MODULAR SMALL SATELLITE DESIGN FOR RESPONSIVE TACTICAL APPLICATIONS

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Abstract

The requirement for small tactical satellites capable of satisfying a variety of missions on short notice has been studied as part of the AFSSD-sponsored RESERVES study. An innovative spacecraft design approach is described which makes use of pre-assembled subsystem modules. The design is successfully tested against the requirements of four example missions using three different launch vehicles, demonstating ample design margins in each case. The approach holds promise of providing a simple, flexible, cost-effective solution to the rapid-response requirements of the tactical military world.

1. INTRODUCTION

As part of the Air Force's long-expressed desire for flexible rapid-response space capability for theatre commanders, the Space Systems Division's Deputy Chief of Staff for Development Planning initiated the RESERVES study in 1989. The intention of the study was to define multiple-use payloads and to derive satellite design and operational characteristics which would support the concept of rapid response and multi-mission capability, and to demonstrate the usefulness of such designs in various mission scenarios and to various tactical users.

As a result of an extensive background as a builder of multi-mission spacecraft (e.g.; NASA's Multimission Modular Spacecraft, MMS), and a more recent concentration on various small satellite endeavors, Fairchild Space was selected by ARDAK Corporation to study the issue of satellite design as part of their RESERVES study.

To form the requirements basis for the Fairchild efforts, ARDAK, together with Fairchild and its subcontractor SKW, Inc., derived a group of four example missions and associated payloads, as shown in Table 1. Two of these are projected to be optical surveillance payloads, and two are communications payloads. The optical missions were defined as a Low Altitude Optical Surveillance Satellite (LAOSS), to fly in a relatively low orbit of 250-to-300 km., and a High Altitude Infrared (HAIR) satellite to fly in a 3-hour elliptic orbit with repetitive ground-coverage characteristics. The example communications payloads were defined as an EHF communications augmentation package to fly in a Molniya orbit, and a data relay payload to fly in the 3-hour elliptical orbit described for the HAIR satellite.

This group of missions is well-selected to exersize the full range of probable requirements for responsive satellites. Although RESERVES is not strictly limited to *small* satellites, the emphasis on affordability within the concept of multiple mission readiness puts a premium on low unit costs. For these reasons, small satellites and sub-geosynchronous orbits, which can ameliorate launch costs, were emphasized.

2. SUMMARY SPACECRAFT SPECIFICATION AND THE MULTI-MISSION CHALLENGE

Based on the requirements of the four missions which constitute the example RESERVES mission list, we have established the summary spacecraft specification shown in Table 2.

The challenge for any multi-mission spacecraft is to specify the requirements in such a way as to avoid "overkill", and increased system cost, where the requirements are modest, while still retaining the capability to satisfy the most stringent requirements anticipated. Complicating the cost equation is the basic RESERVES concept of rapid response capability, which anticipates that all

TABLE 1. PAYLOAD REQUIREMENTS

	LAOSS LOW ALTITUDE OPTICAL SURV. SAT	HAIR HIGH ALTITUDE INFRARED SURV. SAT.	EHF COMM.	DATA RELAY COMM.
MASS	70 KG.	70 KG.	120 KG.	90 KG
POWER –orbit average –peak –standby	 200 W 335 W ~120 W	 300 W 435 W ~120 W	 230 W (duty cycle: 1-hr on, 2-hrs. off)	 185 W 300 W
REAL TIME DATA RATE DATA STORAGE REQ.	8 MBPS (1 sec/frame) 8 MB / frame	8 MBPS (1 sec /frame) 8 MB / frame	2.4 kbps/channel ?	12 kbps 7.2 Mb
DIMENSIONS	30" x 30" x 36" L	30" x 30" x 72" L	?	?
POINTING (3) –control –knowledge –stability	 ±0.5°/axis ≤0.1°/axis ±0.001°/s for 1 sec	 ±0.5°/axis ≤0.1°/axis ±0.001°/s for 1 sec	 ±0.5°/axis ≤0.1°/axis N/A	 ±0.5°/axis ≤0.1°/axis N/A
SENSOR FOV ANGLE(S) –MAX. IFOV –SCAN ANGLE(S)	4°–1° 68.2 uRAD – 17 uRAD n/a	4°–1° 68.2 uRAD – 17 uRAD n/a	1.7° SHF ANT. N/A ± 7° 2-axisgimbal	5.1° N/A ±25° 2-axis gimbal
ORBIT –apogee –perigee –inclin.	 285 km 241 km 63.4 deg	 7865 km 500 km 63.4 deg	 40,000 km 500 km 63.4 deg	 7,865 km 500 km 63.4°
COMM. LINK data rate – contact frequency – xmit antenna, power	9600 baud or real time (TBD) (TBD)	9600 baud or real time (TBD) (TBD)	2400 baud x 36 ch. 44/20 GHz up/dwn link 2-ft dish, 10-W TWTA	? ? ?
BOOSTER OPTIONS	PEGASUS	SSLV (TAURUS) TITAN II	SSLV(?) ATLAS II DELTA II	SSLV TITAN II

TABLE 2. SPACECRAFT SUMMARY SPECIFICATION

STRUCTURE	Accommodate to range of boosters: Pegasus to MLV; Modular construction to accommodate a variety of subsystem and payload modules
POWER	Accommodate range of power requirements from 100-to-350 watts orbit-average power with: – fixed power control/distribution units – modular battery accommodation: 20 and 30 A-Hr. NiCd batteries – modular solar array design for specified power range
DATA	Accommodate payload data up to 168 Mbps; realtime or store-and-playback modes up to10 Mbps
DATA STORAGE REQ.	Modular memory: 1-to-4 cards @ 1.2 Gb / card
PROPULSION	Integral monoprop propulsion module containing pre-plumbed tank(s) and thrusters; 6-to-20 kg N2H4
AC&DS	Baseline bias-momentum system able to accommodate wheel sizes of (TBD-to TBD) N-m-s; normal mode: Nadir pointing, but able to accommodate off-Nadir angles up to TBD deg control: 0.1-to-0.5 deg knowledge: 0.01-to-0.1 deg stability: ±0.001 deg/s over 1 sec
COMM. LINK	SGLS link: up to 1 Mbps downlink; up to 100 kbps uplink; X Band downlink: 10 Mbps and 9600 baud (MIST compatible); uplink TBD
BOOSTER OPTIONS	Pegasus, Pegasus II, Taurus, Titan II, Atlas II, Delta II
ORBIT OPTIONS	250-300 km @63.4° 500 x 40000 km @63.4° 500 x 7000 km @ 63.4° 830 km circ @ 99° 300 x 27,500 @ 63.4°
MISSION LIFE	6 Months to 1 year
READINESS LEVEL	72 hours to launch

spacecraft and payload modules which may be required in any eventuality will be stored in inventory for immediate call-up. This will result in pressure to limit the number of different modules in the inventory, and to provide a more uniform capability across the mission spectrum, i.e., to accept some "overkill" in order to limit the inventory. This is the inherent tension in a RESERVES design. Our sensitivity to this tension led to our determination to achieve a fully modular design, to have modules inherently flexible and interchangeable, and to limit the number of different modules which must be held in storage in order to accommodate the full range of requirements. In addition, the architecture must allow for ready substitution by component as well as by subsystem.

3. SYSTEM CONSIDERATIONS

3.1 ENVIRONMENTAL ISSUES

Two environmental issues were of paramount importance to our initial design study:

- radiation environment at the two elliptical orbits of interest
- eclipse environment for the two elliptical orbits, assuming non-specific longitudinal nodes

Both of these issues are of primary interest to the power subsystem design, but will also influence thermal design and the required hardness level of sensors and electronic equipment.

Radiation Environment

The 500 x 8000 km. orbit is of particular concern. For this orbit, radiation levels are quite sensitive to argument of perigee, as seen in Figure 1. The figure shows a maximum for equatorial arguments, and a decrease in radiation level for higher lattitudes, with a max-to-min variation of a factor of ~7. For this orbit, internal doses can be in the range of several hundred krad per year near solar maximum, which will imply the necessity for shielding the more sensitive

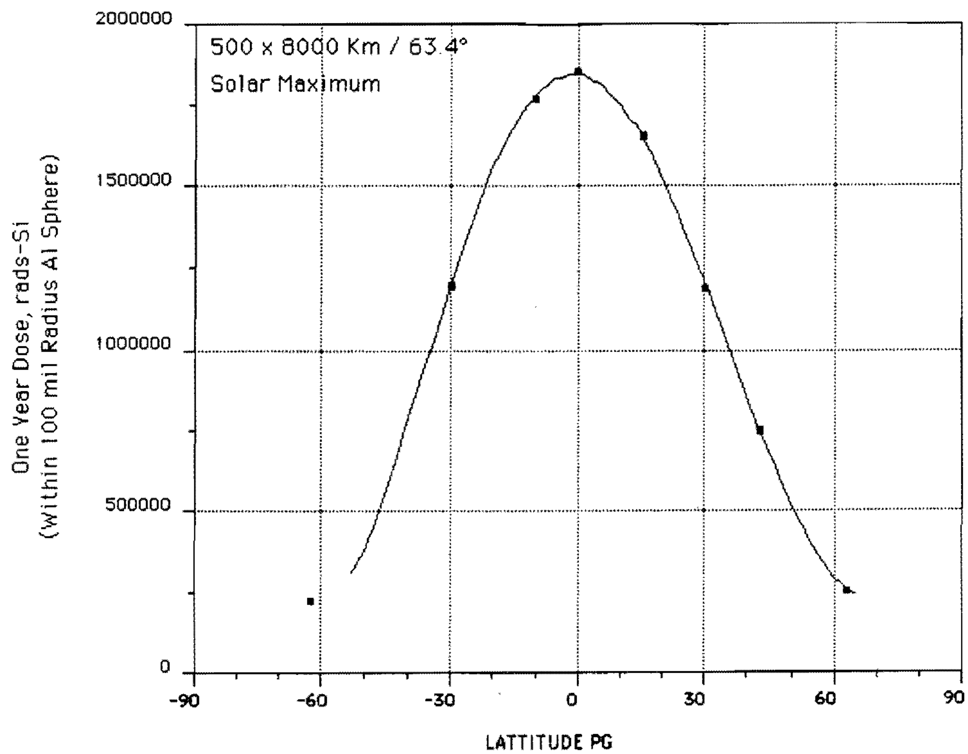


FIGURE 1. EFFECT OF ARGUMENT OF PERIGEE ON RADIATION ENVIRONMENT OF ELLIPTICAL ORBIT

electronic components. Moreover, surface doses are such as to require heavier cover glass for solar cells (e.g. 20 mils) and to causean increased degradation rate for such cells. Thus, this orbit provides an important test for the RESERVES concept in that it offers unique requirements which could proliferate the required inventory of components and boost system acquisition costs. In response to overall systems considerations, we elected to inventory two types of solar cells on a standard substrate, a standard solar panel size and deployment/mechanisms arrangement. On the other hand, radiation-sensitive components are either replaced by suitably hardened components or are shielded at the card level for all of the elliptical-orbit missions.

Eclipse Environment

The two elliptical orbits of interest provide eclipse environments which vary substantially by orbit beta angle. Although the maximum eclipses only occur under very specific orbit conditions, the RESERVES ground rules require that we design to these "worst case" conditions. For the Molniya orbit, the maximum eclipse period is approximately one hour over the 12-hour orbit. For the 500 x 8000 km orbit, the "worst case" is a 41-minute eclipse over the 3-hour orbit.

3.2 DESIGNING FOR READINESS AND RESPONSIVENESS

There are two types of readiness issues which need to be addressed in a RESERVES environment:

1. Rapid final assembly and launch preparation
2. Rapid production capability for reconstituting assets on the ground after a launch surge.

In this phase of the study, we focused our attention on the first of these two important issues. Some key features of the design which were implemented to enhance readiness and rapid buildup for launch are:

- Subsystems and components are modularized together with their supporting structural members

- Modules are completely interchangeable and plug compatible

- Connectors, interfaces, assembly hardware and fittings are standard items

- The spacecraft is designed with built-in-test capability to enable fault isolation during test.

- Provides a standard umbilical interface connector for launch vehicle integration, and a single connector for payload integration.

- Incorporates test points at sensor inputs and outputs, and voltage test points at input and outputs of power converter.

- Mechanical positioning of the payload sensors is carefully designed to allow precision mounting without final alignment.

- Spacecraft attitude reference sensors are mounted to the payload interface structure in order to avoid precision alignment requirements at final assembly.

- The propulsion subsystem is designed as a modular self-contained fully-welded manifolded subsystem, mounted to its supporting structural member.

3.3 SYSTEM DESIGN DRIVERS

A trueism of spacecraft design is unavoidably true in the RESERVES environment: Satellite mass related to booster capability and orbit requirement is the single most important design driver. The added requirement for responsive launch capability and all-azimuth, multi-latitude capability , leads to a concentration of attention on the Pegasus air-launched booster and to the Small Standard Launch Vehicle (SSLV) capability as embodied in the Taurus vehicle at this time. MLV class boosters were catalogued and classified for such missions as exceed the SSLV capability, but no such missions were addressed in this phase of the study.
One level below mass on the system design-driver scale are satellite average power and booster fairing volume limitations in this class of booster. Fairchild experience with small satellite designs for the Pegasus and Taurus-class boosters allowed us to scope the mass and average power range likely to be available to payloads for the various orbits of potential interest to RESERVES, as shown in Figure 2. The "orbit-raising propulsion" is that which may be available from an integral Hydrazine propulsion system if such is already required for attitude control or orbit trim. In such cases, the addition of fuel for orbit-raising can result in significant mass-allowance increases on orbit, as shown in Figure 2.

ORBIT \ BOOSTER	PEGASUS: WITHOUT ORBIT-RAISING	PEGASUS: WITH PROPULSION	TAURUS: WITHOUT ORBIT-RAISING	TAURUS: WITH PROPULSION
SUN-SYNCHRONOUS @ 450 N.MI. POLAR (DMSP ORBIT) INCLINATION=98.74°	MASS=180 kg. payload:10-25 kg; AVE. POWER= 20-50 watts	MASS=310 kg payload: 50-100 kg; AVE. POWER= 200-400 watts	MASS=950 kg payload: 200-450 kg; AVE. POWER= 500-1100 watts	MASS= AVE. POWER=
ELLIPTICAL @ 63° INCL. PERIGEE=244 km (130 N.MI.) APOGEE=285 km (selected to maximize ground-track repeats)	MASS=270 kg payload: 20-80 kg; AVE. POWER= 100-250 watts	MASS= AVE. POWER=	MASS=1070 kg payload: 300-550 kg; AVE. POWER= 350-550 watts	MASS= AVE. POWER=
MOLNIYA; 8-HOUR PERIOD PERIGEE=300 km APOGEE=27,558 km INCLINATION=63.4°			MASS=360 kg payload: 125-200 kg; AVE. POWER= 250-350 watts	MASS=400 kg payload: 150-225 kg; AVE. POWER= 275-400 watts
PERIGEE = 500 km. APOGEE = 7000 km. INCLINATION = 63.4 °	MASS=69 kg; payload: 5-10 kg; AVE. POWER= 20-30 watts		MASS=635 kg payload: 175-350 kg; AVE. POWER= 350-600 watts	MASS= AVE. POWER=

ENERGY-LIMITED REQUIRES MORE DETAILED STUDY

FIGURE 2. RESOURCES AVAILABLE TO PAYLOAD (TYPICAL)

Booster fairing volume limitations can be a significant design driver. The various standard fairings are summarized in Table 3. Obviously, the most severe diameter limitations are offered by the Pegasus and Taurus vehicles, and this has dictated a particular mechanical/structural design approach for such satellites at Fairchild. This has been developed under Fairchild IR&D and is here applied in a RESERVES-unique manner.

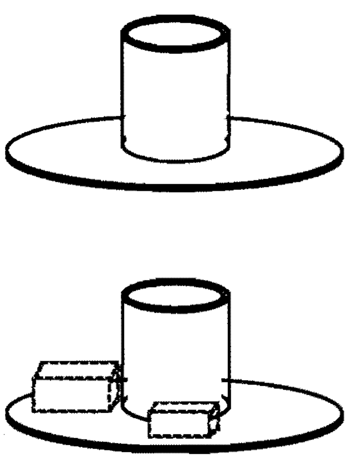
TABLE 3. APPLICABLE BOOSTER FAIRINGS

BOOSTER	FAIRING diam. inch	FAIRING length inch	FAIRING extensions inch
PEGASUS	46	72	
SSLV (Taurus)	50	96	+ cone
LPLS (Lockheed)	~60	~40	+ cone
ATLAS II	~120	154	
DELTA II	97 100	80 80	+ cone + cone (+86Dx90L neck)
TITAN II	~120	72	+ 60 ext. (+96 nose)

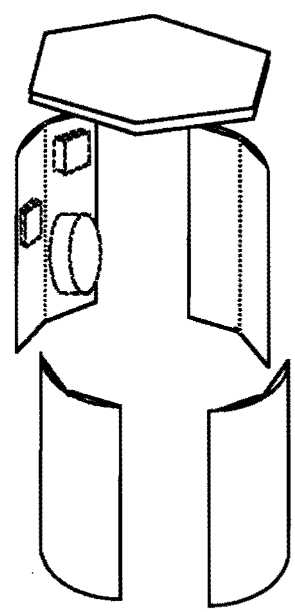
4. SPACECRAFT DESIGN CONCEPT

4.1 CONFIGURATION

The large disparity in shroud diameters of potential interest to RESERVES (see Table 3) led to initial consideration of a design which might be adaptable to a variety of spacecraft diameters. Such an approach is illustrated in Figure 3a. This features a structural design which is most common in spacecraft; i.e., a central thrust tube to react the major launch loads, with most spacecraft equipment mounted outside of this tube. Shelves radiating outward from the thrust tube could be provided in a range of diameters upon which to mount equipment. A module can consist of a section of tube with one shelf of equipment; modules can then be bolted together as required to complete a spacecraft, with all electrical connectors and harnessing within the central tube.



A. THRUST TUBE & SHELVES



B. LOAD-BEARING SHELL & COVERS

FIGURE 3. CONFIGURATION CONCEPTS

This approach was compared with one in which the launch loads would be reacted out by the outer shell of the spacecraft, suitably stiffened by top and bottom (and, in larger sizes, internal) shelves or covers, as illustrated in Figure 3b. In this case, most components are mounted to the inner surfaces of the walls of the structure. Modules would consist of cylindrical segments and covers, with their attached components.

Although the former approach offers the possible advantage of using larger shroud diameters more effectively, this was seen to be too constraining in the important 40-to-50 inch small-satellite diameters, leaving too little space for equipment, and structurally heavier in this size range.

The structural-shell design offers a clearly better solution in the 40-to-50 inch diameter size range. For larger size boosters, the design has the added advantage of being modular in the whole: Two complete spacecraft sized to fit within the Pegasus shroud (46-inch diameter) should be able to fit side-by-side within a 100-inch shroud (e.g.: Delta II), and three such spacecraft can fit in a triangular cluster within a 120-inch shroud (e.g.: Delta II [alt.], Titan II or Atlas II). Since most RESERVES missions appear to require multiple satellites in the same orbit for frequent coverage of specific regions, this potential advantage may be economically decisive. Therefore, it was decided to proceed with the latter design approach. A further advantage is that this approach takes advantage of Fairchild IR&D, under which this structural design concept and the tooling necessary for the fabrication of these structural shells and covers has been developed.

4.2 MODULARITY

The Fairchild-developed tooling is sized to produce a structural shell of 32-inch diameter, with 8 internal flats bonded to the cylindrical shell through a honeycomb layer. The tooling allows the bonding of a complete semi-cylinder containing 4 internal flat walls, or any subsection containing 1, 2, or 3 segments.

Conceptually, therefore, it is possible to modularize the design into cylindrical segments and end covers. A possible configuration would be:

- Power module (1-to-3 segments)
- Attitude-Control module (1-to-3 segments)
- Data Handling and SGLS Communications module (1 segment)
- High-Data-Rate Communications (or mission-unique) module (1-to-2 segments)
- Propulsion module with booster interface (end cover)
- Solar-Array module with payload interface and attitude reference sensors (end cover)

This configuration is shown in Figure 4, containing the modules

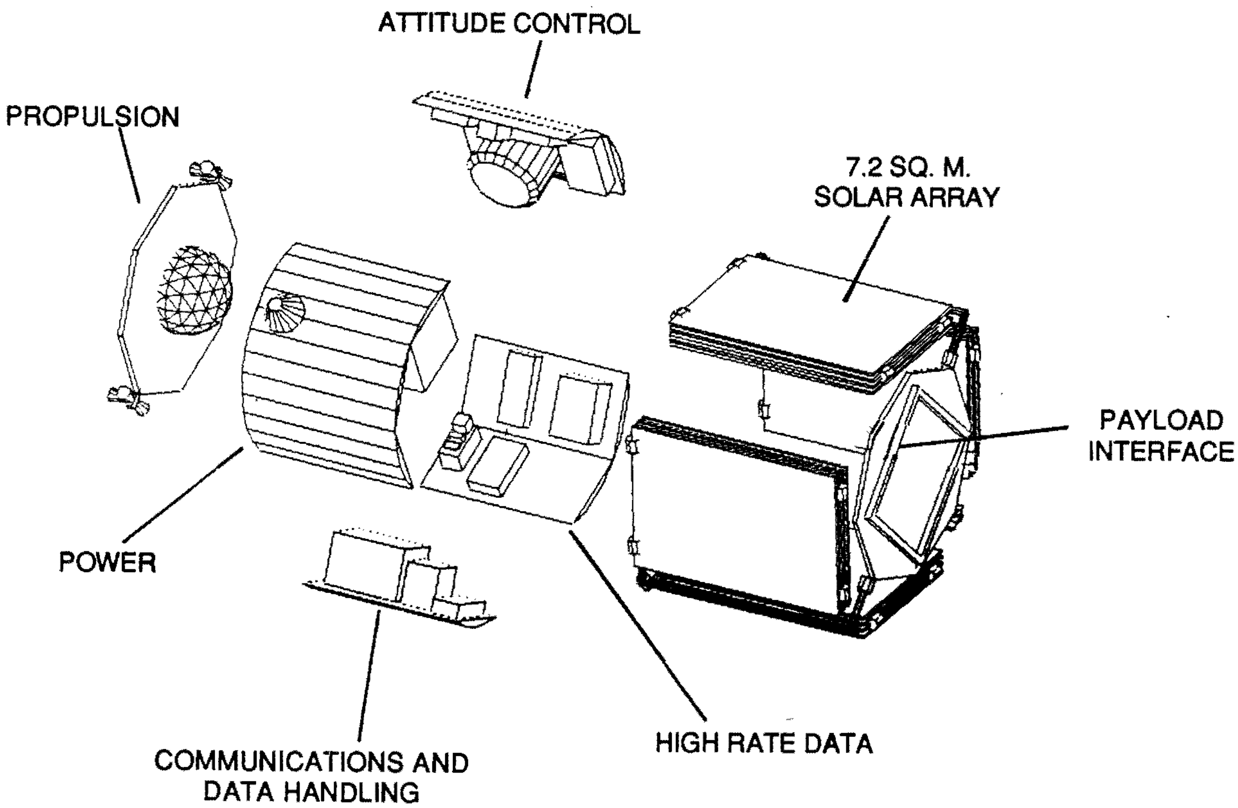


FIGURE 4. SPACECRAFT MODULES

required for the baselined missions. Two versions of the power module and two versions of the solar-array module are required. For the HAIR (and possibly LAOSS) mission the high-rate communications module is included. Figures 5a. and b. show the HAIR-configured spacecraft in assembled condition with the payload-attach plate removed for visibility.

4.3 FLIGHT CONFIGURATION

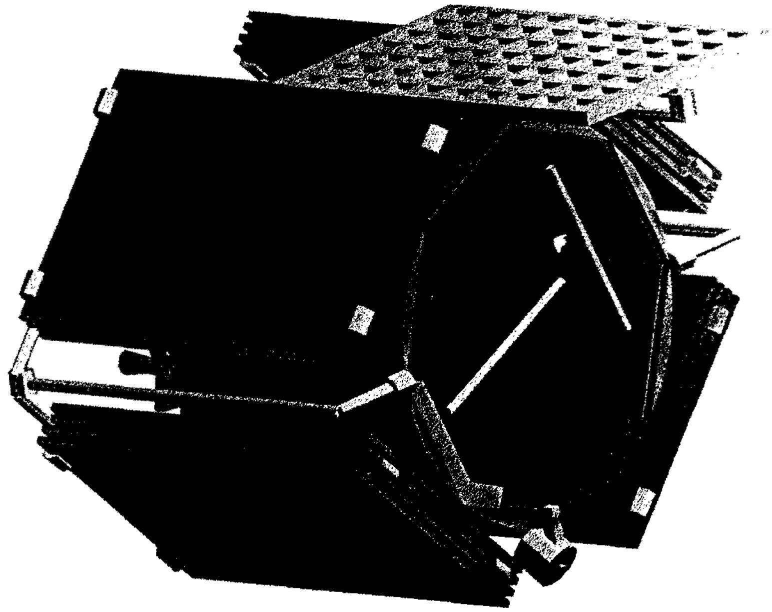
The assembled configuration for the LAOSS and HAIR missions is shown in stowed configuration in the Pegasus and Taurus shrouds in Figures 6 and 7 respectively. Figures 8 and 9 show each spacecraft in deployed configuration. As currently configured, flight direction is along the longitudinal spacecraft axis. Flight direction can be controlled as well such that the longitudinal axis is Nadir-directed, or held to some fixed angle to Nadir. A flight direction with the spacecraft longitudinal axis along the orbit normal would also be possible with a slightly different mounting arrangement for the bias-momentum reaction wheel in the attitude control module.

Because of the large range of sun angles which are allowed within the spectrum of missions to be served, a dual-gimballed solar-array is shown. The pitch-axis gimbal is located at the root of the structural mount to the spacecraft, while the tranverse-axis gimbal is located at the midpoint support of the array.

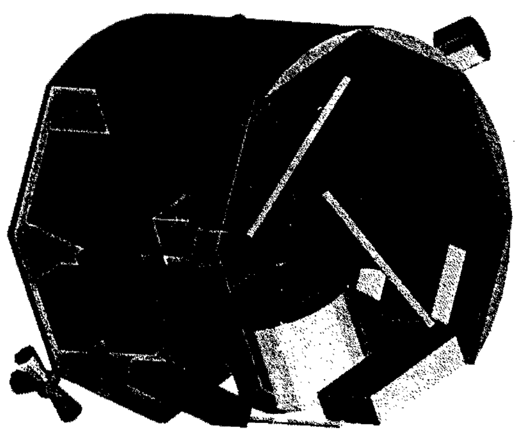
4.4 SYSTEM BLOCK DIAGRAM

Communications and Data Handling

The various modules of the system are further detailed in the system block diagram shown in Figure 10. The central "brain" of the system is the Communications and Data Handling module, which contains the powerful Fairchild-developed Telemetry/Command Processor (TCP). The TCP, contained in a single card-cage box, is a microprocessor-based modular system which controls a MIL STD1773 fiberoptic data bus for all on-board communication. The TCP is based on a MIL STD 1750A microprocessor chip, which will



A. TRANSPARENT TOP COVER



B. TRANSPARENT TOP AND SIDES

FIGURE 5. SPACECRAFT ASSEMBLY

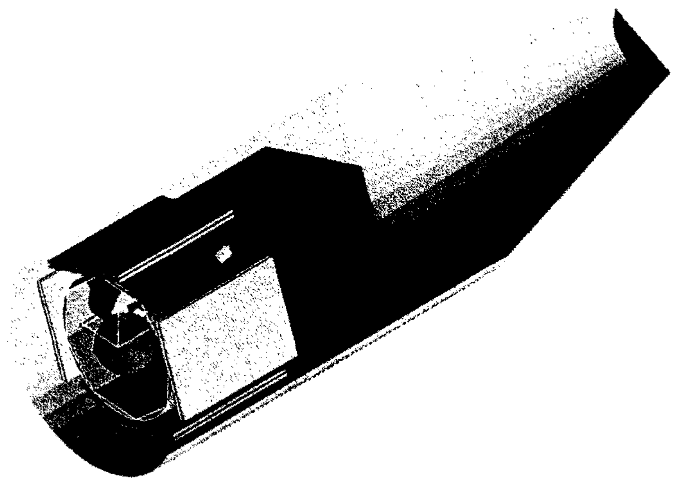


FIGURE 6. LAOSS SPACECRAFT IN PEGASUS SHROUD

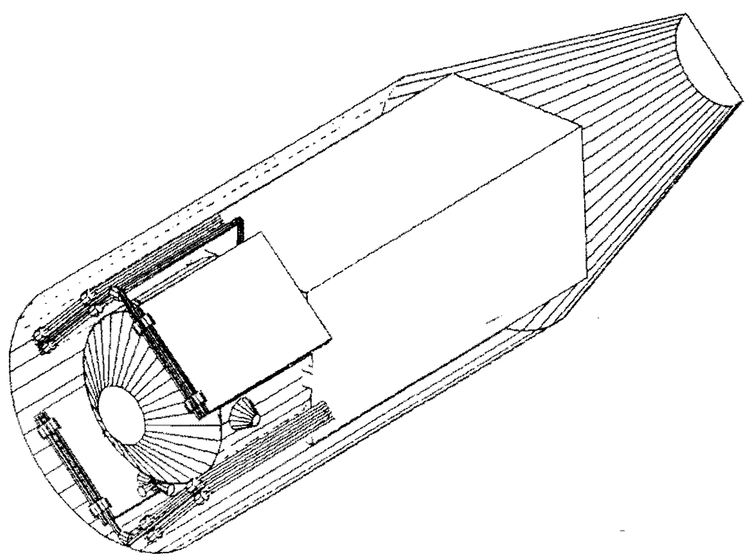


FIGURE 7. HAIR SPACECRAFT IN TAURUS SHROUD

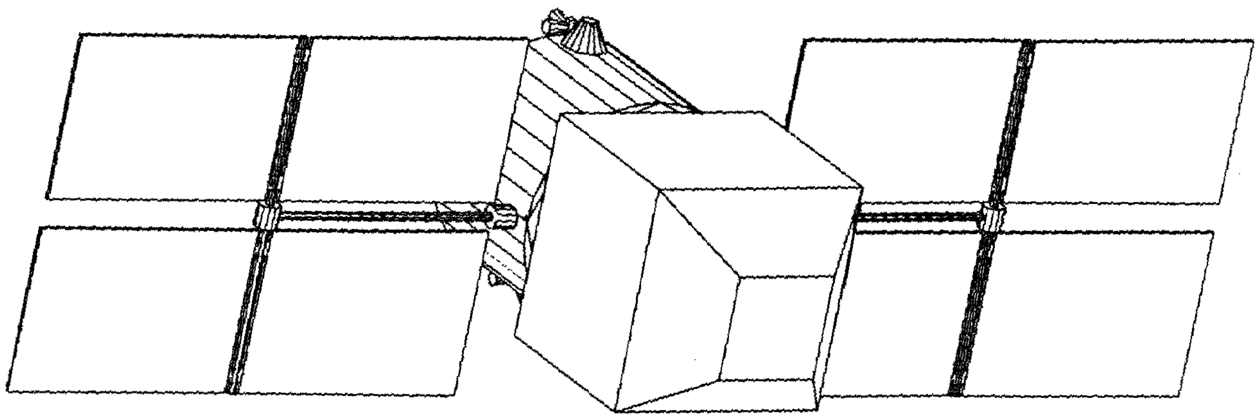


FIGURE 8. LAOSS FLIGHT CONFIGURATION

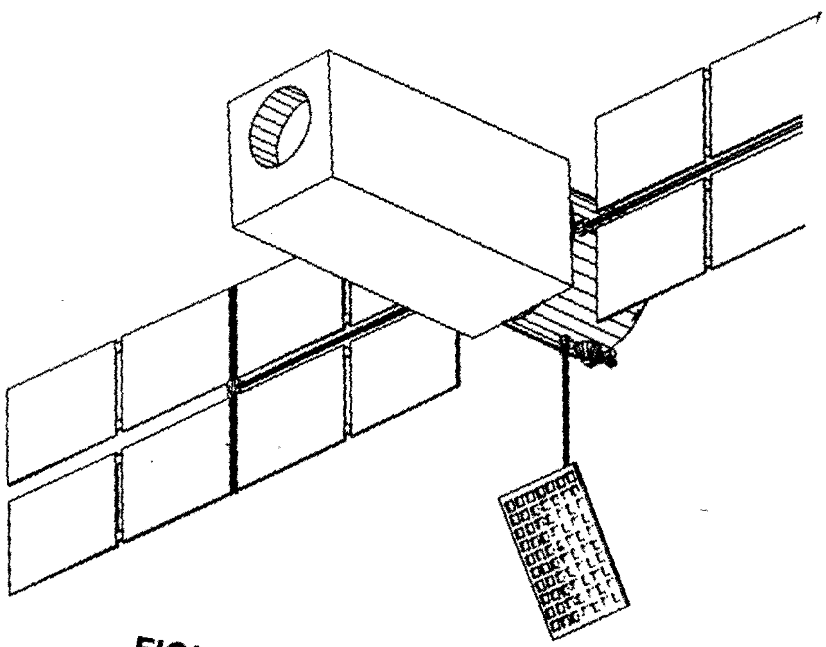


FIGURE 9. HAIR FLIGHT CONFIGURATION

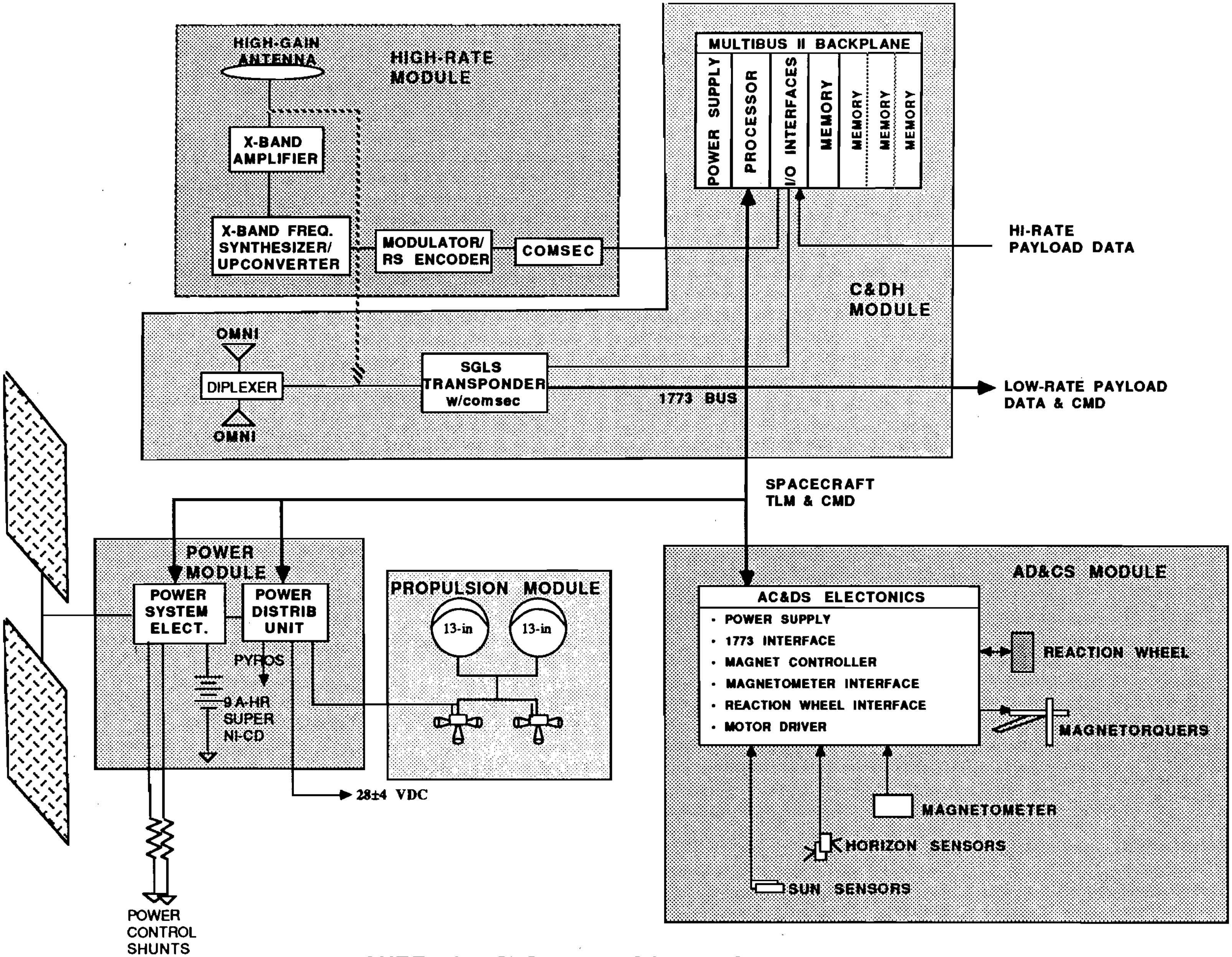


FIGURE 10. SYSTEM BLOCK DIAGRAM

handle up to 2 MIPS of data-processing throughput, and is extremely mass-, power- and cost-effective.

The system is used as a unit integrated with the Fairchild-developed solid-state memory system. This system is also available as a stand-alone replacement for magnetic tape or optical disc recorders. As used in the TCP application, the memory card(s) is mounted to a common (Multibus II) backplane with the processor card, the interface control card, and the power supply card. This configuration allows the memory cards to function as the data storage system for spacecraft control software, telemetry, and delayed commands. Typically, these functions consume only a small portion of the available memory, leaving substantial memory for payload use.

As a result of ongoing development at Fairchild, the next generation of memory card, expected to be available in 1-to-2 years, is expected to have 1.2 Gbits of memory per card, of which over 500 Mb will be available for high-rate data storage.Thus, a single memory card will be able to store over 15 high-resolution frames of HAIR data, a substantial margin for the anticipated mode of operation. The CCD output of the HAIR sensor can be read directly into the system at the real-time rate of 168 Mbps. Once read into the system, the raw CCD data can be fetched by the processor card for data-processing and compression, multiplexed with spacecraft system data such as time and pointing direction, and formatted for downlink transmission at a "real-time" rate of 10 Mbps, or at a "fax" rate of 9600 baud.

The memory units are typically hardened to 30-to-40 krads, and consequently may require some additional shielding for the high-altitude missions to be launched near solar maximum. A one-cm. thickness of shielding has been included in the mass summary for these cards.

The basic S-Band communications package is provided for standard low-rate telemetry and satellite control commanding, with a standard low-power (e.g.; 1-to-4 watt) SGLS transponder feeding and receiving signals from a pair of 3-dB omnis mounted on either side of the spacecraft. This communications equipment flies with all missions and provides communication through the CSOC. Data rates of 10-to-1000 kbps are readily supported over this link, depending on satellite altitude.

Tactical users equiped with a 2-meter S-band remote terminal can receive mission data from the LAOSS satellite at moderately high rates, e.g., at 1 Mbps, owing to the relatively short range of this transmission. This will allow a high-resolution LAOSS frame to be transmitted in some 30 seconds after lock-on, or a 9600 Baud fax transmission of a low-resolution (say weather) scene in some 3.5 minutes, if the satellite pass is sufficiently high to remain in view this long. A restricted command set may be available to the tactical commander to enable him to command this transmission at his discretion. Alternatively, a RESERVES interface team can coordinate this request through the RESERVES command headquarters, with the actual command delivered over the CSOC link.

High-Rate Module

The optional communications package is the High-Rate Module, which is intended for surveillance/recon missions, e.g.; LAOSS or HAIR. (The modular nature of our design will allow this package to be mounted on the satellite for any mission , if mass and power will allow.) The high-rate module will allow essentially real-time image transmission at 10 Mbps from the apogee of the HAIR orbit, of from any point in the LAOSS orbit. Because of power limitations, the duty cycle of transmission may be limited. (In our example, it is roughly 10-minutes per orbit.) It is configured to be compatible with existing X-band MIST (Modular Interoperable Surface Terminal) equipment installed on many surface vessels and ground sites, thus providing the much-desired backward compatibility with tactical airborne reconnaissance systems. A phased-array antenna is shown to accompany the High-Rate module for the HAIR application. From

LAOSS altitudes, the omni antenna will suffice for this X-band transmission.

Attitude Control and Determination

The attitude control and determination requirement outlined in the mini-spec (Table 2), can be satisfied to its essentials by a bias-momentum reaction-wheel system combined with magnetic torquers and a number of sensors, as shown in the diagram of Figure 10. The direction of the wheel axis determines the spacecraft pitch axis. As depicted in Figure 5, this determines the flight direction to be along the spacecraft longitudinal axis, with the solar-array axis at a 30° tilt in the roll direction. Other mounting options for the single wheel are possible if another flight attitude is preferred. Off-nadir pointing in the pitch direction is achievable over a limited range, but less easily achieved in other directions for long periods. As now conceived, the optical payloads do their own pointing, as do the communications payloads and subsystems. Therefor, off-nadir operation is not now seen as a clear requirement and a fixed momentum wheel is baselined at present.

The sensor complement consists of a fore- and aft-looking horizon sensor pair, mounted to the payload interface plate, a pair of fan-beam sun sensor mounted so as to intersect the sun at least once per orbit, and a three-axis magnetometer to assist in use of the magnetic torquers for momentum unload. The horizon sensors provide continuous knowledge of roll and pitch orientation to an accuracy of approximately 0.08°. To hold roughly the same order of magnitude in yaw requires a sun-sensor update at least once per orbit.

During normal operations, all axes are held to within approximately 0.5° by the action of the momentum wheel of the proper size. A control-loop bandwidth of 0.01 Hz will provide rate stability of at least 0.002 degrees per second. It may be assumed that such potential sources of jitter as solar-array or high-gain antenna tracking will be quiescent during periods of optical imagery.

Power Control and Distribution

The power control system is an outgrowth of Fairchild's development for the NASA Small Explorer power subsystem. It is a Direct Energy Transfer system based on the use of sequential linear shunts to reduce internal power dissipation.

This approach was selected by NASA for its simplicity and low mass and power consumption. Its features are:

- Battery connects directly to the bus for low impedence
- Ampere-Hour C/D charge control for extended battery life
- Bus overvoltage, undervoltage, and overcurrent protection
- Battery undervoltage, maximum charge rate, and overtemperature protection
- Low bus ripple; low EMI generation

Propulsion Module

The propulsion module features 1-to-4 Hydrazine tanks and two rocket-engine modules together with various valves, tubes, lines, and pressurant, fully assembled, welded, and mounted to the lower closeout deck. Each fuel tank can hold 7.5 kg of propellant for orbit trim, orbit adjust, or drag makeup purposes. Because the AC&DS system contains its own magnetic torquers for momentum unload, propulsion is not normally required for attitude control purposes.

Based on our initial evaluation of orbit requirements, injection accuracies of the various boosters appear adequate for the placement of these satellites in the desired orbits. Over time, orbit drift may result in a trim requirement, as well as drag makeup for the LAOSS satellite. Therefore, a single tank propulsion system is included in each design assembly, although this may not be absolutely required in all cases for the limited mission times projected in the mini-spec.

4.5 POWER & MASS

The spacecraft power summary is shown in Table 4. The derived average spacecraft power is then added to the payload power requirements for the four baselined missions to derive the satellite Power-Energy balance, as shown in Table 5. In both mass & power summaries, we include the X-Band high-rate transmit equipment for both optical surveillance missions. The X-Band receiver components are omitted under the assumption that uplink commands are transmitted via the standard SGLS S-Band link.

The results displayed in Table 5 demonstrates the feasibility of performing the four missions with two battery sizes and two solar array sizes and cell types. Positive margins are available both in battery and in solar array size. The higher efficiency GaAs/Ge cell is selected for the LAOSS and HAIR missions in order to conserve mass and volume to assure fitting within the Pegasus and Taurus capability. On a system-level cost trade, this is preferred to using the standard 8-mil silicon cell which is provided for the other missions, if this results in requiring the next size booster in each case. However, such system-wide cost trades will clearly be very sensitive to the actual performance cababilities of the various boosters, and must take into account the level of inventory required for the expected distribution of missions to be flown. Such considerations are recommended for future study.

The mass summaries for the 4 baselined missions are shown in Table 6. Of particular interest is the use of Pegasus for the LAOSS mission; this appears to be quite feasible with an adequate mass margin available. The use of Taurus for the HAIR and Data Relay Communications missions is assured with an even greater mass margin.

The EHF Comm. mission is an excellent example of a cluster launch. Three such spacecraft can be launched on a single Atlas II into the desired Molniya orbit with an excellent mass margin.

TABLE 4. SPACECRAFT POWER SUMMARY

SUBSYSTEM/COMPONENT	POWER, W	
POWER	10	
- PSU		7.5
- PDU		2.5
C&DH	12	
HIGH-RATE COMM.*	10	
- X-band diplexer [a]		0.25
- PSK modulator		16.40
- PSK demodulator [a]		15.00
- X-band freq. source		20.00
- 10 MHz reference osc.		1.20
- X-band up/downconverter		18.00
- X-band power amp (20 WRF)		75.00
- X-band LNA [a]		3.00
- X-band switch [a]		0.25
AC&DS	22	
- electronics		10.0
- mom. wheel		6.0
- horizon sensors		0.6
- sun sensors		3.6
- magnetometer		0.7
- magnetorquers		1.0
THERMAL	0	
PROPULSION.	0	
HARNESS	1	
TOTAL	55 W.	

* 10% DUTY CYCLE ASSUMED
[a] RECEIVER COMPONENTS

TABLE 5. SATELLITE POWER AND ENERGY BALANCES

MISSION	ORBIT PERIOD	MAX. ECLIPSE	LOAD POWER	DISCH REQ.	BATTERY		SOLAR ARRAY			
					REQUIRED @ 40% DOD	PROV-IDED	POW'R REQD	1-YR EFF. IN ORBIT	AREA REQ	PROV-IDED
	min.	min.	watts	A-HR	A-HR	A-HR	W	%	SQ. M.	SQ. M.
LAOSS	89.8	37	255	5.6	14	20	570	12.8*	3.3	3.6
HAIR	180	53	355	11.2	28	30	640	7.1*	6.7	7.2
EHF COMM	721	60	275	9.8	25	30	475	5.5^	6.4	7.2
DATA REL.	169	51	230	7.2	18	20	470	5.1^	6.8	7.2

* 8-MIL GaAs/Ge CELLS
^ 8-MIL BSFR Si CELLS

TABLE 6. MASS SUMMARY

SUBSYSTEM \ MISSION	LAOSS	HAIR	EHF COMM	DATA COMM
POWER	89	124	124	124
C&DH	16	21	16	21
HIGH RATE COMM	6	41	0	0
AD&CS	20	20	20	20
STRUCTURE	12	12	12	12
PROPULSION (WET)	15	15	15	15
THERMAL	8	8	8	8
HARNESS	20	20	20	20
BUS SUBTOTAL	185	260	215	220
PAYLOAD MODULE	70	70	120	90
TOTAL SATELLITE WEIGHT	255	330	335	310
BOOSTER	PEGASUS	TAURUS	ATLAS II	TAURUS
NO. SATELLITES/LAUNCH	1	1	3	1
MASS TO MISSION ORBIT	330	625	1,750	625
MARGIN, KG	75	295	746	315
MARGIN, PERCENT	29	89	74	102

5. CONCLUSIONS

The requirement for small tactical satellites capable of satisfying a variety of missions on short notice is challenging and stressing. An innovative spacecraft design approach is described which makes use of pre-assembled subsystem modules which can be assembled and checked out on short notice. The design has been successfully tested against the requirements of four example missions using three different launch vehicles, demonstrating ample design margins in each case. Of particular note, a low-altitude optical surveillance mission which fits within the constraints of the Pegasus launcher is shown to be feasible. The approach holds promise of providing a simple, flexible, cost-effective solution to the rapid-response requirements of the tactical military world.