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Arno G. Ledebuhr, Joseph F. Kordas
Lawrence C. Ng, Mark S. Jones
Oliver D. Edwards, John C. Whitehead
Richard J. Gaughan, Michael D. Dittman

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A. G. Ledebuhr, J. F. Kordas, L. C. Ng, M. S. Jones,
O. Edwards, J. C. Whitehead, R. J. Gaughan, and M. D. Dittman
Lawrence Livermore National Laboratory
P.O. Box 808, L-043
Livermore, CA 94550
(925) 423-1184
ledebuhr1@llnl.gov

Abstract. This paper summarizes the work at Lawrence Livermore National Laboratory in the development, integration and testing of the critical enabling technologies needed for the realization of agile micro-satellites (or MicroSats). Our objective is to develop autonomous, agile MicroSats weighing between 20 to 40 kilograms, with at least 300 m/s of Δv, that are capable of performing precision maneuvers in space, including satellite rendezvous, inspection, proximity operations, docking, and servicing missions. The MicroSat carries on-board a host of light-weight sensors and actuators, inertial navigation instruments, and advanced avionics. The avionics architecture is based on the CompactPCI bus and PowerPC processor family. This modular design leverages commercial-off-the-shelf technologies, allowing early integration and testing. The CompactPCI bus is a high-performance, processor independent I/O bus that minimizes the effects of future processor upgrades. PowerPCs are powerful RISC processors with significant inherent radiation tolerance. The MicroSat software development environment uses the space flight proven Vx-Works, a commonly used, well tested, real-time operating system that provides a rapid development environment for integration of new software modules. The MicroSat is a 3-axis stabilized vehicle which uses cold gas N_2 for ACS and a novel pressure-fed, non-toxic, mono-propellant hydrogen peroxide propulsion system for maneuvering.

Introduction

Competition in the consumer electronics marketplace continues to drive component miniaturization and consolidation reducing cost, mass and power consumption while improving system performance. This technology push should provide the capability to build significantly smaller and smarter micro-satellites (MicroSats) with wet masses between 10 to 100 kilograms. This paper describes on-going efforts at Lawrence Livermore National Laboratory to develop the critical enabling technologies needed for the realization of autonomous agile MicroSats that are highly autonomous in function, less than 50 kg in mass, and possess a robust orbital maneuvering capability of >300 m/s of Δv. LLNL has developed a preliminary design of a near-term MicroSat for a variety of Rescue Mission applications and has constructed two prototype vehicles for ground test experiments.[1] This paper will discuss representative mission applications, enabling vehicle technologies, and integrated vehicle testing approaches.

Potential Missions

Potential missions for MicroSats center on space "logistics" missions such as rescue and servicing, that will require vehicles with the ability to perform a variety of mission functions autonomously or semi-autonomously. These include, rendezvous, inspection, proximity-operations (formation flying), docking and robotic servicing functions (refueling, repowering or repairing). Each of these mission functions require key technical capabilities. For example, rendezvous with a space asset by performing orbit matching requires precision Δv maneuvers. Inspection of a space asset by flying to different view points of an inspection geometry requires precision MicroSat positioning, precision pointing, tracking and imaging. A satellite rescue might involve docking, repairing or refueling the satellite, followed by a departure, and post-rescue inspection. The rescue mission requires precision guidance, navigation, and control; precision ranging; high resolution imaging; and some type of micro-robotic manipulator. For example, a variety of robotic arms could be used to enable the MicroSat to perform a physical dock with a target satellite. Once docked, a precision 6 degrees-of-freedom manipulator (actuator) could be used to align and plug an external connector into a targeted satellite's umbilical connector for data collection, diagnostic measurements or repowering. Other space logistic operations such as the collection and de-orbiting of hazardous space debris (junk) or the interdiction of asteroids or comets in a planetary defense system, requires precision vehicle guidance, navigation and control and a precision endgame homing strategy. Formation flying, flying in concert with a space object or another MicroSat, requires station keeping and positioning, and precision state vector estimation.

Recently LLNL has studied the technologies and requirements for a satellite Rescue Mission.[1] Table 1 contains a preliminary analysis of a timeline for this representative rescue mission. It assumes a Pegasus launch for the MicroSat with orbit injection errors as outlined under Day 1's General Comments. The estimated Δv for each MicroSat maneuver is listed. It is assumed that the servicing required is a battery recharge and processor restart.

Table 1 Preliminary Rescue Mission timeline for Rendezvous, Inspection, Docking and Departure.

Day	Event	ΔV (m/s)	General comments
1	• Pegasus launches and deploys MicroSat • *MicroSat performs practice rescue mission with payload interface adapter* • MicroSat orbit determination	15	Pegasus with HAPS (hydrazine auxiliary propulsion system) can place MicroSat with the following (3σ) injection errors: *± 5 km @ apogee; ± 5 km @ perigee; 0.5° in ascending node; 0.05° in inclination*
2	• MicroSat corrects ascending node using out of plane Δv • MicroSat Orbit determination	50	Out of plane Δv causes orbital plane to rotate about the radius vector and thus corrects the ascending node angle
3	• MicroSat initiates correction of inclination angle at the ascending node • MicroSat Orbit determination	20	Out of plane Δv @ ascending node changes the orbital plane inclination ; MicroSat is expected to within 1-10km behind target satellite
4	• MicroSat initiates apogee and periapsis corrections • MicroSat Orbit determination	3	Correct apogee error to within 1 km of target satellite and avoid collision
5	• MicroSat initiates perigee correction • MicroSat Orbit determination	2	Correct perigee error to within 1 km of target satellite and avoid collision
6	• MicroSat proceeds to a point 100m behind target satellite	50	Image target and conduct station keeping; relay imagery to ground
		140	**Total Δv required for orbit match and rendezvous including a complete practice docking**

7	• MicroSat initiates a 100m circular inspection of target • Preliminary estimate of target axis of rotation and rotation rate	25	Relay imagery and estimates to ground for detailed analysis
8	• MicroSat proceeds via a circular transfer orbit to 10m range from target satellite • Refine target spin axis estimation • Initiate a 10m circular maneuver about target spin axis and track the landing site	8 2	Image target and relay pertinent data to ground station
9	• MicroSat proceeds to 3-5m range from target • Execute a circular maneuver about target	1 2	MicroSat tracks landing site, analyses imagery, and orients itself for landing
10	• MicroSat closes in and docks	2	MicroSat grabs on target satellite flange structure
11	• Align MicroSat to target connector	1	Send imagery to ground and wait for up-link command to proceed
12	• Insert MicroSat adapter into target satellite connector	2	MicroSat uses a six degree of freedom fixture to insert pin connector and vision based imaging software to guide adapter
13	• Mate and revive lost satellite		Trickle charge satellite battery, re-activate satellite computer, and report reactivation diagnostics to ground
14	• Departure from satellite	2	Move away to 50 - 100m behind satellite
15	• Observe target satellite deployment maneuver	5	Mission accomplished
		50	**Total Δv for Inspection, Docking, and Departure operations**

Spacecraft Technologies

The LLNL MicroSat is an adaptation of the lightweight spacecraft technology developed for the Clementine I & II missions. It features a lightweight, Seeker Head, state-of-the-art avionics, H_2O_2 divert propulsion system with N_2 gas for attitude control, VxWorks based real-time controller with a collection of technology based software for precision guidance and control and image analysis. Key modifications from the Clementine missions include the addition of stereo cameras for passive ranging and 3D imaging, a micro-impulse radar for precision docking, miniature grappling arms for docking and a 6DOF robotic arm for satellite servicing. Also added is a mating camera to provide vision feedback to guide the robotic arm, a GPS receiver for precision orbit determination, and spot lights for illumination. Figure 1 shows a conceptual design of this system based on current prototype MicroSat vehicles. The following paragraphs briefly describe the enabling technology for the development of the agile MicroSat.

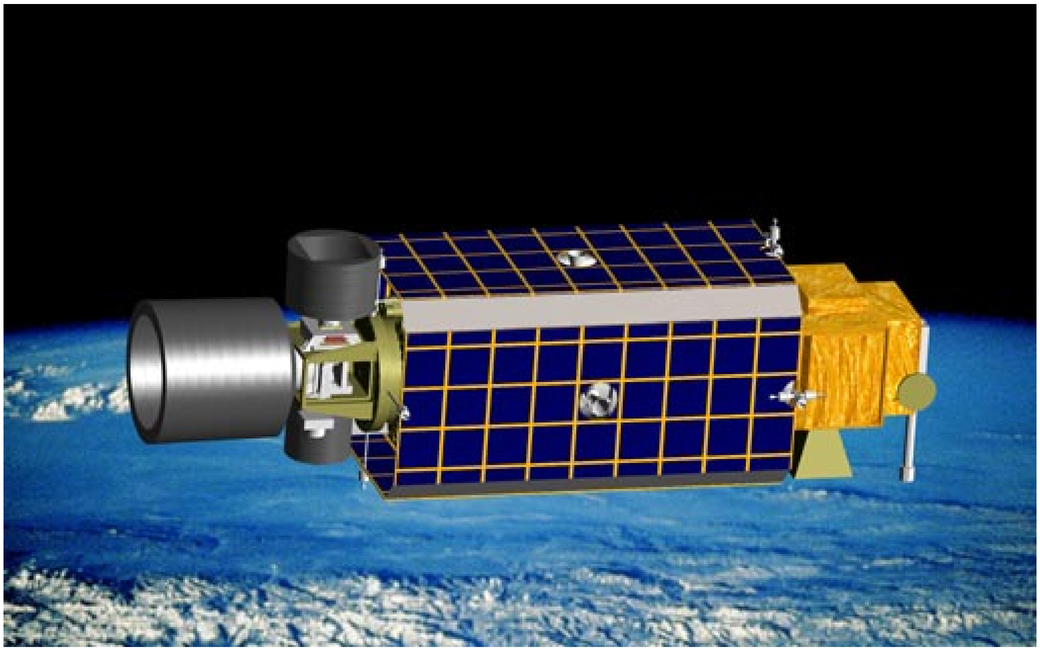

Figure 1 A conceptual MicroSat designed for a rescue mission.

MicroSat Spacecraft

Figure 2 shows an exploded view of the current design of the MicroSat. The Seeker Head Assembly consists of a suite of imaging sensors and provides the GN&C input to the vehicle. These sensors include a high resolution acquisition camera for long range detection and standoff inspection, a medium field-of-view inspection camera for closer-in inspection, one of two small wide-field-of-view (WFOV) color docking cameras for stereo imaging and passive ranging, a WFOV Star Tracker for inertial navigation, a lightweight IMU, a compact GPS unit and a miniature micro-impulse Docking Radar. The IMU operates in conjunction with the Star Tracker to provide attitude quaternions for precision vehicle guidance and control. The MicroSat's modular state-of-the-art avionics suite is based on a compactPCI bus and uses a PowerPC 603e processor and several I/O modules that support the rest of the vehicle sub-systems. Other sub-systems include a pair of rechargeable NiCad battery packs, solar array panels for battery charging, coupled ACS jets for in-place rotation, H_2O_2 divert thrusters and N_2 jets for precision translation control. Also included is a mini-SGLS transponder that is AFSCN compatible, a set of grappling arms for docking and a 6DOF robotic actuator for satellite servicing (arms and actuator not shown in this view).

In order to achieve centimeter level of position control of the MicroSat, during docking maneuvers, we make use of the small minimum impulse bit that is available from the cold gas propellant. Our current design goals are: 4 N-s for H_2O_2 diverts, 0.01 N-s for cold gas diverts. The current MicroSat ground test vehicle is <25 kg and our final design for a flight qualified MicroSat with a full complement of subsystems is expected to weigh less than 40 kilograms.

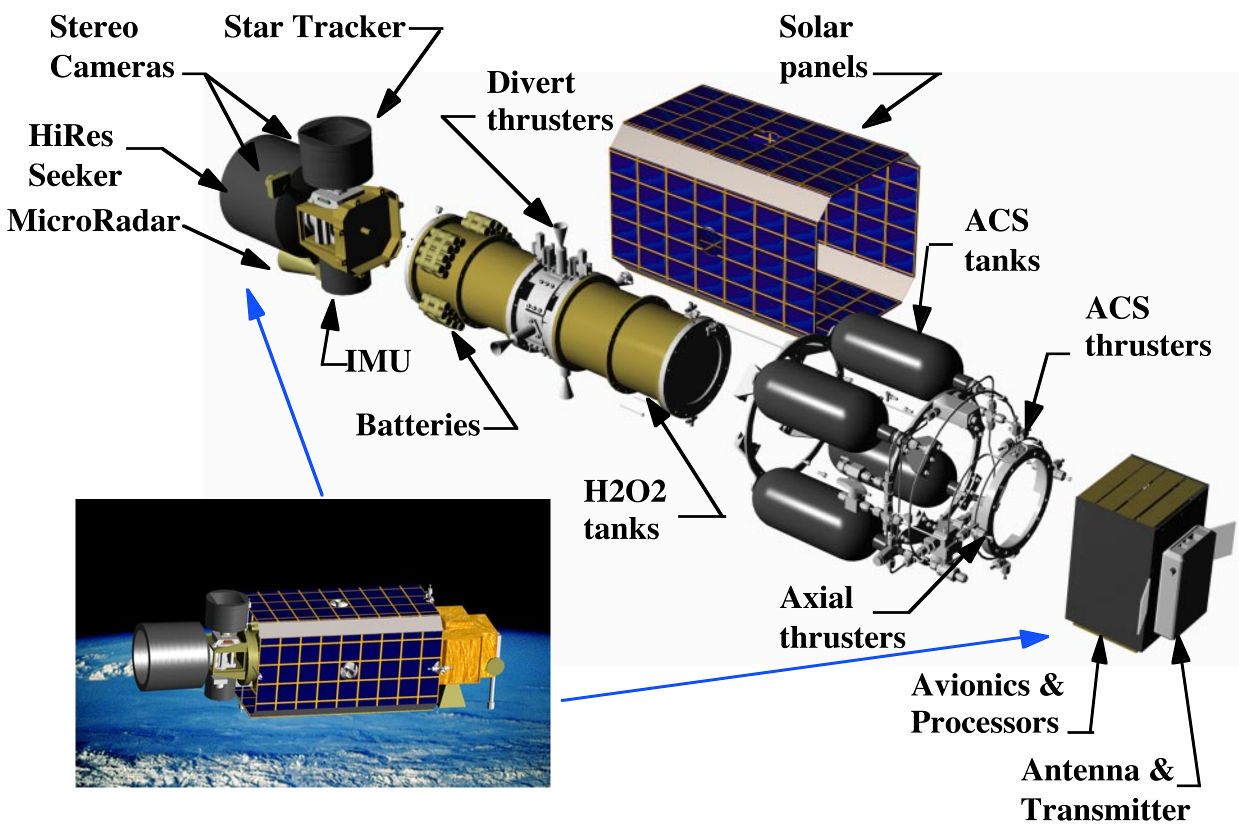

Figure 2 An exploded view of the MicroSat design for the rescue mission (Some components not shown).

The original worst case minimum thruster burn period was assumed to be one second (assumes a cold start) for the H_2O_2 divert system and five milliseconds for the nitrogen system. Table 2 below computes a preliminary set of requirements for the minimum impulse bit (MIB), minimum resolution in velocity, minimum resolution in position, and maximum vehicle acceleration. based on these timing assumptions. The coupled cold gas ACS can provide precision translation, by firing two matching 1N ACS jets simultaneously, a mission requirement used primarily during precision orbit matching, docking, un-docking and departure maneuvers. Preliminary testing of the H_2O_2 divert system indicates that once the thrusters have been warmed-up the response times are typically 10X shorter. For example our prototype 20 N vacuum thruster provided 14 N thrust at sea-level with less than a 0.1 second response time when warm. This yielded an MIB of approximately 1 N-s for the first generation H_2O_2 divert system[2].

Table 2 Precision vehicle control requirements.

Thrust (N)	pulsewidth (sec)	MIB (N-sec)	Δv (m/s)	Δp (m)	Accel (mg)	comments
4	1	4	0.1	0.05	10	H_2O_2 (cold start)
2	0.005	0.01	0.00025	6.25E-07	5	N_2 gas
Wet Mass	<40 kg					

Propulsion

LLNL's current approach to propulsion for micro-spacecraft is to work with nontoxic propellant. We have demonstrated that hydrogen peroxide, when used on a scale appropriate for micro-spacecraft, permits cost-effective bench-top testing of custom-developed hardware. Unlike conventional space propulsion systems, a hydrogen peroxide (H_2O_2) micro-propulsion system can readily be tested at the system level, even after spacecraft integration.

While H_2O_2 has a lower specific impulse than hydrazine, 2.5 times the Isp of cold nitrogen is readily achieved. In addition, H_2O_2 has over 3 times the density of 5,000 psi nitrogen. The combined effect of Isp and density makes monopropellant H_2O_2 many times more capable than cold gas used alone, which is currently the only choice for custom-designed micro-propulsion systems.

Recently during the second half of FY97, LLNL researchers designed, fabricated, tested, debugged, and demonstrated a miniature divert propulsion system (see Figure 3 below) in support of the Clementine II asteroid intercept mission plan[1,2]. The results of ongoing R&D (e.g. long term tank storage tests) indicate that custom-designed H_2O_2 propulsion is well suited to LEO missions, for spacecraft weighing 10 kg to 50 kg total.

There are several options for using H_2O_2, including self-pressurizing tanks which feed liquid to catalytic thrusters, while also delivering a steam-oxygen mixture to warm gas attitude control jets. A relatively small amount of kerosene can be carried to double the specific impulse (nontoxic bipropellants). Figure 4 shows the block diagram of a first generation pressure-fed H_2O_2 propulsion subsystem.

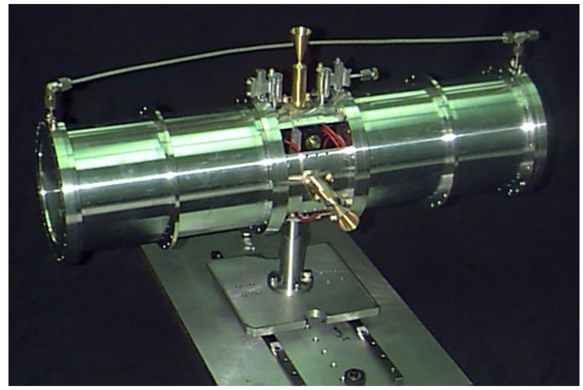

Figure 3 Demonstrated H_2O_2 miniature divert propulsion technology.

The total maneuvering requirement is 250 m/s, including a maneuvering reserve. The propulsion system is being designed to 300 m/s, for a 20% propulsion margin. The breakdown is 275 m/s with the liquid system and 25 m/s with nitrogen (ACS impulse requirements have been converted to an equivalent delta-v based on vehicle mass after liquid burns are complete). The micro spacecraft total mass allowance is 40 kg, of which 9 kg of 85% H_2O_2 would be required to achieve 275 m/s at Isp=130 s. Delivering 25 m/s after liquid burnout requires 1.89 kg of nitrogen at Isp=50 s. This includes an additional 150 grams of nitrogen to pressurize the liquid tanks. Initial hardware mass estimates are 5 kg for the "tank-as-structure" liquid system (already demonstrated with prototype hardware), and 4 kg for the gas components. Thus, total wet propulsion mass is just over 20 kg.

A preliminary estimate of the vehicle mass budget is shown in Table 3 below. A total vehicle wet weight of 39.9 kg with a design margin of 7.2 kg is shown at the current conceptual design stage. Our goal is to achieve a total wet weight of less than 40kg. The table shows that we should be able to meet this goal and that there is also adequate margin to mass balance the vehicle.

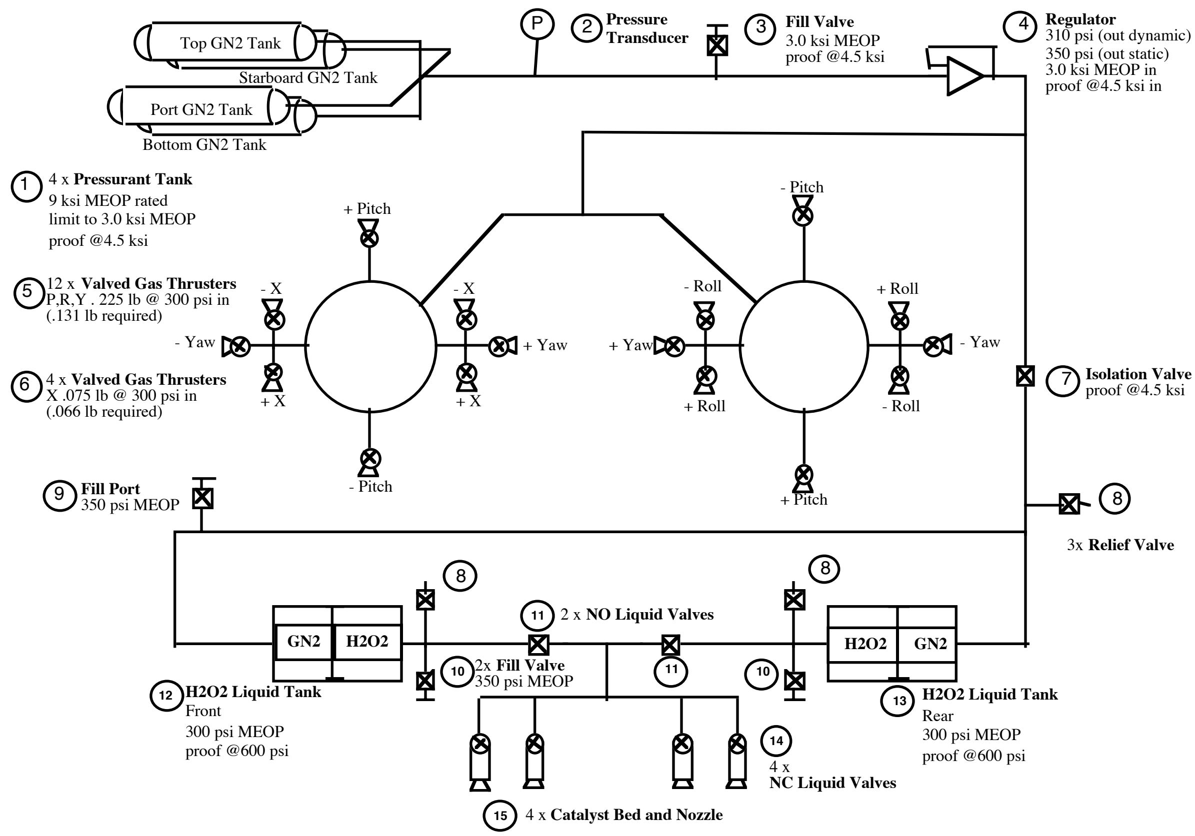

Figure 4 First generation block diagram of a pressure-fed MicroSat propulsion subsystem.

Table 3 Preliminary estimate of the MicroSat vehicle mass budget.

Subsystem	Mass Estimate (g)	Contingency (g)	Total Mass (g)
Seeker Assembly	**6310**	**560**	**6870**
Clementine I Star Tracker Assembly	330	30	360
Acquisition Camera Assembly	1400	0	1400
Inspection Camera Assembly	330	30	360
Docking Camera Module Assemblies #1	100	50	150
Matting Endoscopic Camera Assembly	150	50	200
IMU and Accelerometer Assembly	800	0	800
GPS Receiver Assembly	300	100	400
Microimpulse Radar	300	100	400
Battery Pack Assembly (2)	2200	100	2300
Seeker Head Mechanical Structure	400	100	500
Avionics Assembly	**5480**	**1420**	**6900**
Seeker Avionics Assembly	2000	500	2500
Power Conditioning Module	1000	200	1200
Valve Driver Module (2)	230	20	250
Comm Module Assembly	750	400	1150
Matting Actuator Assembly	1500	300	1800
Docking Camera Module Assemblies #2	100	50	150

Propulsion/Solar Array Assembly	**12700**	**2550**	**15250**
H2O2 Propulsion Assembly	5000	500	5500
GN2 ACS Assembly	4000	750	4750
Wire Harness	900	100	1000
Thermal Control Assembly	500	500	1000
Solar Array Assembly (4)	1000	100	1100
Grappling Arms for Docking (4)	600	400	1000
Vehicle Mechanical Structure	500	150	650
Docking Illuminators (2)	200	50	250
MicroSat Dry Weight (g)	**24490**	**4530**	**29020**
H2O2 Fuel (g)	6350	2650	9000
GN2 Fuel (g)	1890	0	1890
MicroSat Wet Weight (g)	**32730**	**7180**	**39910**

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Sensors

The sensor suite or Seeker Head consists of a Star Tracker for attitude determination using stellar navigation, a high resolution Acquisition camera for long range (20 km to 0.1 km) target detection and stand-off inspection, a medium resolution Inspection camera for closer-in inspection (in the 100 m to 10 m range), one of a pair of wide-field-of-view color Docking cameras for proximity-operations (in the 10 m to 0.1 m range), a microRadar for active close-in (< 10 meter distances) providing precision range and range-rate measurements, a compact GPS receiver for coarse position measurements, and an IMU for inertial navigation. These sensors are lightweight and compact. The imaging cameras (Acquisition, Inspection and Star Tracker) were originally developed for ballistic missile defense applications. A schematic of the MicroSat sensor suite is shown in Figure 5. Versions of the Acquisition (HiRes) camera, Inspection (UV/Vis) camera and the Star Tracker cameras, have previously flown in the successful Clementine I Lunar mapping mission[3,4,5].

WFOV Star Tracker

A key sensor in the MicroSat is a wide-field-of-view (WFOV) Star Tracker which provides inertial orientation of the vehicle. The Star Tracker camera in conjunction with Stellar Compass software can provide a quaternion pointing accuracy of 450 μrad. The Star Tracker field of view is large enough to contain several bright stars in any orientation. Single images are processed to identify unique stellar patterns and provide the determination of the inertial orientation of the MicroSat in real-time. The Star Tracker lens was specified and built to have a 42° x 28° field of view. This sizing results in a pixel IFOV of 1.3 mrad, which is small enough to provide the requisite quaternion accuracy. The collection aperture of the lens is maximized for the greatest possible light gathering capability. At F/1.25, $m_V = 4.5$ G0 stars provide an integrated star signal that is 15 times the electronic noise from the focal plane. This level of signal gathering capability, matched with the wide field of view, ensures a 99.9% probability that 5 stars above minimum threshold will be available for the algorithm set for all possible quaternion pointing vectors. This allows the Star Tracker to handle the “lost in space” condition with a single star image frame and no other a priori knowledge of attitude. A preliminary analysis of the imaging sensor requirements to support the various phases of a typical rescue mission is given next.

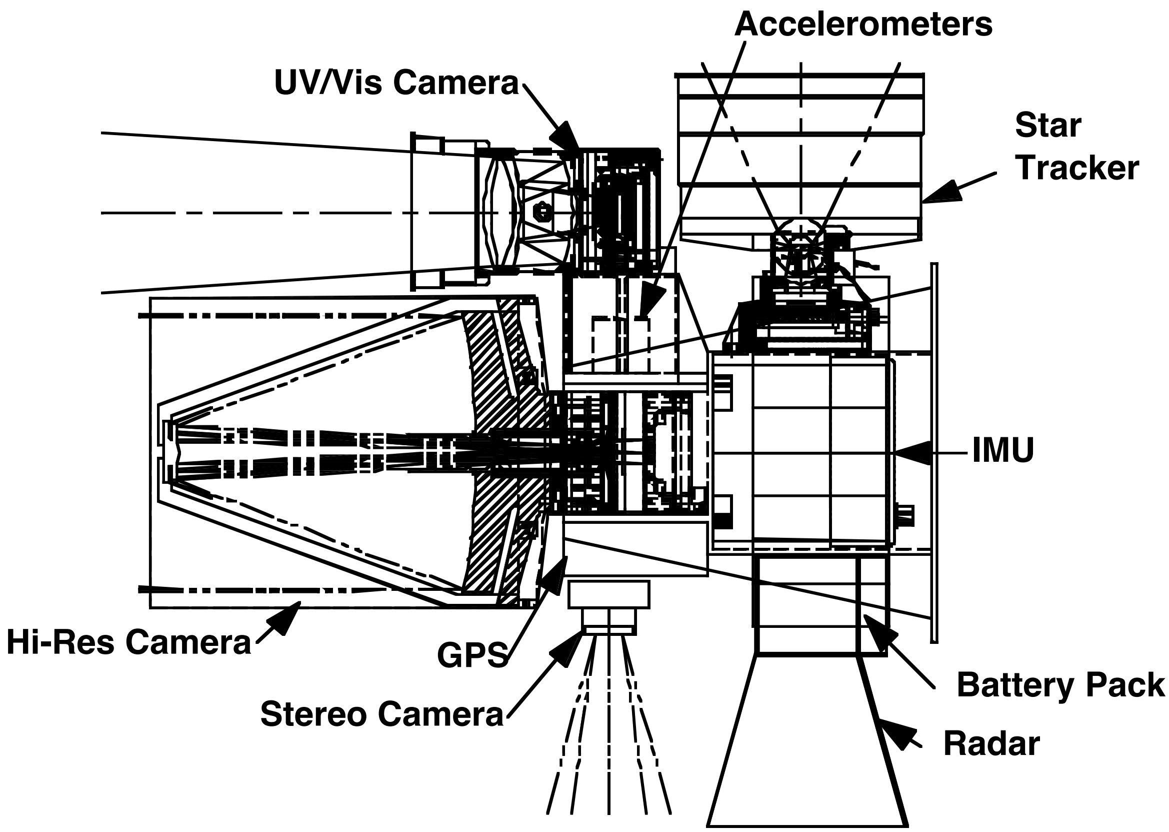

Figure 5 Multi-function MicroSat sensor suite.

Sensor Requirements

The top-level optical system requirements, shown in Table 4 below, support operational phases of our representative rescue mission. In the Rendezvous phase from 20 km to 100 meters, the High Resolution Acquisition sensor will be required to identify the target and provide centroids to the guidance system. This sensor will provide the first detailed images to the ground at a few kilometers out. At several hundred meters out a transition will be made to the Medium Resolution Inspection camera which will be used to determine the target spacecraft's rotational parameters. The 100 m to 10 m closing portion of the Inspection phase will again require input from the optical sensors, and the 10 m inspection will produce more detailed images for ground evaluation. At 10 meters the vehicle will rotate 90 degrees (pitch or yaw) and will then switch to its pair of color Docking cameras. The sensors will again be required to provide information to the guidance and navigation control system during the 10 m to 10 cm closing portion of the Docking phase. For the docking maneuver the optical sensors will allow features to be identified, and in the terminal phase they will determine the precise location on the target where the hard dock will be made (launch vehicle Marmon interface flange). After the docking is complete, the sensor system will provide images for the Servicing phase of the mission including providing images that support the mating of the MicroSat to the target satellite's umbilical connector and confirmation when the connection is completed. A small Mating camera (Endoscopic), attached to the Mating Actuator (manipulator mechanism) will provide imagery of the precise alignment and orientation of the individual pins in the connector. After the servicing is complete and

Table 4 Top Level Optical System Requirements.

Mission Phase Label	Range	Sensor Function	Operational Phases
A	20 km - 100 m	acquisition/tracking	Rendezvous
B	100 m - 10 m	imaging/tracking/guidance	Inspection
C	10 m - 10 cm	imaging/guidance	Docking
D	10 cm - 1 mm	imaging/guidance/alignment	Servicing
E	10 cm - 100 m	imaging/tracking/guidance	Departure

the MicroSat switches to the Departure phase, it will un-dock and depart the satellite. The sensor system now will produce new inspection images to confirm the successful servicing operation.

Each mission phase levies a particular set of requirements on the optical sensor system. These requirements are best met with a sensor system optimized for each mission phase. An alternate approach would be to utilize a zoom lens system to provide overlapping capability. Future designs will explore this option. Our current optical system mission support is summarized in Table 5. The Star Tracker provides MicroSat attitude information in all mission phases, and is used to support both the identification of the satellite spin axis in inertial space, and MicroSat guidance and navigation. The Acquisition camera provides the long range acquisition of the target spacecraft and provides initial imagery of the target. This camera supports the terminal portion of the Rendezvous phase and can provide data to support an autonomous rendezvous operation. The Inspection camera provides additional target information to the guidance and navigation computer, and also supplies detailed target images at 100 meter to 10 meter ranges. The Docking camera system produces close-in imagery of the payload interface flange (docking region) of the target satellite. This system may be used in either stereoscopic or monoscopic mode. Stereo imagery provides passive range data for the GN&C system. The docking camera is also used to assist in mating the umbilical connector. The Mating camera is mounted on the 6DOF manipulator and will support the fine alignment of the two connectors. The Departure phase will essentially be a time reversal of the Docking and Inspection phases and can be met with these sensors.

Table 5 Optical System Mission Support.

Camera	Effective Range	Mission Phases Supported
Star Tracker	N/A	A,B,C,E
Acquisition Camera	20 km - 100 m	A
Inspection Camera	1 km - 8 m	A,B
Docking Camera System	10 m - 10 cm	C,D,E
Mating Camera System	10 cm - 1 mm	D

Avionics

The MicroSat avionics architecture is based on the latest PowerPC processor and CompactPCI bus as shown in Figure 6. The PowerPC family has significant inherent radiation tolerance. The CompactPCI bus is a high-performance, processor independent I/O bus which will minimize the effect of future upgrades in processors. The system supports modern, well-used and well-tested embedded software development environments. This design allows rapid code development, debugging, and testing, as well as hardware integration. The chosen architecture and processor will provide a high performance solution for multiple MicroSat missions. Its modular design leverages COTS technologies, allowing early integration and test.

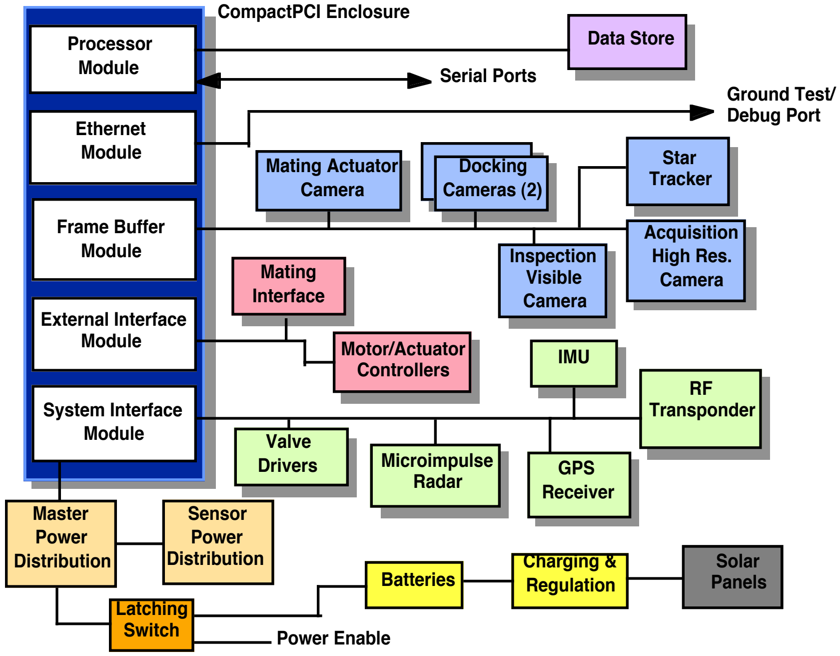

Figure 6 CompactPCI architecture.

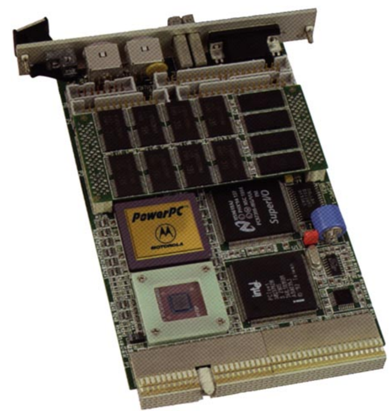

Figure 7 PowerPC Processor.

It provides a clear path to performance upgrades. In addition, this approach and type of architecture are being embraced by major aerospace system providers.

Processor Module

The MicroSat processor module contains a high-performance PowerPC 603e RISC CPU (100MHz-300MHz) that utilizes the 33 MHz, 32 bit data path (132 Mbyte/sec max) CompactPCI bus, a high bandwidth I/O bus that is an industry-standard ruggedized version of the desktop PCI bus used in virtually all desktop computers, MPC106 PCI bridge/memory controller with built in memory single bit error correction and double bit error detection and low power modes, 32 Mbytes DRAM, 4 Mbytes flash EEPROM, 8 Kbytes PROM, a real time clock with 4 Kbytes non-volatile RAM, two high-speed UART channels, interval timers, and watchdog circuit. The flight processor will be a COTS module with modifications for thermal management and radiation tolerant parts as needed.

The MicroSat local data store will be connected to the processor's IDE controller port as indicated above. This will be a commercially available Flash Disk that has built-in error correction capability. The size of the disk selected will be based on the worst-case telemetry data requirements. The docking phase presents the most stressing conditions. In Table 6 we show data store requirements for a representative rescue mission. Estimates are provided for two different docking camera designs, using two different CCD size options (different pixel number, size and formats, designated A and B).

Table 6 Data storage requirement estimates for a representative Rescue Mission.

Component/subsystem	Unit Count	Frame size		Frames	Pixels	Bytes	Bytes	Comp.	Data Store	Minutes	Storage Req.
		Horiz	Vert	/sec	/sec	/pixel	/sec	Ratio :1	Bytes/sec	Active	Bytes
Non-image data									5,000	20	6,000,000
Star Tracker	1	576	288	0.1	16,589	1	16,589	2	8,295	20	9,954,000
Docking Cameras (A)	2	512	512	10	5,242,880	1	5,242,880	20	262,144	20	314,572,800
Docking Cameras (B)	2	256	256	10	1,310,720	1	1,310,720	20	65,536	20	78,643,200
SubTotals (A)					5,259,469		5,259,469		275,439		330,526,800
20% Margin											66,105,360
Total (A)											396,632,160
Download Minutes @ 1 Mbps											56
SubTotals (B)					1,327,309		1,327,309		78,831		94,597,200
20% Margin											18,919,440
Total (B)											113,516,640
Download Minutes @ 1 Mbps											16

Frame Buffer Module

The digital frame buffer module is designed to provide a high-performance image acquisition and data handling interface between the CompactPCI™ bus and high-speed digital cameras. The features of the module are as follows: frame formats to 4096x4096, 8 to 16 bit pixels, pixel clock rates to 20Mhz, two independent video channels (simultaneous acquisition), multiplexed operation for cameras sharing channels, Automated Imaging Association (AIA) digital camera compatibility, Synchronous Addressable Serial Interface (SASI) camera controllers, 32 Mbytes SDRAM image storage, Image compression (Lossless ≈ 2:1, Lossy > 10:1), and a CompactPCI bus interface with Direct Master or Slave operation and two independent DMA controllers. The frame buffer is a LLNL design that will contain rad-tolerant controllers and will incorporate thermal management.

RF Transceiver

The RF transceiver under development for MicroSat applications is a SGLS-signaling AFSCN-compatible unit that is being developed by the Naval Research Laboratory (NRL). This unit was started as part of the Clementine II program and was originally designed to provide a telemetry link between the asteroid impact probe and the Mothership bus during its fly-by of a near earth (earth-crossing) asteroid. This new transceiver design is expected to have dimensions of 14.5 cm x 14.5 cm x 4.6 cm and a mass that is under 2 kg. It is designed to operate on SGLS Channel 10, with a command up-link data rate of 2 kbps (1799.76 Mhz) and a telemetry down-link data rate of 1 Mbps (2247.5 Mhz). For a 1 Mbps telemetry rate the total estimated radiated output power is 3 watts and the total power consumption is 25 watts. More detailed information may be obtained by consulting the NRL Naval Center for Space Technology specifications SSD-S-CM013 and SSD-S-CM017.

GPS Receiver

Several commercially available GPS receivers are available for this application and a light weight (<2 kg) space qualified unit is available from Boeing North America's Rockwell division. Initial testing will be carried out with off-the-shelf commercial units, one of which will be selected for flight qualification during the later development phase of a flight experiment program. Ground test experiments will be carried out with differential GPS to determine the accuracy and utility of the position and attitude data derived from this subsystem. Closed loop positioning (docking) experiments will be performed to determine the overlap in functionality between the various sensor systems (Star Tracker, Micro-Radar and Docking cameras).

Power Distribution System

In order to minimize the size and mass of the MicroSat solar array assembly and battery pack, a power management scheme is required which allows components and subsystems to be powered down when not needed, and which allows for periods of recharge between periods of activity that cause the battery charge level to fall below nominal value.

The master power distribution module contains a low power consumption controller for the purpose of spacecraft power management. The controller is capable of monitoring and controlling the power to all of the spacecraft loads. A communications link to the system processor allows the system processor to make mode changes to the power management controller, send power down commands, monitor the power system condition and provide a way for the controller to signal the system processor of power system alarms. During the power down period the battery modules will be recharged by the solar cell array. The recharging rate will be a function of the particular mission orbit flown, which will determine the average solar incidence angle and effective exposure time.

Battery Module

Representative power system design for a near-term first generation agile MicroSat is as follows. Each battery module would contain a 23 cell, rechargeable, high capacity Nickel-Cadmium battery pack. At a nominal output voltage of 28 Volts, the module will produce 50 Watt-Hours (Wh) of power at a mass of 1,100 grams. The high capacity Nickel-Cadmium cells have approximately 40% more capacity than that of standard cells. Discharge characteristics include nearly constant voltage, low internal impedance and short term discharge rates of up to 100 times the amp-hour cell rating. The cells can withstand over charging and discharging. They feature long storage and service life, and they are rugged and reliable. Nickel-Cadmium technology has a long history and is well characterized. Future efforts will look to qualify Li-ion technology for MicroSat applications since these cells have a 2X increase in energy storage density (specific energy in Wh/kg). Alternate technologies include miniaturized fuel cells which promise a >4X improvement over Li-ion cells.

Solar Array Module

The solar array module would consist of an array of high efficiency, advanced technology dual-junction gallium arsenide solar cells. Each panel will produce a maximum of 25 Watts output per module (actual power output will of course be a function of the angle of incidence of the sun). In this first generation design there would be four body fixed modules that enclose the propulsion tank structure of the MicroSat, see Figure 2. The solar array module consists of a honey comb substrate to which the solar cells and their cover slides are attached. Each cell has an individual protection diode and is wired to provide a 28 volt output. A mass estimate for this assembly, based on real hardware, is approximately 224 grams per module. The cover slide protects the cell from handling and radiation. The benefit of this dual junction cell technology is that each of the cell junctions, simultaneously convert a different part of the solar spectrum into electrical power. Conversion efficiencies of 21.5% have been achieved.

Power Management

Table 7 below summarizes the electrical and power budget for the MicroSat to support a representative rescue mission. While the absolute maximum power consumption (with all equipment on and operating at maximum capacity) is estimated to be 300 W, the maximum instantaneous consumption per mission phase is about 170W. The actual consumption per phase will be less when the duty cycles of various components are considered. For instance, the RF subsystem will only be active when the MicroSat has an opportunity to communicate with a ground station. The previously described on-board battery packs (two) will provide 100 Wh of continuous usage. The solar array is designed to recharge the battery in about eight hours. Thus the MicroSat will need to recharge its battery packs after each maneuver. The power budget estimated in Figure 7 is based on relatively near-term "in-hand" technologies and sub-systems and is currently considered a conservative estimate. Future flight hardware will be able to take advantage of technology improvements that reduce power consumption, providing greater operational flexibility for future missions.

There are two means by which the MicroSat will enter a power-down mode. The first is initiated autonomously by the system processor as a result of preprogrammed mission operations or as a result of detecting battery depletion via the master power distribution module. The second means is receipt and confirmation of a power-down command from the ground station.

Table 7 Electrical Power Budget and Profile.

Subsystem	Absolute Maximum(W)	Maximum per Mission Phase: Rendezvous	Inspection	Docking/ Departure	Servicing
Processor Module	12	12	12	12	12
Data Store	2	2	2	2	2
Frame Buffer Module	10	10	10	10	0
Star Tracker	5.5	5.5	5.5	5.5	0
Acquisition Camera	7.5	7.5	0	0	0
Inspection Camera	5.5	0	5.5	5.5	0
Docking Cameras (2)	3	0	3	3	0
Mating Camera	1	0	0	1	0
External Interface Module	5	0	0	5	5
Motors/Actuators	30	0	0	15	0
Satellite Charging Load	50	0	0	0	50
System Interface Module	8	8	8	8	8
IMU	10	10	10	10	0
Microimpulse Radar	6	0	6	6	0
RF Subsystem	75 (10 Mbps)	12 (128 Kbps)	25 (1 Mbps)	25	12
GPS Receiver	1	1	1	1	0
Valve Drivers	30	6	6	4	0
Power Distribution (75% eff)	35	18	20	24	10
Thermal Control		TBD	TBD	TBD	TBD
Subtotal		92	114	137	99
Margin (25%)		23	29	35	25
Total	297	115	143	172	124

When powering down the spacecraft, the system processor will send a power-down command to the power management controller. The power management controller will set the appropriate time into the wake-up time comparator of the real time clock circuit, and then will remove power to all loads, make a battery voltage measurement, and put itself to sleep. After being awakened by the real time clock at the programmed time, the controller will make a battery voltage measurement, turn the system processor on and report the battery condition with an estimate of the spacecraft power available to the system processor. During the power down period the battery will be recharged by the solar cell array.

Thermal Management

Thermal management on the MicroSat will be accomplished by using industry standard passive heat transfer mechanisms augmented by tank heaters for the liquid fuel during battery recharge mode. When the MicroSat is performing LEO operations it will encounter the solar flux of nominally 1353 W/m2. The MicroSat will produce an additional 150 W for about a 30 minute interval during typical operational modes. This power level is reduced to a few watts for approximately 8 hours during battery charge mode and then the operational cycle can be repeated. Multi-Layer Insulation, (MLI), sun shades, heat sinks, radiators and surface coatings will be employed for passive thermal control. MLI will be used around the sensor suite as a general purpose insulator from incident sun loading and internal heat loss.

The Solar Array, module encapsulates the H_2O_2 and N_2 tanks and provides insulation from incident sun loading and from internal heat loss. Cooling radiators, integral to the Solar Array module provide a passive means to provide thermal control of that assembly. A radiator panel assembly will be used to thermally control the Avionics module. Unshielded areas will be covered with MLI. The MLI and radiator assembly also serve to provide some radiation shielding for the electronics.

GN&C

The core capability of the MicroSat to conduct precision rendezvous maneuvers and docking with a satellite is defined by its guidance, navigation, and control subsystem. This subsystem contains functional software that interacts with the various on-board hardware, processes the signal, selects the appropriate mission scenario, and issues a set of maneuver commands to actuators and valve drivers to meet the specified mission objectives. Figure 8 shows a general block diagram of the intrinsic GN&C function. The hardware components include: GPS for orbit estimation, MIR radar for close in ranging, Stereo and HiRes cameras for imaging and 3D positioning, Star Tracker for inertial orientation, RF COM subsystem to relay and receive data, and an IMU for vehicle stabilization and positioning. In addition, the actuator hardware includes attitude control jets, cruciform and axial divert thrusters, and the 6DOF vision based servo arms to provide a hard dock and to insert an adapter to the satellite service pin connector.

The GN&C function will need to control the spacecraft to better than one milliradian in orientation, five millimeters in translation, 10mg acceleration for performing orbital maneuvers and 1mg for close in inspection and docking. An initial set of the IMU performance requirements is listed in Table 8. Several candidate IMUs can support these requirements. The LN-200 seems to be a good candidate, however its 1milli-gee bias is marginal during docking maneuvers. This can be remedied by either procuring a more accurate set of accelerometers or making use of the millimeter position resolution of the microRadar.

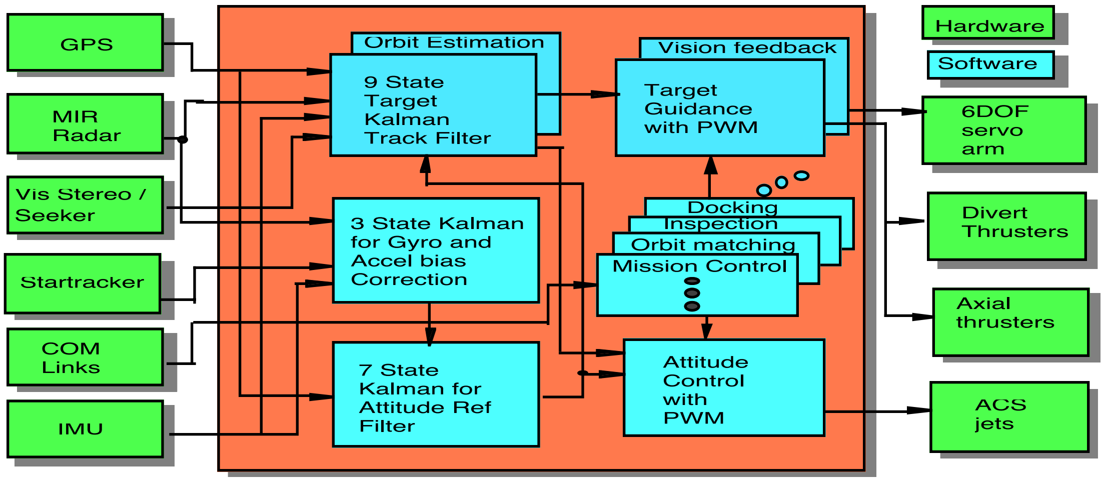

Figure 8 MicroSat rescue mission guidance and control block diagram.

Table 8 IMU Requirements and performance comparison.

IMU Parameters		**Rescue mission Requirements**	**LN-200**	**Systron Donnor μ-IMU**	**Honeywell 1305**
Accel Performance	**Unit**				
Operatinal range	g	< 40	40	100 +/-	50 (?)
Bias	mg (1σ)	< 1	1	5	3
Bias stability	mg	< 1	1	0.5	
Noise	μg/rt_Hz	< 50	50		50
G sensitive drift	μg/g2	< 100	17		50
Scale factor	ppm (1σ)	< 300	< 267	< 1500	600
Axes misalignment	mrad (1σ)	< 0.1	0.1	1.5	0.1
Data rate	Hz	> 50	400	65	400
Gyro Performance					
Operatinal range	deg/s	> 100	1000	100 +/-	500(?)
Bias	deg/hr	0	0	> 100 (?)	0
Bias stability	deg/hr	< 5	3	5	1
Random walk	deg/rt_hr	< 1	0.07	(200 ?)	0.125
Angle noise	μrad	< 1	1	26	7.3
Quantization	μrad	<1	0.24		13.5
G sensitive drift	deg/hr/g (rms)	< 5	1	2	1
Scale factor	ppm (1σ)	< 500	300	1600	300
Axes misalignment	mrad (1σ)	< 0.5	0.3	5	0.1
Data rate	Hz	> 50	400	65	400
Mechanical & Electrical Interfaces					
Start up time	sec (to stabilize)	< 60	5	30	
Supply voltage	Vdc	15+/-	15+/-	5 +/-	15 +/-
Power	W	< 10	10	5	12
volume	cm3	TBD	700		500
Size	inches	TBD	3.5Dx3.35H	4.3x2.5x1.75	
Weight	gm	< 500	700	650	450
Electrical I/F			1Mbps SDLC	128 Hz digital serial	
Environmental Requirements					
Temperature	deg C	54- to 85+	54- to 85+	40- to 85+	
Vibration	g (rms)	< 10	11.9	8	
Shock	g	< 200	90	200	
Prepared by: Larry Ng					
revised: 13Jan98					

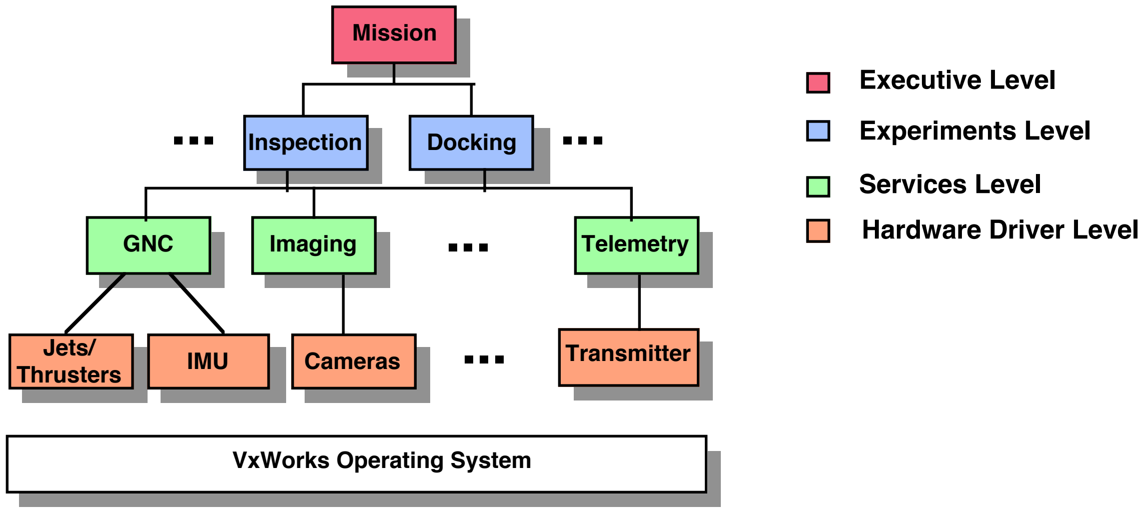

Figure 9 Hierarchical organization of mission software.

Table 9 Summary of processor requirements.

Processor	Required	Available	Use/Rationale
Throughput	90 MIPS	100MIPS	50% Contingency Factor
RAM	15 MB	32 MB	Code + Data. 50% contingency
RRPROM	3 MB	4 MB	3 code images. 50% contingency
ROM	4 KB	8 KB	Bootstrap. Estimate
NVRAM	2 KB	4 KB	Checkpoint State. Estimate

Software

The MicroSat software development environment uses the VxWorks real-time operating system (RTOS). VxWorks is a commonly used, well tested, RTOS that provides a rapid development environment for integration of new software modules. It is also portable among many processors. It has been used in space applications including JPL's Mars Pathfinder and the Clementine I spacecraft. LLNL has more than ten years of experience in building software for spacecraft applications. Many software modules for GN&C, imaging, target tracking, and other real-time operating codes can be adapted for the representative satellite mission.

In general, LLNL's software development philosophy is to maintain maximum flexibility for multiple reuses and to test the integrated software with realistic hardware-in-the-loop experiments - as soon as and as often as possible. Figure 9 shows the hierarchical organization of the mission software. Table 9 summarizes the memory and throughput requirement for the inspection/rescue mission. As indicated, we have built in 50% contingency margins in the processor and memory allocation. The peak software throughput requirement is 90 MIPS out of a 100, including a 50% contingency factor. The peak requirement occurs during docking, and is dominated by image processing functions.

Integrated Vehicle Testing

Ground performance testing is the key to the success of a MicroSat mission. It is crucial to be able to repeatedly practice and test the integrated vehicle's ability to perform precision orientation and translational maneuvers. These tests should include maneuvers to achieve orbit matching, endgame chase, inspection, docking, satellite servicing, and un-docking. Ideally, one would like to have a 6 Degrees of Freedom (DOF) test environment. However, in most cases a 5 DOF or 4 DOF environment is sufficient.

In order to support the testing of integrated MicroSats, LLNL has developed 4 DOF and 5 DOF dynamic air bearing ground testing facilities.[6] The 4 DOF facility is an air rail with 3 degrees of rotational freedom and one degree of translational freedom. The 5DOF facility is an air table with 3 degrees of rotational freedom and two degrees of translational freedom. These facilities enable low cost repeatable end-to-end performance testing of completely integrated MicroSat testbed vehicles, and full-up performance acceptance testing of final flight hardware and software before launch.

Air Table

One of our test vehicles on the air table is pictured in Figure 10. The vehicle sits on a hemispherical air bearing supported by 3 linear air bearings. The hemispherical air bearing allows $\pm 15^{\circ}$ pitch, $\pm$ 360° yaw, and $\pm 15^{\circ}$ roll while the 3 linear air bearings enables the vehicle to translate in 2 dimensions. This facility has been operational for over 10 months and has had over 500 experimental runs performed on it including 3 DOF tracking, 5 DOF tracking, 360° yaw maneuvers, and precision translational maneuvers.

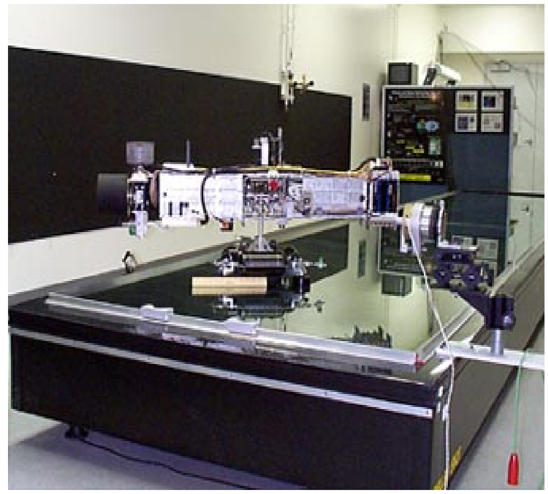

Figure 10 Forerunner Test Vehicle on 5 DOF Air Table.

Air Rail

The Engineering Test Vehicle (ETV100) is pictured in Figure 11 on the air rail. The air rail as pictured allowed only one degree of translation and was used in this configuration to test the ETV100's hydrogen peroxide divert capability. The air rail is being upgraded to enable full 4 DOF movement and will be used to practice high speed diverts, tracking, and docking maneuvers.

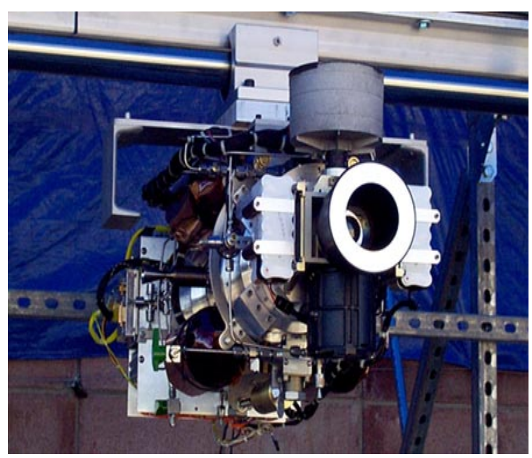

Figure 11 Engineering Test Vehicle (ETV100) on 4 DOF Air Rail.

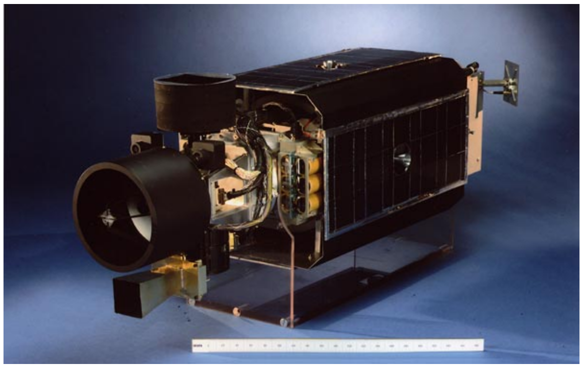

Figure 12 shows a full scale engineering model of the agile MicroSat.

Summary

To date LLNL has developed a preliminary conceptual design for an autonomous agile MicroSat for on-orbit logistics operations. Figure 12 shows a full scale engineering model of this vehicle. On going efforts are focused on integrated ground testing of prototype MicroSat vehicle testbeds and the development of advanced versions of our non-toxic propulsion systems that will provide multi-kilometer per second levels of on-orbit Δv capability.

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Biography

Dr. Arno Ledebuhr earned an undergraduate degree in Physics and Math in 1976 from the Univ. of Wisconsin and masters and doctorate degrees in Physics from Michigan State University in 1982. Dr. Ledebuhr spent the following four years at the Hughes Aircraft Company and earned 13 patents in projection display technology. He has been at Lawrence Livermore National Laboratory since 1986 and led the development of advanced sensors for the Brilliant Pebbles interceptor program and the design of the Clementine sensor payload. In 1996 he was the Clementine II program leader and is currently the MicroSat Technologies Program leader.

Joseph Kordas earned his B.S. in Physics from St. Procopius College and his M.S. in Biophysics from Michigan State University in 1974. He has worked at Lawrence Livermore National Laboratory since 1974. His assignments included lead experimenter for separator systems Laser Isotope Separation Program, sensor engineer Brilliant Pebbles Program, lead electronics engineer for sensor development Clementine Program, and Clementine II Deputy Program Leader. Currently, he is the Deputy Project Leader for the MicroSat Technologies Development Project.

Dr. Larry Ng received his B.S. and M.S. degrees in Aeronautics and Astronautics from the Massachusetts Institute of Technology in 1973, and a PhD degree in Electrical Engineering and Computer Sciences from the University of Connecticut in 1983 under a Naval Undersea Warfare Center (NUWC) Fellowship. In addition, Dr. Ng received his commission as an Air Force officer in 1973 and served at the Hanscom Air Force Base in Bedford, MA. His work experience include: four years with General Dynamics Electric Boat Division in Groton, CT., responsible for the development of the TRIDENT submarine digital control systems; seven years at the NUWC where he led the advanced development of the advanced sonar signal processing for the Seawolf submarine. Since 1986, he joined the Lawrence Livermore National Laboratory where he is currently the group leader of the signal/image processing and control group and is focusing his research in micro-spacecraft guidance and control and integrated ground testing. Dr. Ng is a member of several professional societies, including honorary memberships in Sigma Xi, Tau Beta Pi, and the National Research Council. He has published numerous papers in signal estimation, and precision vehicle guidance and control.

Mark Jones has worked in electronics since 1977. He holds a B.S. degree in Computer Engineering from the University of the Pacific, Stockton, CA. Mark came to LLNL in 1984 and has worked on space-related projects including Brilliant Pebbles, MSTI, Clementine, and Clementine II. Currently, Mark leads the avionics effort for the MicroSat Technologies Development Project.

Oliver Edwards earned his BS in Chemical Engineering from Worcester Polytechnic Institute 1980 and his MS in Applied Mathematics from Carnegie Mellon in 1986. He has worked for Exxon Chemical Company doing chemical reactor simulation and Laboratory automation. Currently, he works for Lawrence Livermore National Laboratory doing real-time software systems and Software Project Management.

Dr. John Whitehead earned his undergraduate degrees in both science and engineering from Caltech. He received his doctorate in mechanical dynamics and controls from the University of California, Davis in 1987. At the Lawrence Livermore National Laboratory, he led the development of miniature pump-fed rocket engines. Since 1995, he has contributed toward understanding unsolved propulsion problems, such as SSTO and Mars departure.

Richard Gaughan is the optical systems designer for the micro satellite development program. He has designed, built, and tested a wide variety of optical hardware for space flight systems.

Michael Dittman earned his undergraduate degree in mechanical engineering from California Polytechnic State University, San Luis Obispo. He received his masters in mechanical engineering, controls and dynamics from San Jose State University in 1993. Since 1987 at Space Systems/Loral he was a lead mechanical engineer for many satellite operations including alignments, assembly, integration and test, control mechanism design and development and spacecraft systems. Beginning in 1997 at Lawrence Livermore National Laboratory he has contributed to the design and development of micro satellite technologies including low pressure warm gas ACS systems.

Technical Information Department • Lawrence Livermore National Laboratory
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