronics module performs many of the calculations on board the spacecraft, eliminating false alarm rates, etc., and only telemetering data to the ground that has been partially analyzed.

The first flight instrument of this design is currently being fabricated and tested by Visidyne and will be delivered to the United States Air Force Phillips Laboratory in mid 1994. Space flight, as part of the USAF Space Test Program (STP), is planned for the 1996 time frame.

SENSOR TECHNOLOGY: ***Photocell detection of scattered laser light***

Instrument Generic Name: (PME) Particle Monitor Experiment

Detection Method: Near-forward scattering of AlGaAs laser light

Particle Size Detection Range: 0.5 to 10 µm (Type A); 10 to 250 µm (Type B)

Consortium Contact: Dr. O. Manuel Uy, JHU/APL, 301/953-5334, Fax 953-6119

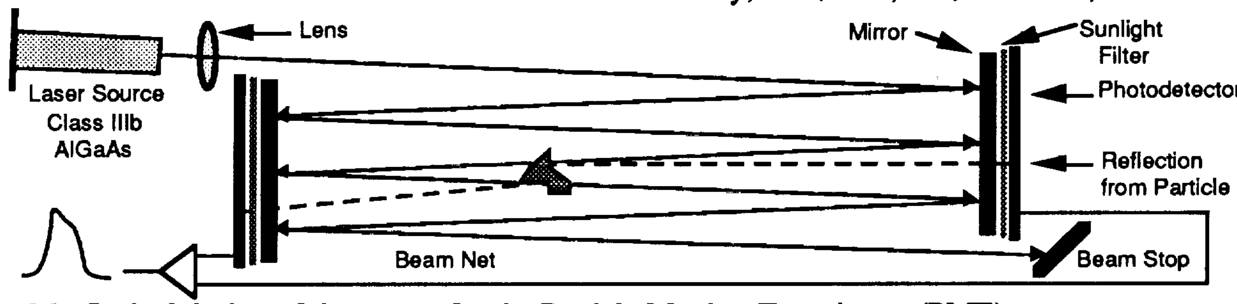


Figure 26. Optical design of the sensor for the Particle Monitor Experiment (PME)..

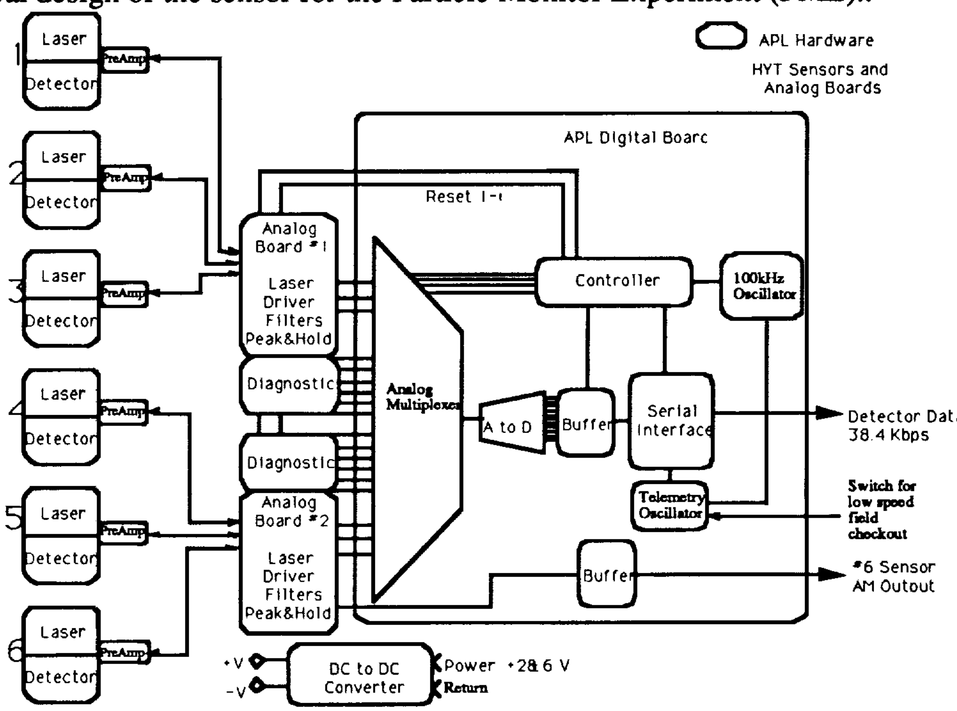


Figure 27. Particle Monitor Experiment (PME) Block Diagram

<u>Nominal Instrument Specifications</u>

dimensions	Sensors: 6 x 18 x 2 cm, Electronics: 20 x 33 x 9 cm
active sensing area	42 cm^2 (6 sensors at 7 cm^2/sensor), 421 samps/sec, 12 bit res.
mass	20 Lbs, includes 6 sensors, electronics, and wiring harness
spacecraft electrical interface	28V ± 6V, no system control required
mounting requirements	sensors and electronics bolted to rigid surfaces
power requirements	18 Watts
telemetry requirements	no special requirements, 4 kbits/sec or 38.4 kbit/sec

<u>Nominal Cost Estimates (U.S. $1000)</u>

delivered hardware (only) costs	200-300, dependent on program requirements
pre-launch mission support costs	50, dependent on program requirements
software development costs and time	50, dependent on program requirements; 3-6 months
post-launch mission support	100, dependent on program requirements
<u>**Nominal hardware (only) delivery**</u>	9-12 months, dependent on program requirements

TECHNOLOGY DESCRIPTION AND FLIGHT HISTORY

The Particle Monitoring Experiment (PME) satisfies a requirement for real-time monitoring of the particle environment around an actual operating spacecraft. Particles consist of dust and aerosols associated with retrorockets and other attitude-control systems, water droplets, ice crystals, and other forms of condensation emanating from the spacecraft, dusts and fibers from ground operations, and even particles associated with surface erosion caused by atomic oxygen. It has been suggested that particles are generated by the rubbing of two different materials' surfaces that are in intimate contact with each other each time the spacecraft crosses the day/night terminators as a result of differential thermal expansion.

The objectives of the PME are to record the density of particles around the spacecraft, measure particulate flux (number of particles per unit area), determine their size distribution, obtain their velocity distribution, and, if possible, determine their chemical composition. In less than one year Applied Physics Laboratory developed the PME to satisfy the above objectives for the VIP launch known as the "Piggyback Flight".

The basic particulate sensor is a light-scattering device built by High Yield Technology (HYT) of Sunnyvale, California. The sensor contains an AlGaAs laser diode source (Class IIIb) that projects a 20 mW laser beam at a wavelength of 780 nm, a beam-forming lens, two reflecting mirrors, a beam stop, and two photocell detectors with sunlight filters (Fig. 26). The mirrors reflect the laser beam back and forth to produce a nearly continuous plane of laser light between the two mirrors with a Gaussian beam profile. When a particle crosses this plane of light it scatters the laser to either or both of the photo-detectors located just above the mirrors. The count rate of the particles crossing the light plane of the sensor is directly related to the particulate flux, $\boldsymbol{J = nv}$, where n is the particle number density and v is the particle velocity. After calibration of the PME using latex spheres of known sizes, densities and velocities, particle fluxes can be measured and velocities inferred. Particle sizes can be binned according to peak heights or intensities of the photo-detector signals. This approximation is based on the assumption that all detected particles are spherical in shape and have the same index of reflection as the latex calibration spheres. Velocities may also be inferred by measuring the peak width of the signal outputs. Because of the assumptions involved, we cannot infer any information about the composition or shape of the particles.

The sensors were manufactured and calibrated by HYT for detection of particles within a size range of 0.5 to 10 μm (Type A), and 10 to 250 μm (Type B). Each sensor contains two photo-detectors whose outputs are summed and amplified. The signal is then sent to a peak-and-hold circuit which saves the largest peak during each 2.3 ms sampling interval. The "peak" value is digitized to 12 bit resolution and inserted into the telemetry stream. The sample and hold then resets and begins holding for the next interval. If multiple particles scatter light during a 2.3 ms interval, only the largest signal will be counted.

Six PME sensors were mounted on the VIP spacecraft and controlled by one Flight Electronics Box (Fig. 27). Four of the sensors were located on booms and the other two were located on the main payload structure. In addition to the digital signals used to measure particulate flux, an analog output from one sensor was telemetered to the ground via a separate analog channel and correlated to the digital measurement to determine pulse height and width. Statistics from the pulse height and width are used to estimate the particle size and velocity distributions around the VIP spacecraft. The PME was flown in the Fall of 1991 at an altitude of 1200 km and a flight duration of approximately 30 minutes and performed as advertised. The unit proved extremely robust and even survived a vehicle hardware malfunction. Data was collected from lift-off through reentry. The particle measurement data is currently classified but appears to confirm payload physicists theories. The digital data correlated well with the analog channel.

New Developments

(1) A self-contained monitor and sensor in a sensor-sized housing is being developed. This would utilize self-contained on-board Analog-to-Digital conversion and bit-streaming compatible with standard space flight telemetry systems. This will entail flight hybridization in which APL has extensive experience and promises to produce a low cost, versatile PME.

(2) A "daisy-chained" unit is also being analyzed to allow several modules on a flight vehicle. APL is also investigating pulsed sheet lasers for higher resolution and speed measurements.

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