MULTI-PURPOSE SATELLITE BUS

By

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Abstract

This paper presents design considerations of a Multi-Purpose Satellite bus (MPS). The project was sponsored by the NASA/Universities Space Research Association (USRA) Advanced Design Program. A multi-purpose satellite bus was selected as the study topic from a Statement-of-Work generated by the Defense Advanced Research Projects Agency. The estimated beginning-of-life weight of the MPS bus is 150 kg. A detailed analysis with two dissimilar payloads (a meteorological payload and a communication payload), both having a three year life, demonstrates the flexibility of the bus. The MPS bus was designed to mate with the Pegasus Air Launched Vehicle and the Taurus Standard Small Launch Vehicle. The satellite bus design is presented with emphasis on configuration requirements due to the dissimilar mission demands.

Introduction

The Multi-Purpose Satellite bus (MPS) is designed to accommodate a number of payloads for a variety of missions. Payloads envisioned for this spacecraft include meteorological, communications, surveillance and tracking, target location, and navigation areas benefiting the military, science, and technical communities [Ref. 1]. The MPS bus will accommodate various payloads in assorted mission areas with a minimum of modifications. The MPS bus is designed to be an autonomous 3-axis stabilized spacecraft with the payload to be attached to the Earth-viewing face.

Two dissimilar payloads were selected for the

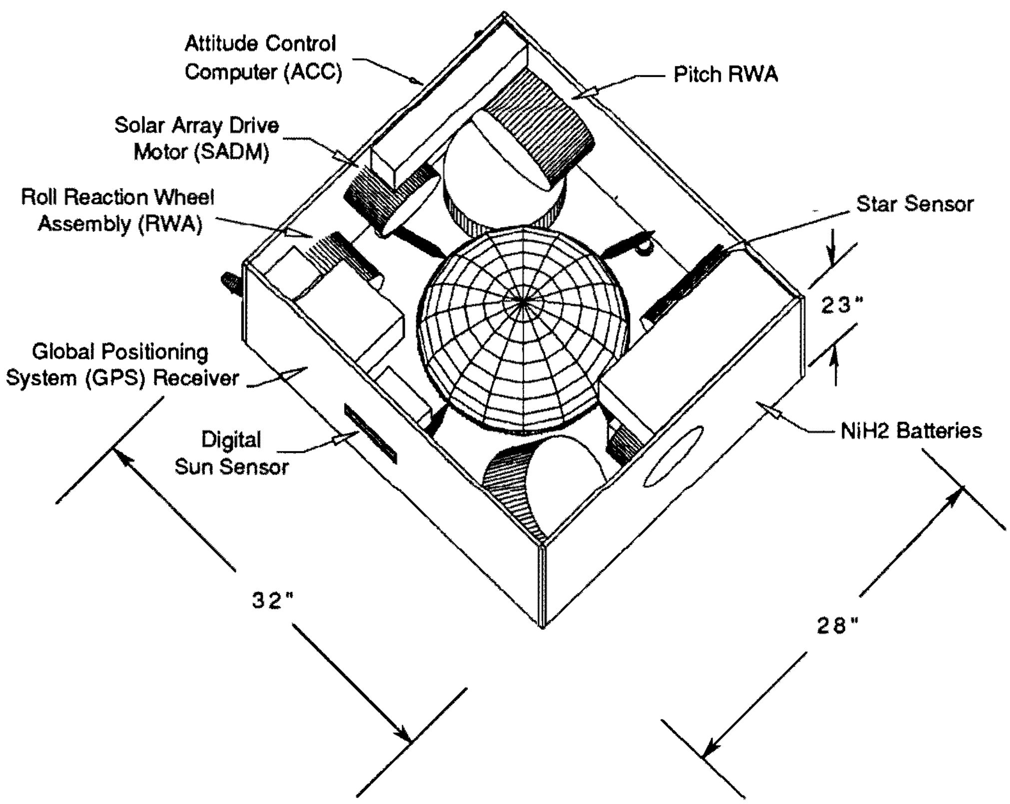

Figure 1. MPS Configuration.

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design having different orbit and pointing requirements. The first payload is the Advanced Very High Resolution Radiometer (AVHRR), a meteorological payload. The second payload is a communications payload operating in the Extremely High Frequency (EHF) bands. Both payloads have a three year life. The MPS bus design provides flexibility to fulfill both mission requirements. Figure 1 shows the configuration of the MPS bus.

The AVHRR payload is launched into a low-Earth, sun-synchronous orbit by the Pegasus Air Launched Vehicle (ALV). The orbit is at 833 km [450 nmi] altitude with 0830 descending or 1530 ascending nodes and 98.75° inclination. The AVHRR payload requires 0.01° pointing accuracy and is attached on the Earth face (+ yaw face). Pointing requirements restrict rotation of the spacecraft about its yaw axis.

The EHF communication payload requires a highly elliptic Molniya-type orbit for long excursions above the northern hemisphere for high latitude coverage. The orbit selected is an 8-hour Molniya orbit with 500 km [270 nmi] perigee and 27,000 km [14,578 nmi] apogee. The EHF payload is launched aboard the Taurus Standard Small Launch Vehicle (SSLV) because its high-energy orbit cannot be attained by the ALV. A critical inclination of 63.43° is imposed. The pointing requirements of the EHF payload is 0.5° accuracy; and also has no restriction for rotation about the yaw axis. The EHF payload is also attached to the Earth-viewing face.

AVHRR Orbital Analysis Results

A summary of orbital parameters follows in Table 1 for the two payloads. Two concerns motivated the orbital analysis for the AVHRR. The first consideration was to prevent sunlight from shining on the AVHRR thermal radiators. The second study determined the angles between the normal vector of the spacecraft's faces and the sun vector. The rotation angles of the solar arrays and the worst-case view by the solar arrays of the sun could then be calculated. The worst-case eclipse duration for the AVHRR payload is 29 minutes. Orbit analysis for the EHF payload is given in Ref. 2.

Figure 2 shows how the sun angles on the spacecraft faces vary as the satellite moves through one revolution on the first day of winter for the 0830 pm ascending node case. The sun angles on the (+/-) pitch faces are constant over a single orbit because it is sun-synchronous. The pitch face sun angles vary with season from 141° to 131° for the (+) pitch face and 39° to 49° for the (-) pitch face. The (+) pitch face allows placement for thermal radiators since it never sees sunlight.

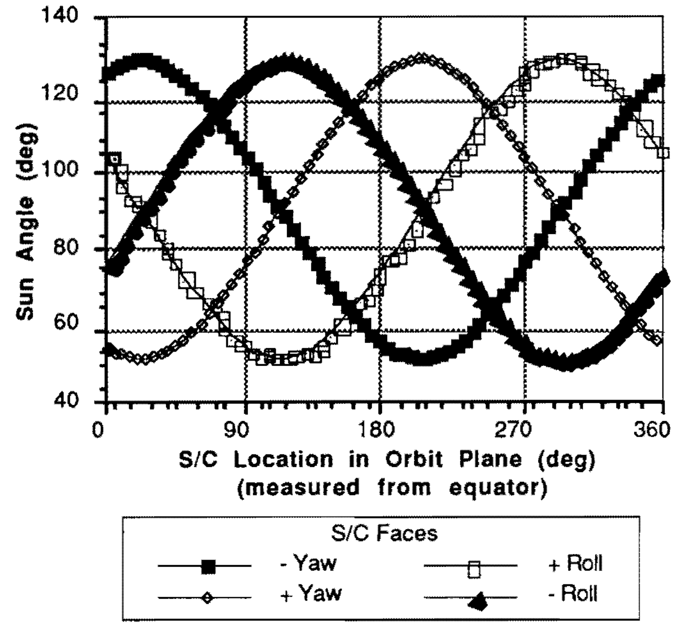

Figure 2. Sun Angle on Spacecraft Versus Orbital Position

Further analysis of the sun angles yields a worst-case 50° sun angle for the solar array. The solar array rotation angle profile is shown in Fig. 3 for seasonal variation. The spacecraft will require either a sensor to detect sun location or continual updated profiles because of the varying nature of the rotation angle profile.

TABLE 1. SUMMARY OF ORBITAL PARAMETERS

Payload	AVHRR	EHF Communications
Orbit Type	Sun-synchronous	Molniya
Period	101.5 min	8 hr
Semi-major Axis	7212 km [3894 nmi]	20,307 km [10,965 nmi]
Eccentricity	0.0	0.661
Inclination	98.75°	63.43°
Ascending Node	3:30 PM/8:30 PM	N/A
Argument of Perigee	N/A	270°

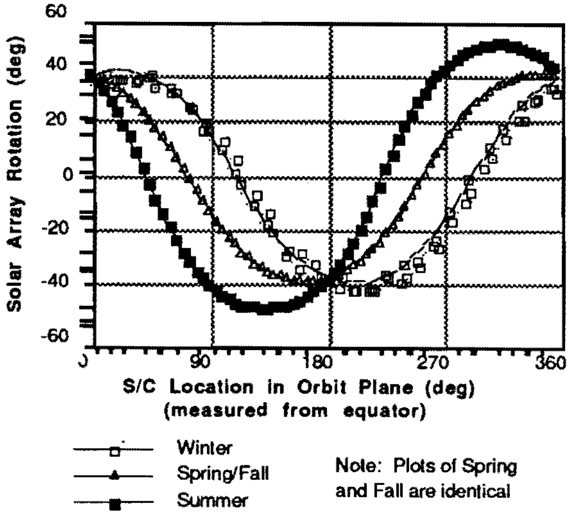

Figure 3. Solar Array Rotation Angles Versus Orbital Position and Season

Orbital maintenance for the AVHRR payload was deemed unnecessary because of similar meteorological satellites which do not require it. The Defense Meteorological Satellite Program (DMSP) uses the same orbit as the AVHRR with a similar payload and requires no orbital maintenance during its lifetime.

The EHF payload because of less stringent pointing requirements can be positioned to provide solar arrays normal to the sun's rays since the spacecraft is free to rotate about its yaw axis. The orbital analysis was more concerned with eclipse duration and the possibility of orbital maintenance requirements. The eclipse duration for the EHF payload is given in Table 2.

The EHF payload orbit was analyzed for zonal harmonics J_2 through J_7 resulting in a 500 year period for rotation of the perigee. Perigee will move less than 2.5° for the satellite's mission life. Change in inclination and eccentricity over the mission duration is also negligible.

Spacecraft Configuration

The configuration of the MPS bus is designed for flexibility for the two payloads as well as other payloads. The design is modular where the payload attaches in a "bolt-on" fashion to the Earth face. The design of the attitude control subsystem is also modular with a Basic Sensor Subsystem (BSS) and a Precision Sensor Subsystem (PSS). The bus is a 3-axis stabilized, nadir pointing, dual solar array spacecraft. A hydrazine propulsion subsystem is provided. The propulsion subsystem is not required for the two payloads studied and is used as a backup system for momentum dumping.

A single degree-of-freedom solar array assembly was chosen for mechanical simplicity. The EHF payload allows a second degree-of-freedom by rotation of the spacecraft about the yaw axis; however, payloads similar to the AVHRR mission will not allow the spacecraft rotation about the yaw axis causing a sun angle up to 50°. This circumstance requires that the solar array be oversized to provide sufficient power to take into account the cosine effect.

Dual solar arrays were implemented providing two wings of two solar arrays each. This design, having the arrays extend from the plus and minus roll faces, provided a challenging requirement in preventing reflection from the solar arrays onto the thermal radiator of the AVHRR which must face space.

Radiation is a major concern for the EHF payload since the spacecraft travels through the Van Allen radiation belts. The effect of radiation degradation for the EHF mission is approximately the same as the cosine effect for the AVHRR mission. The solar cell output for the EHF mission is given in Table 3 for beginning-of-life (BOL) and end-of-life (EOL).

TABLE 2. ECLIPSE DURATION FOR EHF PAYLOAD MISSION

True Anomaly at Eclipse Entry	70.6°
True Anomaly at Eclipse Exit	131.7°
Eclipse Duration	52.1 min.

TABLE 3. SOLAR CELL PARAMETERS

Parameter	BOL		EOL	
	Absolute	Relative	Absolute	Relative
I_{SC}	44	1	29.5	0.670
V_{OC}	584	1	483	0.827
P_{max}	19.8	1	11.3	0.571
V_{mp}	492	1	391	0.795
I_{mp}	40.24	1	28.9	0.72

The solar array area provided by the MPS bus is 2.81 m^2 [30.2 ft^2] which is larger than the determined area needed by either payload. An additional array can be added to each wing if a future payload demands. This would require the use of the Taurus SSLV as the launch vehicle to accommodate the increased payload volume.

Thermal analysis was performed using the PC-ITAS® software program from ANALYTIX Corp. A model with 145 surfaces and 165 nodes suggests that thermal control can be accomplished using passive means only. Optical Solar Reflector (OSR) material is used for radiating heat to the space sink.

Attitude Control

The attitude control subsystem is a modular design for payloads having either moderate (0.5° accuracy) or precision (0.01° accuracy) pointing requirements. A modular design was deemed necessary in order to lower cost where pointing requirements are of the order of 0.5°, as in general coverage satellites. The Basic Sensor Subsystem (BSS) consists of a conical Earth sensor, digital sun sensor, gyro assemblies, reaction wheel assemblies, attitude control computer, GPS receiver, and magnetic torque rods. Three primary reaction wheel assemblies are used with a fourth skewed to provide redundancy. The magnetic torque rods are used for momentum dumping with six 0.89N [0.2 lbf] thrusters providing backup. Figure 4 shows the block diagram of the BSS.

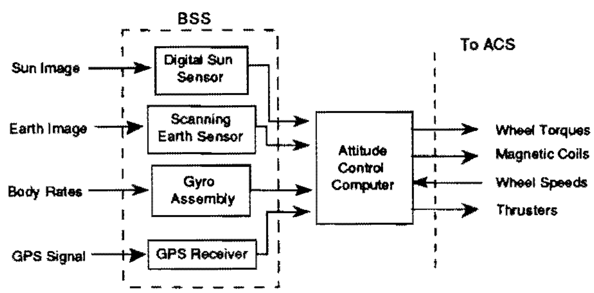

Figure 4. Functional Block Diagram of Basic Sensor Subsystem

The Precision Sensor Subsystem (PSS) augments the BSS with the addition of a celestial sensor assembly (CSA) for attitude determination. The CSA is a strap-down star mapper with a 10.4° field-of-view previously used aboard the DMSP Block 5D-3 satellite. The CSA requires daily updates of the 80 brightest stars that will be in view. The PSS uses the BSS for backup and initial attitude determination to eventually provide the accurate 0.01° pointing required by the AVHRR payload. The block diagram of the PSS is depicted in Fig. 5.

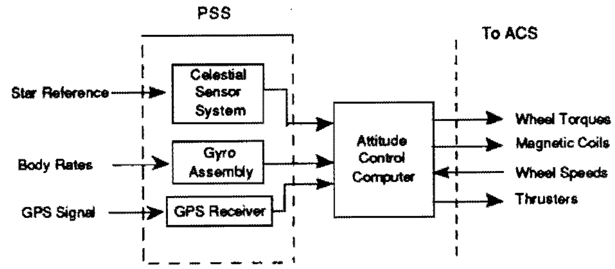

Figure 5. Functional Block Diagram of Basic Sensor Subsystem

The wheel speed is used as feedback and to determine if momentum dumping is required. When the momentum reaches the maximum for the wheel, the torque coils are commanded on to dump the excess momentum. Thrusters are fired to slow the rate to within acceptable limits in the case of excessive rate buildup.

The yaw motion of the satellite in the Molniya orbit is modeled as in HILACS [Ref. 2]. This study considers only the attitude control of the meteorological payload. Solar, gravity gradient, magnetic, and aerodynamic disturbance torques were modeled in a first order approximation for the design. The spacecraft was modeled as a rigid body with non-rotating, rigid solar arrays. An analysis was done using MATLAB® on a PC for resulting errors in pitch, roll, and yaw. The performance satisfied the 0.01° pointing requirement of the AVHRR payload. Figures 6, 7, and 8 show the resulting error in the pitch, roll, and yaw axes respectively over one orbit.

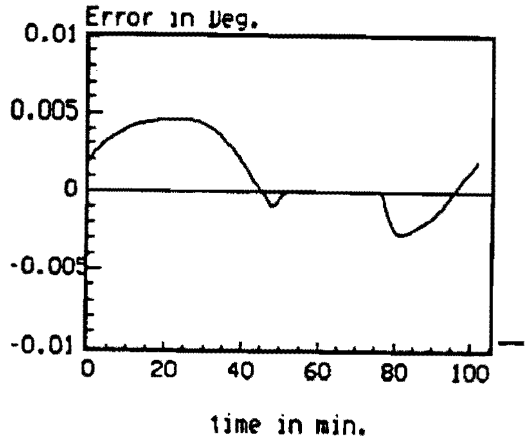

Figure 6. Pitch Error for One Orbit

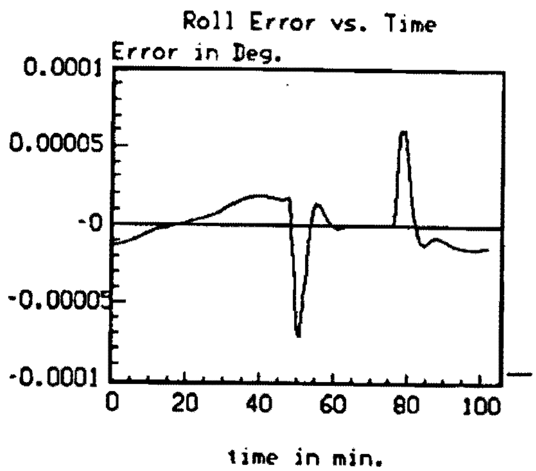

Figure 7. Roll Error for One Orbit

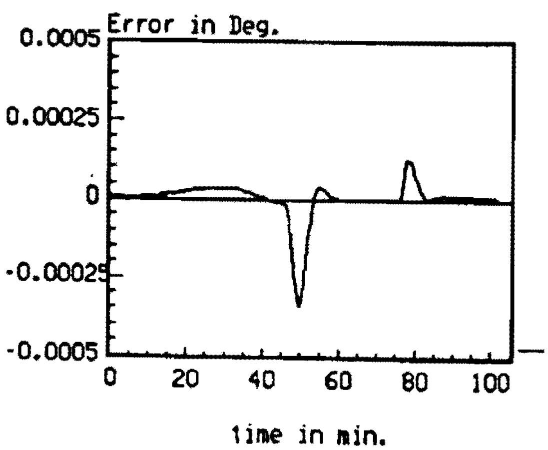

Figure 8. Yaw Error for One Orbit

Electrical Power

The electrical power subsystem provides a 28 V bus from silicon solar cells, a 12 amp-hour nickel-hydrogen battery, and power control electronics. Silicon cells were chosen for reasons of cost and reliability. The AVHRR payload requires continuous power totalling 201.8 W for the satellite. The EHF communication equipment operates only when the satellite is 20° above the horizon having a total 237.8 W for the satellite. Power control electronics are similar to the HILACS satellite [Ref. 2].

The NiH2 battery was chosen because of the high number of charge/discharge cycles the satellite may experience. The low-Earth orbit mission of the AVHRR payload will require over 15,000 cycles over the three year design life. The EHF payload may only be 1000 cycles. Power consumption during eclipse is 100.6 W for the AVHRR payload and 80.7 W for the EHF mission. Charging rates for the battery depends on the the duration of the sunlight period and the amount of power removed. The AVHRR payload requires a higher charging rate because of its shorter sunlight period and higher battery power consumption. Table 4 displays a summary of the battery charging requirements, where C is the capacity of the battery.

TABLE 4. BATTERY CHARGING SUMMARY

	AVHRR	EHF
Charge Required	76.8 W	30.7 W
Charging Rate	C/4	C/10
Charge Time	59 min.	6.5 hrs.
Available Sun	64 min.	7.1 hrs.

Conclusions

The Multi-Purpose Satellite bus (MPS) is a spacecraft design amenable to a number of payloads which may require different orbits, pointing accuracy, and electrical power consumption. The spacecraft was designed to accommodate, specifically, the two diverse missions of the AVHRR payload and EHF communication payload. The design fulfills the mission requirements of the two payloads with launch capability by either the Pegasus ALV or the Taurus SSLV. The design can support future payloads for military, science, and technical missions with minimal modifications.

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