A family of generic earth pointing satellites

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Abstract

This paper outlines a family of generic earth pointing satelllites. The smallest configuration is sized to fit a Get Away Special (GAS) container (2.5 cubic feet). The basic core of the largest member is sized to be scout launch vehicle compatible (18 cubic feet). The smallest member can easily accomodate 125 pounds of payload and generate 200 watts of electrical power. The largest can carry 500 pounds of payload and generate 1 Kilowatt of power. This family of earth pointing satellites is based on modular components that allow upward compatibility. To keep cost per unit volume low, tried and proven components are used which reduce testing of the integrated satellite. The modular design and standardization of power, and command and telemetry functions allow for ease of integration and system level checkout. The familly limitations and their solutions are discussed as well as tradeoffs of various structural fabricatioon methods (metallic vs. composite vs. plastic).

THE LIGHT CASTLE CONCEPT

Definitions

The authors partition a satellite into four major components:

a. mission payload
b. mission support (power, ground-to-spacecraft interface, and command and telemetry functions)

c. attitude control system
d. a structure to mount all the above

Mission payloads provide the reason for building satellites. Each mission payload has unique requirements for power, ground-to-spacecraft interface, spacecraft orientation, and volume. The power function is made up of two parts -- source (photovoltaic arrays and/or batteries) and regulation. Ground-to-spacecraft interface consists of an RF link which requires a transmitter and receiver. Command and telemetry has two functions -- command decoder and telemetry formatter. Attitude control systems insure stable equilibrium orientation of the satellites and come in many forms ranging from passive to fully active. The sole purpose of the structure is to house the above components so that they will survive launch environments and provide shielding and thermal control as necessary.

Philosophy

The job of a spacecraft integrator is to combine the components into a satellite that will maximize mission objectives at minimum cost. Integrating components is easy, but doing it in a cost competitive environment makes the job harder. One might think that an all purpose satellite would keep cost down; however, the result is either a very complex, costly vehicle (gold plated cadillac) or an all purpose satellite that is so limited in capabilities that mission objectives are sacrificed. The authors advocate a highly integrated approach to satellite building. A basic set of standardized structural elements are used. These are configured and integrated with mission payload, mission support functions, and attitude control to meet mission objectives at minimum cost. A stable of standardized components (power, command and telemetry, and attitude control systems) are made available to support mission payload requirements. Using this approach, once a spacecraft integrator has examined the mission objectives, a satellite can be easily configured. The basic Light Castle configurations are derived from the authors' views on "ideal" satellite components, which are described below.

Structure. The ideal structure is designed to support launch loads with minimal weight and to allow integration of the mission payload into the structure with minimal impact to other integration functions. To accomplish the latter, integration of the mission payload and attitude control/mission support functions to the structure is run in parallel until acceptance testing begins. A bus similator is made available for mission payload integration prior to acceptance testing. This approach can work only if mission payload/spacecraft interfaces are held to a minimum. Acceptance testing begins only when the satellite is in flight configuration.

Attitude control system. A majority of mission payloads require an earth pointing attitude. Sometimes the attitude control system may equal or exceed the mission payload in volume and cost. To minimize the size of the attitude control system, the gravity gradient principle coupled with a passive control system is combined to make a cost effective system. The ideal shape of a gravity gradient satellite has its earth pointing axis greater (in moment of inertia) than its other two axes ($I_z >> I_{x-y}$); a cylinder satisfies this requirement. Solar cells are mounted either on the body of the cylinder or on panels perpendicular to the major axis of the cylinder. Historically, gravity gradient satellites have deployed solar panels perpendicular to the earth pointing axis. This requires a deployable boom with an attached mass along the major (earth pointing) axis to insure the proper ratio of inertia is met. The authors instead prefer a dumbbell configuration be used. The latter allows a partitioning of the mission and support functions into separate modules. The near earth portion of the dumbbell is soley for mission payloads. The other end is used for support functions. The solar panel is integrated into the deployment structure that separates the two modules.

Mission support functions. All mission payloads need a ground-to-spacecraft interface and command and telemetry system to allow the ground control over all satellite functions. Also, electrical power must be generated and regulated for use on the vehicle. These functions must be housed efficiently. To control size, a single housekeeping processor controls all the support functions as well as any active attitude control functions. The housekeeping processor also performs autonomous functions which use telemetry data input to self detect onboard anomalies. The latter function has applications both during ground testing, for quicker fault isolation, and on orbit, to enable the ground to quickly determine subsystem state of health.

The RF link is contained within the support module unless the mission payload requires an RF link to be housed in the mission module. This link is shared by the mission payload and housekeeping processor.

Power regulation is performed separately in each module. Each module is connected to its dedicated portion of the solar array. Since the satellite has no yaw control, four solar panels are mounted between the mission and support modules to guarantee one panel is sun facing at all times. This approach is taken (versus a sun tracking array) to decrease component and subsystem test time costs. Solar array output is tailored for each spacecraft by adding more photovoltaic area to the panel.

FAMILY DESCRIPTION

The GAS-Sized Configuration

The GAS-sized family member (Fig. 1a) consists of a mission module and support module connected by a deployable structure. Its mission module can

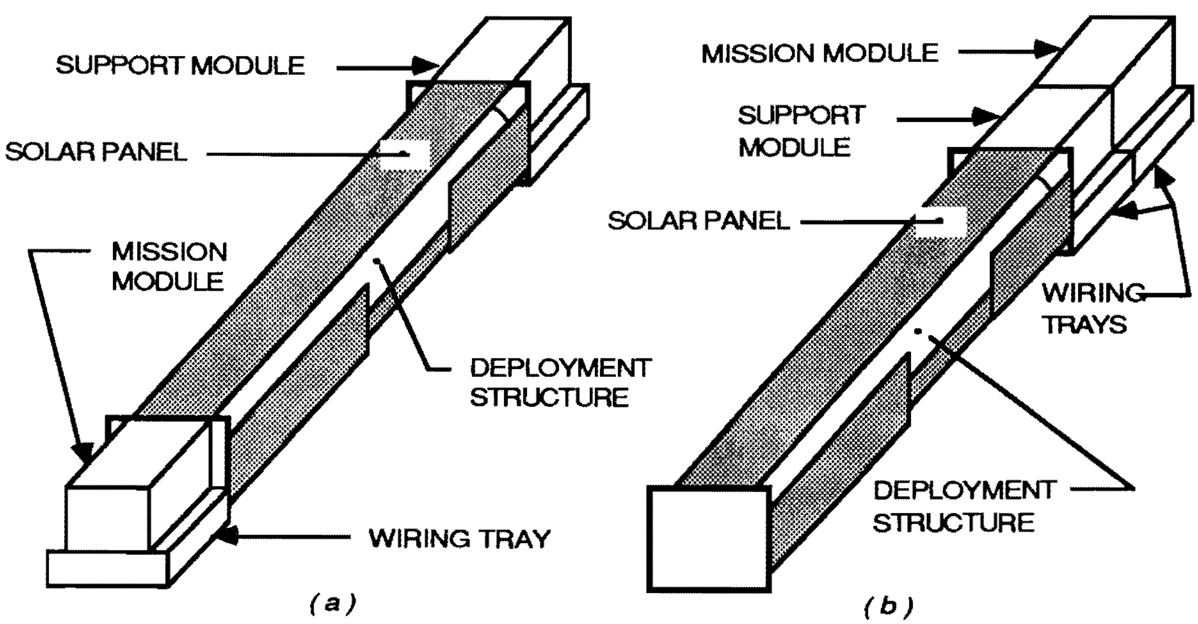

Figure 1 GAS compatible family members

accomodate up to 1.25 cubic feet (0.75 cubic feet structural element with an optional 0.5 cubic feet wiring tray) of payload. It is made up of a rectangular frame with three close out panels and a wiring tray. Mounting bosses are provided on each end of the mission module which allow ease of interfacing the mission module to other payloads or to the support module. The support module contains the "housekeeping processor" (command, telemetry, and other housekeeping functions), and the power regulation equipment. The electrical interface between the mission module and support module is limited to power and a serial interface to the housekeeping processor. The solar panels are made up of four rectangular, polymeric substrate panels that resemble a rolled window shade in their nondeployed state. Photovoltaic power is provide by either amorphous solar cell technology or conventional 2x1 cm crystalline solar cells mounted on the substrate. The entire length of the deployment structure is 20 feet; either a tubular boom or a deployable truss mechanism may be used.

An alternate configuration (Fig. 1b) of the GAS-sized family member places the support and mission payload modules next to each other. The free end of the deployment structure requires a suitable mass to insure the proper ratio of inertia is maintained. This mass could be used for an antenna or a sensor device which requires separation from the other onboard electronics.

The Scout-Sized Configuration

The basic core of the scout-sized bus is made up of an octagon shaped truss (Fig. 2) with four structural elements at 0.75 cubic feet each (dimensions identical to the GAS-sized mission module less wiring trays) and four triangular shaped bays mounted on the perimeter at 0.50 cubic feet each. Another 1.1 cubic feet is available in the center bay of the truss bringing the total volume to 6.1 cubic feet per truss.

One to multiple trusses form the mission module which houses the the mission payload. Similarly, one to multiple trusses form the support module which contains the mission support functions. Depending on mission support requirements, additional mission payload space could be available in the support module. A deployment structure 75 feet in length separates the mission and support modules. Again, either a tubular boom or a deployable truss may be used. The solar panels are configured similar to the GAS-sized solar array (four rectangular panels).

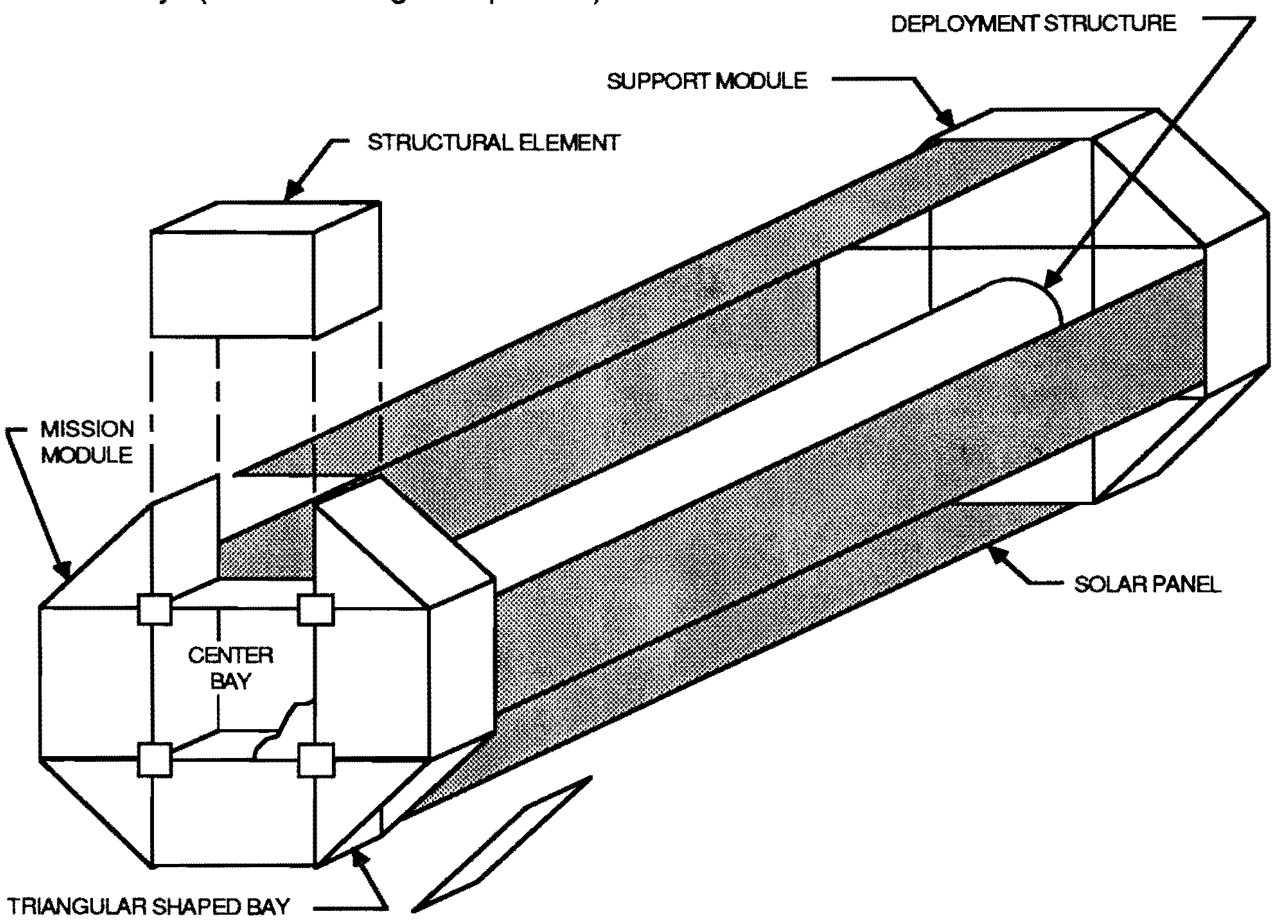

Figure 2 scout compatible generic bus

Other Configurations

Varied module configurations are possible. The 0.75 cubic feet structural elements of the scout-sized configuration or the 1.25 cubic feet modules of the GAS-sized configuration may be interchanged with three optional modules of the same dimensions (Fig. 3) These options are:

a. a black box

b. multiple black boxes

c. a card cage

The black boxes may house payload or support functions. By using cards, the density of payload per structural element is increased by eliminating box housing. (Three sides of the card cage are outward facing allowing for heat dissipation).

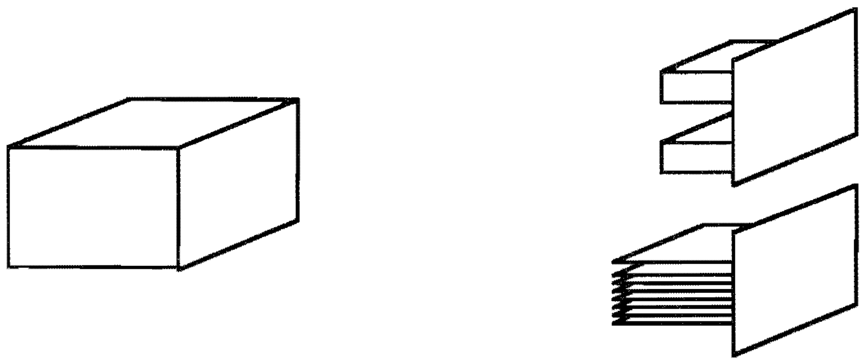

Figure 3 payload interchangeability

The interface between the mission payload and support modules is simplified by having only a single power line -- a serial interface to pass commands and telemetry data. If the mission and support payload outgrows a GAS container, it is possible to connect up to four 1.25 cubic feet modules together (Fig. 4) and still use the GAS-sized deployment structure in conjunction with a customized ejection mechanism. Scout-sized modules may be similarly stacked. Two scout-sized modules may be added to the bottom of the first with no adverse effect on the spacecraft ratio of inertia. Provisions can be made for deploying antennae/sensors away from the side of a satellite and yet still remain within the prescribed launch envelope.

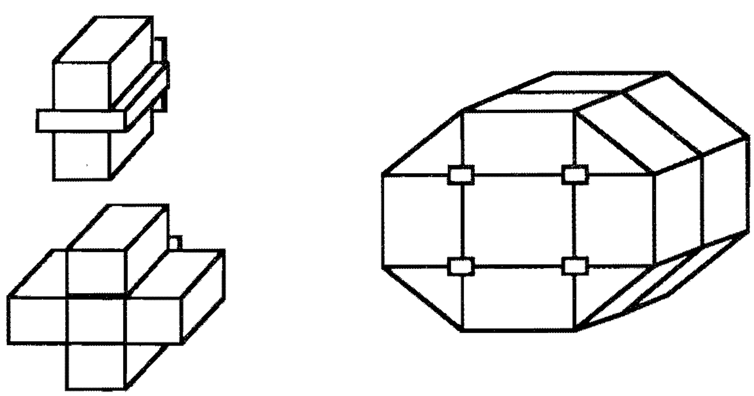

Figure 4 various module configurations

Two standard length (20 or 75 ft) deployable structures may be used. Varied amounts of solar power are provided by varying the area on the "window shade" (solar panel; see The GAS-Sized Configuration) having active cells. Using the 20 foot structure with the standard crystalline solar cell (140 watts per sq.m) generates 200 watts of power from the GAS-sized array. Using the 75 foot structure with the crystalline solar cell generates 1 KW output from the scout-sized solar array.

LIMITATIONS OF THE LIGHT CASTLE FAMILY AND SOLUTIONS

Solar Array Orientation

Each Light Castle family member has its solar panels integrated with the deployment structure that places them in the orbital plane. The ideal beta angle for the this solar array configuration is 90°. Actual beta angles vary from 0° to 70° during the year (Fig. 5). Because of the solar array orientation, its power output approaches zero as the beta angle approaches zero.

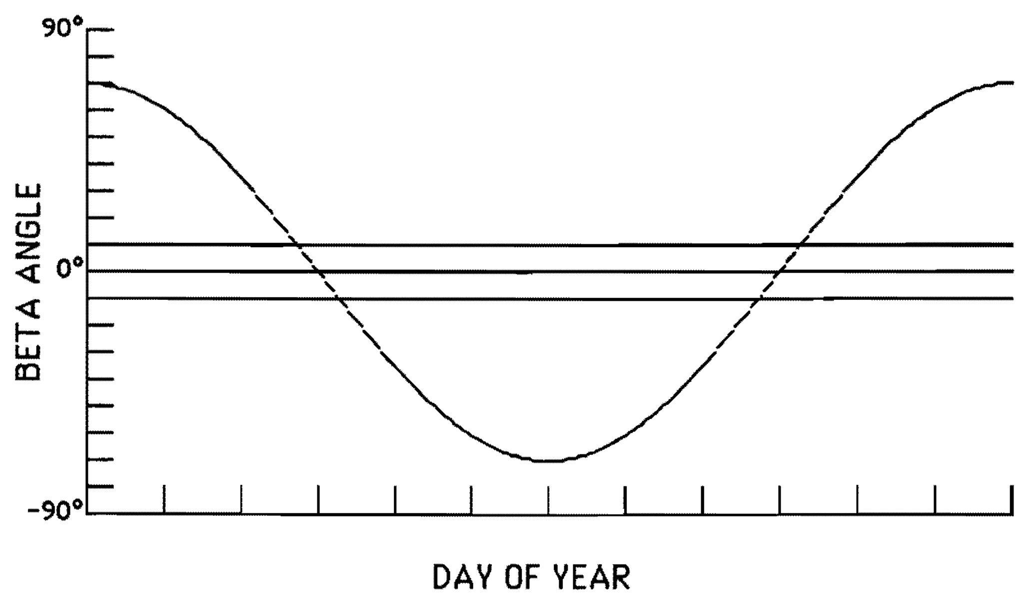

Figure 5

The effect would be as if an additional eclipse season occurs. To compensate for this, the Light Castle array is enlarged which improves the attitude control of the satellite (by increasing the gravitational torque). The Light Castle approach is more attractive then mounting the solar arrays perpendicular to the orbital plane (fixed or sun tracking arrays). The latter configuration requires rigid solar array panels and more a complex deployment mechanism. Sun tracking arrays have the smallest possible solar panel size. These arrays additionally require motor drives and control electronics. Both the fixed and sun tracking configuration costs far exceed the cost of increasing array size.

Predicted Attitude Control Performance

The design of the attitude control system is based on simplicity, with minimal moving parts. This approach is taken to control satellite cost. The dumbbell shape of the LIGHT CASTLE satellite structure is subjected to gravitational torques which have a significant effect on satellite dynamics. Each family member takes advantage of this force to maintain pitch and roll stabilization. The gravitational torque restores the satellite to an earth pointing attitude, but the satellite will oscillate unless damped. Damping is performed by a minimum of two hysteresis rods mounted and aligned on the spacecraft structure. Historically, this method has achieved a pointing accuracy of 10°. If tighter pitch and roll or a yaw axis control is required, the hysteresis rods will be replaced by electromagnetic assemblies and associated electronics. A pointing accuracy of 1° is predicted for this system. The cost of hysteresis rods is two orders of magnitude greater than the cost of electromagnetics. Each family member requires the booster to place the satellite in final orbit. The booster or satellite can despin the bus before earth capture takes place.

TRADEOFFS OF VARIOUS STRUCTURAL FABRICATION METHODS

Vehicle cost is to be controlled during all phases of construction. The unit price per structural element is the raw material cost plus labor cost. Material for the modules should be easy to work with in the lab environment to control cost (i.e., structual modification). Three materials are discussed:

a. aluminum
b. honeycomb composite
c. plastic

Historically, aluminum has been the natural material selection. It meets strength and heat dissipation requirements and is easily worked in the lab. However, initial fabrication of the aluminum structure is labor intensive; touch labor cost grows linearly with the number of vehicles produced.

Honeycomb sandwich material is the most desirable in weight and strength. However, it requires high non-recurring costs (NRE) up front (planning, patterning the honeycomb, etc.). Touch labor rates for assembly equal or surpass those for aluminum.

Plastic has the highest NRE of the three materials and has only half the thermal coefficient of heat transfer of aluminum. However, it is more easily worked in the lab environment than both aluminum and honeycomb composite. Raw material costs for plastic are also considerably cheaper. If satellite orders reach a sufficient level (10+), plastic becomes an attractive alternative with a lower unit cost per satellite than aluminum or honeycomb composite satellites.