A SMALL SATELLITE BUS DESIGNED FOR LOW EARTH ORBITING EXPERIMENTS

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INTRODUCTION

The concept of a small satellite bus, designed to support a variety of low earth orbit payloads, has led to the detailed design of such a bus and its launch vehicle adapter.

The basic bus, which weighs approximately 170 lbs, is sized for launch on either Pegasus or Scout. It provides three axis stabilization (gravity gradient boom plus constant speed momentum wheel), solar power (40 watts minimum after 1 year in polar orbit), an S-band TM transmitter (l watt output) and command receiver, a microprocessor-controlled telemetry encoder/formatter (usable at data rates up to 300 kbps), a command system providing logic level, digital word and relay outputs, a NASA-standard tape recorder (up to 700 Mbits of storage), and a simple thermal blanket/thermostatic heater temperature control system. Minimal redundancy is employed, consistent with the missions visualized for small experimental satellites.

Considerable flexibility of design is vested in the bus-launch vehicle adapter section. The adapter mechanical design varies depending on launch vehicle. A 30 lb. allotment is provided for Pegasus or Scout, and a detailed design has been carried out to validate this allotment. Future small satellite launch vehicles can also be accommodated by varying the mechanical design, although adapter weight may vary. For those payloads requiring higher apogee than the chosen launch vehicle can provide, the design allows incorporation of an additional solid (Star 6B, ~30 lb wt) or a monopropellant (MMH, ~55 lb wt) propulsion stage within the adapter section. In either system, the amount of propellant carried can be tailored to the mission; in the monopropellant case, the restart capability allows orbit tailoring if desired. In either case, the spent propellant casing/tank becomes a part of the stabilizing boom end mass.

The bus is designed to support interchangeable payloads of up to 85 lbs, although the availability of tailorable on-board propulsion allows a weight tradeoff here. For purposes of the design effort, it was assumed that the payload would be launched on top of the bus, and then oriented downward after stabilization in orbit. With this assumption, a payload volume 25 in. in diameter and 20 in. high can be accommodated within either the Pegasus or Scout fairing.

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This paper provides a more detailed description of the bus, discusses its expected performance, describes the options considered, and provides a rationale for the system choices made.

SYSTEM OVERVIEW

The small satellite bus has been designed to support a variety of payloads and mission requirements in a straightforward and economical manner. In this process, design heritage from many other APL-developed spacecraft has been utilized. A top level block diagram has been generated for the purpose of illustrating the major bus subsystems. This block diagram is included here as Figure 1. The spacecraft bus has been divided into seven major subsystems. These major subsystems will be described individually in the more detailed material which follows.

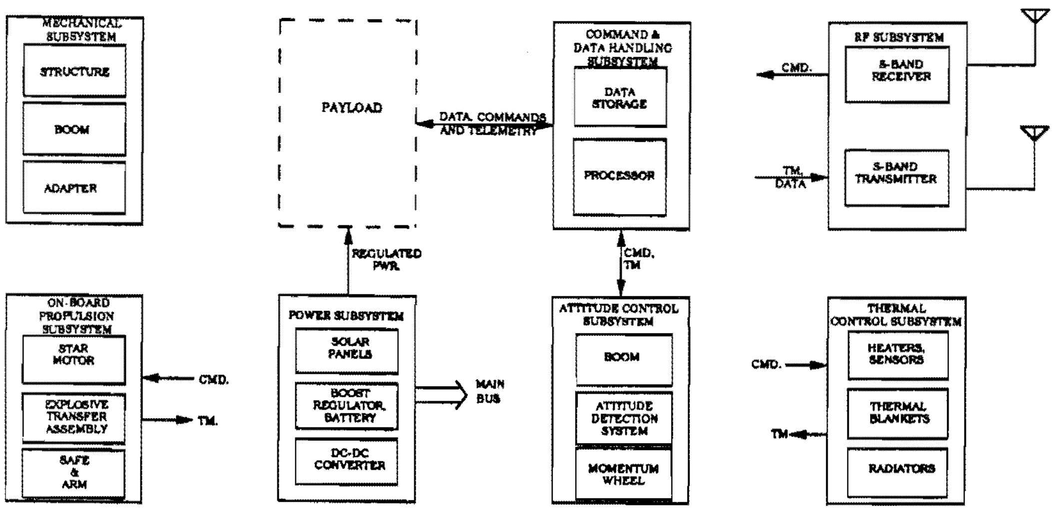

Fig. 1 Small Satellite Block Diagram

MECHANICAL SUBSYSTEM

A basic requirement of our study was that the spacecraft (bus plus payload package) be compatible with either Pegasus or Scout. Since the Scout fairing is smaller than the Pegasus fairing in all dimensions, the spacecraft is designed to fit within the Scout fairing.

The spacecraft bus structure is a simple, lightweight aluminum framework, with four aluminum honeycomb side panels which act as stiffeners. The major spacecraft electronic packages mount to the inside of these panels prior to integrating the panels into the framework. One panel is intentionally left blank to facilitate interior access. Figure 2 is a drawing showing this arrangement. Attached to the bottom of the spacecraft bus volume are three digital solar attitude detectors (DSADs) as well as the ASTROMAST® boom (to provide gravity gradient stabilization). The DSADs and the boom all fit within a cylindrical section which is part of the vehicle-spacecraft adapter section. If an on-board propulsion system is to be

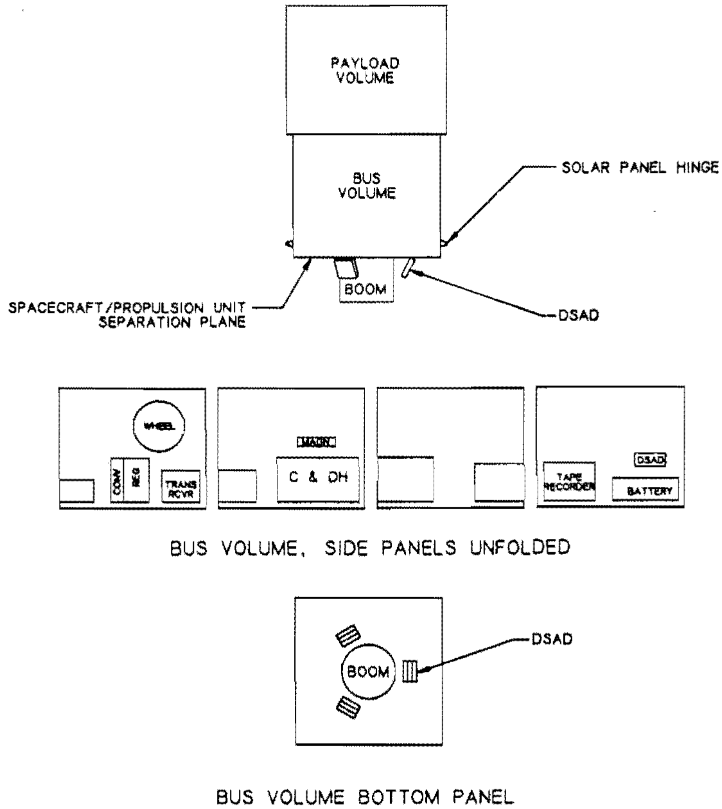

Fig. 2 Small Satellite Mechanical Layout

used for orbit adjustment, it is also housed in this cylindrical section. Two solar panels mount on hinges which are attached to the bus side panels. The hinges are the same, regardless of the launch vehicle; however, the solar panel deployment spars are fairing geometry dependent and therefore launch vehicle-dependent. The spent propellant casing/tank and part of the cylindrical section become part of a gravity gradient boom end-mass after propellant depletion and boom extension. Figure 3 illustrates the configuration just described at orbit insertion; the on-orbit configuration is shown in Figure 4. The simple design principles employed in this basic bus structure are similar to those employed by APL in its NASA-sponsored Small Astronomy Satellite (SAS) series.

In the design process, a detailed weight-size tally was kept current for both the Pegasus and Scout versions of the small satellite, in order to support the mechanical design. The final version of the Pegasus tally is included here as Table 1.

ON-BOARD PROPULSION SYSTEM

To reach orbital altitudes or inclinations, or to loft a spacecraft mass, which cannot normally be achieved with the standard three stage Pegasus

Table 1
Weight-Small Satellite on Pegasus

SUBSYSTEM	SIZE (in.)	WEIGHT (lbs)		
1.0 PRIMARY STRUCTURE		30		
1.1 Structure			30	E
2.0 SECONDARY STRUCTURE		10		
2.1 Mechanisms			10	E
3.0 RF		8		
3.1 Transmitter/Receiver	6x6x5		6	E
3.2 Rx Antenna			0.5	E
3.3 Tx Antenna	1 dia x 10		1.5	E
4.0 C&DH		27.9		
4.1 Command/Power Sw/Telemetry	8.75x12.75x8		14	E
4.2 Tape Recorder	6x7.75x8		13.9	H
5.0 ATTITUDE		18.5		
5.1 Wheel & Electronics	8 dia x 2.75		7.8	E
5.2 Z Torque Coil			2.2	E
5.3 Magnetometer	1.38x1.38x6.05		0.4	E
5.4 ASTROMAST & Rate Limiter	8.5 dia. x 7		4.5	E
5.5 DSADS (3)	3.175x3.175x.8		1.5	E
5.6 DSAD Electronics	4.5x4.87x2.12		1	E
5.7 Hysteresis Rods (2)			1.1	E
6.0 POWER		35.5		
6.1 Battery	3.5x5.5x10.3		10.5	E
6.2 Converter	2x7x7		3	E
6.3 Regulator	4x7x7		5	E
6.4 Solar Array	2@15x60x1		17	E
7.0 PROPULSION		27.9		
7.1 Propellant			15.2	H
7.2 Star 6B Inert, S&A			12.7	E
8.0 THERMAL		7		
8.1 Thermal Blankets			6.6	E
8.2 Heaters/Radiators/ Thermostats, etc.			0.4	E
9.0 HARNESS		15		
9.1 Harness			15	E
10.0 INSTRUMENT		73		E
11.0 ADAPTER & MISCELLANEOUS		27		
11.1 Adapter			15	E
11.2 Clamp			2.5	E
11.3 Balance Weights			5	E
11.4 Despin Weights			3	E
11.5 Adapter Harness			1	E
11.6 Separation Timer			0.5	E
Total		279.8		
Contingency (15% of items marked "E")		37.61		
TOTAL WITH CONTINGENCY		317.41		

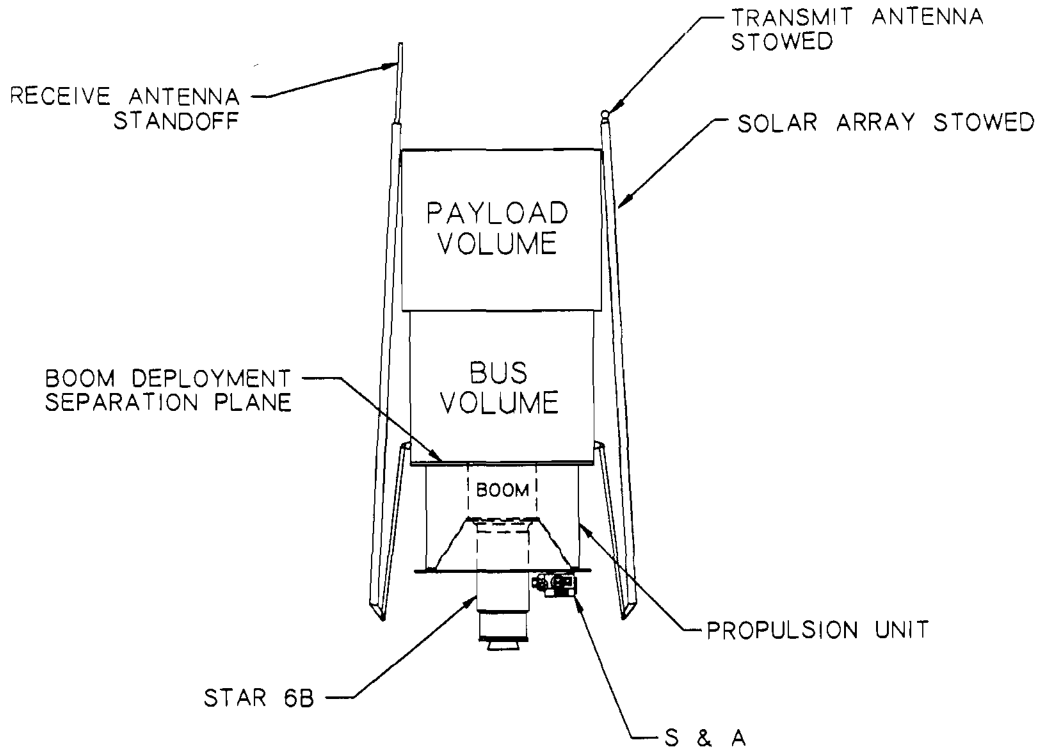

Fig. 3 Small Satellite Orbit Insertion Phase (Pegasus)

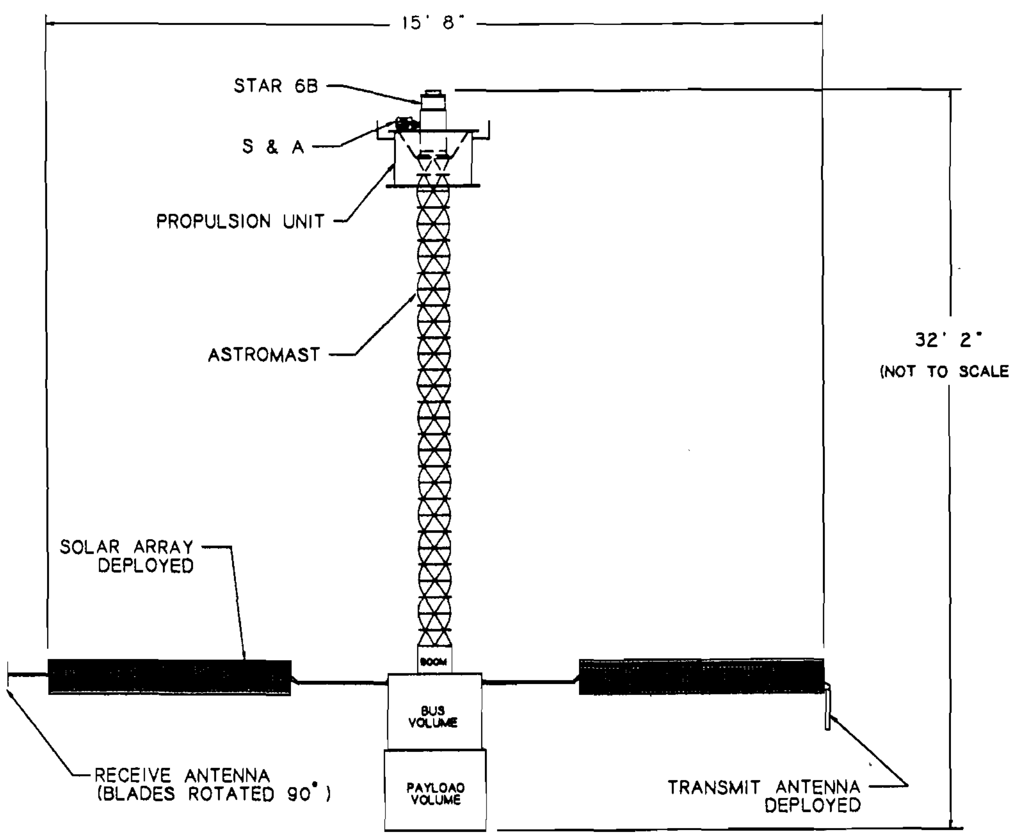

Fig. 4 Small Satellite Orbit Configuration (Pegasus)

or four stage Scout vehicles, the spacecraft has been designed to house an auxiliary propulsion system in the launch vehicle-satellite adapter section. In the case of Pegasus, the spacecraft would be placed in a transfer orbit, and then the auxiliary propulsion system would be employed as an apogee kick motor to achieve the final orbit. Tight control of the orbit injection point by the Pegasus third stage control system allows acceptable dispersion errors to be obtained with an on-board Star 6B solid rocket. In the case of Scout, the fourth stage and spacecraft are spin stabilized during the unpowered coast to apogee, and a restartable hydrazine orbit adjust system is more desirable in order to achieve the desired orbit and correct for the greater dispersion errors. The phases of Pegasus launch scenario are:

1. Pegasus third stage burnout

2. Coast to apogee
   - o Orient spacecraft spin axis for AKM burn
   - o Spin up spacecraft

3. At apogee
   - o Separate from Pegasus third stage
   - o AKM ignition

4. Following AKM burnout
   - o Despin spacecraft with a yo-yo device
   - o Stabilize spacecraft attitude

During phase 2, the Pegasus third stage guidance and cold gas systems are used to provide control (this capability is not available in the Scout launch vehicle option). The entire flight profile is accomplished autonomously by the Pegasus third stage flight computer and timed commands on the spacecraft.

In the case of Pegasus, a Morton Thiokol STAR 6B rocket motor was chosen to provide the total impulse required for this apogee maneuver (on other missions, a STAR 12 motor can be accommodated). This motor was developed as a spin-up and propulsion motor for re-entry vehicles. The design incorporates an aluminum case and carbon phenolic nozzle assembly. The propellant load of the STAR 6B can be varied between 5.7 and 15.7 lbm to satisfy a variety of mission requirements. A Safe and Arm Device (S&A) which has flown successfully on Delta, Ariane, and Shuttle will also provide ignition for the STAR 6B.

The greater dispersion errors associated with the Scout's unguided fourth stage require substantial changes in both the propulsion system and launch scenario. A restartable monopropellant hydrazine Orbit Adjust and Transfer System (OATS) which has previously flown on the NOVA Navy navigation satellites is proposed to meet the orbit insertion requirements of the Scout launch vehicle. The phases of a Scout launch scenario are:

1. Launch into a parking orbit, separate from the fourth stage.

2. Despin spacecraft and deploy solar panels.

3. Use magnetic torquers to spin-up the spacecraft (about 5 rpm).

4. Observe orbit parameters.

5. Orient spacecraft spin axis for OATS burn with magnetic torquers.

6. Command OATS burn; loop back to step 4 as required to achieve a reasonably circular orbit.

7. Upon achieving the desired orbit, the remaining fuel is vented.

This procedure requires ground support for the orbit insertion activities as well as additional time to observe orbit parameters between iterations.

ATTITUDE CONTROL SUBSYSTEM

The attitude determination and control system for the spacecraft has been selected to provide reasonable instrument pointing and stability accuracy, commensurate with the requirements of many typical sensors; that is, less than 4° peak pointing errors with ground-based attitude reconstruction accurate to 1° or better. It is a simple system based on proven techniques utilizing passive gravity-gradient restoring torques to maintain a nadir pointing orientation. Gyroscopic stabilization, achieved through the addition of a constant speed momentum wheel, augments the gravity-gradient stability to provide passive 3- axis attitude stabilization of the spacecraft. Attitude determination is accomplished by combining measurements of the spacecraft-sun vector and the local magnetic field vector. Minimal redundancy is provided within the system. This attitude system is based on the system used on the APL-designed NOVA series of Navy navigation satellites. The NOVA's have typically achieved peak pointing errors of 2-3°.

The attitude measurement system selected to meet these requirements includes a set of two-axis digital sun sensors (DSADs) and a vector magnetometer. Three DSADs with fields of view of ±64° are mounted around the spacecraft 120° apart on the boom side of the spacecraft with their normals canted down from the boom axis by approximately 64°. These sensors will provide slightly more than hemispherical coverage on the zenith pointing side of the spacecraft. The illuminated sun sensor will measure the solar orientation relative to the sensor normal to an accuracy of ±0.5°. A vector magnetometer is used to measure the magnitude and direction of the local magnetic field vector relative to the spacecraft, providing the second attitude measurement. The magnetometer and DSAD outputs will be compared on the ground with modeled sun and magnetic field data to extract the three-axis attitude of the spacecraft. Because of uncertainties in the magnetic field model, spacecraft attitude determination accuracy will be limited to approximately 1°.

The attitude control system is a passive system consisting of an 8-meter ASTROMAST® gravity-gradient boom, a constant speed momentum wheel, passive magnetic hysteresis rods, and a Z-axis magnetic coil. In the mission configuration, the spacecraft will be nominally oriented with its boom axis along the local vertical with the instrument nadir pointing, and the positive spin axis of the wheel aligned with the normal to the orbit plane. Gravity-gradient restoring torques provide two-axis attitude (pitch & roll) control, aligning the spacecraft's axis of minimum moment-of-inertia (along the deployed boom and referred to as the Z- axis) such that the instrument maintains its nadir-pointing attitude. The bias momentum of the spinning wheel gyroscopically couples yaw motion with roll allowing completely passive, three-axis stabilization. Magnetic hysteresis rods provide passive damping of the attitude motion enabling the spacecraft to achieve 2-3° pointing performance. To assist with the gravity-gradient capture and initial attitude stabilization of the spacecraft, an electromagnetic coil is included in the system, oriented such that the magnetic dipole that it produces is along the spacecraft Z-axis. The magnetic coil provides magnetic stabilization prior to boom deployment and enables active, ground-commanded libration damping following boom deployment. Should the spacecraft inadvertently become stabilized upside down, the Z-coil may be used to assist in the re-inversion process.

Several other candidate attitude control systems have been studied for the small spacecraft bus. Of these, the least expensive system having acceptable performance is baselined; it is similar in most respects to that employed in the Navy's NOVA navigation satellites. An active system, consisting of a variable speed momentum wheel and an IR earth scanner, can be accommodated by the spacecraft bus if desired. Such a system is capable of approximately ±0.5° accuracy, but at the cost of additional weight, power and complexity. A system of this type was flown on the APL-designed MAGSAT satellite.

RF SUBSYSTEM

The small satellite bus RF subsystem has been designed under the assumption that a 5-meter ground receiving antenna and a 10-meter ground transmitting antenna are typical of those used in support of a mission of this type. An S-band transponder is used on board the spacecraft for command reception and data/telemetry transmission. The system consists of a receiver and dipole antenna for uplink communications and an S-band transmitter, connected to a shaped beam bifilar helix antenna, for downlink transmissions. The transmit and receive antennas are mounted on the ends of opposite solar array panels so as to provide coverage of the visible earth once the spacecraft attitude is stabilized.

The receiver is a phase-locked double conversion superheterodyne tuned to an assigned frequency in the 1760-1840 MHz uplink band. The receiver is designed to be compatible with the U.S. Air Force Space Ground Link Network's (SGLN) remote ground stations. The modulation is FSK/AM/PM. The uplink data rate is 2 kbps. A 1 kHz triangular wave is amplitude modulated onto the tones for data synchronization and the composite signal is phase modulated onto the RF command carrier.

The transmitter is designed to operate in the 2200-2300 MHz frequency band, and consumes approximately 17 watts when powered. On orbit, the transmitter would normally be powered only when the spacecraft is over the receiving station so that the average power dissipated per orbit will be relatively low. The transmitter is phase modulated by a composite waveform consisting of biphase-level instrument data at up to 300 kbps that are convolutionally encoded (rate 1/2, constraint length 7), and biphase-space telemetry data at 1.6 kbps that are phase-shift-keyed onto a 1.7 MHz subcarrier (Convolutional encoding is used to increase downlink margin. This encoding is easy to implement in the spacecraft, and the signal can be conveniently decoded on the ground). APL has employed an S-band system having these modulation characteristics on the APL-designed SDIO DELTA-183 sensor research satellite.

The spacecraft antennas are small and lightweight to allow mounting on the ends of the solar array panels from which clear fields of view may be obtained. The antennas provide coverage over the visible earth once the spacecraft attitude is stabilized. Prior to spacecraft attitude stabilization, the reception of commands at the spacecraft and the reception of telemetry at the ground station will depend on the attitude of the spacecraft relative to the ground station and on the coverage provided by the transmit or receive antennas. The receive antenna has a characteristic dipole pattern with coverage over approximately 75 percent of the sphere about the spacecraft, providing a reasonably high likelihood of receiving a command from the ground station regardless of spacecraft orientation. The achievable antenna gain is approximately -10 dBic RHCP, over 75 percent of the sphere about the spacecraft. The transmit antenna is a shaped beam RHCP quadrafilar helix. Shaping the beam compensates for off-axis slant range loss. The antenna is mounted on the end of a solar panel and positioned to radiate along nadir. The antenna beamwidth is sufficient to cover the visible earth when the spacecraft attitude is maintained within ~4 degrees of nadir.

Uplink and downlink margin calculations have been carried out for the RF subsystem described above. These calculations indicate a command margin of 22 dB, a received carrier margin of 24 dB, and a received data margin of 6.1 dB, for passes 5° or more above the horizon.

COMMAND AND DATA HANDLING SUBSYSTEM

A microprocessor-based C&DH subsystem processes uplink command signals, and will store and format payload data and housekeeping telemetry information for transmission to the ground. The subsystem provides tape storage for 700 Mbits of instrument data, maintains the mission and universal time clocks, and provides autonomous spacecraft control with uploadable software.

The command portion of the C&DH system has the following specific capabilities:

1. Process 2 kbps uplink command messages to activate real-time relay, pulse, and short data transfer commands.

2. Provide delayed command capability to permit spacecraft commanding without ground station contact.

3. Reconfigure the spacecraft in response to a low voltage sense flag from the power system electronics.

4. Provide spacecraft power fault protection and autonomous control with uploadable software.

All command forms are available on either a real-time or delayed basis. Delayed commands are tagged with the correct mission elapsed time for execution, and are confirmed by requesting a command system memory readout. The command system performs several checks (including a Hamming code check) on all uploaded command messages before they are processed. Delayed command messages are checked prior to execution. Command messages which fail any of the bit checks are not executed.

The command system inputs processed housekeeping data from the Data Handling Subsystem (DHS), compares the data to a set of (uploaded) limits, and takes preprogrammed action to correct detected fault conditions. The command system samples a low voltage sense switch (LVSS) flag from the power system, and sheds noncritical spacecraft loads in the event of an undervoltage condition. All autonomous control and LVSS functions are individually maskable by ground command.

The command system consists of a single (non-redundant) command processor and a power switching unit. The command processor consists of three circuit boards, one of which contains a radiation-hardened, low single event upset rate microprocessor (any one of several can be accommodated, depending on actual payload requirements) and its associated electronics. The power switching unit consists of one board containing the relays needed to accomplish spacecraft and instrument power and signal switching functions.

The data handling portion of the C&DH system has the following specific capabilities:

1. Gather, format, and output real-time housekeeping data at 1.6 kbps using a fixed telemetry format.

2. Store 0.2, 0.4, or 0.8 kbps housekeeping data in its internal solid state memory during payload operational periods.

3. Dump the flight recorder and housekeeping memories upon command.

4. Maintain and distribute mission elapsed and universal time.

5. Provide a launch vehicle data interface.

6. Store full orbit attitude housekeeping data.

Data handling system operation is controlled via a 32-bit data command from the command processor. The data handling system processes housekeeping data received in the form of single ended analog voltages, differential analog voltages (e.g., current measurements), digital telltales, and serial digital data. Serial digital data is gathered by the data handling system using a four line interface consisting of a "Read Out Gate" signal from the DHS, a DHS output clock to the data source, a frame pulse from the DHS to the data source, and a data line.

POWER SUBSYSTEM

The small spacecraft bus employs solar cells for primary power collection, and uses a boost regulator to charge a Ni-Cd battery stack for energy storage. An unregulated battery bus and a regulated DC-DC converter provide power for the bus and payload subsystems, respectively. Battery cells are sized to operate at a low depth-of-discharge, ensuring battery life commensurate with the mission. An up-to-date tally of spacecraft power requirements is kept to assist in the detailed power system design, as indicated in Table 2.

The baseline design uses two solar panels. The two panels are deployed at a fixed angle of rotation with their axes lying along the -X and +X (velocity vector direction) spacecraft axes. The rotation angle, defined as the angle between the panel normal and the spacecraft Z axis, is chosen to optimize array output for the desired spacecraft orbit. For a polar, 1000 km circular orbit, this angle is approximately 30°. The array orbital average power output under these conditions is shown in Figure 5.

It is possible to improve this orbit average power output by several means, if future payloads require additional power. These include; use of a peak tracking scheme in the boost regulator (adding about 15% more power); adding two more solar panels in a fixed orientation, normal to the first two (adding about 40% more power, at a weight penalty of about 17 lbs);

Table 2
Small Satellite Power Requirements

LOAD	(WATTS)	DUTY CYCLE	ORBIT AVG POWER (WATTS)
Battery	3.0	1.00	3.0
Boost Regulator	6.4	0.63	4.0
DC-DC Converter	4.9	1.00	4.9
Heaters	5.0	0.00	0.00
Receiver - On	3.0	0.02	0.06
Receiver - Standby	2.0	0.98	1.96
C&DH	7.0	1.00	7.0
Momentum Wheel and Elec.	5.0	1.00	5.0
Magnetometer	1.5	1.00	1.5
DSADs	1.2	1.00	1.2
Z-Coil	10.0	0.00	0.0
Subtotal			28.6
Payload	50.0	(1)	
Transmitter	17.0	(1)	
Recorder - Standby	2.0	(1)	
- Record	10.0	(1)	
- Playback	19.0	(1)	
Subtotal			10.4

NOTE(1): Duty cycle of these components is mission scenerio-dependent.Orbit average power available is 10.4 watts.

or making the two panels rotatable (adding about 19% more power, but at a cost in complexity).

The battery is made up of twenty-two 3-ampere-hour (Ah) Nickel-Cadmium (Ni-Cd), hermetically sealed cells having a discharge voltage of 28.0 volts. Typically, cells of this type, with an appropriately designed retaining structure, will have an energy density of about 10 Wh/lb and a true capacity of about 105 Wh.

The battery is sized to supply a peak load current in the range of 4 amperes, which is well within the current delivery capability of the selected Ni-Cd cells. The system can survive at least one cell failure with only a slight impact on payload operation time. Predicted battery lifetime (in number of charge-discharge cycles) is well in excess of 1 year for the spacecraft temperature variations typical of a precessing (i.e., non-sun synchronous) orbit.

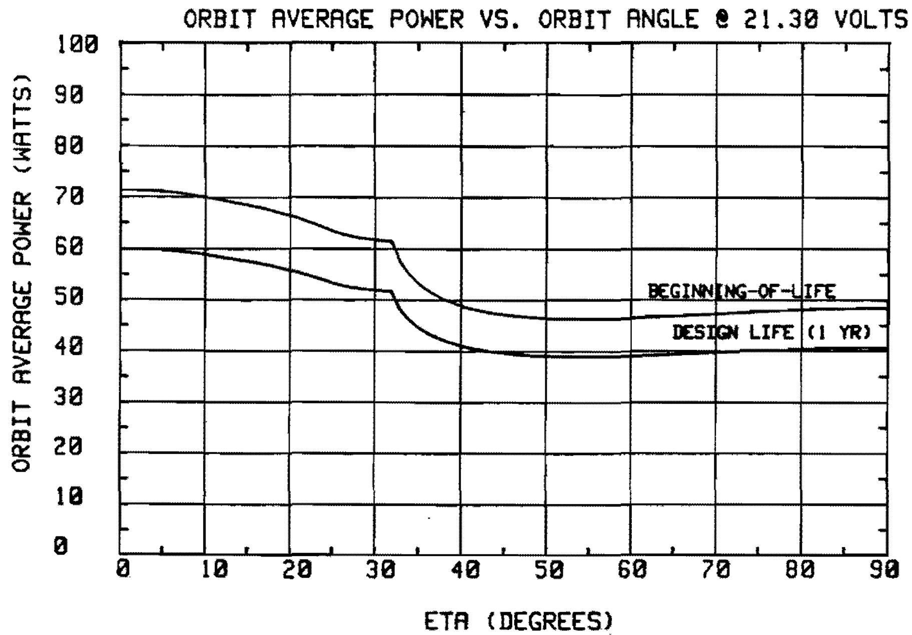

Fig. 5 Small Satellite Orbit Average Array Power vs. Orbit Plane Angle

THERMAL CONTROL SUBSYSTEM

The small satellite thermal design uses radiators and multilayer insulation blankets to minimize the thermal dependency on external heat inputs and to bias spacecraft temperatures to an acceptable level during periods of nominal orbit average internal dissipation. The design assumes that white paint is used as the radiator coating since it is inexpensive and has optical properties (α=.21, ε=.85) suitable for this application. Where coupling to space or other surfaces is undesirable, a standard twenty-layer blanket with a beta cloth outer layer reduces effective emittance to about .03. A thermostatically-controlled heater circuit is provided for those components which may require cold protection.

Those bus packages and internal structure having no view to space are coated with a high emittance paint (such as Chemglaze Z306) to enhance radiative heat transfer. Heat is radiated to space from a portion of the external panel surfaces. The ratio of panel radiator area to blanket area required to meet the design specifications may be determined by thermal analysis once the mission is defined. The boom side of the spacecraft bus is blanketed to the extent allowed by the mechanical interface.

Since the solar panels have cells on both sides, very little can be done in the way of thermal control. Instead, they are designed to tolerate a wide temperature range (-80°C to +80°C). To shield the spacecraft from these large temperature swings, the solar panel mounts are conductively isolated from the spacecraft body. Thermal blankets on the spacecraft will tend to reduce radiative couplings.

The boom is a self-contained system; like the solar panels, the boom will experience large temperature swings and is conductively isolated from the spacecraft body.

The thermal design of the payload relies on thermal capacitance to limit temperature rise during its operational periods. If necessary, the orbit average temperature of the payload may be biased towards the cool end of the allowable range during nonoperating times to increase the operating time limit of the payload.

A 35-node steady state computer model of the spacecraft was developed to support the conceptual thermal design effort. To simplify the thermal analysis, the payload and the spacecraft bus were assumed to be thermally isolated from each other. However, conductive coupling will help to dampen temperature fluctuations within the payload electronics; both methods result in a workable thermal design. Steady state runs were performed for various orbit attitudes to determine the overall hottest and coolest orbit environments for a "typical" payload. A transient model was also created, using satellite mass information. Based on these analyses, thermal design limits for each of the major satellite subsystems have been established. These limits are indicated in Table 3; in general, the limits are reasonable for current space hardware.

Table 3
Small Satellite Thermal Design Limits

Component	Design Limits (°C)
Spacecraft Bus	
General Electronics	-23 To +55
Battery	0 To +30
Recorder	0 To +35
Payload Electronics	-23 To +55
Antenna Array	-30 To +65
Solar Panels	-80 To +80
Boom	-20 To +70 Stowed -60 To +80 Deployed

CONCLUSION

The approach outlined in this paper has resulted in the design of a small satellite bus useful for supporting a variety of low earth orbit payloads. The design is based on an extensive background of experience in small earth satellites by the Applied Physics Laboratory. Because of this experience, the approach can draw on a heritage of the Navy's Transit, TIP, NOVA and Geosat programs, and NASA's Beacon Explorer series, Small Astronomy series, MAGSAT and AMPTE-CCE spacecraft.