# DESIGN OF A SMALL SATELLITE FOR USE IN ASTRONAUTICS EDUCATION

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The Naval Academy is pursuing a small satellite project to give midshipmen hardware and ground control experience. The concept is for a lightweight, gravity-gradient stabilized satellite to be deployed from a Get Away Special cannister; the payload would be two small, low-resolution imagers (one visible and one infrared). The 40 foot diameter ground station antenna being installed at the Naval Academy can receive a power of 10 milliwatts from a satellite that uses an omnidirectional antenna and is in a typical Space Shuttle orbit. Approximately 5 to 7 watts of power is constantly available to the payload of up to 10.4 lb (4.7 kg). Thermal control is largely passive, but some heating will be required during the longest eclipses. Once in orbit, the spacecraft would be used in astronautics courses for such assignments as orbit determination, decoding telemetry, and processing images.

## INTRODUCTION

Few astronautics students have the opportunity to gain experience with spacecraft hardware or learn ground control by using a satellite in orbit. Proposed small satellite networks for the military services would place control of these assets in the hands of the local commander, so it would be helpful for future users to learn ground control at the undergraduate level. The Naval Academy is pursuing project Midsat (MIDshipmen SATellite) to place a spacecraft in orbit that would be used in the undergraduate astronautics track curriculum in the Aerospace Engineering Department. A

lightweight satellite is optimal for this mission due to simplicity and low cost. The purpose of this paper is to determine the feasibility of such a project.

CONCEPT

Midsat is a small, low-powered vehicle with an imaging payload. The baseline launch option is the small Get Away Special (GAS) cannister. The maximum allowed payload weight is 100 lb (45.4 kg), the payload diameter is 19.75 inches, and the payload height is 14.13 inches[1]. Although designed for a GAS/Space Shuttle deployment, the satellite could use other vehicles offering available space to piggyback payloads. Since the Naval Academy is located at 39° North lattitude, the only acceptable orbits are those with higher than 39° inclination. A satellite at this inclination will pass overhead with enough time in view for downlinking telemetry. The Spacelab 1 STS mission used a 57° orbit, but with an altitude of only 133 nm. Midsat requires a 200 nm (371 km) circular orbit for reasonable satellite lifetime. The Shuttle is able to achieve 200 nm at 57°, so this assumption seems within reason. This orbit has a period of 92 minutes and a velocity of 7.685 kilometers per second. The maximum time that the satellite will be within view of the ground station is approximately 10 minutes[2], so most passes will be of the order of five minutes. The primary problem with this orbit is that the lifetime is still short. Charts prepared by the Navy Space Systems Activity[3] show that the orbit lifetime for this design would be approximately 300 days. At most, the satellite could be used in classes for one academic year. Small satellite concepts such as Orion[4] (Naval Postgraduate School) that include propulsion systems overcome this difficulty at the cost of increased weight and complexity.

Midsat is intentionally simple and small so that the bus could be assembled and tested using Naval Academy facilities. Apart from the ground station that is scheduled for completion late in 1988, facilities under development include a vibration facility for GAS-sized spacecraft that is planned for installation in 1989 and a high-altitude chamber (perhaps a thermal vacuum chamber if funds permit) for 1990.

The payload will be two simple, low-resolution imagers such as CCD cameras. Visual systems will have the most impact upon the midshipmen because they relate better to something that can be seen. One camera sensitive to visible wavelengths and one to infrared would make possible

imaging of cloud cover both day and night. Cloud cover is always of great interest to the Navy, so it represents a good teaching subject. If each camera had a 100 by 100 array of pixels with 8-bit resolution, a single image would be 80,000 bits. More than 10 images could be stored in a megabit of memory.

Figure 1 shows the layout of the satellite body. It is essentially a shell that is 16 inches (40.7 cm) in diameter to allow for more than one inch of clearance between the spacecraft and the walls of the GAS cannister. The body is 12 inches (30.5 cm) in height to allow room at the bottom of the GAS cannister for a spring-loaded deployment mechanism. A gravity-gradient stabilization boom with a length of 60 ft (18.3 m) is attached at the top of the body. This boom is identical to the one used by the OSCAR-class Transit navigation satellites[5]. A block of metal approximately the same mass as the spacecraft body will be attached at the tip of the boom.

SPACECRAFT SYSTEMS

Each of the spacecraft systems must be considered in order to develop a complete design. The following paragraphs describe the communications, attitude control, power, thermal control, structure, command and telemetry, and mass calculations.

Communications[6,7]

The facility that makes use of a low-power satellite possible is the 40 ft (12.2 m) diameter antenna ground station that is presently being installed at the Naval Academy[8]. With such a high gain, this antenna can easily receive low-power broadcasts from satellites using omnidirectional antennas. At the present time, transmitting capability for the ground station is not planned, but could be added if requirements change. The slant range is given by the expression

$$SR = \sqrt{(R_e + h)^2 - R_e^2 \cos^2 \varepsilon} - R_e \sin \varepsilon$$

where $R_e$ is the radius of the Earth (6378 km), h is the altitude (371 km), and $\varepsilon$ is the elevation angle. With useful passes limited to those that appear higher than 10° above the horizon with respect to the ground station, the maximum slant range is 1235 km.

To tip mass

Gravity-gradient boom
(60 ft long)

Antenna

12 in

Solar cells
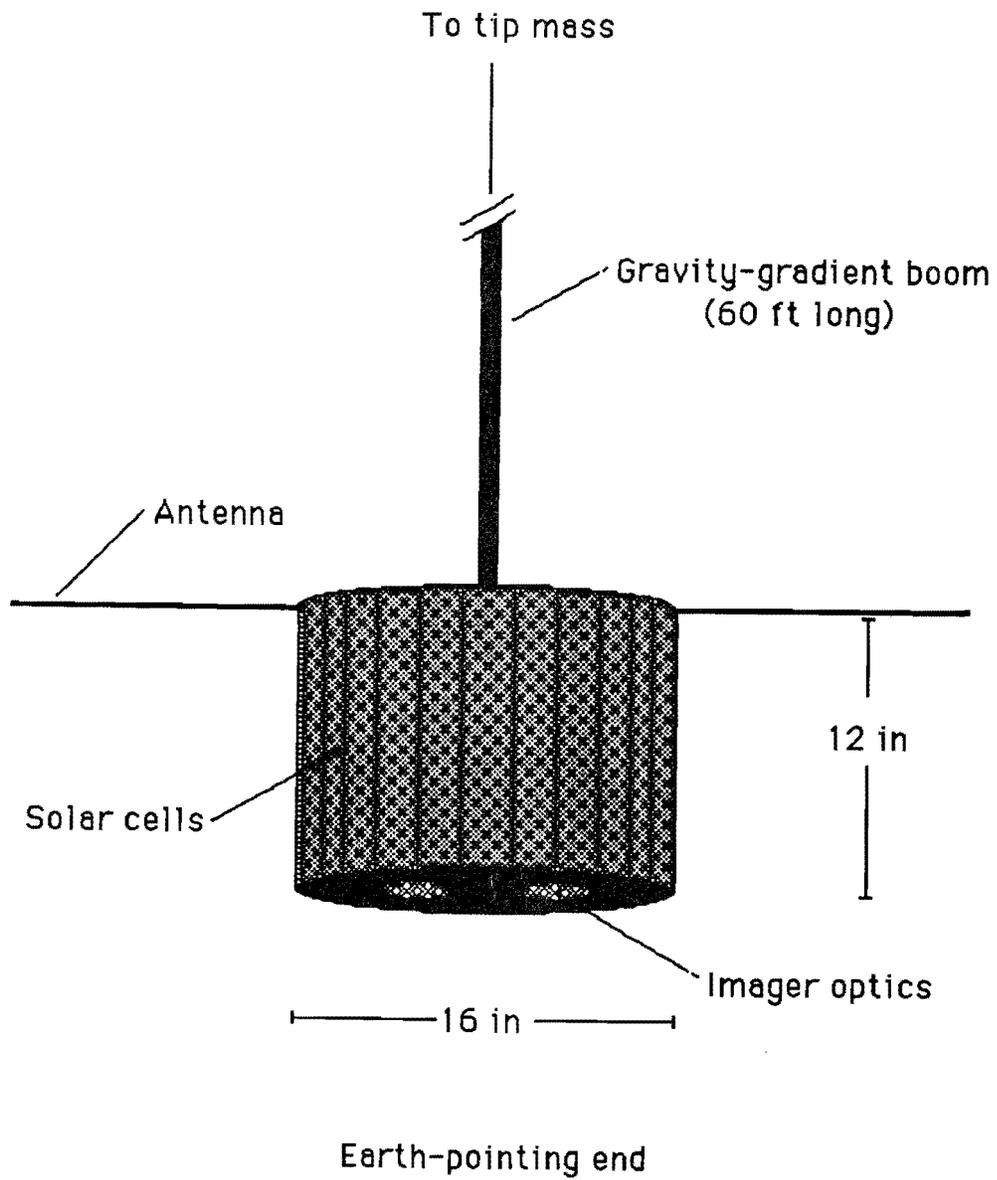
16 in

Imager optics

Earth-pointing end

Figure 1. Midsat Configuration

Path loss for the signal is a function of the slant range and signal wavelength. For 2.2 GHz (the ground station S-band capability), the wavelength is 0.1362 meters. Path loss in decibels is given by

$$L_s = 10 \log\{(\frac{\lambda}{4\pi\,SR})^2\}$$

and is -161 dB for the maximum slant range.

For the downlink calculation, see Table 1. Spacecraft transmitted power was assumed to be 1 W, or 0 dBW. The antenna on the spacecraft is omnidirectional, so it has no gain. Incidental losses (atmospheric loss, polarization loss, pointing loss, etc.) were assumed to total -10dB, slightly worse than for a typical downlink. The usual 3dB margin is included. The size of the ground station antenna is most striking when its gain is calculated. The expression for antenna gain in decibels is

$$G = 10 \log\{\eta(\frac{\pi D}{\lambda})^2\}$$

where G is the gain, $\eta$ is the efficiency (assumed to be 0.8), D is the diameter of the antenna, and $\lambda$ is the wavelength. For 2.2 GHz, the gain is +48dB. Noise density in dBW/Hz is given by

$$N_0 = 10 \log(kT)$$

where k is Boltzmann's constant (1.38 x $10^{-23}$ W/Hz K) and T is the noise temperature. The noise temperature for the ground receiver was assumed to be 800K (based upon typical cases) which yields a noise density of -200 dBW/Hz. A bandwidth of 10 kHz was chosen; note that one million bits can be transmitted in less than two minutes at this rate. In decibels, 10 kHz is 40 dB Hz. Subtracting the bit rate and adding the noise density to the received power results in Energy per bit/$N_0$ (i.e., signal-to-noise ratio) of +34 dB. The level required to achieve bit error rates of $10^{-5}$ using pulse-code modulation is no higher than +13dB. Transmitted power could be reduced to 10 milliwatts (-20 dBW) and the signal-to-noise ratio would still be sufficient (+14 dB) to achieve the $10^{-5}$ bit error rate. Another possibility would be to transmit enough power so that other ground facilities besides the one at the Naval Academy could recieve the signal.

Table 1

DOWNLINK CALCULATION

| | |
|---|---|
| S/C Transmitted power | 0 dBW |
| S/C Antenna gain | 0 dB |
| Margin | -3 dB |
| Incidental losses | -10 dB |
| Path loss | -161 dB |
| Receiver antenna gain | +48 dB |
| | |
| Received power | -126 dBW |
| | |
| Noise density | -200 dBW Hz |
| Bit rate | 40 dB Hz |
| | |
| Energy per bit/$N_0$ | +34 dB |

The uplink calculation is shown in Table 2. The ground station transmitted power was assumed to be 10 W, well below the capacity for a ground station of this size. The noise temperature was assumed to be 1500K for a more noisy spacecraft receiver. The noise density is slightly larger than that for the downlink, but the increased transmitter power more than compensates. Using the same bit rate as for the downlink, the result is Energy per bit/$N_0$ of +41 dB. The uplink will have a bit error rate better than $10^{-5}$ even at power levels as low as 10 milliwatts.

Attitude Control[5]

Gravity gradient stabilization is simple, uses no power, does not require attitude sensors, and has been proven stable to within 10° by the OSCAR-class Transit satellites. Ten degrees of Earth-pointing accuracy is sufficient for the low-resolution payload. The largest disturbing torque for a low-orbit satellite will be that due to atmospheric drag. The expression for torque is

$$T_{aero} = \frac{1}{2}\rho v^2 C_d A d$$

Table 2

## UPLINK CALCULATION

| | |
|---|---:|
| Ground transmitted power | +10 dBW |
| Ground antenna gain | +48 dB |
| Margin | -3 dB |
| Incidental losses | -10 dB |
| Path loss | -161 dB |
| S/C Receiver antenna gain | 0 dB |
| Received power | -116 dBW |
| Noise density | -197 dBW Hz |
| Bit rate | 40 dB Hz |
| Energy per bit/$N_0$ | +41 dB |

where $\rho$ is the atmospheric density ($10^{-11}$ kg/m$^3$ for 371 km altitude), $C_d$ is the drag coefficient (typically 2), A is the cross sectional area of the spacecraft body (0.124 m$^2$), and d is the moment arm (half the length of the boom, or 9.1 m). Assumptions for this model are that drag will only be exerted on the spacecraft body and that the mass at the end of the boom is is equal to that of the spacecraft body. The torque is 6.66 x 10$^{-4}$ Nm. Solar pressure torque is calculated by substituting solar pressure for the dynamic pressure and drag coefficient factors of the aerodynamic torque equation. The solar pressure is approximately 6 x 10$^{-6}$ N/m$^2$, so the torque is 6.77 x 10$^{-6}$ Nm, about 1% of the aerodynamic torque.

The spacecraft is modelled as two masses, each of 22 kg, connected by a massless boom of length 18.3 m. The mass moment of inertia for this idealized configuration is

$$I = \frac{1}{2}mL^2$$

where m is 22 kg and L is the boom length. The moment of inertia for this model is 3684 kg m$^2$. The gravity gradient torque is

$$T_{GG} = -\frac{3}{2}\frac{GM_e}{R^3} I \sin 2\theta$$

where G is the universal gravitational constant, $M_e$ is the mass of the Earth, R is the radius of the orbit (radius of the Earth plus altitude), I is the moment of inertia, and $\theta$ is the angle of the spacecraft subtends from the local vertical axis. With an angle of 10° (the maximum expected pitch or roll angle), the gravity gradient torque is 0.0025 Nm, almost 4 times the aerodynamic torque. Therefore, the gravity gradient system will stabilize this spacecraft.

The tip mass is actually budgeted to be 20 kg, slightly smaller than that of the spacecraft body mass so that there will be mass allowed for deployment hardware in the GAS cannister. If the mass is made of a high-density metal such as tantalum, it will be a cube that is less than 11 cm on each side. The boom used in this analysis is made of a beryllium-copper tape that forms a tube when unfurled. It has a weight of 0.8 lb (0.363 kg) for the length used. Magnetic hysteresis rods will be used to damp the oscillations of the spacecraft. Two of the slender nickel-iron alloy rods will weigh about 0.13 lb (0.06 kg). This stabilization technique does not control yaw, but yaw stabilization is not needed for the low-resolution imaging payload. The boom could be a problem in that this design does not include any system for orienting the spacecraft before deployment. One concept is to use a retractable boom so that more deployment attempts can be made in case the spacecraft is captured upside-down[9].

Power[10]

Midsat has solar cells mounted on the cylindrical body and on the top surface (the one to which the boom is attached). The effective area of the side panels is the rectangular projection of the surface, approximately 0.12 m$^2$. The top panel has approximately 0.10 m$^2$ of usable area for solar cells since a 0.2 m square area is reserved for stowing the tip mass before deployment. The worst case orbit for power generation is when the line that is perpendicular to the orbit plane is itself perpendicular to the Earth-Sun line. If $\Omega$ is the angle between the longitudinal axis of the

spacecraft and the Earth-Sun line, the equation that governs effective panel area is

$$A_{eff} = 0.12m^2 \sin \Omega + 0.10m^2 \cos \Omega$$

where the first term represents the side area and the second term the top area. The satellite is in sunlight during 55.7 minutes of the orbit, so the angle spent in sunlight is (55.7 min/92 min) x 360° = 218° (3.8 radians). To find the average effective area over an orbit, the above equation must be integrated from -109° to +109° while noting that the top is completely shadowed between 90° and 109° on both sides of the Earth-sun line. Dividing this integral by the angle in radians gives the average panel area over the orbit of 0.136 m². With an area utilization factor of 0.85 (i.e., not all of the area is taken by solar cells), the average effective solar cell area is 0.12 m². Multiplying this area by the mean solar intensity of 1353 W/m² and the typical effeciency of solar cells of 12% gives the average solar cell output of 19.5 W while in sunlight.

A working value for mass estimates for solar panels (cells and substrate) is 6 kg/m². The area of the side of the cylinder is 0.39 m² and the area of the top not taken by the boom and tip mass hardware is 0.11 m² for a total panel area of 0.5 m². Therefore, the solar panel mass for the spacecraft is 3.0 kg. Degradation of the cells is neglible for the short lifetime expected for this low orbit. The temperature of the solar cells should be low for the best cell efficiency, so a maximum temperature of 20°C will be specified. DC/DC converters will be needed to provide power at the proper voltages for the various systems. Mass and volume estimation guidelines for the converters yield approximately 1 kg mass in a cube 3 cm on a side.

Nickel-cadmium batteries were chosen for power storage. A typical spacecraft operating voltage of 28 V was used. The goal for the system was to deliver 10 W to the load constantly throughout the orbit. During the sunlight portion of the orbit, slightly more than 10 W goes from the solar cells through DC/DC converters (which are about 80% efficient) to the load. The remaining power goes to charge the batteries. For 28 V, a total of 22 cells at 1.29 V/cell must be used. The average current delivered by the batteries during eclipse is 10W/28V = 0.36 A. The equation for determining battery capacity in A-hr is

$$C = I\,t\,/\,dd$$

where I is the average current in A, t is the eclipse time in hours (36.3 min or 0.605 hr), and dd is the depth of discharge (chosen to be 0.2 for longer battery life). The capacity required is 1.09 A-hr; the design uses a standard size 1.0 A-hr battery, so there will be slightly less power available during the longest eclipse times than desired. This shortfall should present no problem since power can be budgeted. In fact, power budgeting makes possible short peaks of 20 W or more available to the payload. If the communications system requires a transmitted power of 10 milliwatts with a 10% efficiency transmitter, the power required is only 0.1 W, so communications will have little impact upon the power system. Since there is much flexibility in the power system, no formal power budget is included here. Assuming a 20% packaging weight, the weight for the batteries was found to be 2.64 lb (1.2 kg). The volume required is approximately a cube that is 4 in (10 cm) on a side. The batteries need to be kept in the temperature range of 0°C to 30°C, preferably less than 15°C for the best efficiency.

## Thermal Control[11,12]

Midsat will rely upon passive thermal control to keep the temperature of the spacecraft within allowable limits. The heat inputs to the spacecraft in sunlight are solar radiation, solar energy reflected off the Earth, and Earth thermal radiation. The heat absorbed is given by

$$q_{abs} = A_{proj}\, \alpha_{sc}\, Q_{sun} + A_{bot}\, \alpha_{bot}\, (0.3\, Q_{sun} + 237\, \frac{W}{m^2})$$

where the first term represents the solar heat input from the surface facing the sun and the second term the input from the Earth through the bottom of the spacecraft. $A_{proj}$ is the projected area of the surface facing the sun; throughout the orbit, it is approximately the projected area of the cylindrical side of 0.124 m$^2$. The absorbtivity values ($\alpha_{sc}$ for the solar cells and $\alpha_{bot}$ for the bottom surface) were assumed to be 0.84, the value for black paint. The bottom surface would be painted black while the top and sides are covered with low-reflectivity solar cells, so the cells will be considered roughly equivalent to black paint in this thermal analysis. In fact, solar cell absorptivity is 0.75 (emissivity is 0.82)[13], rather close to the values used. $Q_{sun}$ is the mean solar radiation of 1353 W/m$^2$. $A_{bot}$ is 0.130 m$^2$. The factor of 0.3 in the equation is the average Earth albedo, the

measure of how much solar energy is reflected back into space. The term 237 W/m$^2$ is the average power per unit area of the thermal radiation from the Earth, a significant input for a satellite in a low orbit. The power input for the spacecraft in sunlight was calculated to be 211.1 W/m$^2$.

All spacecraft surfaces radiate heat. The area of the entire cylindrical side is 0.39 m$^2$, the top and bottom are each 0.13 m$^2$, for a total area of 0.65 m$^2$. The expression for radiated power is

$$q_{rad} = A_{total}\, \varepsilon\sigma T^4$$

where $\varepsilon$ is the emissivity of the spacecraft surfaces, $\sigma$ is the Stefan-Boltzmann constant (5.669 x 10$^{-8}$ W/m$^2$K$^4$), and T is the temperature of the spacecraft. An emissivity of 0.8, the value for black paint, was used. For an object in thermal balance, the heat absorbed equals the heat radiated. If the spacecraft reaches thermal balance while in sunlight, the equilibrium temperature is 290.8K (17.6°C), within the acceptable ranges for the solar cells and batteries.

When the spacecraft is in eclipse, the only heat being absorbed is that due to the thermal radiation of the Earth. With only that heat input, the equilibrium temperature will be 172.1K (-101°C), too low for many of the spacecraft systems. For the the longer eclipse times, it will be necessary to sacrifice power for heating. Those components that are sensitive to low temperatures (batteries and electronics) will need to be insulated so as to retain heat during the longer eclipses. A mass of 1.0 kg is budgeted for heating elements and insulation.

## Structure

For initial analysis, the structure was selected was a 0.125 inch (0.32 cm) thickness cylindrical shell with the top and bottom being 0.125 inch thickness circular plates. Four T-section stringers were included inside the shell for addition stiffness and to provide points to which to attach the battery cells, transmitter, payload, and so on. The material chosen was 6061-T6 aluminum, a low-cost, weldable alloy. The acceleration loading for a GAS payload is 10 g's in the longitudinal direction and 6 g's in the lateral direction[1]. A safety factor of 1.5 was included in the stress calculations. Assuming the entire weight of the spacecraft was supported in the longitudinal direction by the shell and stringers, the stress was 206

psi, well below the yield strengths of the alloy of 35,000 psi. For the lateral case, all the spacecraft weight was assumed to be supported by the four stringers. The maximum stringer stress was 23,900 psi, still below the alloy strengths. The lateral case is quite conservative since both the shell and top and bottom plates will help to support lateral loads. Dynamic and acoustic loading will probably be higher than the listed static loads, but the structure is overdesigned so that it will be able to function properly. The mass for the entire structure is 6.1 kg. This mass is likely excessive, but a more detailed structural analysis (i.e., a finite element model) will need to be completed in order to determine where structure can be trimmed.

## Command and Telemetry[14]

Low power consumption, high-capability processors are commercially available and were not analyzed in detail for this paper. One megabyte of dynamic RAM storage is included on the processor boards. By limiting the amount of data collected between passes over the ground station and by storing the data dynamically, the design avoids the weight, volume, and power requirements for a mass storage unit such as a tape recorder. Two processor/memory boards, each 20 cm square, are included with a budgeted mass of 1.5 kg for both. Temperature and voltage sensors are placed in several locations on the spacecraft, but the amount of data generated by them is small compared to the image data. A small transmitter and receiver, each a cube 10 cm on a side and of total mass of 1.0 kg, is included. Two dipole antennas are used, so 2.0 kg of mass is budgeted for them as well as deployment mechanisms.

## Mass Budget

The mass budget for the spacecraft body is listed in Table 3. Masses for each system were estimated in the previous sections. A total of 22 kg (slightly less than half the 45.4 kg GAS cannister payload) was allotted for the body, so 4.7 kg is left for the payload. The tip mass for the gravity-gradient boom is 20 kg, slightly less than that of the body, so that 3.4 kg can be used for deployment hardware.

## USE OF THE SATELLITE IN CLASSES

Once Midsat is in orbit, it can be used in several classes. In the USNA Aerospace Engineering junior-level course in astrodynamics, midshipmen

can be assigned a lab exercise to use the ground station to determine the orbit of the satellite. If a stable clock is included on the processor board, the ground station can be used to run Doppler shift satellite navigation problems. A spacecraft operations course could be established; assignments would include decoding Midsat telemetry, determining spacecraft health, sending commands to the spacecraft, recovering imaging data from the spacecraft, and processing the downlinked images. The midshipmen would gain valuable operations experience for which the only other source is actual spacecraft operations.

## Table 3

### MASS BUDGET

| Item | Mass (kg) |
|---|---|
| Structure | 6.1 |
| Battery cells | 1.2 |
| Solar cells and substrate | 3.0 |
| DC/DC converters | 1.0 |
| G-G boom, rods, mechanisms | 0.5 |
| Antennas, mechanisms | 2.0 |
| Transmitter, receiver | 1.0 |
| Processors, storage | 1.5 |
| Heating, insulation | 1.0 |
| Payload | 4.7 |
| TOTAL | 22.0 |

The spacecraft would be an asset for classroom demonstrations and laboratory work. A full-size model, structural test article, or even backup spacecraft could easily be moved into classrooms to show how a gravity-gradient boom is deployed, for example. A structural test article could be used in labs for courses such as structural dynamics or finite elements.

# MIDSAT AS A GENERIC SPACECRAFT BUS

The payload has been deliberately left unspecified because any type of low-resolution imaging equipment will be adequate. Once the design was completed, it became apparent that Midsat was, in fact, a generic, low-cost, small satellite bus. It can support a payload of up to 4.7 kg. The payload can be supplied with 5 to 7 watts continuously or with more power for shorter periods of time. The payload will be Earth-pointing to within 10° of the vertical, sufficient for low-resolution imaging. Thermal control, data storage, and communications are provided. Several of these satellites with imagers that operate at different wavelengths could fulfill one of the missions proposed by military space planners[15]. Once the detailed design and verification testing is completed, the Naval Academy could supply the specifications to other colleges and universities that wish to pursue similar small satellite projects.

## CONCLUSIONS

A small, low-powered, lightweight spacecraft bus design that would be tracked by the U.S. Naval Academy ground station has been proven feasible. The design is gravity-gradient stabilized and supports two small, low-resolution imaging devices as the payload. Two potential problems remain: insuring capture of the spacecraft in the proper orientation after boom deployment and reaching a higher altitude so that a longer orbital lifetime can be achieved. Two years or more of life are needed so that the spacecraft can be integrated properly into courses.

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