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DESIGN OF MULTI-MISSION SPACECRAFT BUS

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Abstract

This paper presents preliminary design of a multi-mission spacecraft bus for meteorological and communications payloads. The meteorological payload uses sun-synchronous circular orbit and the communications payload uses Molniya type orbit to provide communications for areas not covered by geosynchronous communications satellites. The launch vehicles are Pegasus for the meteorological payload and Taurus for the communications payload. The spacecraft bus uses three-axis stabilization consisting of three reaction wheel system. The electric power system consists of single-axis tracking silicon solar array and $Ni_2 H_2$ batteries. The propulsion subsystem consists of six hydrazine thrusters and one propellant tank.

I. Introduction

Spacecraft design analysis and testing constitutes a significant portion of total spacecraft development time and cost. In order to reduce spacecraft development time and cost and improve reliability, use of existing spacecraft buses with minor modification is becoming almost mandatory to win a competitive spacecraft procurement. This has led to recent emphasis by spacecraft manufacturers on the development of multi-mission spacecraft buses. A study was undertaken at the Naval Postgraduate School to design a spacecraft bus for meteorological and high latitude communications missions. This paper presents the results of this study.

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II. Mission Payloads

The spacecraft bus was designed for two payloads: Advanced Very High Resolution Radiometer (AVHRR) and Extremely High Frequency (EHF) communications. These payloads were suggested by the Defense Advanced Research Projects Agency (DARPA). The orbit parameters for these payloads are given in Table 1.

AVHRR is a nadir pointing scanning radiometer which is sensitive in the spectral regions from 0.7 to 1.2 microns. It monitors data for day and night cloud mapping, sea surface temperature mapping, and other oceanographic and hydrologic applications. It provides data for High Resolution Picture Transmission (HRPT) at 665 Kbps and Automatic Picture Transmission at 2 Kbps. The HRPT data is at 1.1 km resolution while the APT transmission is maintained for use by ground terminals that do not have HRPT capability.

The EHF payload provides communications at high latitudes for the areas not covered by geosynchronous spacecraft. Communications is done at 2400 bps by means of 32 channels using frequency hopping over 255 frequencies. The signal band width of a single channel is 245 kHZ. Total band width required is 2 GHZ. The antenna/feed horn arrangement was designed by Lincoln Laboratory.

Table 1. Summary of Orbit Parameters

Payload	AVHRR	EHF Communications
Orbit Type	Sun-synchronous	Molniya
Period	101.5 min	8 hr
Semi-major Axis	7212 km	20,307 km
Ecentricity	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°

III Design Considerations

Both payloads, AVHRR and EHF communications, are nadir pointing. However, there are major differences in orbit, yaw axis control, attitude pointing, and thermal control requirements. The AVHRR payload uses sun-synchronous circular orbit and EHF communications payload uses highly elliptical inclined orbit. The AVHRR payload requires fine yaw control. Therefore, either the solar array needs to have two-axis tracking in order to keep its surface always normal to sun rays or single axis solar array should have larger surface to take into account the cosine effect of the angle between the solar array surface normal and the sun rays. The EHF communications payload allows unconstrained spacecraft rotation about yaw axis. Using spacecraft yaw rotation, a single-axis tracking solar array surface can be always kept normal to the sun rays. The attitude pointing accuracy of AVHRR is 0.01 deg in comparison to 0.5 deg for EHF communication, a difference of more than an order of magnitude. Thermal control of AVHRR is significantly more challenging. Its infrared detector is cooled to a temperature of about 108°K and maintained within 0.1°K. In general, in-orbit requirements for AVHRR payload are more stringent. Therefore, the challenge is to design the spacecraft bus to meet mission requirements for AVHRR payload without making spacecraft bus design unnecessarily complex and costly for EHF communications payload.

IV. Spacecraft Configurations

The selected spacecraft configurations for AVHRR and EHF Communications payloads are shown in Figs. 1 and 2, respectively. The solar array for AVHRR is selected to be single axis in order to avoid increase in complexity and cost for two-axis solar array. The thermal radiator for the AVHRR is mounted on positive pitch surface, resulting it facing deep space with no sunlight throughout the orbit. Tables 2 and 3 give spacecraft mass and power budgets for AVHRR and EHF communications

V. Launch Vehicles

AVHRR mission spacecraft is designed to be launched by Pegasus Air Launched Vehicle (ALV). It fits into Pegasus shroud of 46" diameter. The booster is carried aloft by a conventional transport/bomber class aircraft (B-52, B-747, L-1011). Once oriented along the desired orbit direction, level at approximately

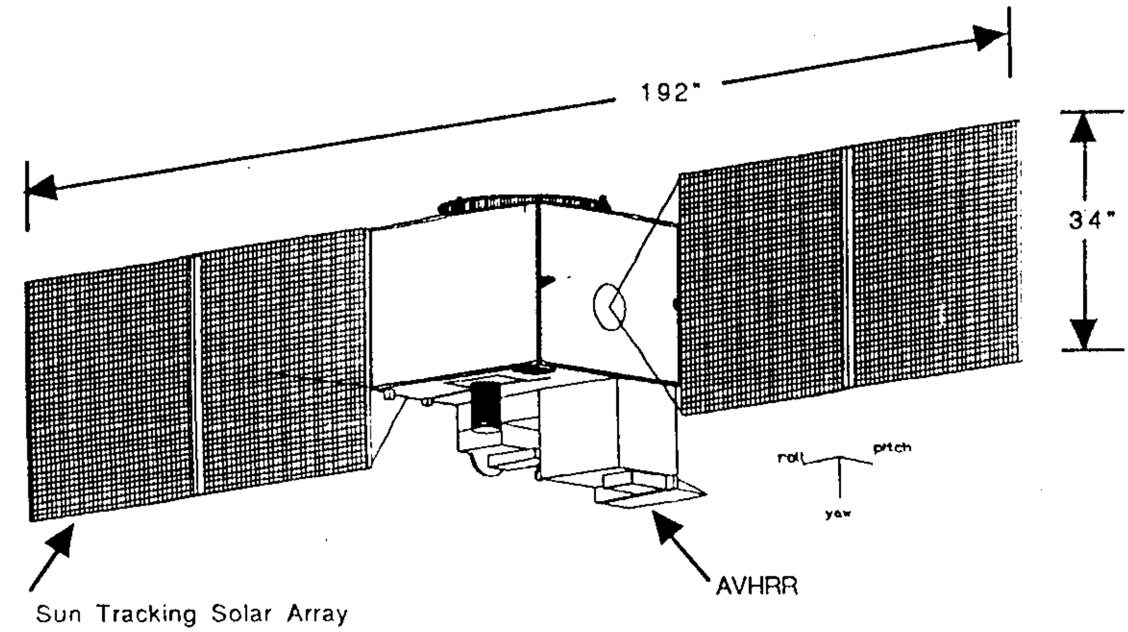

Figure 1. AVHRR Payload Spacecraft Configuration.

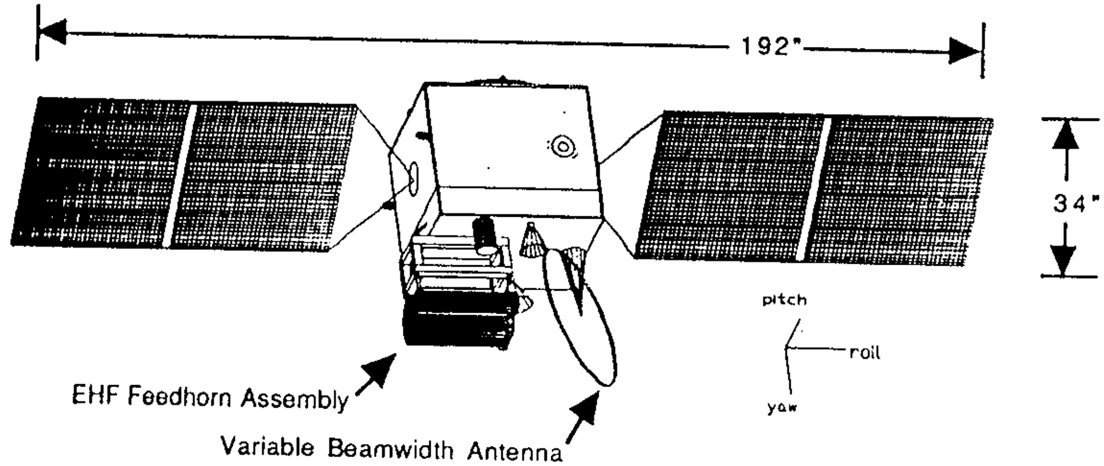

Figure 2. EHF Communications Payload Spacecraft Configuration.

Table 2. Spacecraft Mass Breakdown

Subsystem	AVHRR Mass (kg)	EHF COMM Mass (kg)
Payload	29.5	38.0
Electric Power	37.0	37.0
Propulsion	15.0	15.0
Attitude Control	25.0	21.0
Structure	21.0	27.0
Thermal	2.5	5.0
TT&C	4.5	4.5
Electric and Mechanical Integration	7.0	7.0
Mass Margin	13.5	16.0
Dry Spacecraft Mass	155.0	170.5
Propellant/ Pressurant	11.0	13.0
Spacecraft Mass at Separation	166.0	183.5

Table 3. Electric Power Budget (Normal Operations) Watts

Load	AVHRR	EHF
Payload	28.0	115.0
Electric Power	14.0	14.0
Propulsion	6.0	6.0
Attitude Control	58.5	58.5
Thermal Control	10.0	10.0
TT&C	11.0	11.0
Harness	4.0	4.0
Battery Charging	76.0	25.0
Margin	20.0	24.0
Total	227.5	267.5

42,000 feet, and flying at high subsonic speed, the parent aircraft releases the Pegasus booster.

EHF communications spacecraft is to be launched by Taurus into 8 hour Molniya type orbit because of inability of Pegasus to meet the orbit performance. The Taurus shroud diameter of 50 inch allows addition of a third solar array panel per side if needed. Taurus is a four-stage, inertially guided, 3 axis stabilized, solid propellant launch vehicle. The design incorporates a Pegasus first, second, and third stage atop a Peace keeper ICBM.

VI. Orbit Analysis

The orbit for the AVHRR is a circular sun-synchronous with altitude of 833 km. The orbit is the same as that used by the Defense Meteorological Satellite System (DMSP). The orbit analysis was performed for determining sun angle for satellite surfaces, sun angles on the solar arrays, and ellipse periods.

The sun angle for a particular satellite surface is defined as the angle between the surface normal and the sun vector. If the sun angle is greater than 90°, then the satellite surface does not have incident sunlight. The positive pitch surface does not see sunlight at any time and therefore is used for thermal radiator for the detector. The sun angle for negative pitch surface varies between 38.4° and 48.8°. The variation of sun for the remaining surfaces is shown in Fig. 3 for the first day of winter. The plots of sun angle for the other seasons are similar shape but contain a phase shift and a change in amplitude. Figure 4 shows sun angle profile for positive roll faces for the first day of all four seasons.

For EHF communications, by using spacecraft rotation about yaw axis and solar array rotation, the sun angle

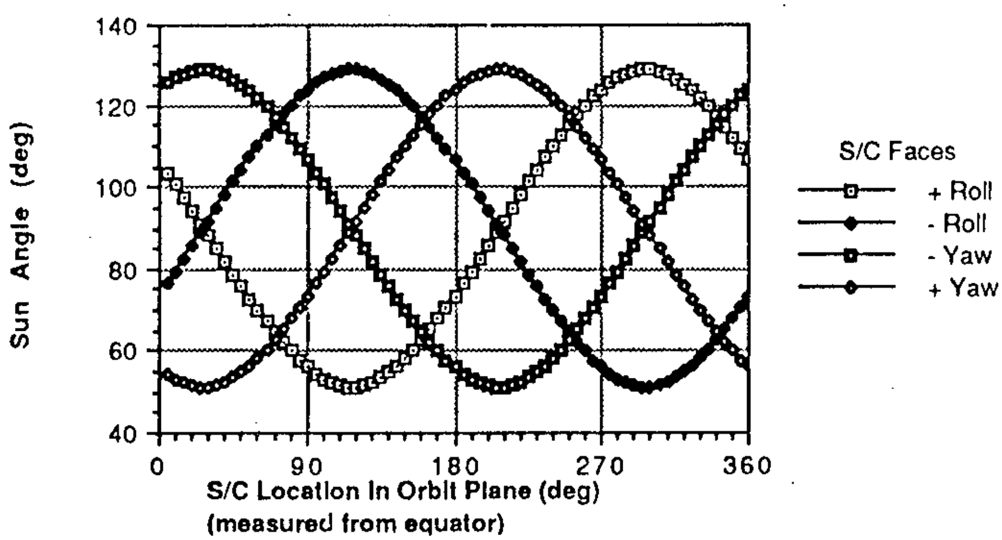

Figure 3. Sun Angles on Spacecraft Faces Vs. Orbital Position for First Day of Winter.

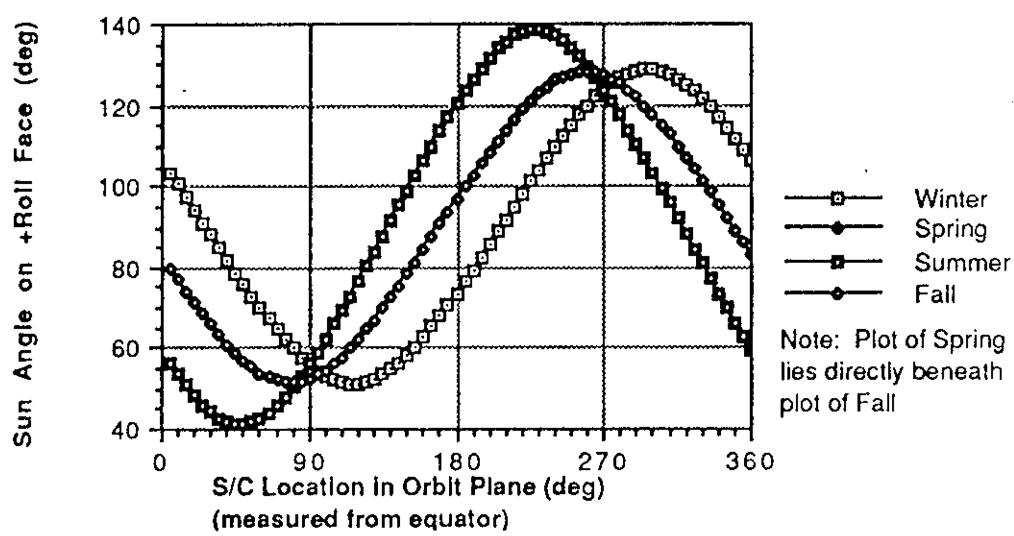

Figure 4. Sun Angles on Positive Roll Faces Vs. Orbital Position.

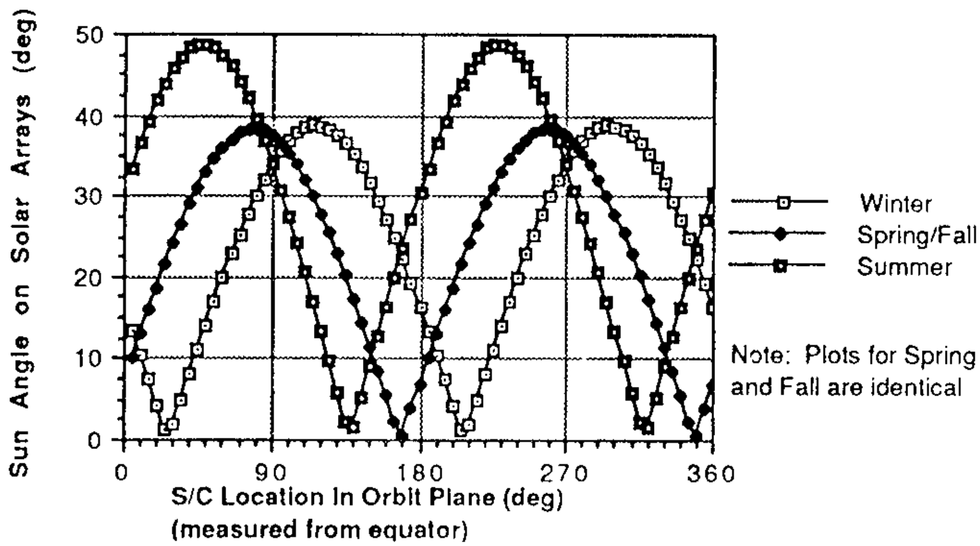

Figure 5. Solar Array Sun Angle Vs. Orbital Position and Season.

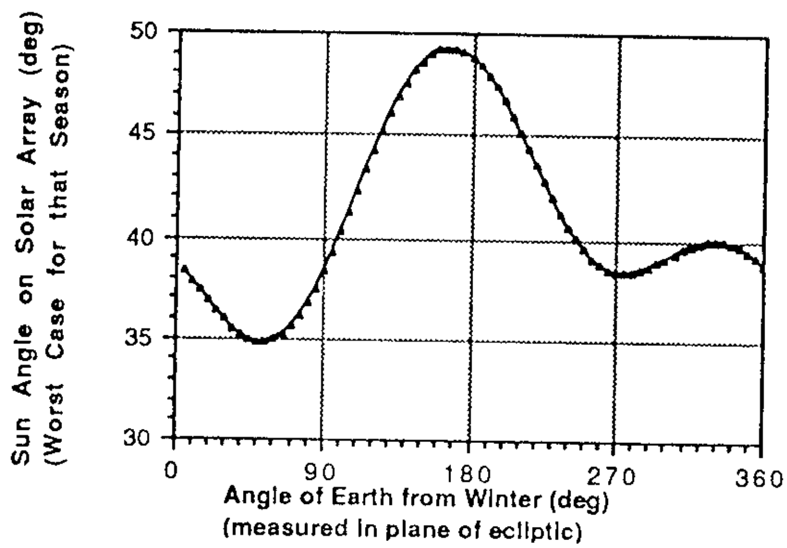

Figure 6. Worst Case Solar Array Sun Angle Vs. Time of Year.

can be always kept zero. For AVHRR mission, an analysis was performed to determine solar array rotation and sun angles. Figure 5 shows solar array rotation as a function of orbital position and season. Figure 6 shows worst case solar angle vs. time of the year. The maximum sun angle is 50°, resulting in reduction in effective solar array area by 36%.

For the EHF communications mission, the orbit inclination is 63.435° to prevent the line of apisode from changing. Perigee is located at the orbit's southern most point to give coverage in the northern hemisphere. Perturbation analysis was performed to determine changes in inclination and argument of perigee over the course of a satellite's life time. During the mission life time, perigee will move less than 2.5 degrees. The change in the inclination and the eccentricity are likewise very small during a satellite's life time. Therefore, station-keeping is not required for EHF communications mission. DMSP uses the same orbit as for the AVHRR and does not perform station-keeping. Therefore, station-keeping is also not planned for the AVHRR mission.

VII. Attitude Control Subsystem

The attitude control requirements for the two missions are significantly different. EHF communication allows the spacecraft to be continually rotated about yaw axis to keep the solar array surface normal to sun rays. To satisfy this requirement, three reaction wheel control system is used. One wheel is along each axis and a fourth skewed wheel is used to provide redundancy. The wheel desaturation is provided by magnetic torque rods. Attitude determination accuracy for AVHRR is 0.01 deg. in comparison to 0.5 deg. for EHF communication. The dual objective is met by using two sensor systems: the Basic Sensor System (BSS) and Precision Sensor System (PSS). The BSS obtains pitch and roll attitude errors from earth sensor assembly (ESA) and yaw error from the gyros with updates. The PSS system is similar to that used for Defense Meteorological Satellite Program. It uses three strapped down gyroscope to provide an inertial reference for measuring short term changes in attitude and a strap down star sensor for providing long term attitude updates to bound the effects of gyro drift.

An on-board data processor computes the desired satellite attitude using stored star data and ephemeris data, abbreviated star catalogs and ephemeris tables are periodically transmitted to the spacecraft from the ground and the comparison of the desired attitude with the sensed attitude yields error signals that are used for correcting the spacecraft attitude to within 0.01° (3 sigma). EHF communications and BSS payloads use BSS and PSS, respectively. Thrusters are used as back-up for wheel desaturation.

VIII. Electric Power

The electric power subsystem (EPS) consists of silicon cell solar arrays and Ni_2H_2 batteries. The bus will be fully regulated at 28V by employing a shunt regulator for the periods of solar array operations and will use a boost regulator during the periods of battery operation.

Solar Array

The spacecraft bus has two symmetric solar arrays with single axis rotation about roll axis. Each solar array consists of two panels. For EHF communications, the solar array surface is kept normal to sun rays by solar array rotation about its axis and spacecraft rotation about yaw axis. For AVHRR mission where yaw axis rotation is not allowed, the angle between the solar array surface normal and the sun rays reaches a maximum of 50° . This results in decrease in effective solar array surface by 36%. The solar cells are 2.5 cm X 6.2 cm K-7 silicon cells.

Radiation effects and shielding requirements were examined for the AVHRR circular orbit and EHF eight hour Molniya type orbit. The apogee of eight hour Molniya orbit extends into the Van Allen belts exposing the solar cells to large fluence and significant degradation. For AVHRR orbit, the level of fluence and degradation is significantly lower. Table 4 gives power degradation in solar cell performance for AVHRR and EHF missions for 3 years orbit period. It is shown that power degradation for EHF mission is 43% in comparison to 9% for AVHRR mission. Therefore, reduction in effective area for AVHRR mission is offset by reduced in-orbit power degradation.

Table 4 . Degradation of Solar Cells for 3 years Orbit Period (percentage)

	AVHRR	EHF
I_{sc}	3.0	33.0
V_{oc}	4.8	17.3
I_{mp}	3.1	28.0
V_{mp}	6.3	20.5
P_{max}	9.1	42.9

Batteries

The battery to provide power during eclipse is a 12 amp-hour $Ni_2 H_2$ battery manufactured by Eagle Pitcher. The battery consists of eight two-cell common pressure vessels (CPV). Dimensions of each CPV are approximately 8.89 cm (3.5 in) diameter by 15.2 cm (6 in) height. A $Ni_2 H_2$ battery was chosen because of the high number of charge/discharge cycles. For AVHRR payload, the battery will experience 1000 charge/discharge cycles. The charge rates for AVHRR and EHF are C/4 and C/10, respectively.

IX. Propulsion

The propulsion system consists of six hydrazine thrusters and one propellant tank. The thruster configuration is shown in Fig, 7. The system is used primarily as backup for reaction wheel desaturation, orbit maintenance, and orbit station-keeping. Therefore, it does not have redundancy. The thrusters are of 0.89-N (0.2 lbf) force made by Rocket Research (Model MR103C). The tanks are 0.41 m (16 in) diameter made of titanium alloy by TRW Pressure System Inc. An elastomeric diaphragm inside the tank separates the nitrogen gas pressurant from the propellant. Maximum capacity of the tank is 20 kg.

X. Telemetry , Tracking, and Command

The telemetry, tracking, and command (TT&C) package for the spacecraft bus is designed to be compatible with the Air Force Space Ground Link Subsystem (SGLS) for satellite control. TT&C is designed to operate at super-high frequencies (SHF) that corresponds to channel 1 of the SGLS ground terminal. The TT&C system uses payload antenna and/or the anti-earth face antenna. The anti-earth face antenna is a four element microstrip antenna and has a gain of 2.5 dB. Normally, during launch TT&C will use anti-earth face antenna and once on station, payload TT&C antenna will be switched to. The anti-earth face command receiver will remain active to provide a fail safe in case of satellite attitude control system failure.

XI. Structures

The spacecraft bus is designed to fit within the 1.17m (46 in) diameter Pegasus shroud. A rectangular design

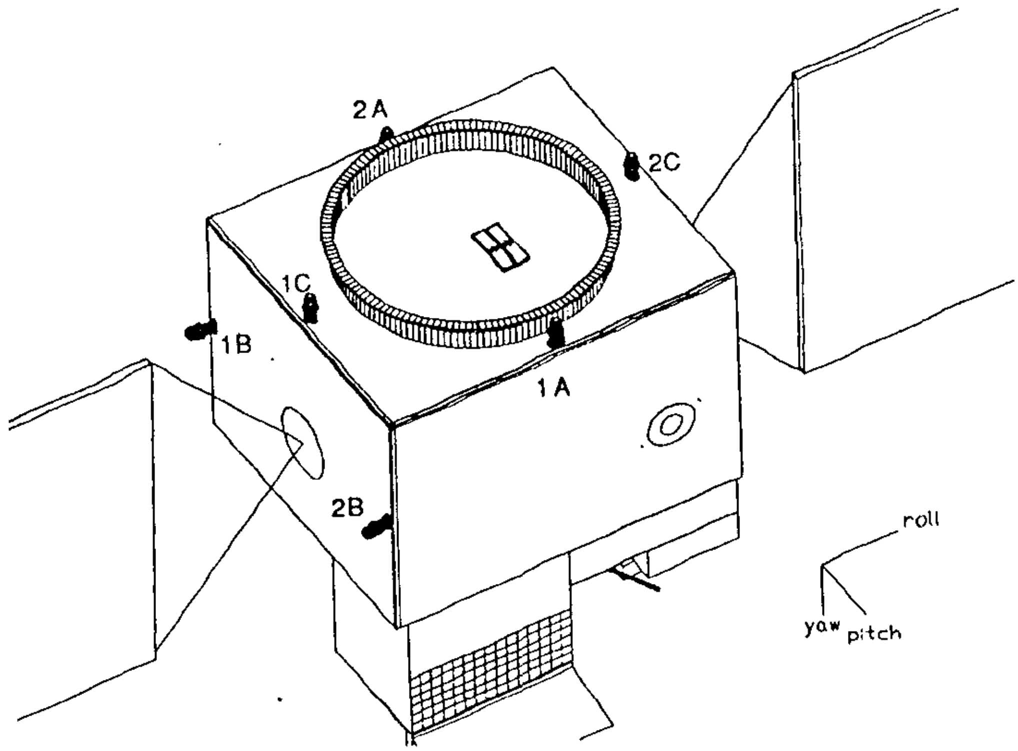

Figure 7. Location of Thrusters.

was chosen for simplicity and ease of assembly. The bus is built on a rectangular frame that is comprised of hollow rectangular cross-section tubing made of 6061-T6 aluminum. Five load supporting honeycomb panels with aluminum face skins are attached to the frame. The entire spacecraft is mounted to Pegasus with a standard Marmon clamp assembly.

XII. Thermal Control

The challenging thermal control task is for the infrared detector mercury-cadmium-telluride (HgCdTe) of AVHRR. It is cooled to a temperature of about 108°K and maintained with 0.1°K of the chosen set point by an active control loop using a heater and have the cone cooler uninterrupted view of space on positive pitch surface for AVHRR orbit.

For EHF communication mission, thermal control is achieved by using passive techniques. The elements of the thermal subsystem are multi-layer blankets, coating, optical solar reflectors (OSR), and heaters. Solar incidence on east and west surfaces is zero. Therefore these surfaces are used as thermal radiators by mounting high dissipating equipment and batteries on them and putting OSR for radiating heat. Other spacecraft bus surfaces are insulated by using multi-layer insulation. Preliminary thermal analysis, by using PC-ITAS software, indicated that the passive thermal control will be adequate for AVHRR mission also.

XIII. Conclusions

A challenge in the design of a multi-mission spacecraft bus is to provide flexibility in the design of subsystems to meet the requirements of different payloads without increasing complexity and cost for simpler payloads. In the present study, requirements of AVHRR payload for attitude control and thermal control subsystems are significantly more stringent. Two-axis solar array and precision sensor system would have introduced unnecessary complexity and cost to the EHF communications missions. Therefore, it was decided to have single-axis solar array for the spacecraft bus and different sensing system for the two missions. The study concludes that although the two missions have major differences in the requirements for several subsystems, the multi-mission spacecraft bus can be used for both missions with only minor modifications.

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