Development of a Responsive Small Spacecraft

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The development of a small spacecraft bus is presented. The approach applied seeks to respond to the requirements of potential users as represented by a large data base of payload missions, while minimizing cost. The development results in a general purpose spacecraft design that is responsive to a wide range of mission requirements, and provides a clear definition of capabilities and interfaces to facilitate user mission design.

Recently there has been a growing interest in small, inexpensive spacecraft to accomplish a range of missions, including those that are experimental, developmental, and operational in nature. This paper defines an approach recently taken to develop the responsive, capable, low-cost, small spacecraft bus seen in Figure 1. The emphasis of this approach was to provide the widest accommodations for potential small satellite users, while minimizing the development and recurring cost.

The development has three distinct phases: requirements definition, concept development,and design. Special attention is paid to the methods used to maximize mission accommodation while minimizing cost. The development activities have been thoroughly documented to insure that the results were portable and to allow flexibility in future design decisions. Each phase concluded with a formal review, in order to evaluate progress and to elicit objective criticism. The result of the development process is a general purpose spacecraft bus, capable of providing a large variety of users with desired housekeeping resources via standardized interfaces. The sizing and design of the bus is based upon identified cost targets, primary launch vehicle, and ground networks, however the bus is adaptable to other uses based upon different ground rule assumptions.

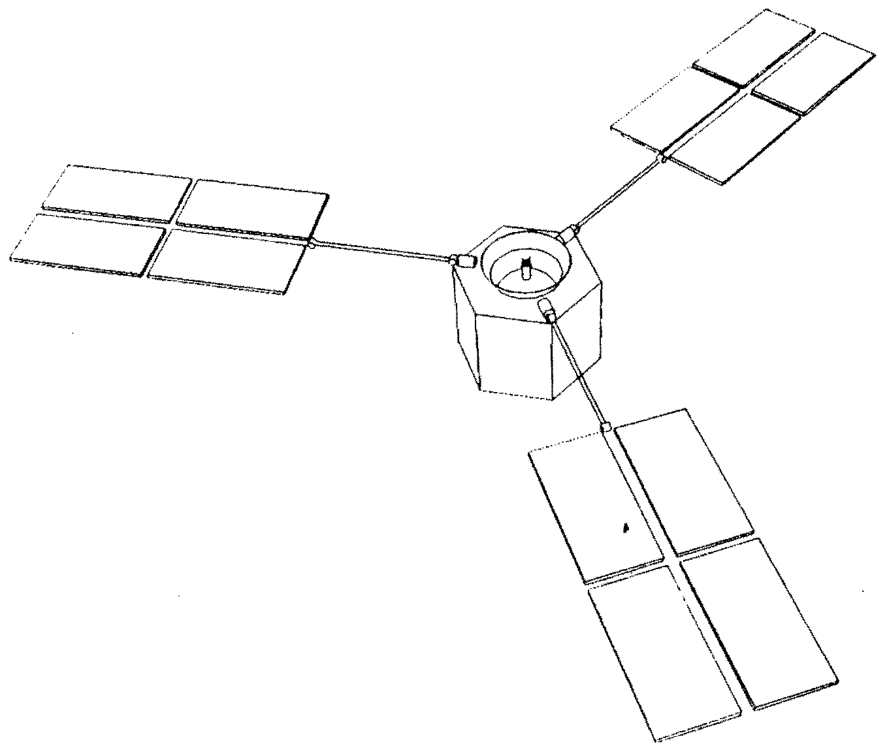


Fig. 1 GE Smallsat

Requirements Definition

The key to the successful development of any system is a clear, accurate, and early definition of requirements. Without such a definition, the development process becomes muddled and the resulting system may be over-designed or lack essential capabilities. Three methods may be applied to definition of spacecraft requirements:

1) define requirements as those performance levels and capabilities derived from engineering judgement,

2) define requirements as those associated with a finite number (1,2, or 3) of well defined missions,or

3) define requirement "envelopes" derived from a large data base of potential users.

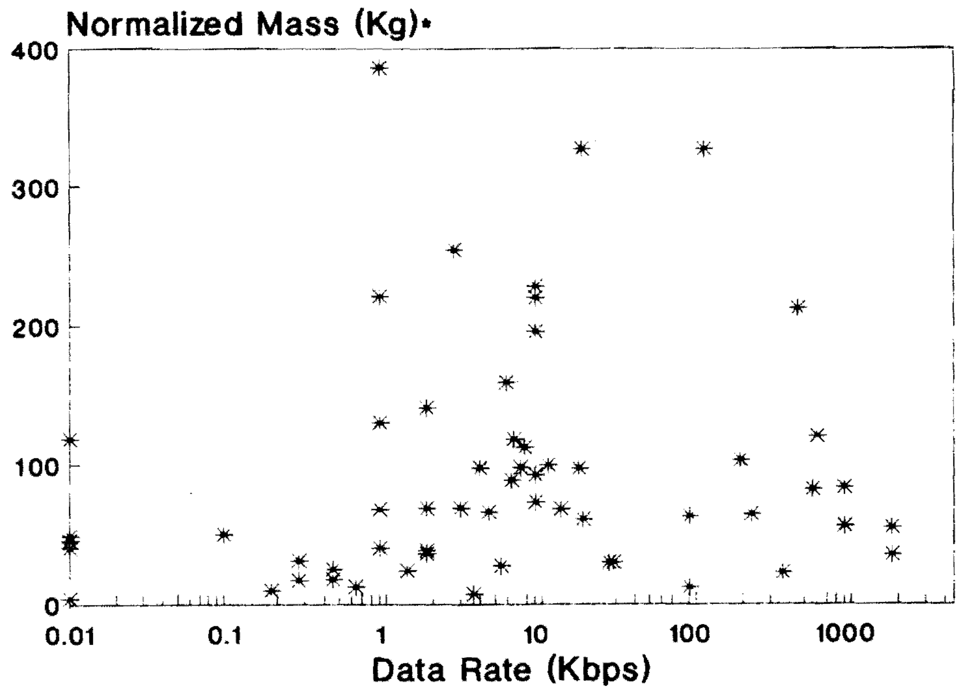
All three of these methods have been used in the past with varying degrees of success, and all are obliged to respond to limits such as lift mass, launch vehicle volume, program budgets etc. The first method, basing requirements upon engineering judgement, is subject to the quality of that judgement, as well as the ability of the user to utilize the resulting capability. The second method of designing to a small number of specific missions, which characterizes large spacecraft programs, relies on the fidelity of the mission definitions. The final method responds to the users' corporate requirements, without allowing individual missions to drive the design. Most requirements definition activities can be characterized by one of these three methods.

The potential missions for small spacecraft share three overriding characteristics: diversity, immaturity, and cost sensitivity. The use of requirement "envelopes" derived from a large data base of potential missions was selected, because it allows the design to be responsive to the wide range of possible applications while guarding against the impact of a few immature missions with demanding requirements.

A data base of potential small satellite payloads and their requirements was generated in order to develop the resulting "envelope" requirements. The potential payloads used came from many sources including: the NASA Small Payloads' Workshop[1], the Air Force Space Test Program[2], the Space Station Payload Engineering Requirements Document[3], a number of universities, and internal sources. The resulting database, that is continuing to grow and be updated, has over 60 missions specified involving over 100 instruments and payload elements. These represent experimental, developmental and operational missions, flying in orbits of various inclinations, altitudes, and eccentricities. They additionally represent both earth and inertially oriented payloads, requiring both spin and 3-axis stabilization.

Once the data base had been assembled, it was analyzed in order to characterize the mission requirements. After accounting for the payloads' orbit, attitude, and stabilization requirements, the primary resource requirements, such as mass, power, data rates and storage, and pointing, were evaluated. As shown in Figures 2a and 2b, no strong correlation was found between payload data rates and payload mass or power requirements. However, from Figure 3, a correlation between payload power and mass requirements can observed. Figure 3 also shows payload power and mass envelopes, that are consistent with the lift capabilities of the 3 major launch options considered, the SCOUT[4], the Pegasus[5], and the SSLV[6]. [These three vehicles were considered because they span the capability of possible small launchers and are perceived to have government agency backing, and are thus more likely to be available.] The percentage of the on-orbit mass that was allotted for payload was selected as approximately 40% for each case, since this represents a historically aggressive percentage for low earth orbit missions. The resulting requirement envelopes and their respective data base accommodation are presented in Table 1 for each considered launch option.

Data Rate Vs Normalized Mass



* Normalized to 500 Km Polar Orbit

Fig. 2a Payload Data Rate vs Mass

Average Power Vs Data Rate

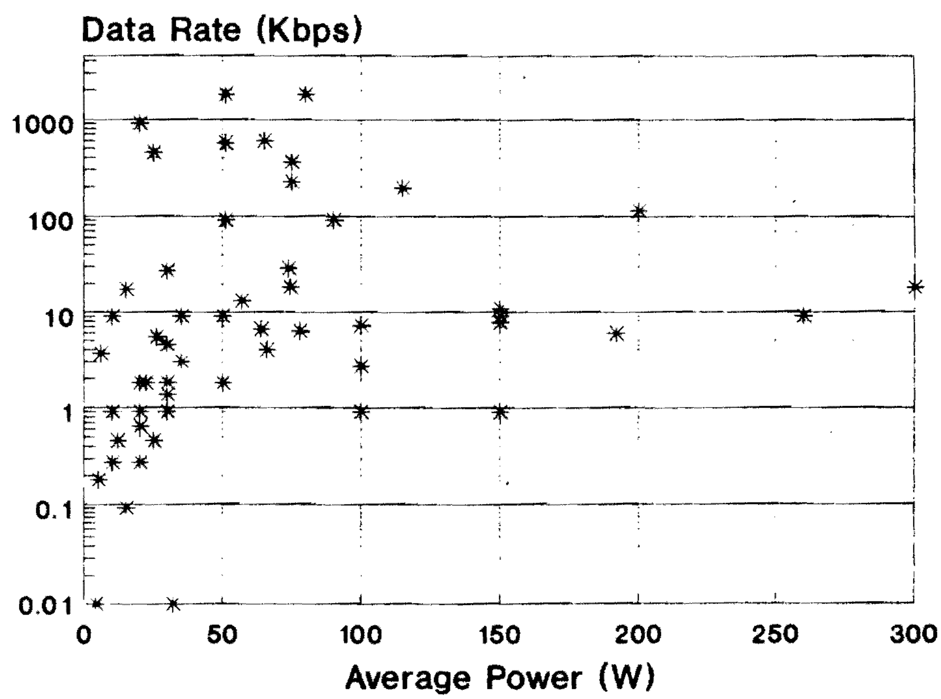


Fig. 2b Payload data Rate vs Power

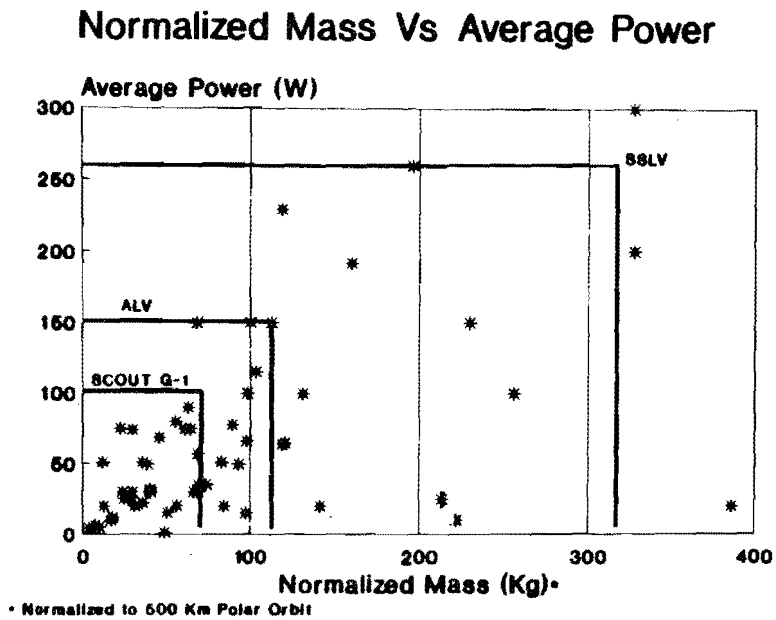


Fig.3 Payload Power vs Mass

PARAMETER	SCOUT G-1	ALV	SSLV
PL VOLUME	0.4 cu m	1.2	2.5
PL MASS	68 kg	111	317
PL POWER	100 W	150	260
PL DATA RATE	2 Mbps	2	2
DATA STORAGE	1.0E+9 bits	1.0E+9	1.0E+9
POINTING ACCURACY*	0.2 deg	0.2	0.2
PERCENT MET	50%	74%	95%

NOTE: PL Mass is for 500 km circular, i=90 deg orbit

* Control of Spin Axis of Primary 2 axes; Improved Accuracy w/PL Error Signal

Table 1 Requirement Envelopes

Two major spacecraft design drivers are launch vehicle and orientation/stabilization. The Pegasus vehicle was selected to serve as the primary launch option, due its accommodation (75% of the data base), cost ($6M per launch) and availability (first launch in 1989). This does not imply that the spacecraft will be incompatible with alternate launch options. The spacecraft bus has the additional requirement to be able to support earth and inertially orientations, as well as spin and 3-axis stabilization. Figure 4 shows that the data base has a significant population in each of these categories, and thus the accommodation associated with the bus design would be reduced significantly if any of these options were not included. [Gravity-gradient stabilization was excluded as it would impose serious design constraints. Additionally, the data base had only 3 missions that could accept Gravity-gradient stabilization.] In order to insure that a system designed to "envelope" requirements can accommodate actual missions, a number of "test case" missions were identified for use in later accommodations analyses.

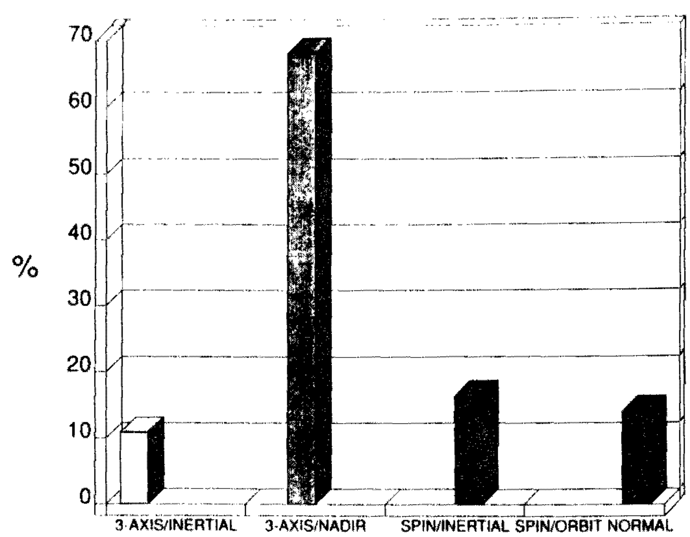


Fig. 4 Payload Stabilization/Orientation

For the development of this spacecraft bus, it was considered insufficient just to strive to reduce cost. Instead a target recurring bus cost was developed, based upon a top-level market assessment. This cost target was then allocated between the major cost elements based upon the distribution resulting from a bottom-up cost estimate for a similarly sized program. This break down of cost to the major elements facilitates a design-to-cost approach to this development. In addition a non-recurring cost target was set to bound the total cost.

Concept Development

The concept development development activities center on converting the system requirements into efficient, responsive design concepts. Activities during this phase included technology assessment, system/subsystem conceptual design, definition of program support, and reassessment of cost targets.

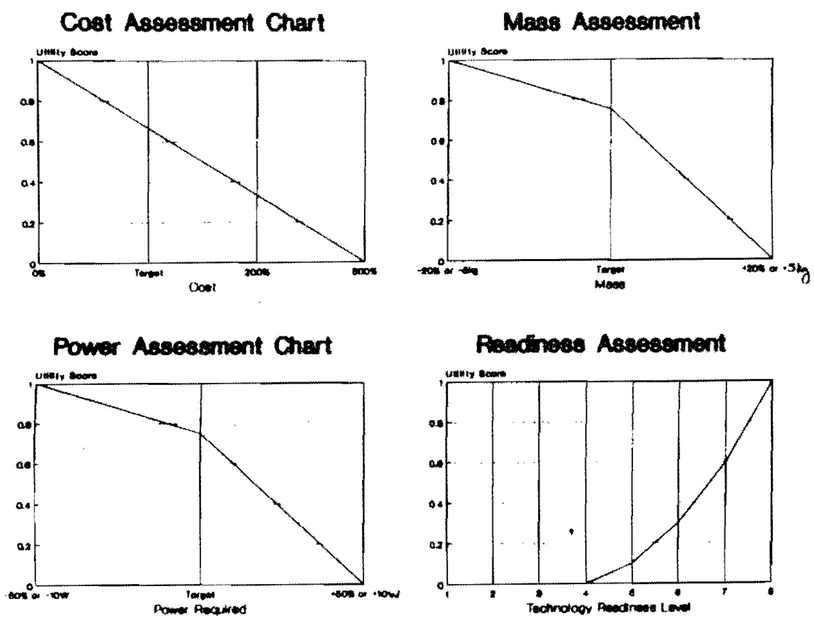


Fig. 5 Technology Assessment Charts

Prior to developing system and subsystem architectures, available technologies and hardware were evaluated for the major system and subsystem functions. This kept the architecture development process from precluding the use of a technology that offered significant cost, mass, or power benefits. Quantitative decision theory methods[7,8] were employed to select between technology and hardware candidates. Each subsystem was given a set of target resource allocations that included cost, mass, and power required. Each candidate technology or piece of hardware was required to support the system performance requirements. Figure 5 shows a set of four assessment charts that each subsystem used for technology/hardware evaluation, each scored with respect to the allocated targets. For example, Table 2 shows the evaluation results for the selection of solar array photovoltaic cells. This approach provided a consistent evaluation process for each subsystem and thus the spacecraft design reflects one set of assumptions.

Solar Cell	Mass (kg)	Readiness	Relative Cost	Score
Silicon	34.8	8	1.0	0.283
GaAs/Ge	31.4	7	3.5	0.048
GaAs/GaAs	23.4	4	2.5	0.141

Table 2 Solar Cell Candidate Assessment

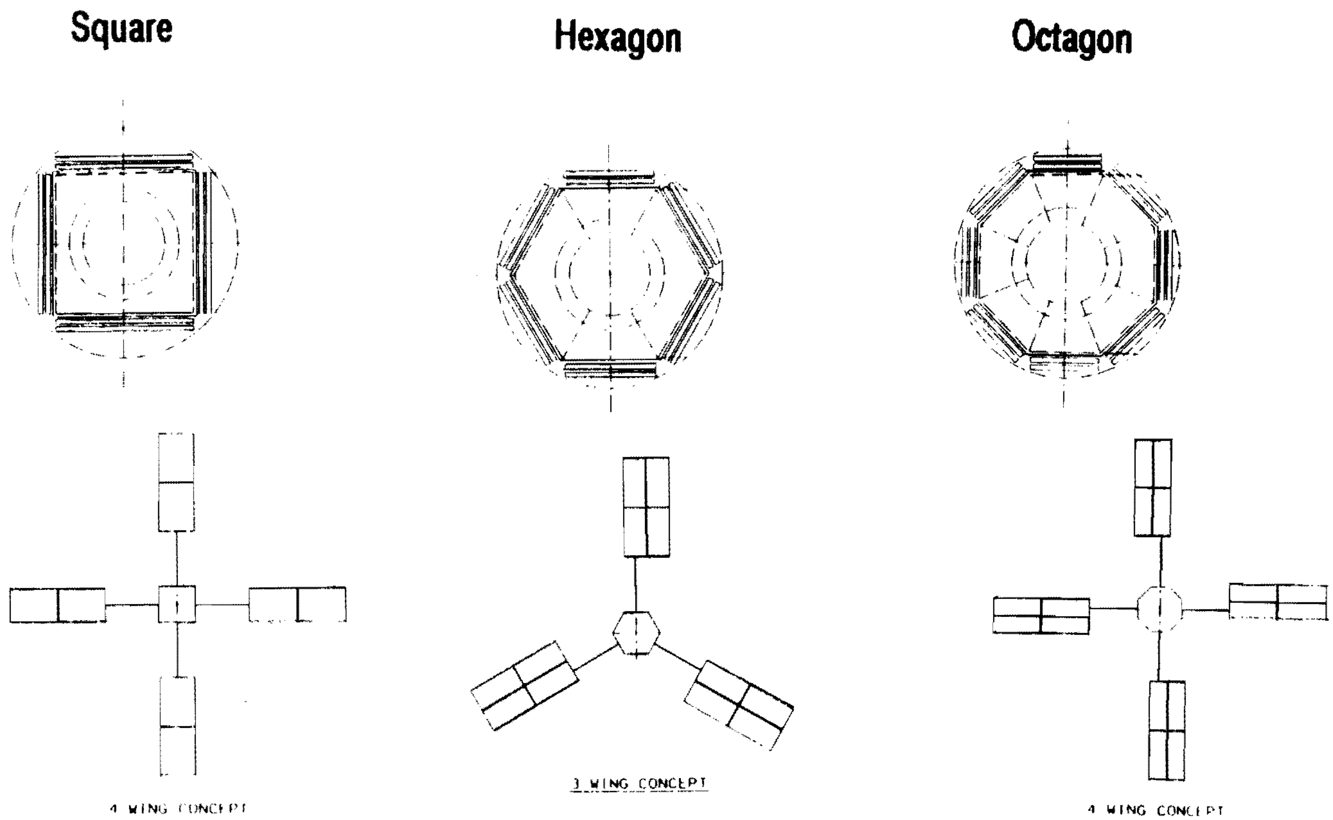
The development of the system and subsystem architectures started with the selected technologies and hardware, and the envelope requirements and mission parameters. The balance between performance and cost was still the key to this activity. A good example of this is the structural development, summarized in Figure 6, where the configuration cross-section polygon (independent of the material) is driven to a larger number of sides to maximize its internal volume, but is driven to a small number of sides to minimize the cost. Notice that the selection of the structural cross-section affects the size, number, stowage, and deployment of solar arrays. For this reason they are a major design consideration in this tradeoff. The result is a hexagonal cross-section in order to allow sufficient volume for bus hardware while minimizing the cost.

Of importance is the ability of the candidate system and subsystem configurations to satisfy the range of mission parameters identified during the requirements definition phase, especially those associated with stabilization, and orientation. This means, for example, that the physical configuration of the spacecraft must support both spin and 3-axis stabilization, the thermal control subsystem must tolerate a wide swing of sun angles, and that the communications subsystem must be able to reconfigure to support both spinning and non-spinning operations.

At the beginning of this project, it was observed that the cost of a spacecraft system was highly dependent upon the program support costs not just on the design, analysis, and hardware costs. Program support includes: integration & test, program management, product assurance, production control, shipping & insurance, and launch site support. These costs are dominated by labor costs, so the most straightforward approach to reducing them is to reduce the time and number of people required to accomplish these tasks. The resulting shorter program schedule is attractive to potential users.

In order to realize these labor reductions, a number of approaches had to be incorporated into the program and into the spacecraft design. After sizing the support staff consistent with the size of the program, the most important impact on the support cost is the streamlining of the bus integration and the payload integration. The bus integration is streamlined primarily through the use integrated avionics to minimize interfaces and procedures. The approach to payload integration is more radical. A payload mounting plate is delivered to the user, who mechanically and electrically integrates the payload and any support electronics to the plate. The payload is functionally tested prior to its delivery to the spacecraft. This minimizes the constraints upon the user, allowing them to tailor their payload layout and interfaces without impacting the bus schedule or cost. The mechanical and electrical interfaces from the bus to the payload are few, standardized, and well defined. This greatly simplifies the integration of the payload to the spacecraft, reducing the associated cost. Finally the use of common engineering support between multiple spacecraft is proposed to increase manpower utilization.

The results of the concept development phase were a consistent set of system /subsystem architectures, sizing, technologies, spacecraft configurations, and preliminary program plans. The original cost targets (and their allocation to various cost items) were also revisited during this phase, in order to reallocate this precious resource and in so doing equally distribute fiscal accountability.



	INTERNAL CROSS-SECTION	REQUIRED # OF SOLAR ARRAY PANELS	RELATIVE MECHANISM COSTS	RELATIVE STRUCTURE COSTS	SPACECRAFT BUS HEIGHT
SQUARE	.505 SQ. M.	8	1	1	.965
HEXAGON	.641 SQ. M.	12	1.35	1.33	.813
OCTAGON	.680 SQ. M.	16	1.43	1.64	.813

Fig. 6 Configuration Cross-Section Tradeoff

Preliminary Design

The final phase of this small spacecraft development project is a traditional preliminary design. This serves as the preparation for final design and build activities to be conducted under contract. This is the phase where the system/subsystem architectures, selected technologies, cost controls, and program plans were reduced to practice.

The system and subsystem designs were evaluated against the identified "test case" missions, in addition to the established "envelope" requirements of the first phase of this process. The use of the "test case" missions served to uncover problems with detailed mission accommodations. The final design accomodated the mass, power and data rate requirements since these test cases were part of the data base from which those "envelopes" were formulated. The test cases did mesure the ability of the design to provide the proper mechanical interface, field of view and launch vehicle compatibility. Mechanical compatibility of the spacecraft with the Pegasus vehicle is demonstrated by Figure 7.

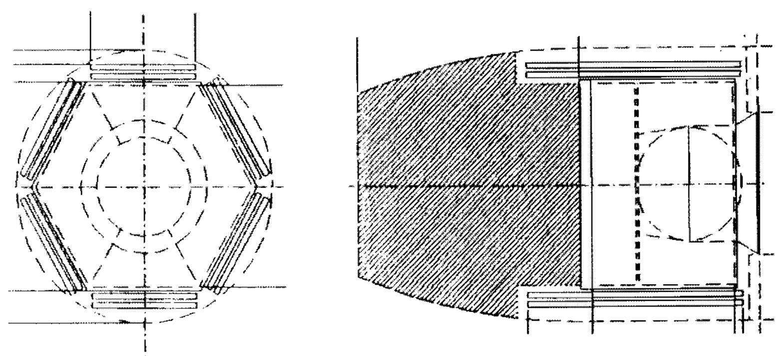


Fig. 7 Smallsat Stowed in Pegasus Fairing

The accommodation of a spinning, inertially pointed mission is shown in Figure 8. This payload is the Soft X-Ray Spectrometer[1], a proposed NASA Small Explorer mission.

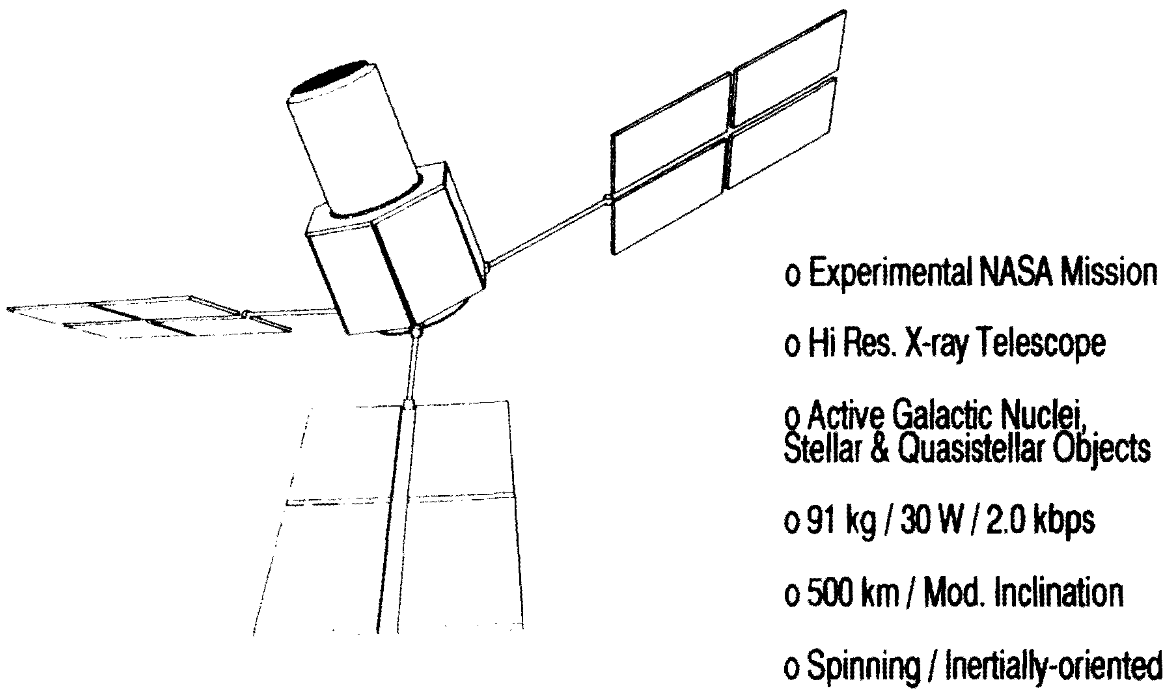
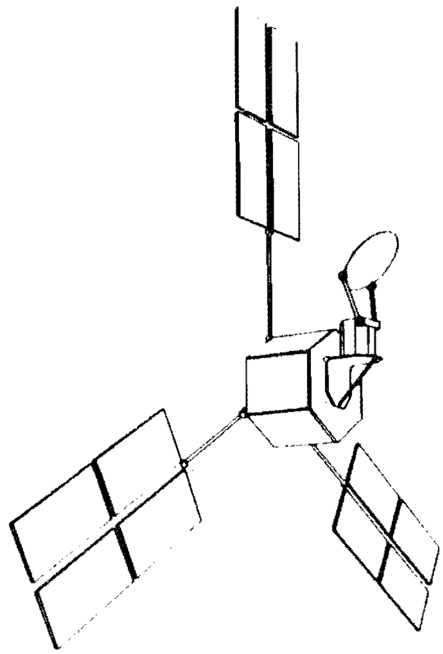


Fig. 8 Soft X-Ray Spectrometer on Smallsat

An earth pointed 3-axis mission, the Special Sensor Microwave Imager (SSM/I) currently flown on Defense Meteorological Satellite Program (DMSP), is shown in Figure 9.



o Operational Mission
o DMSP, NOAA
o 7-channel Radiometer
o 34 kg / 40 W / 3.6 kbps
o 833 km , Sun-synchronous
o 3-axis, Nadir Pointing

Fig. 9 SSMI on Smallsat

The success of the preliminary design phase can be measured by the thoroughness of its system definition and documentation. This allows the design to survive the completion of the development process and be applied readily to potential user missions. In addition to the design and performance documentation, there were four critical documents generated. The first is a Spacecraft Performance Specification that concisely and formally documents the preliminary bus design and its performance characteristics. The second document is a Payload Interface Specification that details the mechanical and electrical interfaces between the bus and the payload. This document includes the specification of certain payload interfaces that are required to facilitate integration and test of the spacecraft. The final critical document generated during the design phase is the User Guide. The User Guide links the payload interfaces with descriptions of the subsystem designs in a way that allows the user to take best advantage of the available bus resources. Finally, a program plan was written which outlines scheduled design, integration and test activities with manpower estimates. A detailed bottoms-up cost estimate was produced in conjunction with all of this design and performance documentation. This estimate substantiated the cost targets established at the beginning of the project.

CONCLUSIONS

The development of a responsive small spacecraft bus design is presented. The approach applied accounts for the diverse, immature, and cost sensitive nature of the potential missions. The use of both "envelope" requirements and "test case" missions insures that the design is responsive to the users' corporate requirements, as well as being capable of accommodating specific missions. The use of quantitative methods in evaluating the development process, thorough documentation of all activities, and formal reviews served to strengthen the resulting design.

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