Copy no. 19

SPAR-R.677

FEASIBILITY STUDY OF A GENERAL PURPOSE SPACECRAFT BUS

VOLUME I
TECHNICAL REPORT

SPAR

SPAR-R.677

FEASIBILITY STUDY OF A GENERAL PURPOSE SPACECRAFT BUS

VOLUME I
TECHNICAL REPORT

Reference: Department of Supply and Services Contract PL36100-4-2010 Serial OPL4-0192

For Department of Communications

MAY, 1975

SPAR aerospace products ltd.
825 Caledonia Rd.
Toronto, Ontario. M6B 3X8
Canada.

SPAR

Telephone — (416) 781-1571 Telex — 02-2054 Sparcal Tor TWX — 610-491-1503 Cable — Sparcal Tor

SPAR AEROSPACE PRODUCTS LTD

SPAR-R.677
VOLUME I

(1)

FEASIBILITY STUDY OF A GENERAL PURPOSE SPACECRAFT BUS

TECHNICAL REPORT

Prepared by: Staff

Approved by: ....
for: P.A. McIntyre
Program Manager, CTS

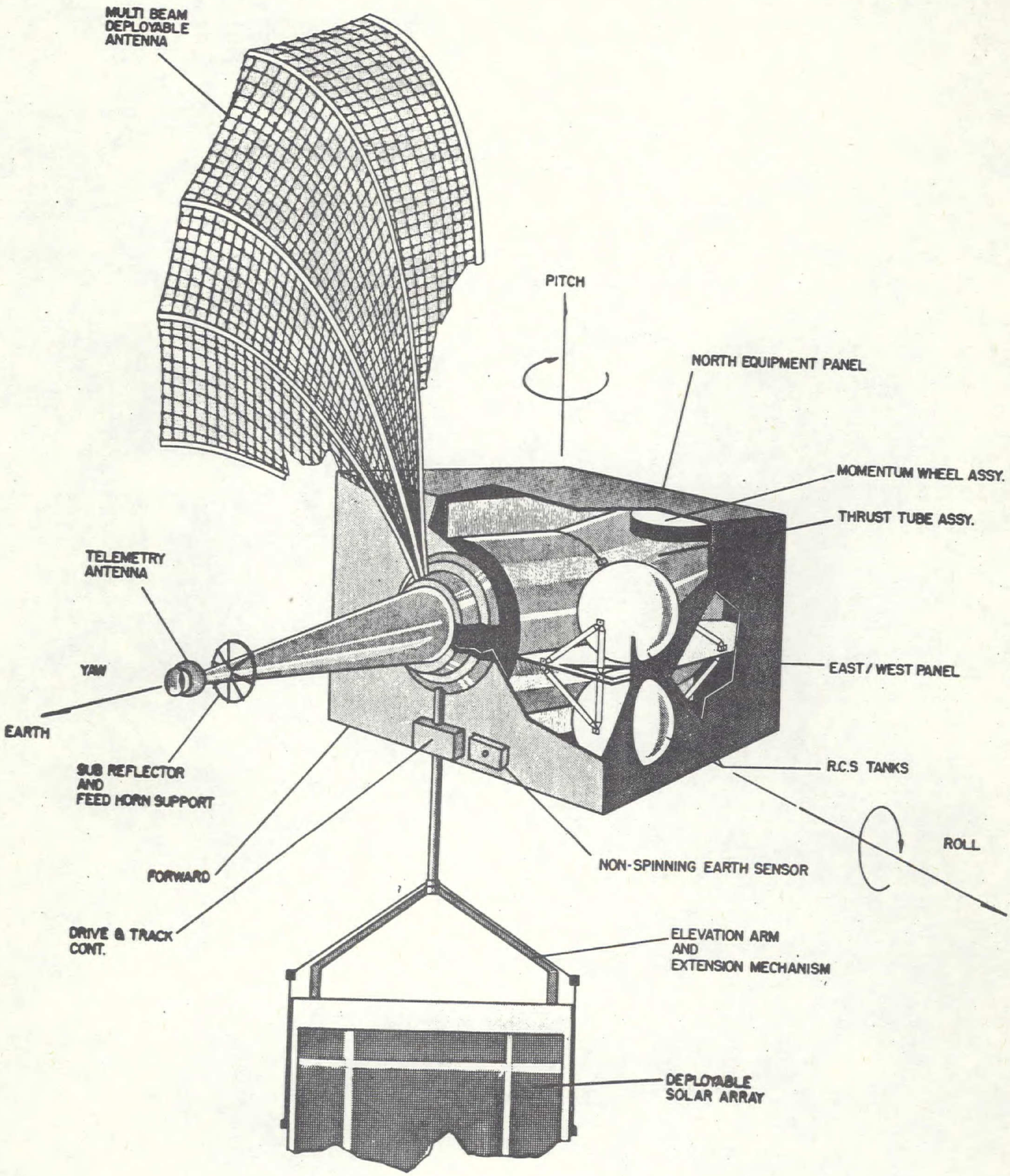
Approved by: Howard S. Kerr
H.S. Kerr
Technical Director

Approved by: ....
E.R. Grimshaw
Director,
Space Programs

Approved by: ....
pp J.E. Lockyer
Vice-President and
General Manager,
Engineering Division

CD/2

May, 1975



GENERAL PURPOSE BUS

PROPRIETARY MATERIAL

In the course of this study and in the preparation of this report, extensive use has been made of Spar and vendors' confidential background data and material. In order to protect the companies' commercial position, it is respectfully requested that the Government of Canada take this into consideration in the dissemination of this report.

SUMMARY

This report, prepared for the Department of Communications (DOC), Ottawa by Spar Aerospace Products Ltd., Toronto, is the result of a feasibility study carried out by Spar under contract to the Department of Supply and Services (DSS), Ottawa, to evaluate the feasibility of designing and fabricating a General Purpose Spacecraft Bus (GPB) capable of carrying five different communications and experimental payloads on a Thor Delta 3914 launch vehicle in the 1979-1985 time frame.

The object of a general purpose Bus is to provide a cost effective method of operating satellites that have the same general mission parameters such as orbit and launch vehicle and are of similar weight and power. The Bus, which is the basic "service module", consists of the spacecraft attitude control, power, telemetry and command, propulsion, structure and thermal control subsystems. If a Bus can be designed to cater for different payloads with little design changes needed for each payload, then considerable cost savings can be effected.

It has been shown, in this study, that such a bus design is feasible for the five payload options defined by the DOC in their S.O.W., and that a spacecraft with this bus and payload combination can be launched on a 3914 Thor Delta launch vehicle in the period 79-80. This report is submitted in four volumes:

Volume I	-	Technical Report
Volume II	-	Specifications and Responses from Vendors
Volume III	-	Implementation Plan and Costs
Volume IV	-	Bus Performance Comparison.

Volume I contains a technical description of the Bus and its subsystems, coupled with the design/analysis and tradeoffs that took place during the study phase for selecting the configurations and subsystem features chosen for the Bus.

In Volume II are found the subsystem specifications for the Bus that were derived from the overall spacecraft Bus requirements supplied by DOC. Some of these have been submitted to vendors for quotation purposes and their responses are available on request.

Volume III discusses and proposes an implementation plan and schedule that could be adopted to deliver a spacecraft for launch in late 1979, assuming a start date of mid-1976.

Volume IV contains a review of the GP Bus design with existing satellite arrangements, and highlights the improved performance achievable with the use of this Bus when comparing parameters such as payload allowable versus power available for both normal operation and eclipse. In addition, this volume discusses, in more detail, further improvements that can be made to subsystem weights and overall lift off weight; it also discusses the present Canadian capability for performing the tasks identified in the Implementation Plan (Volume III).

The Bus configuration studied has considerable advantages over other satellites built in that it has been designed to meet all known future Canadian satellites synchronous orbit payload requirements, and has incorporated features noted from other DOC subsystem and component studies (Sections 5.1 and 5.3) in coming up with the General Purpose arrangement contained in this report. Use has been made of the know-how developed by Spar during the CTS program, and includes the following technology improvements which will be demonstrated and available for a 1979 launch:

- Higher Isp RCS thrusters,
- Improved efficiency solar cell performance,
- Reduced natural frequency requirements for the solar array,
- Hybrid system ACS with magnetic bearing system.

The resultant baseline Bus described herein does have growth areas which can be accomplished at reasonable cost. Some of the more obvious of these are identified in Volume IV.

7/CD/30

ACKNOWLEDGMENTS

The management and staff of Spar wish to express their thanks and appreciation to the personnel from companies listed below who provided support, criticism, and information necessary for us to complete this Bus study in a fairly short time frame:

- Canadian Astro
- Communications Research Centre
- Department of Communications
- Grumman Aircraft Corporation
- Hamilton Standard
- Hughes Aircraft Company
- Telesat
- Thiokol
- University of Toronto Institute for Aerospace Studies

TABLE OF CONTENTS

CD/126

TABLE OF CONTENTS - (Cont'd)

7/CD/127

TABLE OF CONTENTS - Cont'd

LIST OF FIGURES

7/CD/132

LIST OF TABLES

7/CD/133

LIST OF TABLES - (Cont'd) Page

7/CD/134

1.0 INTRODUCTION

Information contained in this study report relates to a three-axis stabilized synchronous orbit satellite Bus capable of carrying various communications transponder payloads specified by DOC.

Typically, these transponders are:

- UHF + 12 channel 4-6 GHz
- UHF + 4 channel 12-14 GHz
- UHF + 7-8 GHz + Experiments
- UHF + 7-8 GHz + L Band
- 24 channel 4-6 GHz
- Other Telesat derivations, i) 8 channel 12-14 GHz; ii) 4 channel 12-14 plus 12 channel 4-6 GHz.

The payload power requirements for the spacecraft have been established from dissipations provided by the Department of Communications (DOC) in their SOW and attachments A and B. It is in the order of 800 watts at end of life, and is compatible with the spacecraft life requirements of six years.

The Bus and its subsystems have been designed to be launched on a Thor Delta 3914 launch vehicle, and for environments associated with a synchronous orbit mission. This mission on a Thor Delta initially places the satellite in transfer orbit, which is followed by geosynchronous orbit injection using an apogee kick motor.

Initially, various configurations of spacecraft were examined; these included:

- Conventional three-axis stabilized spacecraft earth oriented with the antenna mounted on the forward face.
- Three-axis stabilized spacecraft with the antenna mounted on the east or west face and earth facing.

7/CD/33

- Three-axis stabilized spacecraft sun oriented with despun antenna locked on to earth.

Detail information on the above is contained in the text along with detail work done on the Bus subsystems whose main features are listed below.

1.1 Attitude Control Subsystem

Incorporates the established spin stabilized technique for transfer and synchronous drift orbit using sun and earth sensor signalling for attitude data. These sensors also provide spacecraft attitude data for apogee motor firing to affect synchronous orbit injection. In synchronous orbit, the spacecraft is spun down and a 3-axis control system using earth sensing and a momentum wheel system is adapted for the main stability mode.

1.2 Reaction Control System

Consisting of high level thrusters for precession manoeuvres during the transfer and synchronous orbit phase, with low level thrusters for momentum dumping and spacecraft orientation during the 3-axis synchronous orbit phase of the mission.

1.3 Solar Array Subsystem

Consisting of a low power solar cell array providing the required spacecraft electrical power during the spin (transfer/drift orbit) phase, and supplemented with batteries for eclipse operation. This spin phase array is part of a high power deployable solar array providing all required electrical power during the sunlit periods in synchronous orbit with, again, batteries providing power during eclipse operation.

1.4 Thermal Control Subsystem

Consisting of passive thermal control materials supplemented with thermostatically controlled electric resistance heaters (having ground command override) and with a heat pipe system, for maintaining required component temperatures during all mission phases.

1.5 Apogee Motor

Provides sufficient thrust to achieve the required velocity change to inject the satellite into a geosynchronous orbit.

1.6 Structure Subsystem

Capable of carrying the complete spacecraft through the launch environment of the Thor Delta 3914 boost vehicle and into transfer and synchronous orbit. This structure will be designed and qualified to withstand an additional 100 lbs. of payload, thus catering for any launch vehicle capability increase which may occur.

1.7 Telemetry and Command Subsystem and Power Control Subsystem

These subsystems usually form part of the Bus; however, as agreed with DOC the work on these are being covered separately. It is assumed that they will provide:

a) continuous communications with ground stations in the required frequency band for all phases of the mission,

b) regulation and control of solar array and battery electrical power.

Also included in this section of the study are Launch Vehicle/Mission Analysis details and a Mass Properties Analysis. The former deals with spacecraft V's for RCS Sizing over the operating years 1979 to 1985, and the latter is associated with ensuring:

- that a favourable moment of inertia is obtained by the Bus, i.e., greater than 1.1, and;

- that the weight margins associated with a lift off weight of 1,925 are identified using transponder battery and telemetry weights provided by DOC.

2.0 MISSION/LAUNCH VEHICLE

The objective of the General Purpose Bus Satellite program will be to place a number of satellites with different payload configurations, in nominally geosynchronous orbits, stationed over Canada. The general guidelines for the mission are listed below:

a) Orbit Inclination

The satellite design will be such that an existing network of fixed narrow beam ground antennas may be employed for 4-6 and 12-14 GHz transmission. To accomplish this, inclination will be controlled to better than $\pm$ 0.05°.

b) East/West Errors

To accommodate the fixed ground antennas, east/west error will not exceed $\pm$ 0.05°.

c) Satellite Location

The coverage requirements will be met for satellite locations having a longitude between 90°W and 120°W.

d) Repositioning in Orbit

Each satellite will be capable of being transferred from an initial position in orbit to any other position in order to provide a back-up communications coverage in the event of failure of a prime satellite.

e) Service Duration and Satellite Lifetime

The service duration of each satellite will be not less than six years with a design lifetime objective for each satellite of eight years.

R677/71

f) Launch Period

The satellites will be designed for launch between 1979 and 1985. Since orbit inclination perturbations are minimum in the earlier portion of the launch period, off-loading of orbit adjust fuel will allow increased payloads for launches during this earlier period.

g) Launch Window

The launch window will be at least 1/2 hour, any day of the year.

h) Stationkeeping Cycle

The stationkeeping operation will be based on a 21 day cycle.

2.1 Mission Phases

The mission may be divided into the following distinct operational phases:

a) Launch
b) Transfer Orbit
c) Apogee Injection
d) Attitude Acquisition
e) Station Acquisition
f) On-Orbit Operation

Each of these phases are described in detail in the following sections. Calculations of ΔV requirements are presented in Appendix A.

2.1.1 Launch Phase

2.1.1.1 Launch Vehicle

The General Purpose Bus Satellite will be launched by a Thor Delta launch vehicle designated Delta 3914. The Delta 3914 is a three-stage launch vehicle with the first two stages being inertially guided, whereas the third stage is spin stabilized prior to separation from the second stage. The payload delivered into a synchronous transfer orbit will be a maximum for a transfer orbit inclination of 28.3 deg. (approximately) with due east launch from ETR.

2.1.1.2 Launch Profile

The mission starts with a lift-off at a time compatible with the launch window constraints. The first stage burn and a partial burn of the second stage will place the spacecraft into a 28.3° inclined 100 nm circular parking orbit 8.9 minutes after lift-off. After a 15.5 minute coast in the parking orbit, the second stage is restarted. The second and third stage burns inject the spacecraft into an inclined synchronous transfer orbit 25.8 minutes after lift-off. The launch phase guidance, timing of events and T&C planning is the responsibility of the launch authority.

2.1.1.3 Launch Environment

The launch environment defines the spacecraft vibration, acceleration and accoustical design constraints. The Delta Critical Flight Environment is summarized in the General Purpose Bus Specification, Volume II of this report.

The allowable spin-up of third stage is a function of the spacecraft spin axis (on-orbit yaw axis) moment of inertia. Torque to the spin table is imparted by combinations of small solid propellant rocket motors, which provide spin rates from 30 to 100 rpm for spacecraft spin axis moments of inertia ranging from 20 to 170 slug ft^2. The lower limit of 30 rpm is dictated by minimum dynamic stability of the third stage/spacecraft combination during third stage motor burning. The upper limit of 100 rpm arises since the third stage motor is qualified only up to this spin rate. A nominal spin rate of 60 rpm at separation from the third stage will be assumed.

2.1.1.4 Launch Window

The lift-off time is constrained due to several considerations. The potential constraints that may arise in the mission are listed below:

(a) The angle between spin axis and sun vector must be between 60 degrees and 120 degrees for the perigee injection and apogee firing attitudes. This constraint arises due to spacecraft thermal and power design limits. Further, it ensures that the sun is within the field-of-view of the 64° FOV spinning sun sensor used for the attitude determination.

(b) Spacecraft thermal and power design limits impose a constraint on eclipse times in transfer orbit.

(c) The separation angle between the spacecraft-sun line of sight (LOS) and the spacecraft-earth LOS at apogee must be close to 90°. This constraint is necessary only if the attitude in the transfer orbit for apogee motor firing attitude (AMFA) is determined using the spinning earth sensor and sun sensor data obtained in the neighbourhood of the apogee. This constraint is relaxed if orbit information is used in conjunction with earth and sun sensor data obtained during portions of the orbit which are not in the vicinity of the apogee.

The launch window calculations are presented in Appendix A. Provided a midnight launch is chosen, (see Figure A-9) the window is always greater than 1/2 hour for launch any day of the year.

2.1.2 Transfer Orbit

The launch vehicle will inject the spacecraft spinning at 60 rpm into an inclined synchronous attitude transfer orbit. The dispersion assumed for the 3914 launch vehicle with the General Purpose Bus Satellite are:

Apogee Height	$\pm$ 300 n.mi.
Perigee Height	$\mp$ 3.3 n.mi.
Inclination	$\mp$ 0.25 deg.

The spacecraft telemetry and orbit determination will be initiated in the transfer orbit as soon as ground station visibility permits. The spacecraft will be reoriented into an approximate apogee motor firing attitude two hours after injection and attitude determination based on the spinning sun and earth sensor data will be started. To minimize the reaction control fuel required for injection error correction and for final station acquisition, the apogee motor can be reaimed on the basis of orbit determination results to compensate for the launch vehicle dispersions. At least two spin axis precessions will be performed with continuous attitude determination to obtain precise apogee motor firing attitude. There is no need to keep the spacecraft in an intermediate attitude prior to AMFA to satisfy thermal, power and attitude sensing requirements since these are satisfied by the proper selection of launch window.

2.1.2.1 Spacecraft Parameters

The nominal transfer orbit spacecraft parameters used for this study are:

Weight of Spacecraft	1925 lb
Spin Rate	60 rpm

2.1.2.2 Transfer Orbit Dispersions and Orbit Determination

The 99% dispersions in the transfer orbit parameters were listed earlier. It is assumed that extensive use of the NASA tracking stations will be made and for this purpose, a compatible S-band ranging transponder will be carried onboard the spacecraft. Within about three hours from transfer orbit injection, the spacecraft position can be determined within 1 to 3 nautical miles (nm). Nearly continuous orbit determination up to second apogee will result in estimated 3 S uncertainties as follows:

- Perigee Altitude ± 0.45 nm
- Apogee Altitude ± 0.54 nm
- Apogee Velocity ± 0.9 fps
- Orbital Inclination ± 0.003 deg.

The spacecraft will be precessed to the AMFA prior to the second apogee. Past the second apogee, the orbit will be monitored with a limited amount of orbit determination, and attitude touch up manoeuvre performed as required.

2.1.2.3 Attitude Determination

The sensor complement used for spin axis attitude determination during transfer orbit will be composed of a pair of redundant 64° FOV spinning sun sensors (SSA) and a pair of spinning earth sensors (SES). The sensors will provide the following data to the Transfer Orbit Electronics (TOE) for processing and subsequent telemetry to the ground:

- Sun reference pulse.
- Sun elevation in spacecraft coordinates.
- Lead and trailing earth edge pulses from each SES.

After processing the data, the TOE will carry out the following:

- When a train of high thrust engine firing pulses are commanded, the sun reference pulse will be used to reference the individual pulses.

- The sun reference pulse will be transmitted to ground for spin rate monitoring.

- Sun-line elevation in spacecraft coordinates will be telemetered to ground.

- Chord lengths of the earth obtained from each SES will be telemetered to the ground.

R677/76

The TOE will be a subunit of the Attitude Control Electronics Assembly. With the spinning sensors and TOE, if proper separation of sun - S/C LOS and Earth - LOS is ensured by launch window constraint at apogee, the attitude of the S/C can be determined to the following levels of accuracy, by suitable processing of data obtained in the vicinity of the apogee.

Error in Pitch	0.1 degs.
Error in Yaw	0.225 degs.

If the LOS separation constraint on the launch window is too severe, the constraint may be eliminated by determing the spin axis attitude with redundant data sets and proper smoothing.

The sun sensor data will be available at all times (with the exception of eclipse periods) since appropriate geometry will be ensured by launch window constraints. The earth sensor data will be available in the vicinity of perigee and apogee. The minimum data sets that are sufficient for attitude determination are:

- Sun sensor measurements with measurements from a single SES and orbit determination results.
- Measurements from a single earth sensor, separated in time, plus orbit determination results.
- Sun sensor, and measurements from both earth sensors.

2.1.2.4 Spin Axis Reorientation Manoeuvre

A limited calibration of the hydrazine thrusters, valve operation and the response of the nutation damper will be carried out prior to the spin axis precession. The preliminary tests will be finished no later than T + 2 hours and additional calibration of axial thrusters will be obtained during spin axis precession manoeuvres.

The spin axis orientation into apogee motor firing attitude (AMFA) will be initiated at T + 2 hours, after reaction control system tests are completed. The manoeuvre will be computed using the nominal perigee injection orientation as the initial conditions for the attitude and the spinning mode sun sensor as an inertial and timing reference. During the manoeuvre, the spin axis sun-line angle variation will be monitored against a precomputed time history to preclude any false precessions. Should larger than nominal dispersions be observed, the manoeuvre may be deferred until the earth is visible with the spinning earth sensor and a more detailed monitoring is possible. The attained reorientation accuracy will depend on the following:

- Dispersions in the initial attitude at the start of the manoeuvre due to launch vehicle third stage attitude and payload separation dispersions (about 2 degrees 3).

- Accuracy of the axial thruster operation.

- Thrust tailoff leading to a decrease in the magnitude of the planned pulses (5 to 10 percent).

- Dispersions in thruster duty cycle timings, i.e., in the start and cut-off times, resulting in variations in the centroid of the pulses. Angular dispersions of ± 15 deg. for pulse sequences of 50 pulses or less, and ± 1 deg. for pulse sequences of 100 pulses or more are typical.

The thruster performance dispersions will primarily decrease the magnitude of the angle through which the spin axis is precessed, whereas the precession direction, particularly by careful monitoring of the sun angle, will be maintained relatively accurately. At the end of the manoeuvre, the earth will be within the field-of-view of the spinning earth sensor.

At T + 3 hours preliminary orbit determination results will be available and a preliminary AMFA to compensate for the launch vehicle injection errors will be computed. Based on the attitude determination results, a sequence of AMFA corrections will be carried out. Allowing 30 minutes for collecting attitude sensor data and 30 minutes for attitude determination computations, evaluation of the thruster actual performance, thruster firing computations and manoeuvre execution corrections one hour apart are feasible. A final AMFA accuracy of 0.225 deg. can easily be attained one hour prior to the second apogee.

A summary of the precession manoeuvre is given below:

i) All precession manoeuvres performed with the 5 lb axial thruster in a pulsed node with a duty cycle of 45 deg. (for 60 rpm, on-time = 125 msec).

ii) The precession angle from the perigee attitude to the AMFA is about 129.5°.

iii) Manoeuvre rate 0.15 deg/sec.

iv) Precession per pulse 0.15 deg.

Smaller values of precession angle/pulse can be obtained for "touch-up" precession using a duty cycle of 18 deg. (for 60 rpm on-time = 50 msec, precession angle/pulse 0.06 deg).

2.1.3 Apogee Injection

The following factors have been taken into consideration in planning the nominal apogee injection manoeuvre.

i) The duration of the station acquisition phase will not exceed 15 days.

R677/79

ii) The payload delivered into the synchronous orbit must be maximized.

iii) The RCS fuel requirement for station acquisition must be minimized.

An apogee motor firing attitude (AMFA) to minimize the dispersions in the final orbit inclination due to launch vehicle injection errors will be used.

2.1.4 Attitude Acquisition

The acquisition phase starts approximately one day after apogee motor firing. The spacecraft will be spinning at 60 rpm about the yaw axis. A sequence of manoeuvres will be performed during the acquisition phase to despin the spacecraft and orient it such that it is in a three-axis stabilized state from which the on-orbit controller may take over. The attitude acquisition sequence will be carried out by the on-board attitude acquisition controller. The only information required from the ground will be a command to initiate the sequence. The sequence may be initiated at any point in the orbit and is completely compatible with an automated initiation of a reacquisition in the event of a failure which causes the earth image to leave the non-spinning earth sensor FOV during on-orbit operation. The acquisition sequence will use wide angle FOV sun sensors which provide complete 4π steradian coverage.

The pitch, roll and yaw thrusters will be used for control along the momentum wheel with its angular momentum vector pointing along the negative pitch axis. The acquisition method does not rely on rate gyros.

2.1.5 On-Orbit Operation

The nominal on-orbit operations are as follows:

a) East/West Stationkeeping

b) North/South Stationkeeping
c) On-Orbit Three-Axis Attitude Control
d) Momentum Dumping

In addition, orbit determination is necessary throughout the mission to provide information for stationkeeping and spacecraft pointing decisions.

2.1.5.1 East/West Stationkeeping

After acquiring station, the satellite is required to maintain station to within $\pm$ 0.05 deg. The significant perturbations that affect the E/W station of the satellite are:

(a) The triaxiality of earth causes an east/west drift. The fuel required to control this drift is a function of the longitude.

(b) The solar radiation pressure changes the orbit eccentricity and longitude of perigee. The eccentricity leads to a daily longitudinal oscillation.

The solar radiation pressure effect dominates the triaxiality drift. With the chosen correction procedure (see Appendix A), drift correction is obtained as a side effect of solar pressure correction.

2.1.5.2 North/South Stationkeeping

While on station, the satellite is required to maintain an orbital inclination of + 0.05 deg. As shown in Appendix A, a launch in 1975 would require a ΔV of 829 ft/sec for six years whereas launch in 1979 or 1985 will require a ΔV of 884 and 999 ft/sec, respectively. Thus, for earlier launches, off-loading propellant will provide additional payload capability.

2.1.5.3 On-Orbit Attitude Control

The satellite will be three-axis stabilized during its on-orbit operation. The on-orbit attitude

R677/81

controller uses a biased momentum wheel (with its angular momentum vector point north) and a set of thrusters as actuators. The roll and pitch errors will be sensed by the non-spinning earth sensor assembly. No yaw sensing is required except during orbit adjust manoeuvres. The pitch control is carried out by varying the speed of the momentum wheel. The roll and yaw errors are controlled by using the roll information to fire a set of offset thrusters which provide a torque about an axis subtending a small angle with the roll axis, in the roll-yaw plane. The coupling between the roll and yaw axis, due to the presence of the wheel angular momentum, enables the control of yaw with no yaw sensing. A nutation damping scheme included in the roll-yaw controller prevents nutation build-up and consequent limit cycle operation.

At the end of the acquisition sequence, the on-orbit controller will be activated. The on-orbit controller will null the existing attitude errors and will operate in the earth pointing mode. The on-orbit controller will maintain the yaw axis pointed to the subsatellite point to ± 0.15° accuracy in pitch and roll and ± 1° accuracy in yaw, against all disturbance torques. The fuel required for this function depends on the disturbance torques.

2.1.5.4 Momentum Dumping

The angular momentum imparted to the satellite in pitch, by external disturbance torques, will be stored in the momentum wheel, leading to either a decrease or increase of the speed of the wheel from its nominal value. The wheel speed will be maintained to within ± 10% of its nominal value by periodically dumping momentum using the pitch thrusters. The fuel required for momentum dumping will depend on the pitch disturbance torques. The dumping operation will be carried out autonomously on-board.

3.0 GENERAL PURPOSE BUS

Following an initial review of various spacecraft configurations for the General Purpose Bus, a short list of three possible emerged for further evaluation. They were:

I Conventional configuration (see spacecraft arrangement Figure 3-1) which is a three-axis stabilized spacecraft with the antenna mounted on the forward face and earth facing.

II Spacecraft arrangement Figure 3-2 which is similar to the configuration above. It has the antenna on the spacecraft east or west face and earth pointing.

III Spacecraft arrangement Figure 3-3 which is also three-axis stabilized. It is a sun oriented spacecraft with fixed solar arrays and a despun antenna which is earth pointing.

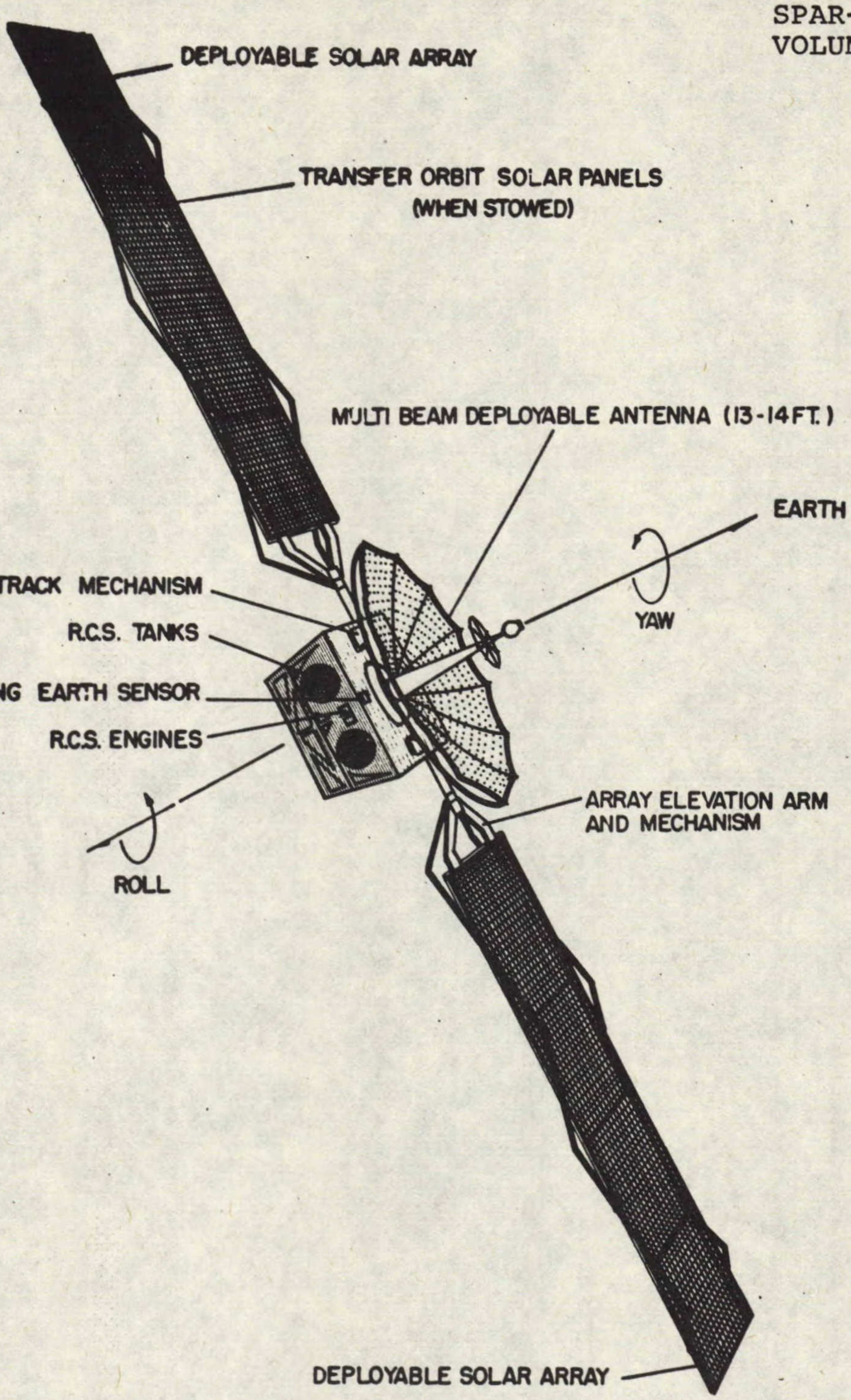
These arrangements and their descriptions, including important features, are discussed in the text that follows. However, it should be stated here that having reviewed the pros and cons, the conventional configuration I was selected for the study and all work in this study was concentrated on this configuration, in order to evaluate, in the allotted timeframe the feasibility of a General Purpose Bus. Although the design study revealed the selected configuration to be extremely competitive from weight, power and performance considerations, it is a recommendation of this report that further evaluation of the two alternative configurations described below should take place at an opportune time in order to ensure the optimum configuration is selected for the General Purpose Bus.

3.1 Spacecraft Configurations Investigated

3.1.1 Conventional Arrangement (See Figure 3-1)

The major features associated with this configuration are identified in the attached sketch and are:

- The multi-beam communications antenna and TT&C bi-cone antenna are mounted off the forward platform which is earth facing.



SPACE CRAFT ARRANGEMENT I. IN SYNCHRONOUS ORBIT

FIGURE 3-1

- The antenna arrangement will need to make allowances for cut outs in its configuration to allow the Attitude Control Subsystem (ACS) non-spinning earth sensor to view earth.

- Other components of the ACS used during the three-axis stabilized phase are mounted on the north/south panel while sensing components for the spin stabilized phase, are mounted from the forward platform and around the 'Z' (spinning) axis of the spacecraft.

- Communications transponder high dissipating components such as TWT's and power amplifiers are also mounted on the N/S radiating panels.

- The deployed solar arrays, capable of supplying over 800 watts at EOL, are stowed on the N/S panels during the spin phase when part of the array is providing power for this phase of the mission. In the deployed state, the array is attached to the drive and tracking units mounted on the forward platform.

- Conventional structure/thrust tube arrangement, consisting of a forward, apogee motor mount and separation rings, with the apogee motor mounted at its interface half way up the thrust tube, and the separation ring defined by the interface requirements of the 3137A adaptor.

- Acquisition sequence in synchronous orbit will adopt the CTS procedure which involves spinning down to less than 2 rpm, activating three-axis control, acquiring the sun, deploying the arrays, and then locking on to earth.

- RCS system with tanks mounted off the thrust tube in E/W direction and high and low level thrusters mounted off the structure and suitably placed for N/S and E/W station-keeping, and momentum dumping.

- The arrangement can achieve a favourable moment of intertia during the spin phase; however, the following areas are sensitive and need to be monitored:

3/CD/42

a) Apogee motor location,
b) Multi-beam antenna, C.G. and weight,
c) RCS tank and fuel,
d) Deployed solar array size/location.

The only major disadvantage with this arrangement is the complex acquisition sequence associated with spinning down from a spinner configuration, and precessing to acquire sun and earth.

3.1.2 Spacecraft Configuration II (See Figure 3-2)

This configuration is basically the same as the conventional arrangement in terms of the Bus subsystems, the main difference being the antenna which is mounted facing east in terms of satellite orientation and thus provides the following improved features:

- Satellite is always locked on to earth, both in the spin and 3-axis phase of the mission since both types of earth sensors are mounted off the east panel.

- Simpler acquisition sequence from spin phase to 3-axis since lock on to earth can occur immediately after spin down to 2 rpm and no re-orientation of the spacecraft is required.

- Acquisition and array deployment can occur immediately after injection into synchronous orbit.

- Utilises the length of the fairing for earth pointing systems, i.e. antenna, non-spinning earth sensor and other experiments.

- Can accommodate both quad helix design antenna and a 5 ft. parabolic dish (ANIK antenna).

- Drive and tracking unit is mounted on west panel with arrays stowed on N/S panel. Similar to conventional configuration.

However, with all these features its main disadvantage is in achieving a favourable MoI ratio with the larger parabolic antenna. To achieve a favourable MoI ratio one would have to use the stowed "Donut" configuration produced by Lockheed with a deployed subreflector. Alternatively, one would have to consider flying unfavourable during the transfer orbit phase of the mission.

SPACE CRAFT ARRANGEMENT II. IN SYNCHRONOUS ORBIT

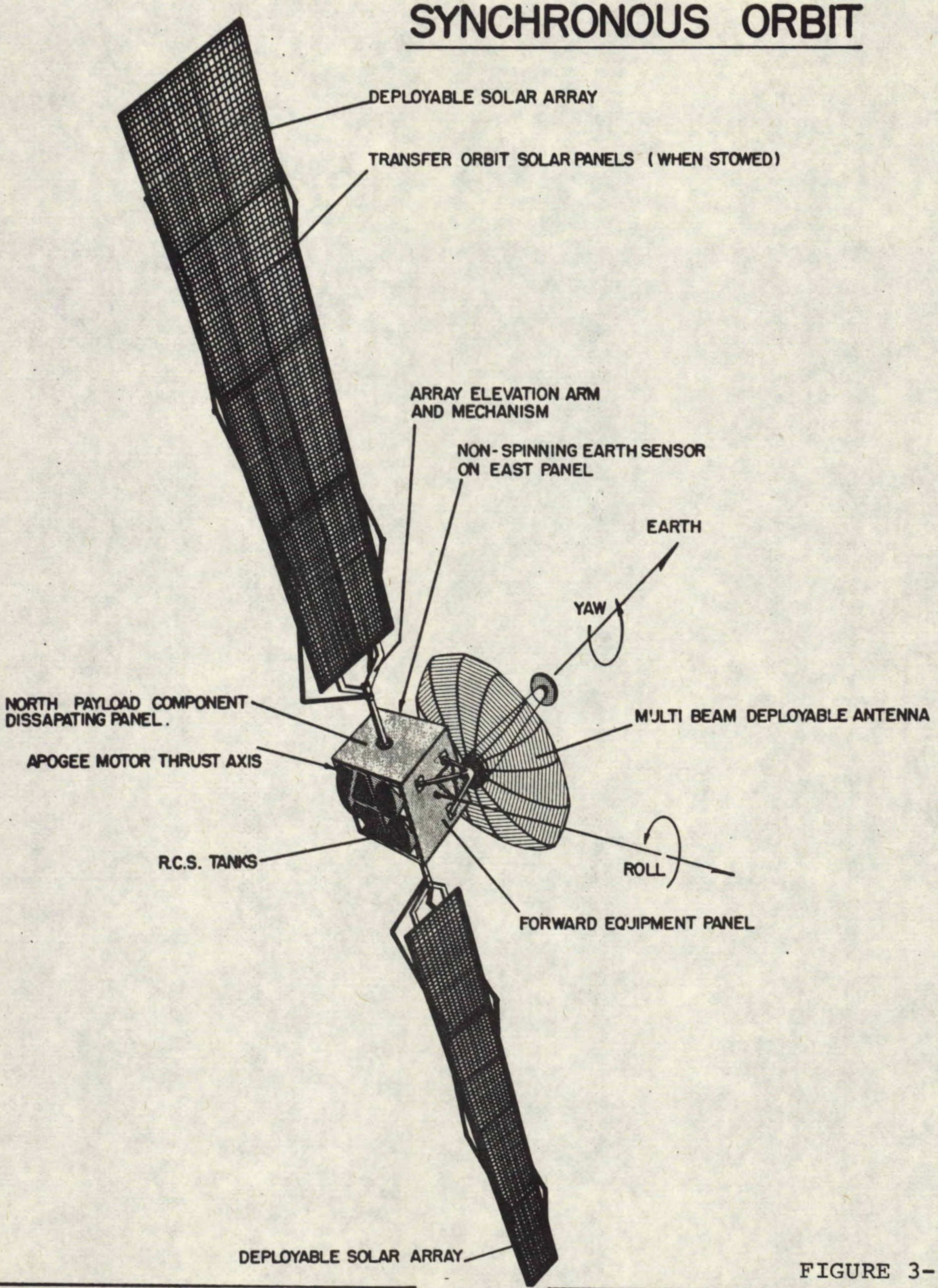


FIGURE 3-2

It should be noted that no rotating joints for the RF waveguide are required with this configuration.

3.1.3 Spacecraft Configuration III (See Figure 3-3)

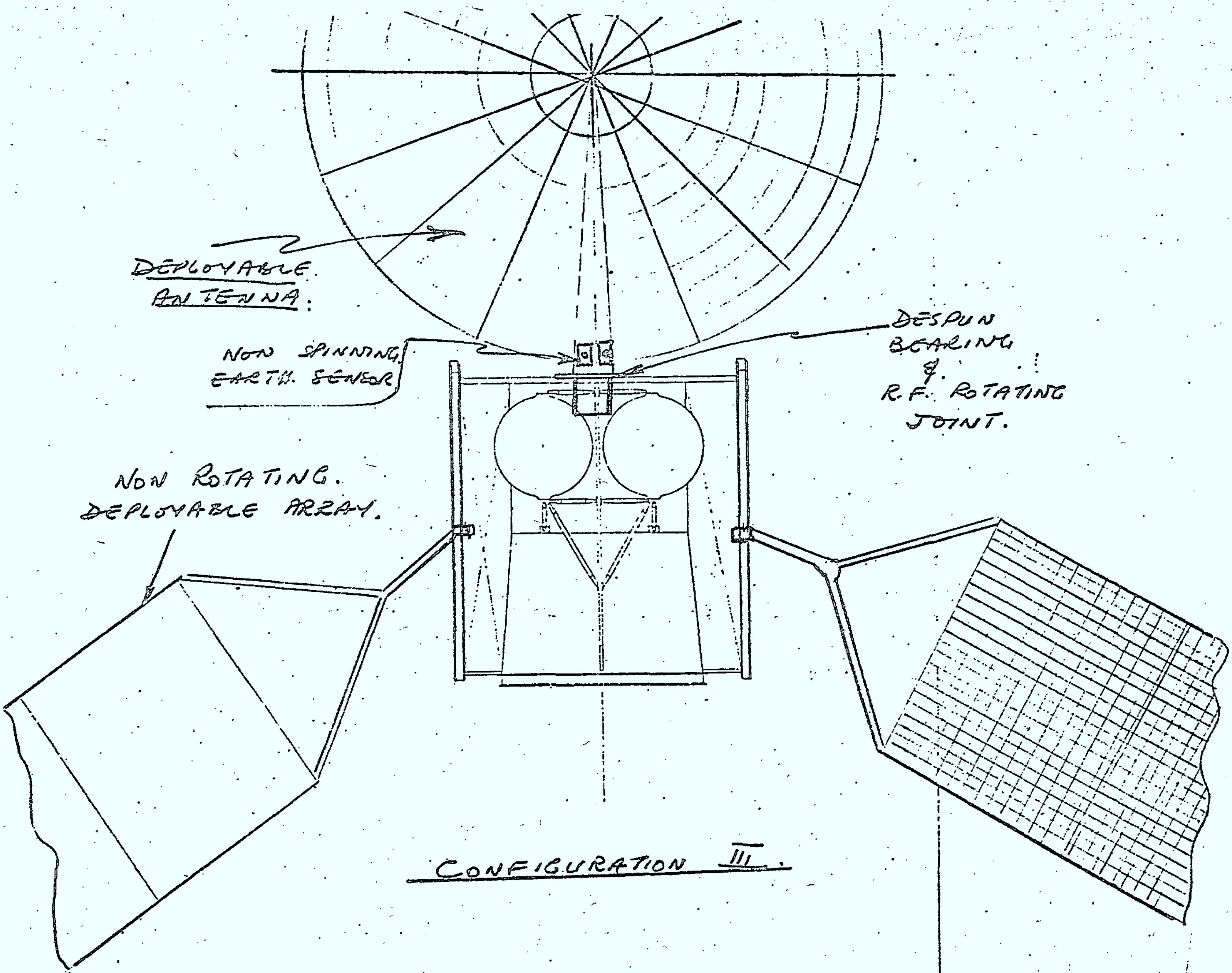
This arrangement was considered as a good candidate initially since, as a sun sensing satellite with a despun antenna mounted from the forward face of the spacecraft which is earth pointing, numerous advantages could be seen with this configuration. To summarize, they are:

- No shadowing of solar panels by communications antennas even with no separation of solar panels from spacecraft equipment platforms.
- Attitude control subsystem improved since
  - a) no flexible joints; also, improved stiffness,
  - b) solar torques reduced.
- Improvement to thermal subsystem since reduced temperature variations on spacecraft; also, additional radiating area available for dissipating components.
- Simplified acquisition sequence since the spin axis and attitude of spacecraft does not change significantly from spin to 3-axis stabilized configuration.

However, again there are areas which cause problems and, consequently, eliminated this configuration. They are:

- Because the antenna is earth pointing one would need a despun bearing with a rotating RF joint and a complex co-ax/signal cable down the middle of the RF joint to provide earth position sensing, and UHF signals. Also, a despun antenna system designed for 1 revolution per day continuous operation over six years is considered a high risk.

CD/44



CONFIGURATION III.

FIGURE 3-3

- Difficult to achieve a favourable MoI ratio. However, with a despun system built in, one could fly unfavourable during transfer orbit using the "gyrostat" principal successfully demonsrated by Hughes, (dual spin spacecraft).

3.2 Description of the General Purpose Bus

The basic elements of the Bus are the space frame and the major subsystems integrated with it (See Figures 3-4, 3-5, 3-6). These include thermal, RCS, attitude control, power subsystem, solar array, telemetry and command, and the apogee motor system which are briefly described in the following paragraphs. Greater detail is contained in the individual subsystem details discussed in Section 5.0.

The space frame consists of a thin walled cylindrical thrust tube coaxial with the satellite yaw axis, a conical adaptor attached to the aft end of the thrust tube which is attached to the third stage of the Thor Delta vehicle, and three major honeycomb sandwich platforms. The thrust tube which is 31.70 in. diameter and 55.37 in. long surrounds the apogee motor engine, and supports all the equipment platform by a system of bulkheads. To the bulkheads are mounted the reaction control system. The forward platform in synchronous orbit is earth pointing, and consequently any payload which is earth oriented such as communications antennas or attitude control horizon sensors will be mounted on this forward platform.

The north and south panels, because of their attitude with respect to the sun, are the main thermal radiating panels and their dimensions are 53.0 in. by 64.0 in. In the configuration shown the north and south panels house all the high dissipating power elements for the communications transponder, attitude control system, the batteries, the power electronics and the telemetry and command system. In addition, the deployed array is stowed on the north and south panels and the drive and tracking system is mounted on the forward platform. This latter system is used to rotate the deployed solar array and keep it pointed towards the sun.

GENERAL PURPOSE SATELLITE BUS (EXPLODED VIEW)

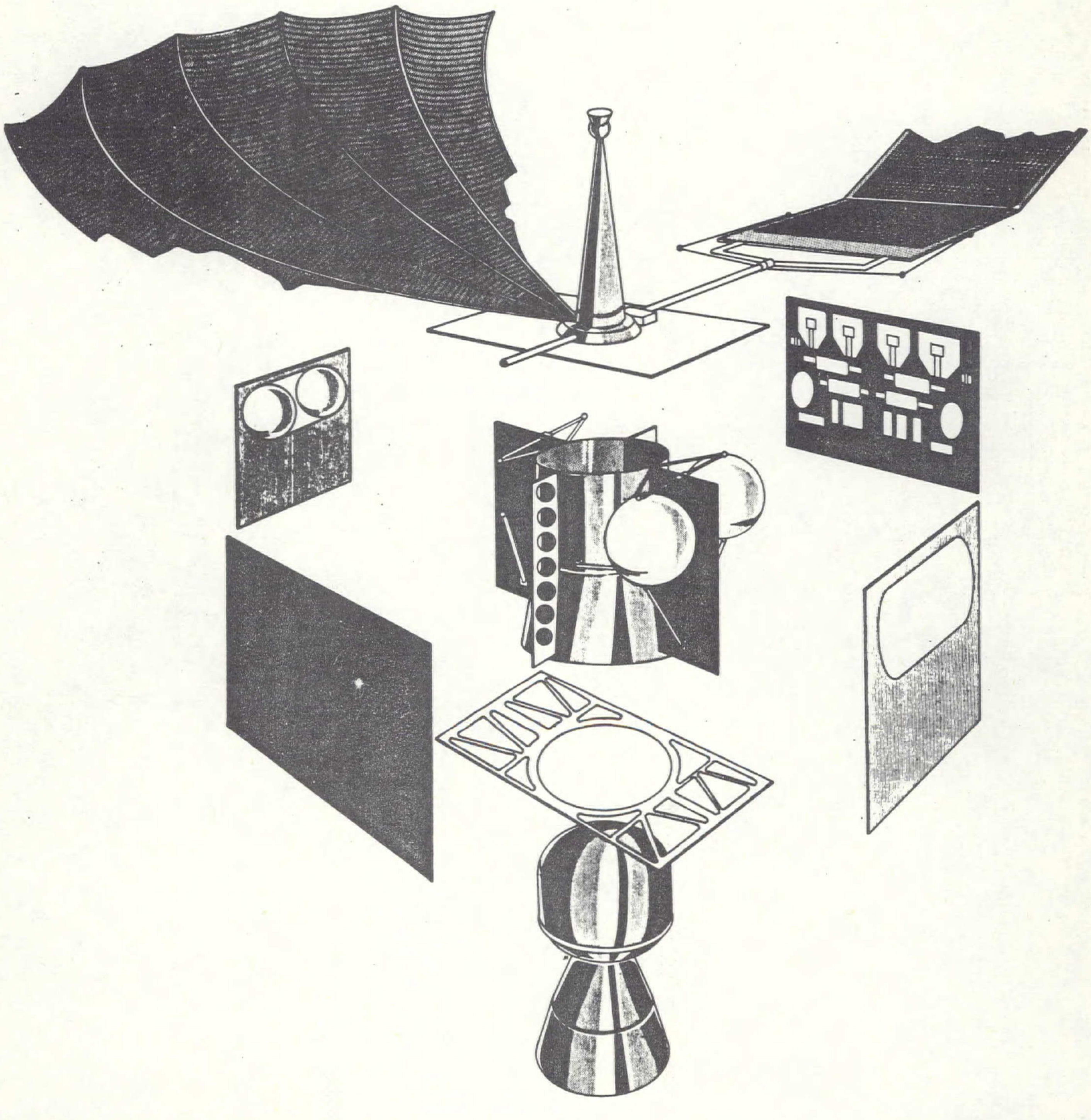


FIGURE 3-4

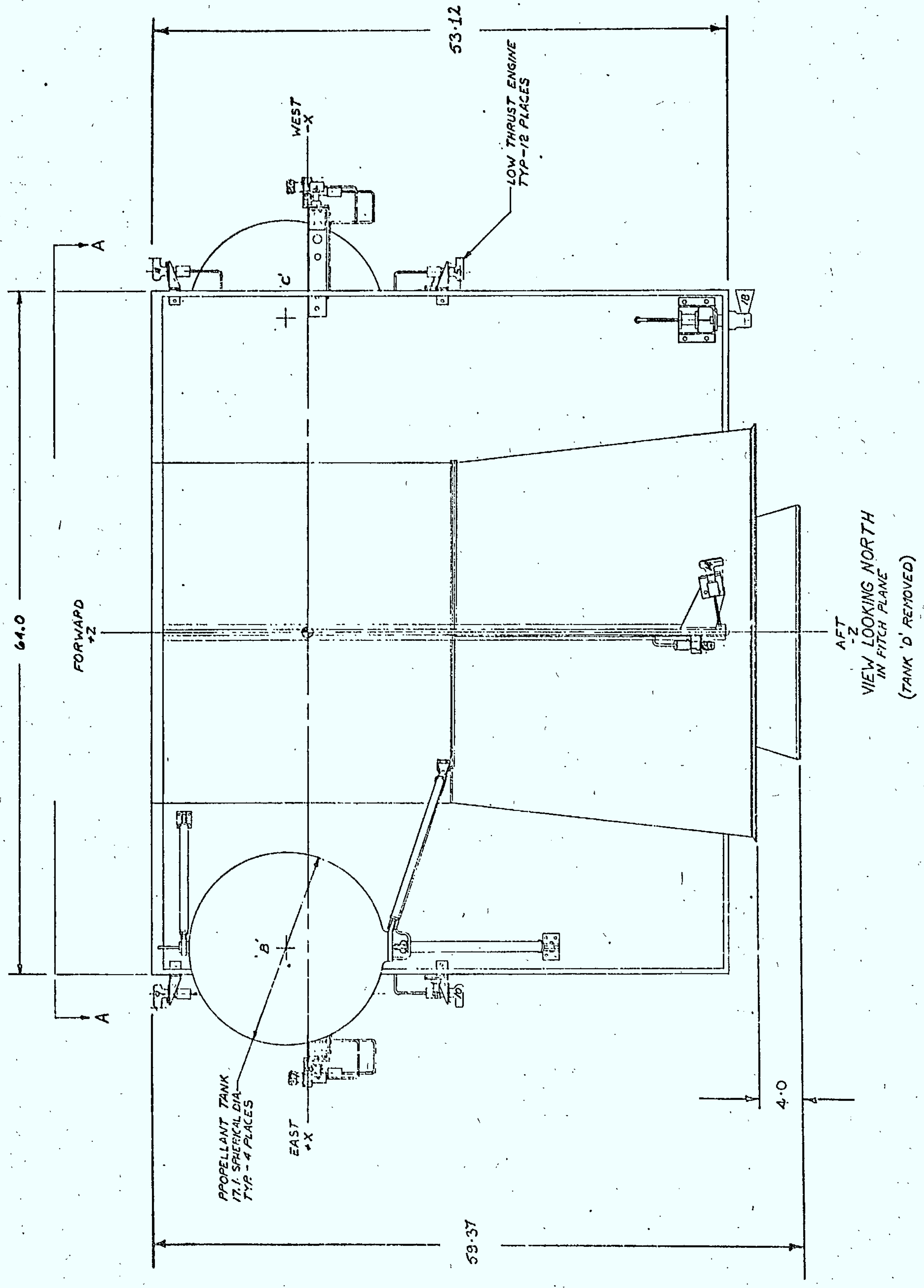
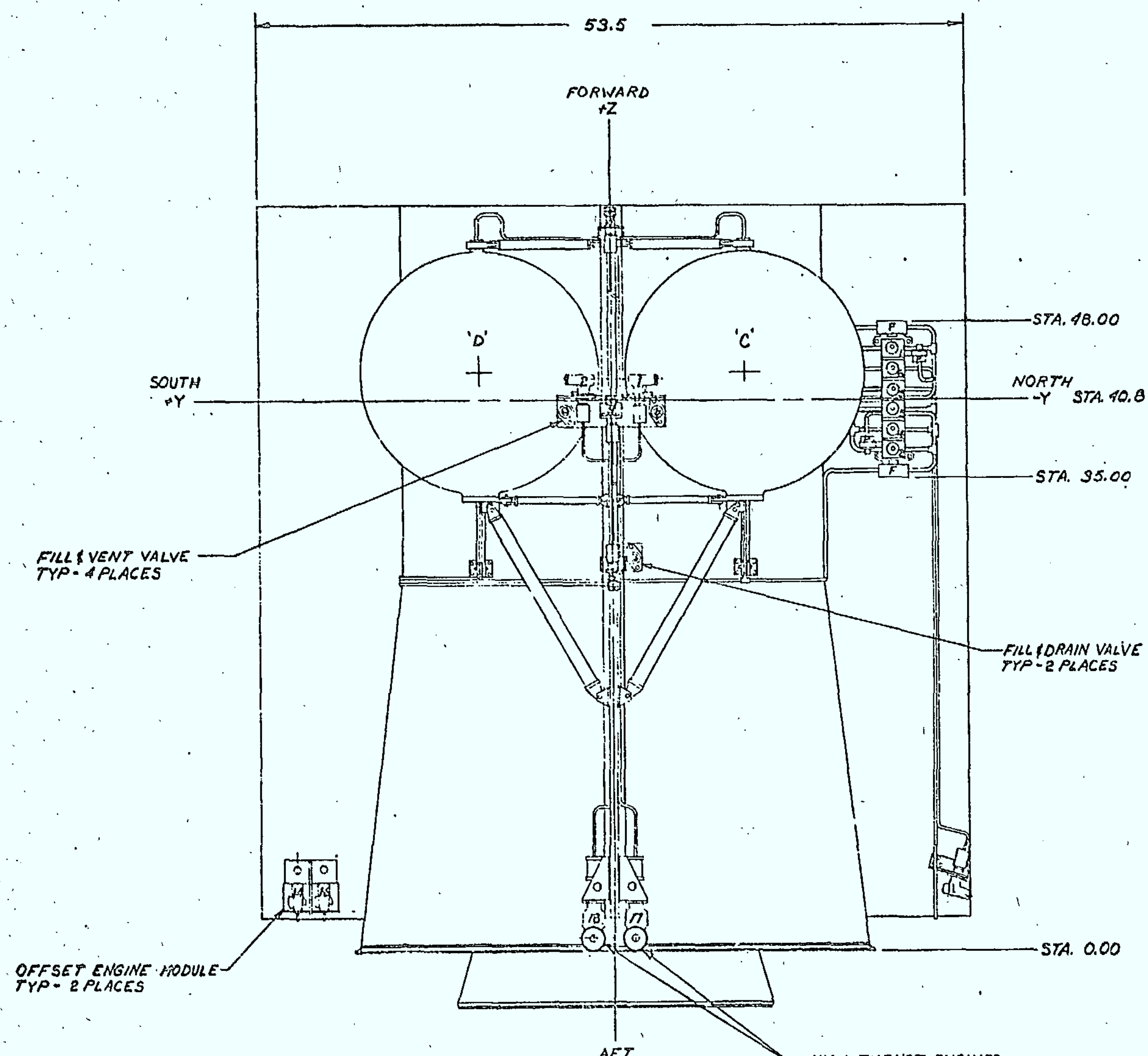

53.12
LOW THRUST ENGINE
TYP-12 PLACES
WEST
-X
A
'C'
64.0
FORWARD
+Z
AFT
-Z
VIEW LOOKING NORTH
IN PITCH PLANE
(TANK 'D' REMOVED)
'B'
A
PROPELLANT TANK
17.1 SPHERICAL DIA
TYP - 4 PLACES
EAST
+X
4.0
59.37

FIGURE 3-5



VIEW LOOKING EAST
IN ROLL PLANE

FIGURE 3-6

The arrays stowed on the north and south panels are the main source of power during transfer orbit, synchronous drift and acquisition, and have mounted on them sufficient solar cells to provide required power for the housekeeping subsystems during these phases of the mission. The solar panel substrate is constructed from aluminum honeycomb with aluminum face sheets. In synchronous orbit the arrays are deployed from the north and south panels and make available power for the entire spacecraft during sunlit periods.

Platform areas available for communications payloads are discussed in Section 3.3.

3.2.1 Structure Subsystem Description

The structure subsystem incorporates design features which are intended to achieve the maximum strength and stiffness characteristics obtainable in a minimum structural weight allowance. Conventional proven and established fabrication techniques, together with assembly techniques that are generally accepted within the aerospace industry, are adopted for the design of the structure. However, improvements on established methods, particularly in the areas of honeycomb cylinders and processes, are being considered wherever such improvements are indicated to be cost and weight effective.

Basically, the structure consists of two major assemblies, namely the primary structure and the peripheral structure. The primary structure is defined as the main thrust tube along with the aft platform and the bulkheads. The peripheral structure is made up of the remaining panels, namely the forward platform, north and south panels, east and west panels, and the interconnecting bulkheads and bracketry connecting these platforms and panels to the primary structure.

The central thrust tube of the primary structure is the main load carrying member. The load paths of all other elements of the spacecraft ultimately terminate in this item. It is a semi-monocoque structure consisting of a rivetted assembly of machined rings, rolled cylindrical and conical skins, assembled on a special fixture and finely machined to meet the accurate tolerance requirements. The three main rings in this tube are the upper ring, to which is mounted the forward platform; the apogee motor ring, which provides a mounting face and attachment point for the apogee motor; and the separation ring which is configured to match the Marmon type separation clamp and the flange of the Thor Delta third stage adaptor.

The dynamic response of the structure will be analyzed using computer programs such as STARDYNE. This program has also been used extensively for predicting spacecraft resonant nodes and frequencies for the satellite and major components.

3.2.2 Thermal Subsystem Description

The technical approach taken in formulating the thermal design has been to use the north and south panels as the primary heat rejection radiators during geo-synchronous orbit, and to minimize all other heat leaks from the spacecraft. During the spin (transfer and drift orbit) phase, when spacecraft heat must be conserved, the north and south panel radiators are each thermally protected and covered by the stowed array. These stowed north and south arrays utilize solar cells on their outer surface to provide electrical power for central housekeeping systems while in the transfer and drift orbit. During these phases (with arrays stowed) the spacecraft spin maintains the array solar cells at acceptable temperatures. While the north and south panels provide radiating surfaces for synchronous orbit, during drift and transfer orbit the aft platform is available to provide the necessary heat rejection capability for operating spacecraft components if required, i.e. TT&C components.

7/CD/47

A critical thermal item in the whole spacecraft is the apogee motor, which must be kept from becoming too cold (a solid propellant is temperature sensitive) before the motor is fired to inject the spacecraft into synchronous orbit. Protection is provided in part by mounting the motor case within the upper thrust tube, and by utlizing the aft thrust tube and its attached superinsulation cover to shield the motor nozzle. Heat leaks are further prevented by the use of a radiation barrier inside the nozzle.

The thermal design will basically be passive making extensive use of thermal blankets. However, due to the high dissipation that occurs within certain payload options heat pipes are adapted and submerged into the upper half of the north, south panel, specifically in the areas of the TWTs and preamp.

As mentioned the apogee motor is mounted in the thrust tube which contains blankets on the inside attached to the structure, and in the lower portion of the thrust tube where blankets are attached to the outer thrust tube arrangement. The blanket design technique adopted was developed on CTS. All blankets are grounded and they consist of outer layers of 0.001 inch (0.0025 mm.) Kapton, aluminized one side only, with aluminum faces turned inward. In between the outer layers of Kapton are cloth separators and double aluminized 0.00025 in. mylar layers. For the apogee motor and nozzle blankets, aluminized Kapton sheets have been used instead of the mylar layers.

Other techniques used to maintain temperature control on this spacecraft include the use of second surface mirrors which are mounted to the north and south platforms.

Electrical heaters will be used in some instances within the various spacecraft subsystems to reduce thermal load variation in the spacecraft.

7/CD/48

Thermal interactions between components and the spacecraft interior are controlled by using specified thermal surface finishes and by controlling the distribution of heat with the aid of thermal doublers. Thermal surface finishes used cover special paints, second surface mirrors, and silvered Teflon.

The analysis activity carried out on the spacecraft includes the prediction of thermal subsystem performances in both the space flight, and flight test configurations. This is achieved by formulating a series of thermal analytical models (TAMs) which represent the designed system. These TAMs are analyzed utilizing a variety of computer programs developed on CTS. Analysis cases include spacecraft bulk model analysis, aft platform analysis, south (experiments) analysis, apogee motor cool-down, north panel and forward platform analysis.

3.2.3 Attitude Control Subsystem (ACS)

This system will be analyzed and designed as an integrated system with the Bus structure RCS and DSA. The ACS controls the attitude of the spacecraft during both transfer and synchronous orbit, together with the transition, (acquisition) phase. During the spin stabilized (transfer/drift orbit) and three axis stabilized phases, the attitude of the spacecraft is determined from data obtained from earth and sun sensors.

The spacecraft is in a spin stabilized mode throughout the transfer and part of the initial synchronous drift orbit phase. The spacecraft is spinning about the yaw (apogee motor thrust tube) axis at a nominal rate of 60 rpm. During this phase the ACS uses spinning earth and sun sensors to provide the data necessary to determine the inertial attitude of the spin (yaw) axis. This sensor data is telemetered down for ground processing. Also included is a passive mechanical damping system which is adequate to damp spacecraft nutations induced or caused by any manoeuvres such as precessing of the spin axis during transfer orbit.

7/CD/49

To complete this precessing manoeuvre, the ACS in conjunction with the reaction control subsystem provides control of the necessary spin axis precession prior to and after apogee motor firing.

Initially for the study a conventional whecon ACS system was selected as the baseline design. However, because of the considerable weight improvement gained a hybrid system using microwave attitude sensing system (MASS) for attitude sensing has been chosen.

The system consists of the conventional components used for spin stabilized phase and reaction wheels plus thruster and MASS for the 3-Axis stabilized arrangement. The configurations show two wheels, but with the use of a currently being developed magnetic bearing system, this could be reduced to a single reaction wheel.

The spin phase components are mounted on the forward platform on the E/W faces. 3-Axis components such as MASS plus any back up Earth sensors are mounted on the forward platform which is earth facing in 3-Axis mode and the reaction wheels with the control electronics are mounted on the north/south panels along the pitch axis.

Further details on the control system are given in Section 5.1. These include trade-off studies that led to the selection of a hybrid system plus control features such as East/West and North/South stationkeeping momentum damping, spacecraft disturbances, i.e. solar radiation pressure, magnetic effects, and antenna torques.

Also included in the ACS electronics are the control features for stopping and slewing the solar array, this unit being an integrated system using computer processing technology.

3.2.4 Reaction Control Subsystem (See Figure 3-7)

The Bus incorporates the reaction control subsystem used to provide control torques during both the spinning and non-spinning phases of the mission. It is mounted on the thrust tube bulkheads of the Bus and integrated with the structure.

The RCS is a dual thrust blow down mass expulsion propulsion system utilizing terpolymer, surface tension tanks. Propellant, neat hydrazine N^2H^4, is isolated and directed within the system by latching valves. Both high and low thrust catalytic N^2H^4 engines are used for the spinning, despin and non-spinning manoeuvres respectively. Line filters are installed in this system to ensure cleanliness within each component. The Electrical Junction Box (EJB) of the RCS interfaces with the ACS and the TT&C subsystems and provides drivers and latches for RCS commands and signal conditioning for the RCS telemetry.

Gaseous nitrogen is loaded into the pressurant side of the two fuel tanks through respective pressurant fill and drain valves, and drives hydrazine through the system. As fuel is used up during the mission, the driving pressure decreases (blow-down) causing a corresponding reduction in thrust from the engines.

Hydrazine is loaded into the propellant side of the two fuel tanks through respective propellant fill and drain valves. Fuel may be isolated within the tanks by shutting latching valves. Pressure transducers monitor individual tank system pressures. There are two high thrust engines and twenty low thrust engines which are isolated into groups by latching valves. These groupings are arranged to provide maximum redundancy associated with minimum mass. Heaters are required on thrust chambers, valves, lines and tanks to ensure adequate performance and to prevent fuel from freezing during spacecraft cool down periods. These heaters are grouped functionally to minimize command requirements.

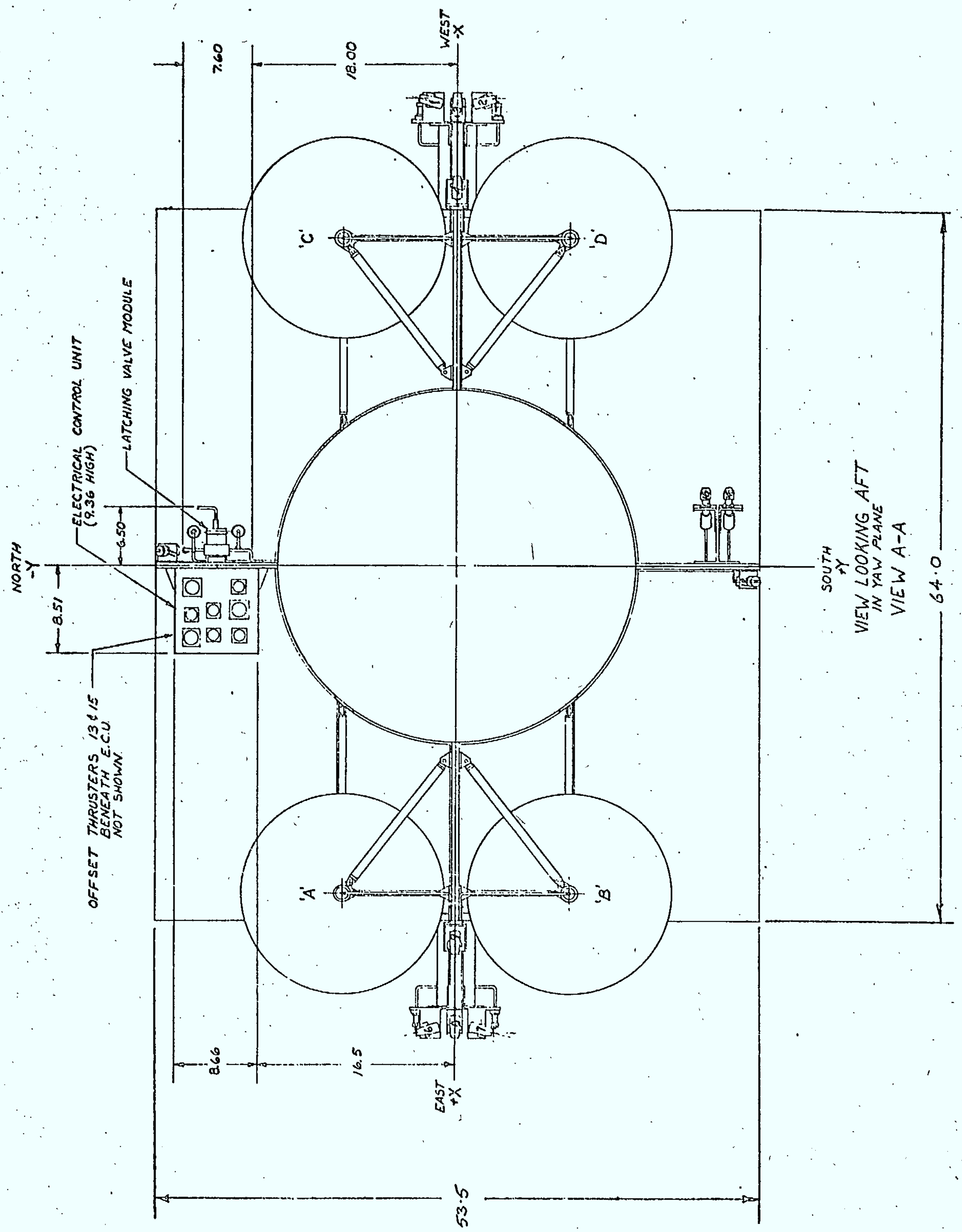

WEST
-X
7.60
18.00
'C'
'D'
LATCHING VALVE MODULE
ELECTRICAL CONTROL UNIT
(9.36 HIGH)
6.50
NORTH
-Y
8.51
OFFSET THRUSTERS 13 & 15
BENEATH E.C.U.
NOT SHOWN.
SOUTH
+Y
VIEW LOOKING AFT
IN YAW PLANE
VIEW A-A
64.0
'A'
'B'
8.66
16.5
EAST
+X
53.5

FIGURE 3-7

The 22 thrusters are used for control as described in the Attitude Control Subsystem. Redundancy exists to some degree for all thrusters if time permits low thrust engines to perform high thrust engine manoeuvres.

3.2.5 Deployable Solar Array Subsystem (See Figure 3-8)

The Deployable Solar Array (DSA) subsystem serves as a source of electrical power that is supplied to the spacecraft power subsystem for conditioning, distribution and battery charging. Power is required during the spacecraft's spinning (transfer and drift orbit) phases, as well as during the non-spinning (synchronous orbit) phase.

The DSA is divided into three basic assemblies:

a) The Solar Cell Array,
b) The Stowage and Deployment System,
c) The Orientation and Power Transfer System.

A general layout of the system is shown in Figure 3-8.

The solar cell array consists of an assembly of n-on-p silicon solar cells with cover glasses, interconnected in series and parallel and mounted on an insulating substrate. Flat conductor cable wiring transfers power from the solar cell strings along the array to the diode boards on the stowage system. The solar cell array is bonded to the stowage system substrate. The array is sized to provide 835 watts of D.C. power at 40 volts at the end of life (8 years).

The stowage and deployment system consists of aluminum honeycomb core/aluminum face sheet panels hinged together with spring loaded hinges that allow the panels to be folded concertive fashion on the spacecraft north and south panels, and lock in place when the panels are deployed in a plane normal to the N/S panels and parallel to the spacecraft pitch axis. Four panels are used per side.

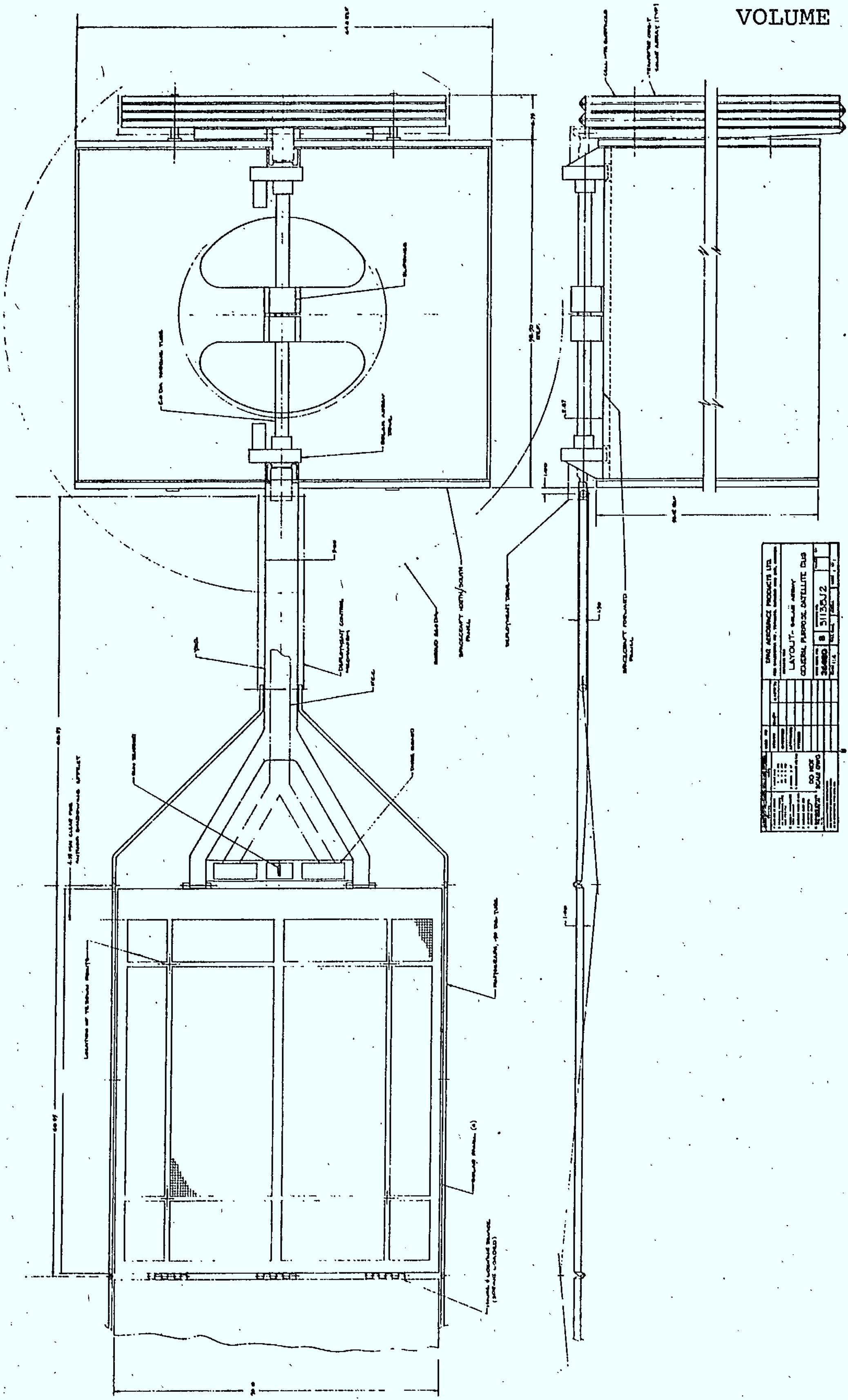


FIGURE 3-8

When stowed, the four panels will be restrained against the spacecraft N/S platforms at four tie-down points. Release will be effected by a central pyrotechnic actuated system. A pantograph system with an electric motor controlled mechanism will control deploympent of the array, the deployment energy being provided by the interpanel hinges.

A metal yoke structure will place the inboard-most panel approximately 60 inches away from the N/S panels to minimize shadowing by the spacecraft's UHF antenna and provide an adequate viewfactor for the heat dissipating spacecraft platforms. This will have provision for mounting diode and terminal boards to interface with the solar array and flat conductor cabling to the power transfer system. Provision is also made for installing an analog sun sensor to be used by the spacecraft attitude control system (ACS) to control rotation of the array.

When deployed, the system will feature a natural frequency of greater than 0.15 Hz to avoid interaction with the spacecraft ACS. When stowed, the outer-most panels will provide an average of greater than 80 Watt of D.C. power at 40V when the spacecraft is spinning at 60 rpm. It will be possible to remove 1 panel per side to provide 625 watt at the end of life in the deployed condition, and still maintain the stowed condition power capability.

The orientation and power system is installed on the outside of the spacecraft forward or earth-facing platform. It consists of redundant drive machanisms containing motors, bearings and position read-out. Slip ring assemblies are used for power transfer. A torque tube or through shaft connects the north and south arrays. The system will be controlled by the spacecraft ACS and will be capable of turning the arrays at the two basic rates of 1 revolution/drag and 15°/minute.

More details of subsystem trade-offs and sizing are contained in Section 5.3 of Volume I and a preliminary subsystem performance specification is contained in Section 1.4 of Volume II of this report.

3.2.6 Apogee Kick Motor

The apogee kick motor is mounted in the thrust tube and is capable of providing the velocity increments sufficient for placing a spacecraft weighing 1925 lbs. into synchronous orbit. The motor size will be optimized to include recent improvements in Isp and motor performance which have been developed in the last two years.

The apogee motor for this satellite will be developed and supplied by a motor supplier and integrated with the structure at the launch facility. Interface details and interface tooling will be made available. Consequently no integration problems are anticipated. Qualification tests on this motor will be conducted which include sea level and high altitude firing tests.

The motor comprises the following major components:

- Case and insulation chamber
- Expansion nozzle
- Ignition system consisting of igniter, pyrotechnic train, and safe and arm device
- Motor attachment flange.

It should be noted that the motor design can be overloaded by 10% additional propellant so without any major redesign a 2125 lb. satellite can be injected into geo-synchronous orbit with the proposed motor.

7/CD/54

3.3 Communications Payload Platforms

The bus studied has made available three major patforms for communication and experiments payload. They are:

a) Forward Platform - for communication's antenna and multiplexer. The arrangement allows for both mounting of a deployed multi-beam antenna 13-14 ft. dia. for UHF, and 4-6 GHz transponder or a quad-helix arrangement for UHF with a 5 ft. parabolic dish and feed horn arrangement for the 4-6 GHz transponder.

b) North and South Platforms - for mounting high dissipating communication payload components such as TWT's, preamps, etc. The study has adopted the concept wherein individual panels for UHF, 4-6 GHz, 12-channel, 12-14 GHZ, 4-channel; and S-band plus L-band and experiments, would be available and interchangeable; these panels would also be fitted with ACS, battery, and other power housekeeping dissipating components.

Further details on the above and other features associated with these panels follow:

3.3.1 Forward Platform (See Figure 3-9)

The forward platform is capable of carrying communication payloads given in weights details in Section 4.2. The control system static earth sensor is mounted on the forward face of this platform; consequently, it is an earth-oriented stable platform with a pointing accuracy of 0.15 in pitch and roll and 1.0° in yaw. An aperture for viewing earth must be made available in any antenna design considered, or alternatively, a stand-off mounting arrangement will be required for this unit. It should be noted that high dissipating components should not be placed on this platform as they distort the platform. If this request is met the platform distortion will not exceed 0.2° in all axes. In addition, a specific area for the drive and tracking units has been allocated on the panel.

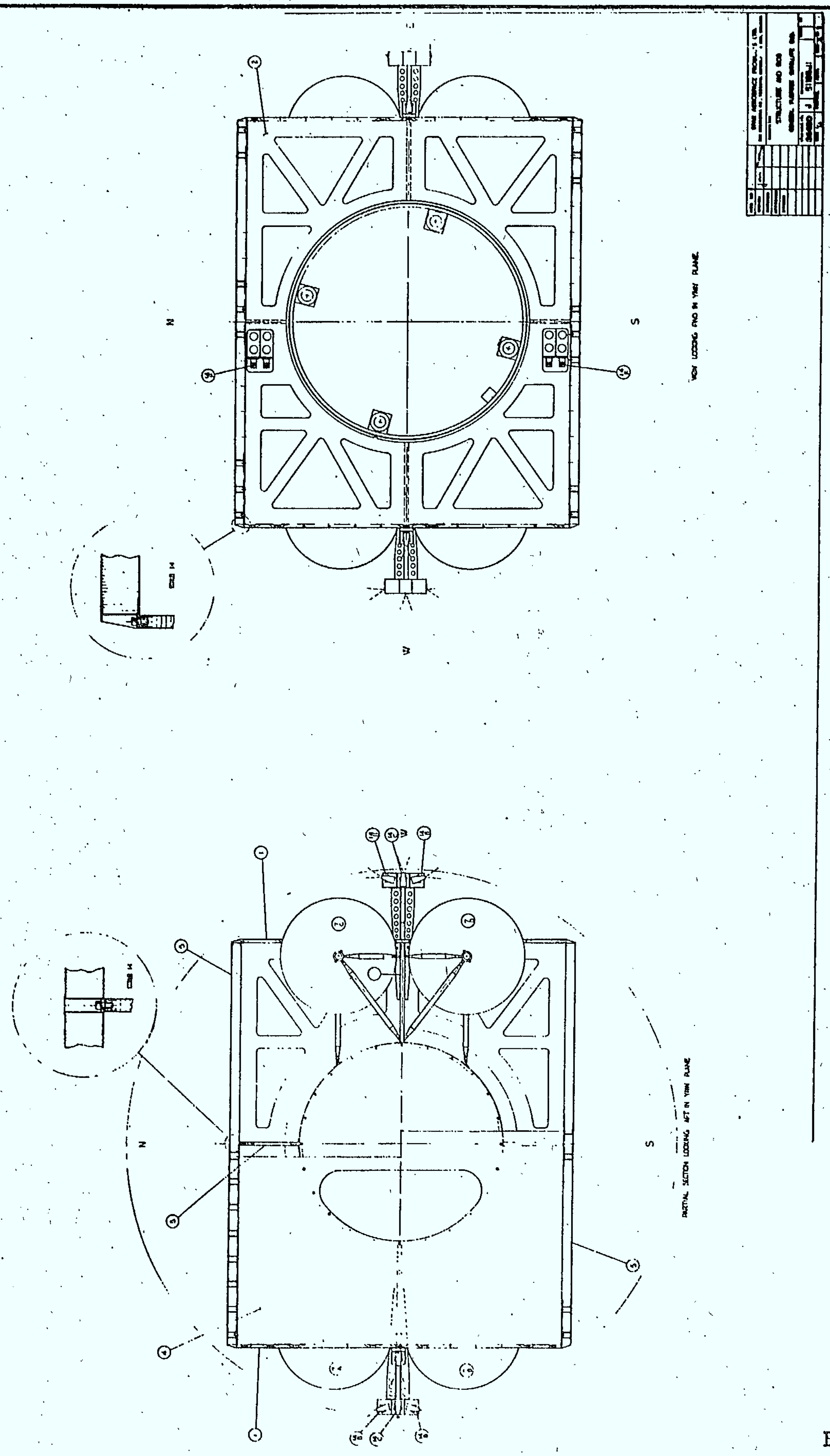


FIGURE 3-9

b) Arranges TWTs and preamps close to the forward platform thus keeping RF losses to a minimum.

c) Thermally viable in the allowable spacecraft envelope established.

Other arrangements have been reviewd where the 4-6 GHz, 12-channel or 12-14 GHZ, 4-channel arrangements are mounted on the same platform as the UHF. The major concern with that type of arrangement was that for thermal reasons larger panels would be required to maintain component temperatures using dissipation data provided by DOC and previous knowledge on component operating limits.

3.3.2 North and South Panels (See Figures 3-10, 3-11, 3-12)

These panels are the main radiating surfaces in synchronous orbit; they are thermally isolated from the inside of the spacecraft, and the panels have been sized to handle the dissipations provided by DOC using heat pipe configurations (discussed with Grumman).

Panel details are provided in the attached figures and in arriving at these sizes and configurations, the following assumptions were made:

a) For the 4-6 GHz, 12-channel transponder, 5-6 watt TWTs, preamp and multiplexer sizes used on ANIK spacecraft were assumed.

b) For the 12-14 GHz, 4-channel transponder CTS 20 watt TWT and preamp sizes were assumed.

c) For the UHF transponder, components were sized based on footprint requirements related to thermal dissipations provided by DOC.

d) For "S"-band, "L"-band and other Telesat (late) requirements (other than payload e_2), it has been assumed that the payloads can be fitted on to the panel sizes configured. If additional dissipation surfaces are required, further heat pipe systems can be adapted to either panel.

In arriving at the panel schemes and arrangements discussed above, considerable thought was given to interchangeability features for north/south panels. The main advantages of these are:

a) Assists in providing constant spacecraft balance between configurations and assists in achieving and maintaining consistent moment of inertia.

7/CD/71

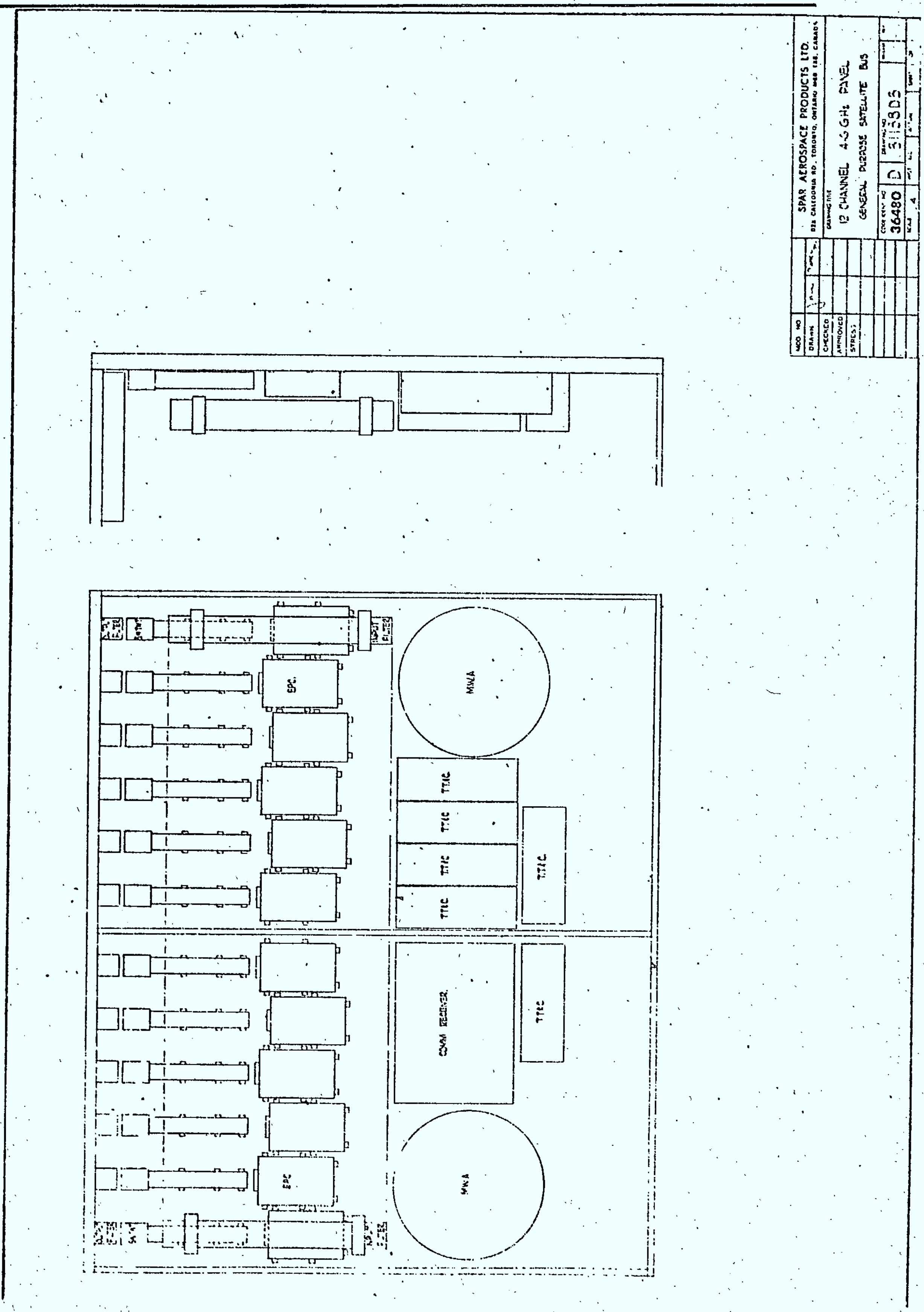

SPAR AEROSPACE PRODUCTS LTD.
12 CHANNEL 4-6 GHz PANEL
GENERAL PURPOSE SATELLITE BUS
36480
D
MWA
TT&C
EPC
COMM RECEIVER

FIGURE 3-10

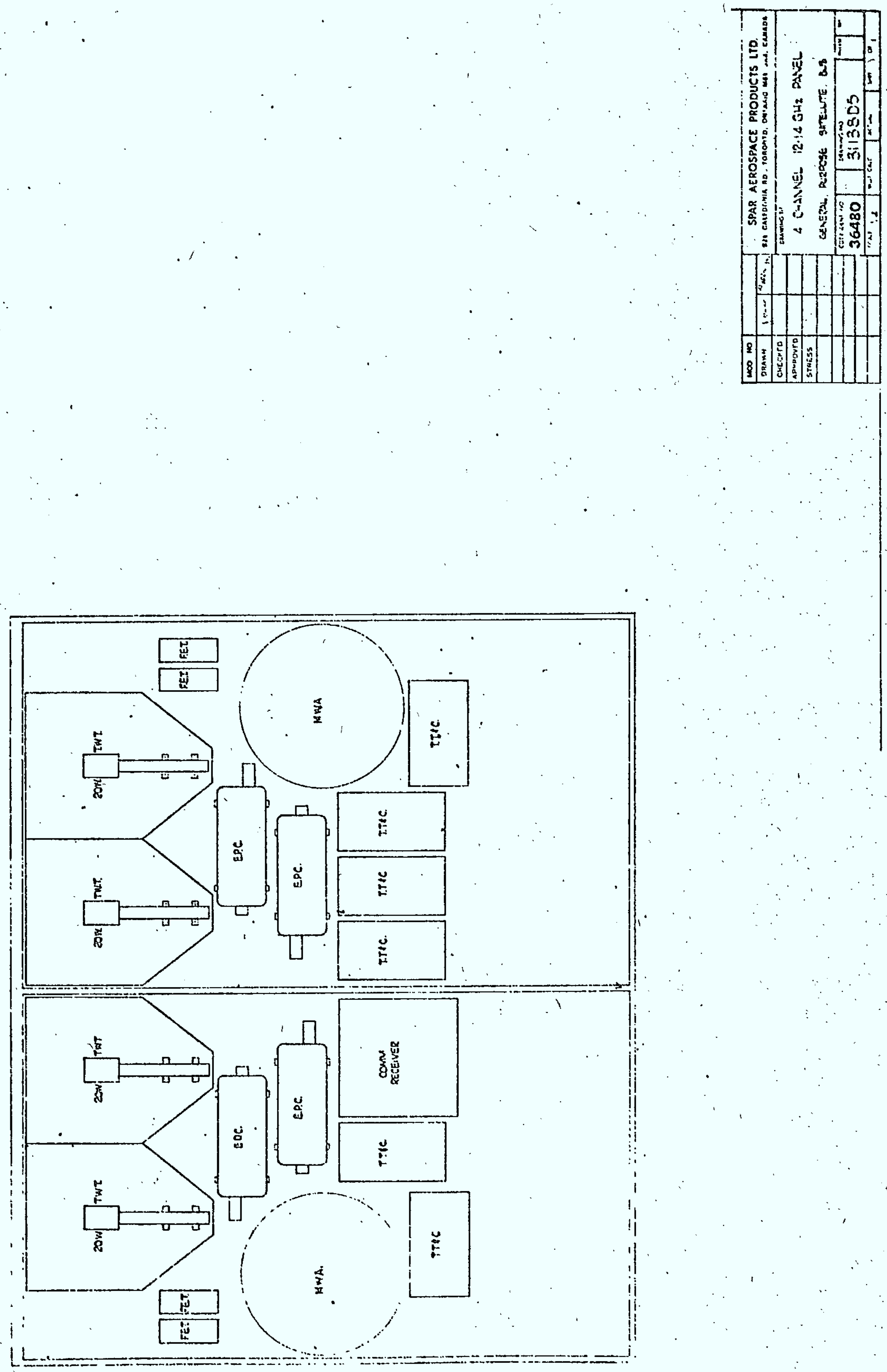

SPAR AEROSPACE PRODUCTS LTD.
4 CHANNEL 12:14 GHz PANEL
GENERAL PURPOSE SATELLITE B.S
36480
3113805
MOD NO
DRAWN
CHECKED
APPROVED
STRESS
F.E.T.
FET
M.W.A.
T.T.&C.
TWT
20W
E.P.C.
COMM RECEIVER

FIGURE 3-11

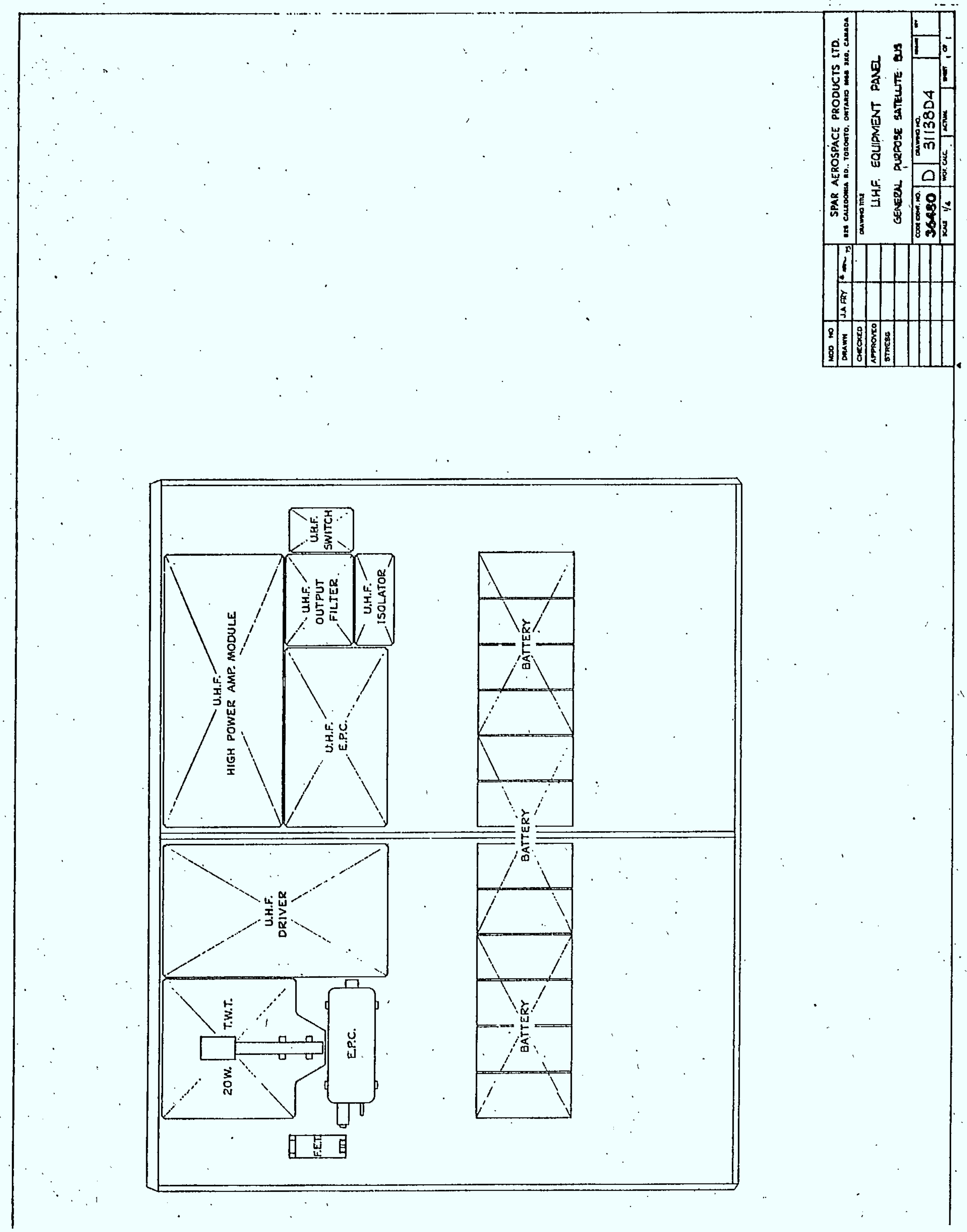

SPAR AEROSPACE PRODUCTS LTD.
U.H.F. EQUIPMENT PANEL
GENERAL PURPOSE SATELLITE BUS
DRAWN
CHECKED
APPROVED
STRESS
U.H.F. SWITCH
U.H.F. OUTPUT FILTER
U.H.F. ISOLATOR
U.H.F. HIGH POWER AMP. MODULE
U.H.F. E.P.C.
U.H.F. DRIVER
T.W.T.
20W.
E.P.C.
F.E.T.
BATTERY
BATTERY
BATTERY

FIGURE 3-12

4.0 BUS SYSTEM DETAILS

This section lists the parameters and requirements ranging from launch vehicle details to pointing requirements provided and agreed to by DOC against which the baseline bus study was produced.

In many cases due to lack of information at the time it became imperative to use assumptions for certain components in order to establish spacecraft sizes and layouts and complete the study. Generally, these were associated with communications payload details. Typically, to establish moment of inertia information for roll to pitch stability criteria, details from a separate antenna survey study conducted for DOC were used, (Harris Multi-Beam Antenna). Also, to size panels, TWT's, pre-amps, multiplexer dissipations and sizes used by ANIK and CTS spacecraft were adopted.

It should be noted that power for payloads listed in items e, e_1 and e_2 were provided by Telesat fairly late in the study and, in particular, e_2 does not form part of the baseline design since its power requirements are outside the boundary of the study which listed an End of Life (EOL) power requirement of 835 watts after six years.

Having made these assumptions the latter part of this section discusses the mass properties of the spacecraft that were derived from the baseline design, configurations and layouts established and listed in Section 3.0 of this volume.

To summarize the mass properties section, all payload options listed by DOC can be launched using this bus configuration, however, the moment of inertia criteria is not met by the design (this is explained in the text).

Further improvements are discussed in Volume IV and if incorporated along with possible third stage kick motor improvements (reference Thiokol) then the options listed by Telesat also become viable for a launch on 3914 with this bus arrangement.

4.1 Parameters and Assumptions Used for "Bus" Study

4.1.1 Launch Vehicle

4.1.1.1 Delta 3914

4.1.1.2 Fairing dynamic envelope = 86" diameter

4.1.1.3 NASA Delta Restraint Manual

4.1.1.4 Adaptor 3731A

4.1.2 Orbit

4.1.2.1 Geosynchronous

4.1.2.2 Stationkeeping: N-S = ±.05°
E-W = ±.05°

4.1.2.3 Adjustment period no less than 21 days

4.1.2.4 Transfer orbit:

Apogee	-	19,525 n.mi.
Perigee	-	100 n.mi.
Inclination	-	28.3°
Delta V	-	6,024 ft./sec.
No. of Transfer Orbits	-	13

4.1.3 Spacecraft

4.1.3.1 Life

In orbit life time	=	6 years
Storage prior to launch	=	3 years

4.1.3.2 Weight

Lift-off weight = 1,925 lbs. (excludes adaptor weight)

4.1.3.3 Stability

Transfer/Drift Orbit: spin stabilized
Synchronous Orbit: 3 axis stabilized

4.1.3.4 Power

825 watts E.O.L. (does not include power for payload e_2 below).

7/CD/58

4.1.4 Payload (See Tables 4.1.4-1 and 4.1.4-2).

4.1.4.1 Options

a) UHF + 12 channel (4-6 GHz)
b) UHF + 4 channel (12-14 GHz)
c) UHF + 1 channel (7-8 GHz)
d) UHF + L Band + 1 channel (7-8) + Exp.
e) 24 channel (4-6 GHz)
e_1) 12 channel (12-14 GHz), 30 W TWT
e_2) 8 channel (12-14 GHz), 4-20 W TWT, 4-50 W TWT

Roll to pitch inertia greater than 1.1 during spin phase.

4.1.4.2 Eclipse Operation

Full eclipse operation

4.1.4.3 Payload Assumptions Used for Sizing Spacecraft

i) 4-6 GHz (12 Channel)

- 5-6 watt TWTS (HAC)
- No transponder redundancy per channel
- Same transponder electrical configuration as ANIK Mk I including EPC multiplexers, filters, etc.
- 12 channels powered E.O.L., including eclipse.
- Dissipations per channel as per ANIK.

ii) 12-14 GHz (4 Channel)

- 20 watt TWT
- No redundancy per channel.
- Similar transponder electrical configuration as CTS, including EPC's, etc.

CD/59

GP BUS PAYLOAD CHARACTERISTICS

TABLE 4.1.4-1

PAYLOAD OPTIONS	ANTENNAE TYPE	ANTENNAE NO. & SIZE	ANTENNAE WEIGHT	ANTENNAE SIZE	TRANSPONDER WEIGHT lbs.	TRANSPONDER DISSIPATION watts	DC POWER
a) UHF/4-6 GHz Transponder (12 channel) (5-6W TWT)	Dep. parabola or parabola + quad helix	13 ft. dia. or 82" dia. x 100" long	55 lbs.	TBD	218.4	S 314.9 E 264.3	S 414.9 E 339.3
b) UHF/12-14 GHz Transponder (20W TWT)	Dep. parabola or parabola + quad helix	13 ft. dia. or 82" dia. x 100" long	55 lbs.	TBD	180.5	S 456.6 E 351.5	S 624.6 E 463.5
c) UHF/7-8 GHz Transponder with auxilary experimental payload	Dep. parabola	13 ft. dia.	55 lbs.	TBD	221.9	S 404.2 E 252.3	S 505.2 E 305.3
d) UHF/SHF/L Band Transponder	Dep. parabola	13 ft. dia.	60 lbs.	TBD	190.4	S 397.2 E 305.3	S 513.2 E 373.3
e) 24 channels 4-6 GHz	parabolic, shaped beam	5 ft. parabolic or equivalent	55 lbs.	TBD	TBD		S 400
e_1) 12-14 GHz Transponder (12 channels) 20W TWT			55 lbs.	TBD	TBD		S 636 E 530
e_2) 12-14 GHz Transponder 4-20 watt TWT 4-50 watt TWT + 50% redundancy			55 lbs.	TBD	TBD		S E 732

SPAR-R.677
VOLUME I

POWER REQUIREMENTS/ASSUMPTIONS

	POWER REQUIREMENTS HOUSEKEEPING	POWER REQUIREMENTS COMMUNICATIONS PAYLOAD
a) UHF/4-6 GHz (12 channel transponder)		S 414.9 watts E 339.3 watts
b) UHF/12-14 GHz transponder		S 624.6 watts E 463.5 watts
c) UHF/7-8 GHz transponder with Exp. payload		S 505.0 watts E 305.0 watts
d) UHF/SHF/L Band transponder		S 513.0 watts E 373.0 watts
e) 4-6 GHz, 24 channel transponder (5 to 6 watt TWT)		
e_1) 12-14 GHz 12 channel transponder (4-20W TWT) (4-50W TWT)		636.0 watts
e_2) 12-14 GHz 8 channel transponder (4-20W TWT) (4-50W TWT)		732.0 watts
TTC	30 watts	
Power	15 watts	
ACS	40 watts	
DSA	10 watts	
Thermal	30 watts	
RCS	15 watts	
TOTAL	140 watts	732.0 watts
Maximum Requirement at End of Life:	i) excluding item e_1 and e_2	765.0 watts
	ii) includes e_2 requirement	872.0 watts

TABLE 4.1.4-2

- 4 channels powered E.O.L. including eclipse.

- Dissipations per channel as per CTS.

iii) UHF

- Dissipation as supplied by DOC

- Panels sizing based on thermal doubler requirements.

iv) L Band and Experiments

- Assume they will fit on panel sizes derived from a), b) or c) above.

v) e_1) Assumptions same as d).

4.1.5 Subsystem Parameters

4.1.5.1 ACS

(a) Operation

Not to require correction more than every five days. Stationkeeping cycle to be 21 days.

(b) 3 Axis Accuracy

	Specified	Value met (during Orbit adjust thrusting)
Roll:	$\pm$.10°	.15°
Pitch:	$\mp$.10°	.15°
Yaw:	$\mp$.3°	1.0°

(c) Spin Phase Accuracy

$\pm$.20° (inclination angle for apogee motor firing).

(d) Alignment Accuracy

Same as CTS and ANIK.

7/CD/60

4.1.5.2 RCS

(a) Fuel: Hydrazine
(b) Life: 6 years, tanks sized for 8 years
(c) Thruster size: i) Low thruster 0.1 lb.in.
ii) High thruster 1.0 to 5.0 lb.in.

4.1.5.3 Solar Array

(a) Power

End of Life, 6 years:	835 watts
Housekeeping:	140 watts
Payload:	640 watts

(b) Tracking Accuracy: $\pm 1°$

(c) Transfer Orbit Power: 80-100 watts

(d) System Voltage: 50V

4.1.5.4 Structure

(a)	Life-Off Weight:	1,925 lbs.
(b)	Design for:	2,120 lbs.
(c)	Launch Loads:	Delta Restraints Manual

4.1.5.5 Thermal

System/Component Dissipations: as provided by DOC and attachments.

4.2 Bus Mass Properties

The mass properties details covered in the text that follows discuss bus subsystem weights and Inertia Ratio details of the complete spacecraft using data for payloads provided by DOC at various meetings during the study phase and as extracted from internal subsystem studies conducted for DOC.

The bus subsystem weight provided in Table 4.2.1-3 "GP Bus Weight Summary" reviews a bus weight using improved 3-Axis satellite ACS, RCS and solar cell technology weights which will be available for a

1979 launch and whose details are discussed in the ACS section Volume I and RCS section Volume II. The resulting bus weight of 1607 lbs and 1587 lbs (depending on payload option power requirements) when coupled with the payload options provided in the DOC statement of work show that all options can be launched on 3914 launch vehicle in the period 1979-1985 the lowest weight margin being +19.0 lbs. This does not include the two Telesat requirements which were submitted later during the study. Volume IV discusses further improvements to other subsystems which with the Thiokol suggested improvements to TE364-4 third stage motor may make the Telesat requirements a feasible payload in the time frame 1979-1985.

With regard to the spacecraft major axis inertias details are provided in Table 4.2.2-1 and the resulting Inertia Ratios show that a figure of 1.03 is met. This is lower than the specified limit; however, as agreed by many specialists a value greater than 1.025 for a spinner is considered acceptable.

4.2.1 Weights (See Tables 4.2.1-3, 4.2.1-4 and 4.2.1-5)

The following is a brief summary of the guidelines used for the determination of the mass properties of the General Purpose Bus.

4.2.1.1 TT&C and Antenna

The telemetry, tracking and command consists of six units on the N/S panels with an omni antenna mounted on top of the deployable dish antenna. The total weight of 35 lb. is based on CTS and other communications satellites, but does not include security electronics.

4.2.1.2 ACS

The attitude control subsystem weight of 54 lb. is based on a Hybrid system using one momentum wheel and magnetic bearing system.

TABLE 4.2.1-3 GP BUS WEIGHT SUMMARY

		WEIGHT IN.LBS.	
SUBSYSTEM		G-P B BASELINE	CONVENTIONAL ACS AND RCS SUBSYSTEM
TT&C & Ant., no security box		35	35
ACS Hybrid		54	76
RCS (Tanks, Pressurant and struts (Fuel - 6 years		218	259
Structure (Mag. Thrust Tube) (including Payload Inserts, excluding RCS Struts and Brackets)		103	103
Thermal		21	21
Apogee Motor (Motor Fired 54) (Propellant & Inserts 903)		957	957.9
Array (D&TM) Plus Solar Panel) 120 lb 800W 100 lb 600W			
Battery & PC (Housekeeping Portion .22 x 140 + 25)		56	56
Harness		35	35
Safe and Arm and Balance Weight		8	8
TOTAL BUS	800W Array	1,607	1,675
	600W Array	1,587	1,651

7/CD/66

TABLE 4.2.1-4 UHF BUS PAYLOAD TYPICAL

ITEM	WT	X	Y	Z	Wx^2	Wy^2	Wz^2	Ixp	Iyp	Izp
TT&C (S)	34	5.67	22.55	18.23	1,093	17,289	11,299	493	8,670	8,347
TT&C ANT.	1	0	0	118.0	0	0	13,924	494	494	0
UHF/SHF DISH	31.5	0	0	88.21	0	0	245,102	17,895	17,895	3,024
ADDITIONAL WAVE GUIDES	23.5	0	0	62.5	0	0	91,797	979	979	392
PAYLOAD S	107.6	0	22.55	40.00	0	54,715	172,160	3,667	39,175	35,669
PAYLOAD N	55.8	0	-22.55	40.00	0	28,374	89,280	1,902	20,316	18,498
BATTERY & PCN	130.8	0	-22.55	21.4	0	66,512	59,901	1,180	30,261	29,866
MW S	20.0	±23.5	22.55	19.5	11,045	10,170	7,605	342	653	342
ACS MW	34	0	0	52.4	0	0	93,356	7,129	10,245	17,283
ARRAY - D&TM	25	0	0	57.9	0	0	83,810	885	885	1,667
- SOLAR PANEL	95	0	±30	28.81	0	85,500	78,852	27,945	47,349	20,180
HARNESS ELEC.	35	0	0	30.6	0	0	32,773	4,713	7,338	11,958
SAFE & ARM & BALLAST	8	0	0	24.0	0	0	4,608	0	0	0
SECURITY BOX	10	0	±22.55	18.23	0	5,085	3,323	0	0	0
THRUST TUBE STRUCT.	36	0	0	20.47	0	0	15,085	13,745	13,745	10,130
NORTH PANEL	14	0	-26	28	0	9,464	10,976	3,277	8,056	4,779
SOUTH PANEL	14	0	26	28	0	9,464	10,976	3,277	8,056	4,779
EAST PANEL	5	32	0	28	5,120	0	3,920	2,147	1,105	1,042
WEST PANEL	5	-32	0	28	5,120	0	3,920	2,147	1,105	1,042
FORWARD PLATFORM	10	0	0	54.4	0	0	29,594	2,083	3,413	5,497
SEP. PLATFORM	5	0	0	1	0	0	5	1,707	1,042	2,728
E/W PARTITION	5	0	±23	28	0	2,645	3,920	1,127	1,220	94
N/S PARTITION	4	±20	0	28	1,600	0	3,136	928	901	27
RCS STRUCT.	1	0	±26	28	0	676	784	0	0	0
MISC. HARDWARE	4	0	0	28	0	0	3,136	1,717	2,249	2.199
RCS TANK & PRESS	65	±30	±10	43	58,500	6,500	120,185	2,106	2,106	2,106
RCS FUEL	153	±30	±10	40.8	137,700	15,300	254,690	4,957	4,957	4,957
THERMAL CONTROL	21	0	0	40.0	0	0	33,600	6,307	8,743	11,900
APOGEE MOTOR	54	0	0	24.0	0	0	31,104	17,150	17,150	6,075
PROPELLANT & INERTS	903	0	0	37.0	0	0	1236,207	91,104	91,104	80,870
TOTAL	1,910.2	0	0	36.36	220,178	311,694	2,749,025	221,403	349,212	285,451

TABLE 4.2.1-5 - WEIGHT SUMMARY: PAYLOAD OPTION DETAILS

PAYLOAD	UHF + 4-6 GHz	UHF + 12-14 GHz	UHF + 7-8 GHz EXPERIMENTS	UHF + SHF & L BAND	24-4-6 GHz	12/12-14 GHz	8/12-14 GHz
DEFINITION	a	b	c	d	e	e_1	e_2
POWER REQUIREMENTS	414.9S 339.3E	624.6S 463.5E	505.0 305.0	513.0S 373.0E	400S 350E	636S 530E	732S 732S
A) PAYLOAD - TRANSPONDER & ANT.	218.4	180.5	221.9	190.4	231	235	235
Antenna included in A	(55)	(55)	(55)	(60)	(55)	(55)	(55)
B) BATTERY TO OPER. A	74.8	102.08	67.1	82.06	77	116	161
C) SECURITY BOX MIL. VER.	10	10	10	10	0	0	0
X TOTAL (A + B + C)	303.2	292.58	299.0	282.46	308	351	396
Y TOTAL (BUS)	1587	1607	1607	1607	1587	1607	1607
TOTAL (X + Y)	1890.2	1899.58	1906	1889.46	1895	1958	2003
LIFT OFF WEIGHT ALLOWED	1925.0	1925.0	1925.0	1925.0	1925.0	1925.0	1925.0
(CONTINGENCY)	+34.8	+25.42	+19.0	+35.54	+30.0	-33.0	-78.0
Note:							
TOTAL ARRAY (INCL. HOUSEKEEPING)	100	120	120	120	100	120	120*
TOTAL BATTERY & PC (56 + B)	130.8	158.08	123.1	138.06	133	172	217

*Higher efficiency cells will be required to meet this.

4.2.1.3 RCS

The reaction control subsystem is based on a four tank system that weighs 65 lbs. including tank pressurant struts and brackets. The fuel weight of 153 lbs. is sufficient for six years, using improved technology RCS engines.

4.2.1.4 Structure Subsystem

The weight of 103 lbs. is for a structure consisting of a magnesium thrust tube and honeycomb panels for all platforms and partitions. A thrust tube made of 1/8" honeycomb panels would save 9 lbs. Replacing the film adhesive in the panels, platforms and partitions with a pre-bonded core material would save an additional 4 lb.

Also included in this weight is 4 lb. for panel inserts.

4.2.1.5 Thermal Subsystem

The weight of 21 lbs. is for a thermal subsystem using heat pipes and thermal doublers to dissipate heat. The use of heat pipes and doublers instead of just thermal doublers results in a saving of 12 lbs.

4.2.1.6 Apogee Motor

The weight of 957 lbs. (fuel 903 lbs., motor case 54 lbs.), is for a spacecraft weighing 1,925 lbs.

4.2.1.7 Solar Array Subsystem

The weight of 120 lbs. includes the drive and tracking mechanism and a solar array large enough to power any one of the proposed communication payloads (except payload option e_2).

4.2.1.8 Battery and Power Control Subsystem

The weight of 56 lbs. listed includes a power control subsystem weighing 25 lbs. and that portion of the battery that is needed to power the housekeeping load of 140 watts during eclipse. The battery weight has been broken out in the above manner as battery power requirement (and hence weight) varies significantly with payload and it is considered to be the one component that could realistically be modified to take advantage of reduction in eclipse power requirement (due to its cellular construction).

4.2.1.9 Harness

The harness weight of 35 lbs. is based on CTS experience.

4.2.1.10 Safe and Arm and Ballast

The weight of 3 lbs. for the safe and arm is based on CTS experience, and 5 lbs of ballast is available. It is also suggested that the S&A be used as ballast and located in the most optimum location to assist spacecraft balance during the early phase of design.

The above weights add up to a total of 1,607 lbs. and 1587 lbs. for the Bus (depending on payload power requirement) to which the payload must be added to obtain the total weight of the spacecraft.

The payload consists of a transponder, antenna, battery (that portion used to operate the transponder payload) and a security box for military payloads (UHF only).
(Weights for payloads are shown on table entitled WEIGHT SUMMARY: PAYLOAD OPTION DETAILS).

4.2.2 Moment of Inertia Ratio (See Table 4.2.2-1)

The minimum moment of inertia ratio (lesser of two ratios Izz/Ixx or Izz/Iyy), for the present spacecraft configuration with payload option "a" is 1.03. This is less than the required minimum of 1.10, however still considered by specialists to be within stability requirements (1.025mm). The changes listed below would improve the moment of inertia ratio. This is essentially a listing of sensitive components, which effect the I.R. significantly and on which sensitivity studies should be conducted to drive the optimum baseline configuration to the value specified if considered necessary.

7/CD/63

MOMENT OF INERTIA RATIO

Table 4.2.2-1

M. of I. Ratio with Antenna Weight at 55 lbs. and CG at Station 77.22

$$Ixx = W\,(y^2 + z^2) + Ixp - Wt\,(\bar{y}^2 \times \bar{z}^2)$$

$$= 3311694 + 2749028\ 221043 - 1910.2 \times 36.36^2$$

$$= \underline{756386}\ \text{lb.in.}^2$$

$$Iyy = W\,(x^2 + z^2) + Iyp - Wt\,(\bar{x}^2 + \bar{z}^2)$$

$$= 220178 + 2749028 + 349212 - 1910.2 \times 36.36^2$$

$$= \underline{793039}\ \text{lb.in.}^2$$

$$Izz = W\,(x^2 + y^2) + Izp - Wt\,(\bar{x}^2 + \bar{y}^2)$$

$$= 220178 + 311694 + 285451 - 0$$

$$= \underline{817323}\ \text{lb.in.}^2$$

$$\text{M. of I. Ratio} = \frac{Izz}{Iyy} = \frac{817323}{793039} = 1.03$$

7/CD/67

a) Move the dish antenna down by shortening the mounting dish base cone.

b) Move the separation plane up.

c) Move the N_2H_4 tanks radially toward the N/S panels.

d) Shorten the stowed length (Z direction) of the solar panels.

e) Place ballast on N and S panels.

f) Move the batteries forward.

g) Shorten the spacecraft by lowering the forward platform which may require a hole in the forward platform for the apogee motor and also requires that the stowed length (Z direction) of the solar panels be shortened.

The most effective changes from the above list are a), c) and g), and realistic modifications in these three in combination would result in a moment of inertia ratio of approximately 1.12.

5.0 SUBSYSTEM TECHNICAL DETAILS

The information provided in this section and Volume II, Section 2.1.2 includes the technical details of the bus subsystem; trade-off studies that occurred during the study and resulted in the selection of specific subsystem configurations, and parameter assumptions used in sizing and configuring the bus.

On certain subsystems it will be noted that the text is not consistent with the final configurations selected since initially it was decided to use state of the art technology in sizing subsystems. Toward the end of the study it was recognized that advanced technology would be available for a satellite launched in the period 79/80 with little risk, and consequently these have been included as part of the baseline design. Subsystems in the category given above are:

(a) ACS baseline will be a hybrid system using MASS.

(b) RCS baseline is using super heated electro-thermal engines.

(c) DSA baseline will utilize violet solar array cells.

5.1 Attitude Control Subsystem Description

The Attitude Control Subsystem (ACS) is configured to provide attitude stablization and control capabilities for the General Purpose Bus Satellite during the following operational phases of the mission:

(a) Spin Stablized Transfer Orbit
(b) Attitude Acquisition Phase
(c) Three-Axis Stablized Phase
(d) Orbit Adjust Manoeuvers

The primary objective of the ACS is to maintain the spacecraft pointing accuracy, during the three-axis stabilized phase, within $\pm$ 0.15° in roll and pitch and $\pm$ 1.0° in yaw. Additional major design considerations are:

(a) Eight year life.

(b) Autonomous Attitude Acquisition.

(c) Stationkeeping accuracy of $\pm$ 0.05° North-South and East-West with orbit adjust intervals scheduled on a 21 day operational cycle.

(d) Capable of a pitch slew manoeuvre of up to $\pm$ 0.5° to accommodate a change of station longitude.

5.1.1 Attitude Control System Trade-Off

For the purposes of a trade-off comparison of potential Attitude Control Systems, a performance specification has been prepared with parameters appropriate for the Multi-Purpose Bus Satellite. This specification is included in Volume II.

The detailed parameters contained within the specification may change somewhat without substantially altering the conclusions reached herein.

This trade-off is based on information gathered as a result of CTS experience and recent vendor surveys, and reflects information available to Spar at this time.

5.1.1.1 Identification of Major Trade Parameters

In order to arrive at a rational choice of candidate control systems, it is necessary to quantify and compare a number of major parameters, including the following:

(a) Accuracy

Both angular pointing accuracy as well as angular rate accuracy must be considered in light of specification requirements.

(b) Power

Power consumption during the on-orbit phase requires the provision of sufficient solar array area and battery capacity to service the attitude control system. Power consumption may be expressed as a "power equivalent weight" by means of the following factors:

i) Array - 0.10 lb/watt
ii) Battery - 0.20 lb/watt

The battery power equivalent weight factor is based on the latest available information for space qualified batteries and the solar array factor is based on a straight line fit between an 835 watt and 625 watt (end of life) "rigid" array design. The total power equivalent weight factor is 0.3 lb/watt.

(c) Weight

For most spacecraft, weight is a very important factor. For purposes of trade-off, the total weight should include not only the control system hardware weight, but also the RCS fuel weight required by the system, the weight of additional RCS engines required for system operation (as, for example, the offset engines required by Whecon), the power equivalent weight and any other overall weight impacts resulting from the system.

(d) Reliability

A basic philosophy normally followed for extended life spacecraft is the elimination of potential single point failures, regardless of the system numerical reliability. This redundancy philosophy has significant weight impact, but has been followed for the candidate systems under consideration.

(e) Cost

Cost is often considered to be as important as weight for comparison purposes. The component cost data available for this tradeoff must be considered preliminary since not all potential component vendors have been fully exercised. In addition, component costs are quite sensitive to program details.

5.1.1.2 Baseline Variables

The independent baseline variables used for this trade study are as specified in Attitude Control Subsystem Specification.

5.1.1.3 Candidate Systems

Three candidate control systems, a momentum bias system (Whecon), a reaction wheel system and a hybrid reaction wheel with thruster system are considered the most favourable candidate systems.

The momentum bias systems have been specifically configured such that yaw sensing on a continuous basis is not required. Until the introduction of the Microwave Attitude Sensing System, continuous yaw sensing has been difficult to accomplish for long life spacecraft. Of the various momentum bias schemes, the Whecon or fixed wheel with offset thruster system, has been chosen to satisfy the performance requirements with minimum cost, weight and maximum reliability. (Reference SPAR-R.664, Technology Survey of Attitude Control Techniques, April, 1975)

A fixed wheel with ground command correction (no on-board controller), though potentially providing the lightest, most reliable system, requires frequent ground correction and is, therefore, not considered sufficiently autonomous for an operational satellite. The remaining momentum bias schemes with additional reaction wheels or gymbals have been configured to provide additional performance features such as roll and yaw offset pointing and high resolution control. Such additional features are not required for the Multi-Purpose Bus Satellite and, for reasons of economy, weight and reliability, have not been considered as potential candidates.

It is suspected that the Whecon system, as originally conceived by Lockheed, has been used in classified missions, though probably of short duration. With modifications included to limit nutation growth and thruster activity, the Whecon system was chosen for the CTS. Subsequently, Whecon (or equivalent) systems have been chosen for the FLEETSATCOM, RCA SATCOM, OTS and the Hughes three-axis satellite.

A modification of the Whecon system proposed for SATCOM includes the use of electromagnets interacting with the earth's magnetic field, both to compensate for secular disturbance torques through current loops in the solar arrays and to provide vernier control with pulsed magnets. Independent trade studies conducted during definition of the CTS control system, as well as other studies, (Reference, SPAR-R.666, Technology Survey of Critical ACS Components for the Multi-Purpose UHF Satellite Project, April, 1975) would indicate that any fuel savings using this approach would marginally affect total system weight. In addition, there is disagreement as to the frequency of magneto-pause crossings at synchronous altitude which could adversely effect the magnet system utility. Because of the marginal performance benefit and increased complexity, the magnet system has not been included with the candidate Whecon system.

Low momentum or reaction wheel systems have been used in the past for various spacecraft missions. General Electric, in particular, has quite an extensive background with these systems and is, in fact, using a three reaction wheel system for the Japanese BSE. The major advantages of these systems is the very fine pointing and rate accuracy attainable as well as the ability to perform offset pointing or other manoeuvres. A major disadvantage is the requirement for continuous yaw sensing, which for the BSE is accomplished with a Microwave Attitude Sensor (MASS) plus earth sensor combination. With the addition of a polarization angle measurement (POLANG) to the MASS, the earth sensor may be dispensed with, resulting in a weight attractive sensor system.

The zero momentum system (without wheel), utilizing control thrusters about each of the control axes, is not a viable alternative for the Multi-Purpose Bus Satellite. The rather large, but purely periodic pitch disturbance torque and small pitch inertia, would result in a pitch control fuel consumption of approximately 60 lb for six years. Since this major inefficiency occurs only about the pitch axis, the use of a pitch reaction wheel to absorb the periodic disturbance torques is a possible alternative. Thus, the hybrid zero/low momentum system using the MASS with POLANG has promise as an attractive control system and is included as one of the prime candidates.

5.1.1.3.1 Common System Components

Each of the candidate systems include a number of common components, used for the spin phase, acquisition and array tracking. These components, whose major parameters are listed in Table 5.1.1.3.1-1, include the following:

(a) Spinning Horizon Sensor

This sensor, employing a narrow field-of-view infrared telescope, scans across the earth disk as the spacecraft rotates about its spin axis. The resulting earth scan signature provides a measure of the earth chord length. With two such sensors, one pointing 5° above and the other 5° below the spacecraft equatorial plane, a measure of the angle between the spin vector and the earth line is obtained from the difference between the two sensor chord length outputs.

(b) Spinning Sun Sensor

The spinning sun sensor provides two outputs:

i) A sun pulse generated as the sun passes through a plane containing the spacecraft spin axis. This sun pulse is used to phase thruster pulses during precession manoeuvres.

ii) Angle between the spin vector and the sun line. This angle and the angle to the earth line together determine the spacecraft spin axis direction.

(c) Array Track Sun Sensor

This sensor, with a relatively narrow fan-shaped field-of-view, is mounted on the solar array and used to provide automatic array sun tracking and initialize a clock driven array tracking operation.

TABLE 5.1.1.3.1-1

COMMON CONTROL SYSTEM COMPONENTS

	Component	Weight (lb)	On-Orbit Power (Watts)	Mission Reliability	ROM COST Non-Recurring	ROM COST Recurring (per S/C)
(a)	Spinning Horizon Sensor	3.0	-	0.998	Detail Information Relating to Costs is Available on Request.	
(b)	Spinning Sun Sensor	1.84	-	0.998		
(c)	Array Track Sun Sensor	0.52	0.1	0.996		
(d)	Acquisition Sun Sensor	0.36	-	0.999		
(e)	Nutation Damper	0.9	-	0.9999		
(f)	Control Electronics	18.0*	15.0*	0.96*		
(g)	Bracketry	1.5*	-	-		
	TOTALS	26.12	15.1	0.96 (On-Orbit Reliability)		

*Estimated Quantities

(d) Acquisition Sun Sensors

These sensors, consisting of clusters of analogue "eyes", provide a complete spherical field-of-view allowing attitude acquisition to occur from any spacecraft orientation.

(e) Nutation Damper

The nutation damper, consisting of a closed tube partially filled with mercury, is aligned along the spin axis and mounted near the periphery of the spacecraft. Passive nutation damping occurs as a result of energy dissipation through surface waves and sloshing of the mercury within the tube.

(f) Control Electronics

The control unit considered here consists of redundant programmable computers and an interface box. The electronics unit is included as a common component since the candidate control systems will have little impact on the major parameters for this unit.

5.1.1.3.2 <u>The Whecon System</u>

A block diagram of the Whecon system is shown in Figure 5-1. The pointing accuracy of this system will be $\pm$ 0.14° in roll and pitch and $\pm$ 1° in yaw. Rate accuracy would be approximately $\pm$ 0.003 deg/sec. In addition to the common components, the candidate Whecon system includes components whose major parameters are listed in Table 5.1.1.3.2-1, as follows:

(a) Horizon Sensor

In order to meet the pitch slew requirement without accuracy degradation, a scanning infrared horizon sensor is assumed for measurement of roll and pitch angles. Since the East/West stationkeeping accuracy is $\pm$ 0.05°, earth chord length changes resulting from

R677/89

WHECON ATTITUDE CONTROL SYSTEM

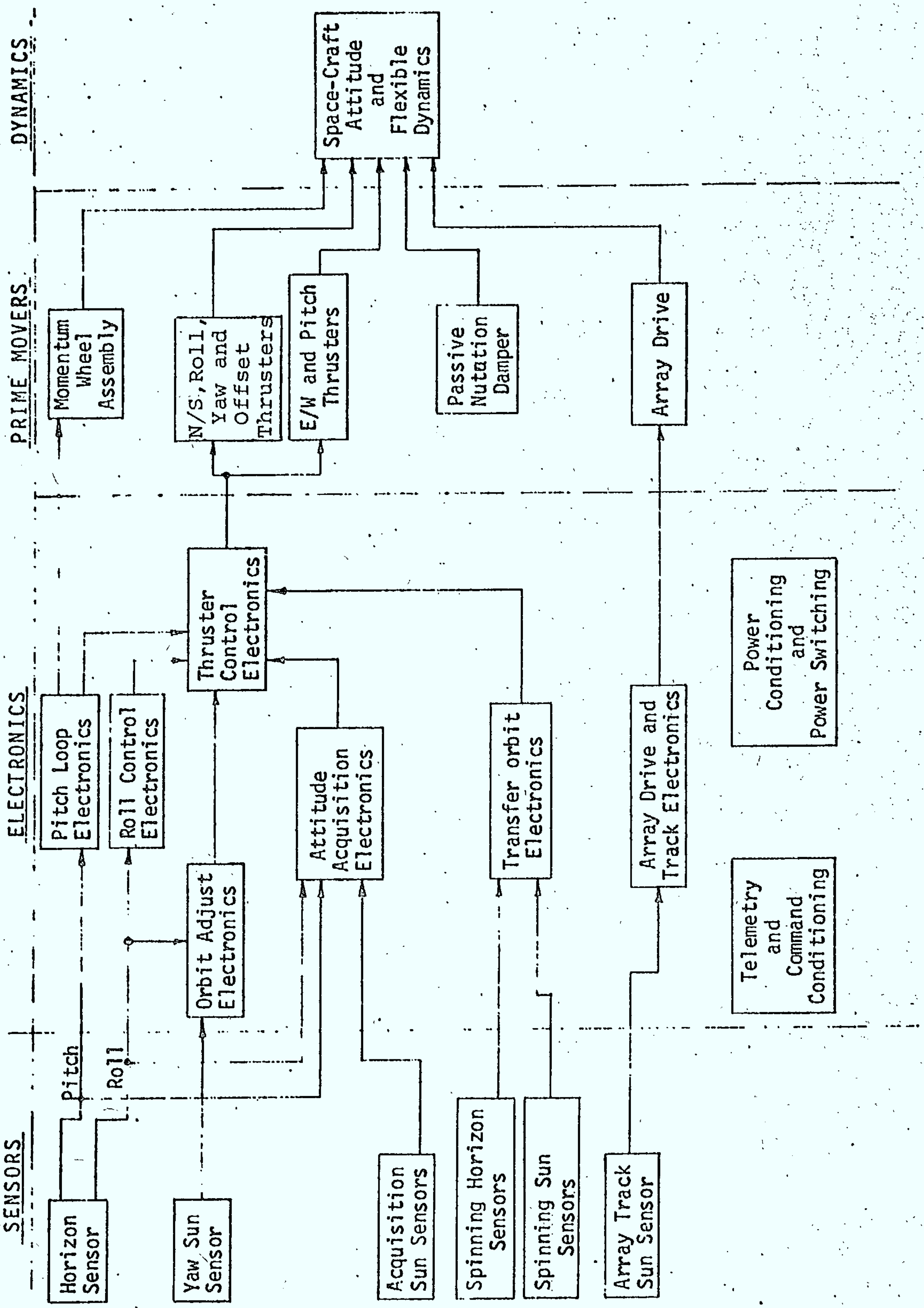


FIGURE 5-1

TABLE 5.1.1.3.2-1

WHECON SYSTEM COMPONENTS

	Component	Weight (lb)	On-Orbit Power (Watts)	On-Orbit Reliability	Cost ($K) Non-Recurring	Cost ($K) Recurring
(a)	Horizon Sensor	12.0	4.5	0.988	Detail Information Relating to Costs is Available on Request	
(b)	Momentum Wheel					
	- Conventional Bearing	34.0	15.0	0.978		
	- Magnetic Bearing	24.5	2.5	0.995		
(c)	Yaw Sun Sensor	4.0	1.0	0.990		
(d)	Common Components	26.12	15.1	0.960		
	Totals:					
	- Conventional Wheel	76.12	45.6	0.918	1,043	1,248
	- Magnetic	66.62	23.1	0.934	1,360	1,236
	Weight Equivalent Power:					
	- Conventional Wheel	13.68				
	- Magnetic Wheel	6.33				
	RCS Weight:					
	- Offset Thrusters	3.0				
	- Fuel (six years)	9.0				
	Total Weight :					
	- Conventional Wheel	101.8 lb				
	- Magnetic Wheel	84.95 lb				

altitude variation will be of the order ± 0.005° thus allowing the CTS sensor to be used without modification. This sensor employs a single scanning field-of-view with chord length comparison to a preset nominal chord length providing cross-scan angle calculation. Two sensors would be employed for redundancy and to eliminate sun interference effects. One sensor would scan 5° above the earth equator in an east/west direction and the other would scan 5° below the equator.

(b) Momentum Wheel

In order to meet the yaw pointing accuracy requirement a wheel of 15 ft.lb.sec capacity is assumed. With conventional ball bearing wheels, redundant units are generally considered necessary to meet a six to eight year life requirement. The CTS momentum wheel with a slightly modified bearing configuration would be used. As an alternative, a magnetic suspension wheel may be considered. Since the bearing wearout mechanism is eliminated, only a single wheel need be used. The levitation system, electronics and drive motor would be completely redundant.

(c) Yaw Sun Sensor

Two digital sun sensor heads measuring the "vertical" (elevation) and "parallel" (slant azimuth) components of the sun line will provide a yaw angle measurement during all periods during which north/south stationkeeping must be performed. The vertical sensor provides yaw angles within ± 30° of the sun line. The parallel sensor is an identical head, rotated through 90°, and will fill the gap during solstice seasons, (but not during equinox seasons), thus eliminating a requirement for a rate integrating gyro. This sensor complement will allow orbit inclination control to ± 0.05°.

5.1.1.3.3 The Reaction Wheel System

The reaction wheel system requires yaw sensing on a continuous basis. The use of sun sensors with rate integrating gyros is not considered sufficiently reliable for a six to eight year mission. Thus, the MASS with earth sensor or MASS with POLANG are the only alternatives apparent at this time. In addition, the MASS with earth sensor combination is not considered a weight effective combination. Thus, only the MASS with POLANG will be considered. The reaction wheel system will include the common components as well as the following as shown in Table 5.1.1.3.3-1:

(a) MASS with POLANG

Since this sensor is not yet developed, only estimates of weight and power can be considered. The presently developed non-redundant MASS without POLANG weighs 8 lb and requires 10 watts. It is estimated that a redundant MASS with POLANG would weigh 16.8 lb and require 12 watts of power.

(b) Reaction Wheels

Conventional ball bearing reaction wheels of 1.5 ft.lb.sec capacity weigh 8 lb with a power requirement of 4.5 watts each. With four skewed wheels, three operational at any time, the total reaction wheel assembly weight would be 32 lb and would require 13.5 watt power.

Magnetic bearing reaction wheels of 1.5 ft.lb capacity would weigh 11 lb each and require 1 watt. Only three wheels would be required.

A block diagram of the reaction wheel system is shown in Figure 5-2. The accuracy of this system will be 0.14° in roll and pitch, and 0.6° in yaw with rate control probably an order of magnitude better than the Whecon system although analysis will be required to justify this claim.

TABLE 5.1.1.3.3-1

REACTION WHEEL SYSTEM COMPONENTS

	Component	Weight (lb)	On-Orbit Power (Watts)	Mission Reliability	ROM COSTS Non-Recurring	ROM COSTS Recurring (per S/C)
(a)	MASS (with POLANG)	16.8	12.0	0.985	Detail Information Relating to Costs is Available on Request	
(b)	Reaction Wheels					
	- Conventional Bearings	32.0	13.5	0.970		
	- Magnetic Bearings	33.0	3.0	0.985		
(c)	Common Components	26.12	15.1	0.96		
	Totals:					
	- Conventional Wheel	74.92	40.6	0.917	1,410 K	1,491 K
	- Magnetic Wheel	75.92	30.1	0.931	1,710 K	1,491 K
	Weight Equivalent Power:					
	- Conventional Wheel	12.18				
	- Magnetic Wheel	9.03				
	RCS Fuel (six years)	4.0				
	Total Weight:					
	- Conventional Wheel	91.10				
	- Magnetic Wheel	88.95				

REACTION WHEEL CONTROL SYSTEM

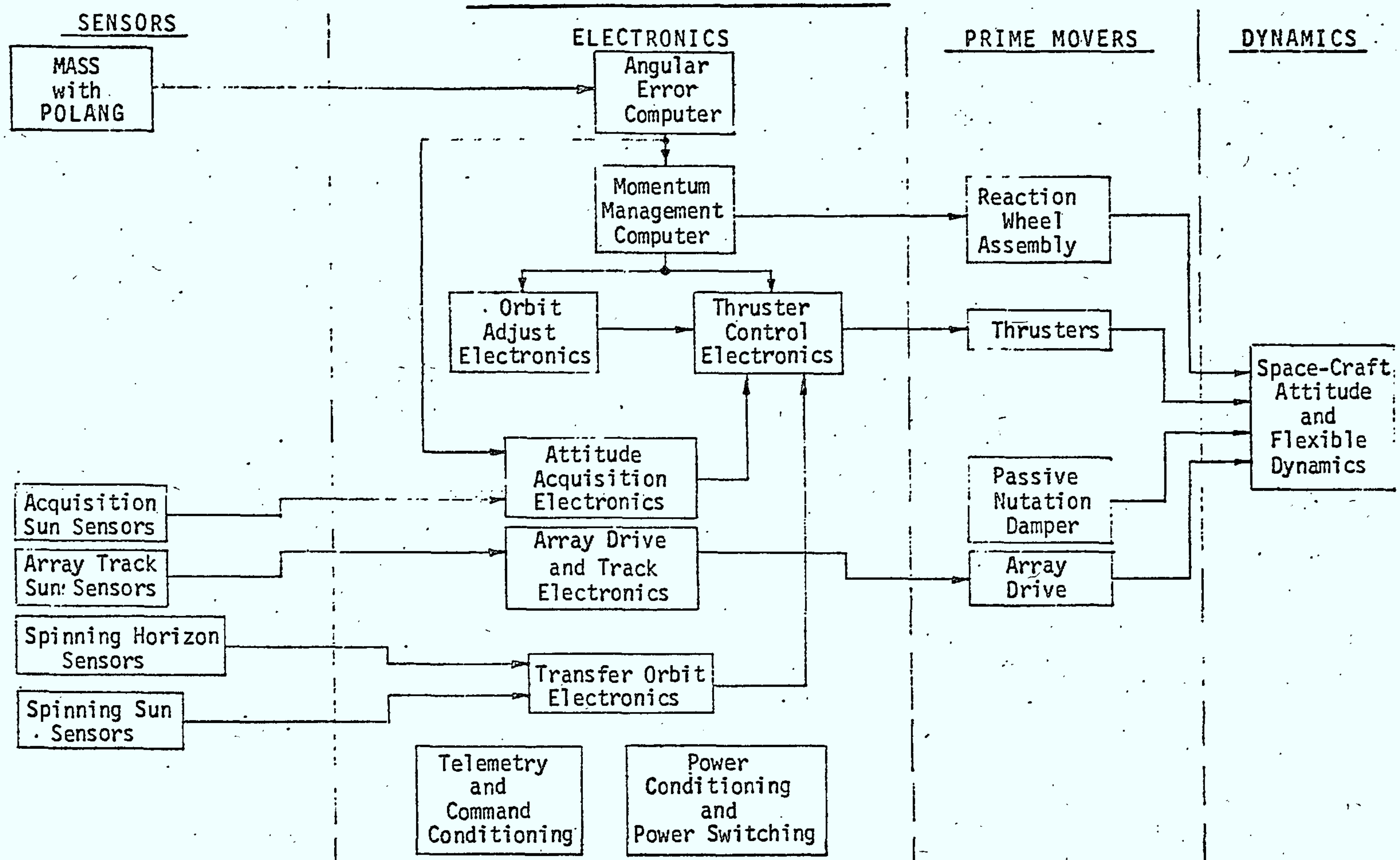


FIGURE 5-2

5.1.1.3.4 A Hybrid System

Due to simplicity and extensive flight background, an all gas zero momentum system has considerable merit. However, for the present configuration, gas control about pitch would impose a serious weight penalty due to the small moment of inertia and large periodic torque. A hybrid system consisting of a pitch reaction wheel and thruster control about roll and yaw is worthy of consideration. In this case, only two conventional bearing reaction wheels or a single magnetic bearing wheel would be required. Attitude sensing would be obtained with the MASS as in the reaction wheel system.

Pointing accuracy for this system would be ± 0.14° in pitch, ± 0.16° in roll and ± 0.7° in yaw. Rate accuracy would be similar to the Whecon system. The component major parameters for this system are presented in Table 5.1.1.3.4-1.

5.1.1.2.4 Trade Study Matrix

Table 5.1.1.4-1 is a summary of the major parameters for each of the candidate systems. The "total weight" shown in this matrix includes the subsystem weight plus weight equivalent power, and RCS weight required for system operation. It is apparent that the reaction wheel system has little to offer over the Whecon or Hybrid systems. The following salient features may be identified:

(a) The use of magnetic bearing wheels results in a significant weight saving over conventional bearing wheels (through elimination of redundant wheels and power requirement reduction) as well as an increased system reliability. The additional cost for a magnetic bearing wheel has a small effect on total subsystem recurring costs since non-redundant wheels may be used.

(b) The development of the Microwave Attitude Sensing system with POLANG would allow use of the Hybrid system which is the lightest of all the candidate systems. If, in addition, the magnetic bearing reaction wheel is considered, the resulting Hybrid system is very attractive. This system is worthy of additional study. In particular, an optimization of the

TABLE 5.1.1.3.4-1

HYBRID SYSTEM COMPONENTS

	Component	Weight (lb)	On-Orbit Power (Watts)	Mission Reliability	ROM COSTS Non-Recurring	ROM COSTS Recurring (per S/C)
(a)	MASS (with POLANG)	16.8	12.0	0.985	Detail Information Relating to Costs is Available on Request	
(b)	Reaction Wheels					
	- Conventional Bearings	16.0	9.0	0.978		
	- Magnetic Bearings	11.0	1.0	0.995		
(c)	Common Components	26.12	15.1	0.96		
	Totals:					
	- Conventional Wheels	58.92	36.1	0.925	1,410 K	1,341 K
	- Magnetic Wheels	53.92	28.1	0.941	1,710 K	1,291 K
	Weight Equivalent Power:					
	- Conventional Wheels	10.83				
	- Magnetic Wheels	8.43				
	RCS Fuel (six years)	12.0				
	Total Weight:					
	- Conventional Wheels	81.75				
	- Magnetic Wheels	74.35				

TABLE 5.1.1.4-1

ACS SYSTEM TRADE MATRIX

System / Major Parameter	WHECON Conventional Bearing Wheel	WHECON Magnetic Bearing Wheel	REACTION WHEEL Conventional Bearing Wheel	REACTION WHEEL Magnetic Bearing Wheel	HYBRID Conventional Bearing Wheel	HYBRID Magnetic Bearing Wheel
Subsystem Weight	76.12	66.62	74.92	75.92	58.92	53.92
Total Weight	101.8	84.95	91.10	88.95	81.75	74.35
On-Orbit Reliability (six years)	.918	.934	0.917	0.931	0.925	0.941
Component Costs ($K)						
- Non-Recurring	1,043	1,360	1,410	1,710	1,410	1,710
- Recurring (per S/C)	1,248	1,236	1,491	1,491	1,341	1,291

RCS interface could result in further total weight reductions.

(c) It has been suggested that the ground beacon required for MASS, even if redundant, cannot be considered sufficiently reliable to exclude a back-up sensing capability. This would imply not only redundant infrared horizon sensors would be required, but the yaw sun sensor and integrating rate gyro would also be necessary to provide a three-axis back-up sensing system. The additional weight of 20 lb for this back-up capability would likely exclude a Hybrid system from consideration. To overcome this suggested deficiency, it is considered likely that redundancy and a preventive maintenance program for the ground beacon could be devised to assure continuous operation.

5.1.1.5 Trade Result

The Whecon control system with conventional bearings has been chosen, at this time, for the General Purpose Bus Satellite primarily because minimum development is required for this system. In addition, considerable experience has been gained on a Whecon system through the CTS program. However, it should be noted that a total spacecraft weight saving of almost 17 lb may be realized if necessary, through the use of the magnetic bearing wheels. Further, a total spacecraft weight saving of approximately 27 lb could be realized with the Hybrid system with a magnetic bearing wheel.

5.1.2 ACS Operation Description

5.1.2.1 Three-Axis Stabilized Operation

The Whecon (WHEel CONtrol) employs a single fixed momentum wheel, a pair of "offset" thrusters and an earth horizon sensor. Gyrocompassing is used to transfer yaw errors into roll errors for detection by the roll horizon sensor. With a wheel momentum sufficiently large compared with the disturbance torque environment, no yaw sensing is required. When the horizon sensor detects a roll error, instead of a pure roll correction, a combined roll and yaw motion is created by thrusters which are offset from the vehicle principle axes. The offset jets reduce the yaw response time to significantly less than one quarter of an orbit period and introduce yaw damping.

The roll response is damped by electronically filtering the roll horizon sensor signal to introduce rate compensation. Nutation limiting and damping logic is incorporated to suppress nutation growth due to thruster pulses, thus eliminating any possible limit cycle motion. This results in efficient utilization of RCS fuel. Pitch control is achieved by varying the momentum wheel spin rate.

R677/99

5.1.2.2 Spin Stabilized Operation

During the transfer orbit phase of the mission, the ACS provides the support necessary for ground based attitude control. The spinning control system comprises spinning earth sensors and spinning sun sensors linked to the transfer orbit electronics, thruster control electronics and a passive nutation damper. All control is exercised by ground processing of the sensor telemetry data and ground commanded thruster generation.

5.1.2.3 Attitude Acquisition Operation

The acquisition phase is the transition between the spin stabilized configuration and the three-axis stabilized configuration. This phase begins with the initiation of despin and extends to the turn-on of the on-orbit controller.

The acquisition control system is autonomous with initiation of the sequence commanded from the ground. A number of ground control overrides are available for non-standard or back-up operation. The initiation command may be sent at any point in the orbit. The sequence is designed such that the same controller and mode control logic is compatible with an automated reacquisition in the event of a failure during the on-orbit phase.

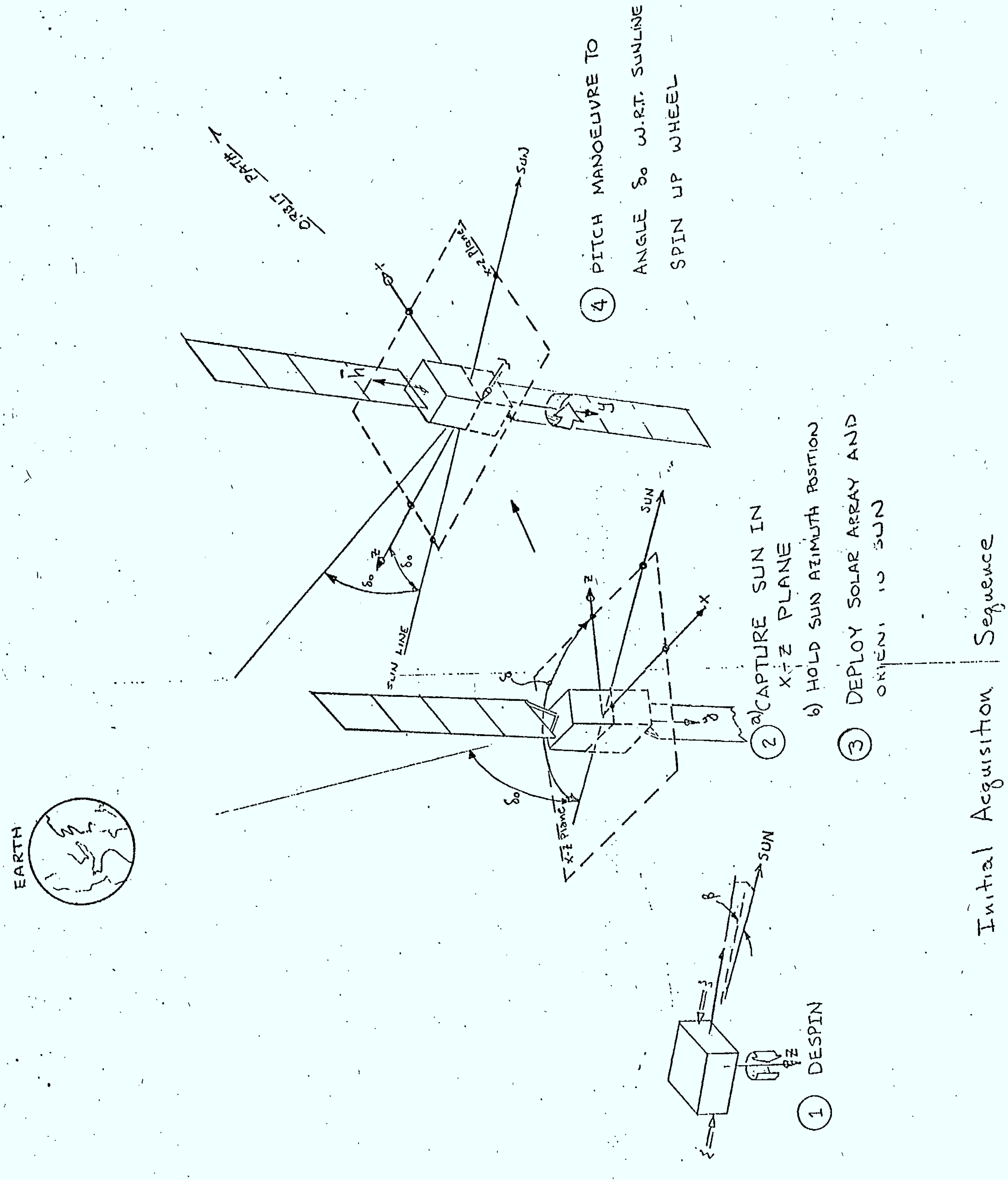
The only sensors required for acquisition are the on-orbit non-spinning earth sensor and a full spherical field-of-view sun sensor. Roll, pitch and yaw thrusters are used as well as the on-orbit controller and momentum wheel.

The acquisition sequence is outlined in Table 5.1.2.3-1 and Figures 5-3a and 5-3b. It consists of a despin to a low rate, followed by a sun capture and hold mode during which the satellite is stabilized to the sun-line while the solar array is deployed and oriented to the sun. When the solar array is tracking the sun, the aximuth reference is changed to the angle $_o$, the angle between the sun line and earth line. This causes an azimuth error which is corrected by the

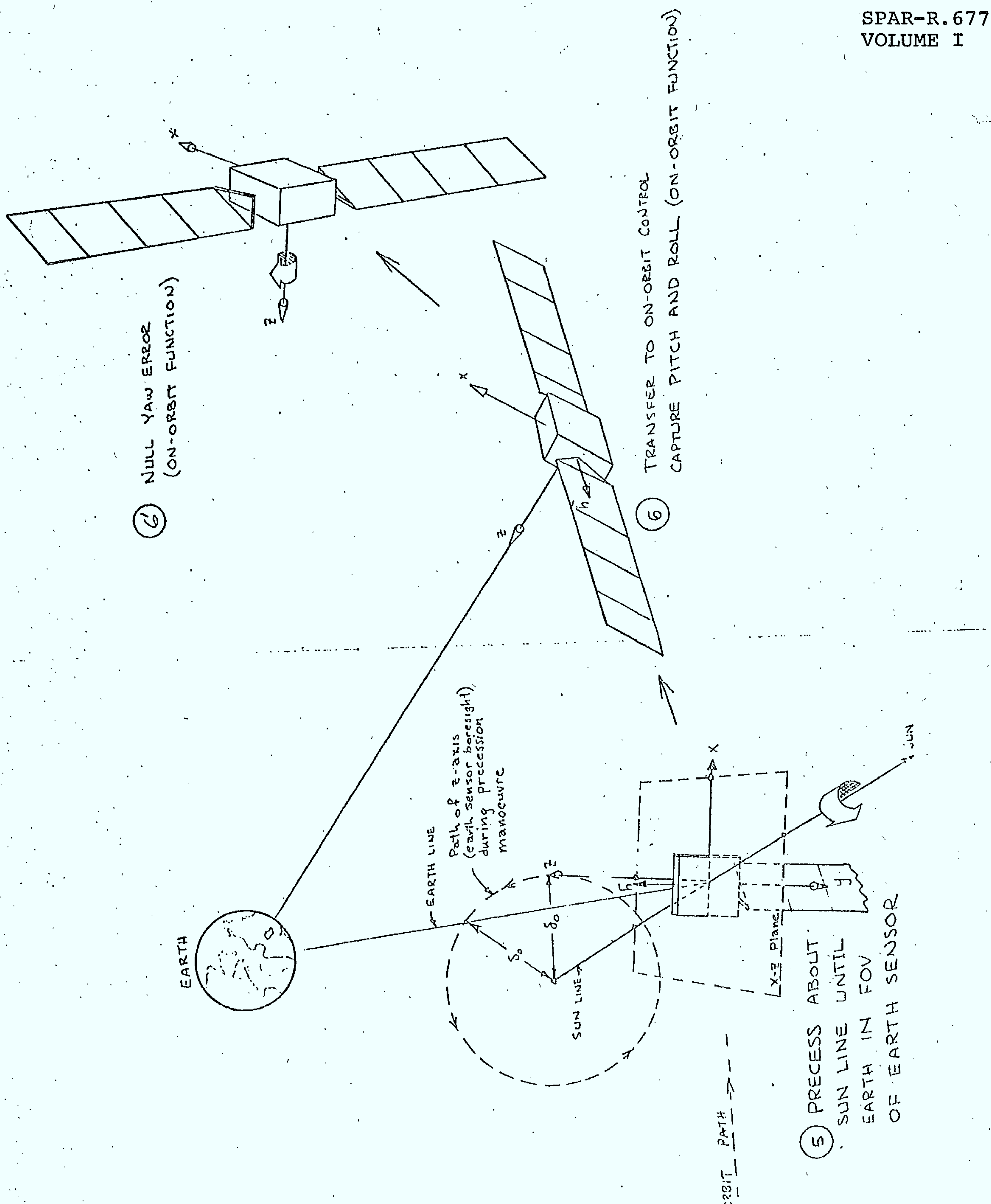
TABLE 5.1.2.3-1

ACQUISITION SEQUENCE OUTLINE

Step		Description	Allotted Time/ Impulse Expenditure
1	Despin	Despin satellite to less than 1-2 rpm.	15 min 246 lbf-sec
2	Sun Acquisition		
	a) X-Z Plane Capture	Use roll and yaw thrusters and elevation channel of AAEA to capture sun-line in A-Z plane.	
	b) Aximuth Capture	When elevation transient has settled, enable azimuth channel and capture to present value of azimuth. Partially spin-up wheel.	20 min. 10 lbf-sec
3	Sun Hold and S/A Deployment	Hold sun at the reference point in the sun sensor FOV. Deploy solar array and orient to sun. Control reaction torques with sun-line controller.	Time fixed by Deployment negligible fuel
4	Pitch Manoeuvre and Wheel Spin-up	Manoeuvre in pitch to place Z-axis at angle ${}_{o}$ from sun-line. Simultaneously bring wheel up to nominal speed.	30 min 7.5 lbf-sec
5	Precession Manoeuvre	Precess about the sun-line using sun-line controller with bias input. Maintain Z-axis at required angle with respect to sun-line.	Up to 90 min 50 lbf-sec
6	Transfer to On-orbit Control	When earth image in earth sensor FOV and attitude and rates within bounds, enable OCEA, disable AAEA.	-



Initial Acquisition Sequence



Initial Acquisition Sequence

FIGURE 5-3b

sun-line controller with a rotation about the pitch axis. Null is attained when the Z-axis of the satellite is at a known angle from the sun-line. Simultaneous to the pitch manoeuvre, the wheel is spun up to nominal speed in preparation for the final precession manoeuvre.

The final precession, to acquire the earth image in the non-spinning earth sensor FOV, is accomplished by pulsing the roll and yaw thrusters in the proper ratio to cause precession about the sun-line. The Z-axis (earth sensor boresight) is thus manoeuvred to intersect the earth. When the earth image is within the FOV, control is transferred to the on-orbit controller which performs the remainder of the corrections to null attitude errors and place the satellite in the on-orbit, three-axis stabilized configuration. When the on-orbit controller assumes control, the acquisition controller is deactivated and reset for reactivation in the event of a failure which would require a reacquisition.

The above acquisition scheme has been configured to provide a positive automatic acquisition or re-acquisition. Should it be considered possible to perform any re-acquisition through ground control, a somewhat simpler initial acquisition scheme may be considered utilizing the "momentum exchange principle" identified in early CTS definition studies. This scheme includes the following steps:

(a) Despin the spacecraft to low speed.

(b) Precess the spin axis normal to the orbit plane.

(c) Energize the wheel and allow the damper to remove nutational energy.

(d) Deploy arrays and rotate spacecraft in pitch to normal on station orientation.

(e) Enable the on-orbit controllers.

This scheme is relatively simple and would reduce the control system complexity with the disadvantages of possibly stressing the wheel bearings at the start of the mission as well as requiring a slightly heavier fluid ring damper rather than the light tuned nutation damper, and requiring additional ground control.

5.1.3 Control System Major Parameters

The major control system parameters are:

(a) Pointing Accuracy

Control Axis	Accuracy During Orbit Adjust Thrusting	Accuracy Design Goal
Pitch	±0.15°	±0.1°
Roll	±0.15°	±0.1°
Yaw	±1.0°	±1.0°

(b) Subsystem Weight - 76.12 lb

(c) Six year reliability - 0.918

(d) The On-Orbit Power Requirement - 45.6 watt

5.2 Reaction Control Subsystem Design Definition

5.2.0 General

This section reports on the project definition for the Reaction Control Subsystem (RCS) for the General Purpose Bus Satellite. The work was performed and is presented in the following sequence:

a) RCS subsystem requirements definition,

b) RCS preliminary design based on CTS technology,

c) RCS preliminary design specification, SPAR-SG.350, and preliminary Statement of Work, SPAR-SOW.071 generated to encompass all viable bidders, except colloid, pulsed plasma and ion engine (electric propulsion), but based on CTS technology,

d) Release of budgetary RFP to viable bidders,

e) Receipt and evaluation of budgetary estimates from viable bidders,

f) RCS present baseline design for GPB,

g) RCS implementation plan and schedule,

h) Comparison of subsystem performance with other spacecraft designs.

Volume I, Section 4.2, reports on items (a) through (c). Volume II, Section 2.1, reports on items (d) through (f). Volume III, Section 2.4, presents item (g). Item (h) is discussed in Volume IV.

5.2.1 Subsystem Parameters and Interface Assumptions

The RCS has been synthesized from spacecraft and mission parameters presented below.

5.2.1.1 Mission

a) Thor Delta 3914 launch vehicle with launch pad capability of 1925 lbm excluding attach fitting and launch vehicle telemetry. Ref. 1, paragraph B.8.3. This yields a spacecraft mass just after apogee motor firing of 1017 lbm taking into account pre-apogee precession fuel usage (AM V = 6023 ft/sec and ISP = 294 lbf sec/lbm).

b) Spin stabilized transfer orbit, 3-axis stabilized synchronous orbit.

c) Pre-apogee precession angle required of 129.5°. Ref. Appendix A. (Including 2% inefficiencies).

d) Post-apogee precession angle required of 68.3°. Ref. Appendix A. (Including 2% inefficiencies).

e) Spin axis moment of intertia prior to apogee motor fire is 179.7 slug ft^2. (scaled for s/c mass on pad of 1925 lbm).

f) Spin axis moment of inertia just after apogee motor fire is 138 slug ft^2 (scaled for s/c mass on pad of 1925 lbm).

g) Spacecraft spin rate in spinning mode is 60$\pm$ 6 rpm.

h) Attitude acquisition shall be performed prior to station acquisition.

i) Attitude acquisition shall be comprised of the following manoeuvres - momentum wheel spinup 60 ft.lbf.sec. at pitch momentum dumping duty cycles.

- 3-axis limit cycle 80 ft.lbf.sec. at very low duty cycle,
- onboard roll-yaw capture 40 ft.lbf.sec. at very low duty cycle.

These requirements have been estimated from CTS data.

j) Station acquisition delta velocity maximum requirements, 99% confidence level shall be 160 feet per second. Ref. 2. Assume that this requirement is split 80 feet per second in-plane and 80 feet per second out-of-plane.

k) On-orbit life requirement shall be hardware capable of 8 years operation but with a present baseline fueled to 6 years operation. Ref. 1 paragraph B.9.

l) Ground shelf life unfueled of 2 years. Ref. DOC/Spar meeting held at Spar 26 February, 1975.

m) Launch dates shall be within time frame early 1979 until end 1985.

n) On-orbit roll-yaw solar torques of 2×10^{-6} ft.lbf. secular plus 5×10^{-6} ft.lbf. cyclic with season (maximum at solstice) which, with the recommended "WHECON" attitude control subsystem, requires "offset" thrusters producing 314.2 ft.lbf.sec. per year continuous pulsing with torque impulse bits less than or equal to 4.0×10^{-3} ft.lbf.sec. (Note: torque impulse bits greater than 4.0×10^{-3} but less than or equal to 8.0×10^{-3} ft.lbf.sec would be allowed but with an associated spacecraft weight penalty (momentum wheel weight) of 10 lbm maximum at 8×10^{-3} ft.lbf.sec., 0 lbm at 4×10^{-3} ft.lbf.sec and linear in between).

o) On-orbit pitch momentum dumping requirement shall be less than or equal to 1 ft.lbf.sec. to be dumped every 21 days.

p) As a design goal, east/west stationkeeping will not be performed more frequently than every 21 days, Ref. 1 paragraph B.4 Impulsive delta velocity required shall be 14 feet per second per year. Ref. 2. Each 21 day manoeuvre shall be performed around 6 a.m. and 6 p.m. orbit slot throughout the year.

q) As a design goal, north/south stationkeeping will not be performed more frequently than every 21 days, Ref. 1, paragraph B.4. Impulsive delta velocity required will depend upon year of launch and mission life as follows:

Year of Launch	6 Year Life	8 Year Life
1975	829 ft/sec	1124 ft/sec.
1979	885 ft/sec.	1217 ft/sec.
Worst case	999 ft/sec (1985)	1320 ft/sec (1984)

Ref. 2. Each 21 day manoeuvre shall be performed around the node(s) of the orbit. These nodes shall rotate in time of day occurrence with a period of 1 year such that at solstice north/south stationkeeping shall be performed around 12 midnight and/or 12 noon orbit slot and at equinox it shall be performed around 6 a.m. and/or 6 p.m. orbit slot.

r) The RCS shall be designed to minimize flight operational complexity and to provide for straightforward ground servicing of fluids and electrics.

5.2.1.2 Attitude Control

a) The RCS shall be designed for use with the "WHECON" Attitude Control Subsystem (ACS). This ACS is as described in Section 5.1.1.3.2 and requires pitch thrusters for wheel momentum dumping and "offset" thrusters approximately 10° offset from roll torque into yaw torque with opposite roll and yaw polarity to perform soft limit cycle roll-yaw attitude control.

R677/51

b) The RCS shall not be qualification tested to provide 3-axis hard limit cycle operation in the event that wheel failure causes the spacecraft angular momentum to drop to zero. Rather, a redundant wheel shall be provided.

c) Maximum allowable thrust level for engines performing station acquisition or keeping manoeuvres shall be .3 lbf.

d) Maximum allowable torque impulse bit and average torque during pitch wheel spinup and momentum dumping shall be 7.5×10^{-3} ft.lbf.sec. and 6×10^{-3} ft.lbf. respectively.

e) East/west and noth/south station acquisition and keeping can be performed with steady state burns accompanied by periodic off-pulse modulation (if necessary) to remove unbalanced torque impulse. Thrusters performing these functions should have minimum moment arms about spacecraft principal axes.

f) Roll and offset thrusters shall not be required prior to solar array deployment.

5.2.1.3 Mechanical

a) The RCS shall be designed to allow the spacecraft an acceptable moment of inertia ratio (≥ 1.1) during the spinning mode and to ensure that spacecraft centre of mass shifts during life due to propellant expulsion are kept small with respect to other unavoidable shifts.

b) The RCS shall be designed to integrate directly onto the spacecraft primary structure (i.e. thrust tube and 4 bulkheads).

c) The allowable dynamic envelope of spacecraft in the launch vehicle fairing shall be 86 inches.

d) A modularized RCS design is preferable to one which is not.

e) Because of the need to use the majority of the north and south panels as mounting area and thermal dissipators for the payload, it shall not be possible to locate thrusters pointing north or south near the spacecraft centre of mass. Indeed thrusters shall not be permitted to occupy north or south panel real estate over their full width and from the forward platform to approximately 10 in. aft of the centre of mass.

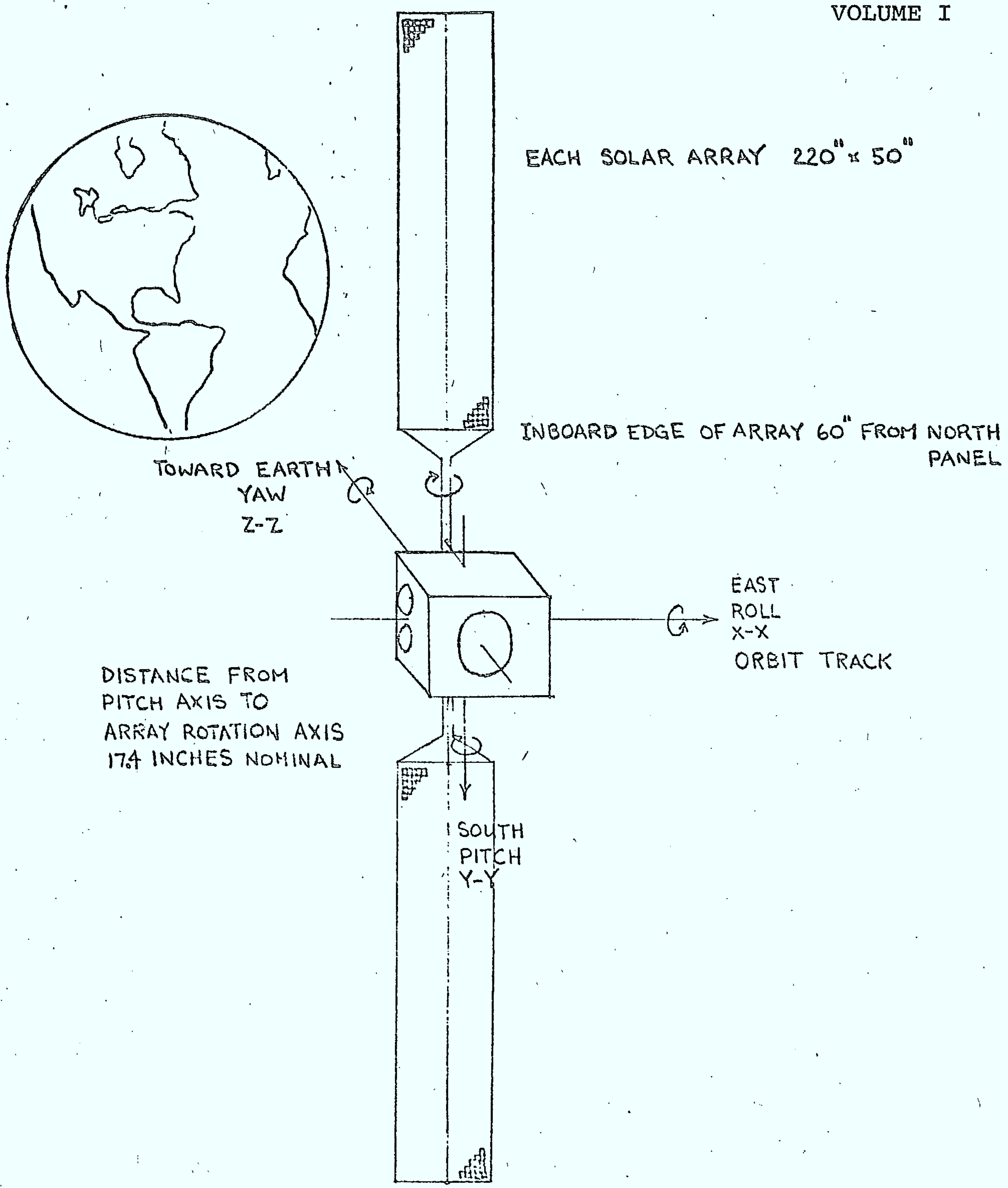
f) The stowed solar array package shall cover the north and south panels from forward platform to aft closure perhaps overhanging at the aft end, and to within 7 inches of the east and west panels.

g) The rigid substrate fold out arrays shall rotate about the north/south axis and shall be hinged approximately 3 inches forward of the front edge of the forward platform. They shall each be 220 inches long and 50 inches wide and their inboard edge shall be 60 inches outboard of the north and south panels. After deployment, these arrays shall rotate at 1 revolution per day with respect to the spacecraft main body. Figure 5-4 shows this geometry and defines spacecraft axes.

h) Over the life of the mission the uncertainty in spacecraft centre of mass location shall be $\pm$ 1.5 inches along the yaw (Z-Z) axis, $\pm$ 1.0 inches along the roll (X-X) axis and much less than $\pm$ .5 inches along the pitch (Y-Y) axis.

5.2.1.4 Electrical

a) The spacecraft housekeeping voltage rail shall be 35 volts $\pm$ .5 VDC during sunlit array operations and may vary more sig- ificantly during eclipse and/or battery operation. Typically, when part or all of the load is being provided by the battery the line voltage will vary according to the battery state of charge as shown below:



SPACECRAFT 3-AXIS STABILIZED CONFIGURATION

FIGURE 5-4

Battery DOD	Battery Voltage
5%	28.2 VDC
10%	27.6 VDC
15%	27.1 VDC
20%	26.7 VDC
30%	26.4 VDC
40%	26.2 VDC
50%	26.0 VDC

ref. Telecon with W. Threinen, 23 May, 1975.

b) RCS shall be provided, if necessary, with regulated 27.5 VDC $\pm 1\%$ power by the ACS, for powers up to approximately 15 watts on-orbit.

c) Any power conditioning required other than that mentioned above shall be the responsibility of the RCS.

d) On-orbit RCS power usage (excluding station acquisition or keeping) shall not exceed 15 watts.

e) Nominal battery capacity, end of life, shall be adequate to operate the max. load payload and housekeeping (800 watts) through the longest eclipse (72 minutes) and not have the batteries discharge past 50 percent depth of discharge (DOD). There shall be three nominally equal capacity batteries. Should one battery become inoperative, the other two shall not discharge past 66% for the condition mentioned above. Ref. Telecon W. Threinen 2 May, 1975.

f) Assume that a constant trickle overcharge will be put on the batteries during sunlit operations with the batteries fully charged. The charge rate for this operation will be C/60 and the array power required shall be approximately 35 watts. If required, during stationkeeping manoeuvres in sunlight, this trickle charge could be removed and the power used by RCS. Additional array power has not as yet been allocated for battery charging at

r677/54

the end of life. RCS must therefore assume that at the end of life, in addition to the 15 watt allocation, only 35 watts shall be available from the array for stationkeeping manoeuvres. Ref. Telecon W. Threinen 2 May, 1975.

g) Battery charge controllers shall be capable of charging all batteries at C/20 and possibly one or more batteries at C/10. Ref. Telecon W. Threinen 2 May, 1975.

h) Assume that the batteries are watt hour and not watt limited. For high power requirements, the wiring size might have to be increased to handle the high current. Ref. Telecon W. Threinen 2 May, 1975.

i) The solar array shall provide 835 watts of power at the end of life and shall be sized to account for degradation over the life. For the rigid substrate array this requires 1160 watts of power at the beginning of life (BOL = 1.39 x EOL) derived from ref. 3, para. 7.4.

j) RCS command and telemetry complexity shall be minimized consistent with subsystem operational requirements.

5.2.1.5 Thermal

a) All RCS components other than engine thrust chambers will be maintained by the spacecraft thermal control to between 40°F and 130°F except for the propellant tanks which will be maintained between 40°F and 120°F.

b) RCS thrust chambers shall be designed to deliver specified performance when exposed to steady state full solar flux of 450 $BTU/hr/ft^2$ and/or steady state cold space with no solar flux.

5.2.1.6 Environment

a) It shall be assumed that acceleration, vibration and acoustic levels will be identical to those used in the CTS project for the 2914 Delta launch vehicle with the following changes:

Acoustics - increase 140 db level in Delta Restraints Manual to 142 db.

Random Vibration - increase axial level of $0.03g^2$/Hz in Delta Restraints Manual to $0.05\ g^2$/Hz.

Ref. 1 paragraph B.8.1.

b) It shall be assumed that Electromagnetic Compatibility requirements will be identical to those used in the CTS project.

5.2.1.7 Weight

a) Every effort shall be made to save RCS weight because of the very significant spacecraft weight problem. This effort will be consistent with choice of developed if not flight qualified hardware. As stated in Ref. 1, paragraph 4.3.4," design effort should accommodate improved state of the art" where warranted.

5.2.1.8 Reliability

a) The design of the spacecraft subsystems should avoid catastrophic single point failure modes. Ref. 1, paragraph B.9.

b) As a design goal, the overall housekeeping reliability for 8 year on-orbit life shall be 0.9.

5.2.2 RCS Design

5.2.2.1 General

The following paragraph reports on the initial RCS design work performed at Spar to satisfy the subsystem requirements. This effort involved interfacing with the other housekeeping subsystems to ensure an optimum overall bus design. Spar wishes to acknowledge the significant contribution made by Vincent Sansevero of Hamilton Standard Division, under subcontract to Spar, towards accomplishing this task.

The baseline design subsequently chosen and tradeoffs performed after receipt of candidate RCS vendor RFP responses are presented in Volume II, Section 2.1. Also presented in Volume II, Appendix B is the Hamilton Standard technical proposal for the CPB RCS.

The initial RCS design has concentrated on maximum use of CTS qualified technology, see Ref. 4. This was done for several reasons. Firstly the dual thrust level catalytic hydrazine, blowdown mass expulsion subsystem represents the current standard in the North American technical community for RCS for Thor Delta Class, 3-axis stabilized, communications satellites. This subsystem type would therefore be a leading contender for GPB. Secondly, before extensive investigation of RCS state-of-the-art could be undertaken, it was necessary to work with the structures design group to size and locate fuel tanks and to locate thrusters. It was timely, therefore, to use readily available CTS information. Thirdly, it was expected that the dual thrust level, catalytic hydrazine subsystem would produce the heaviest design, of those which eventually would be considered, resulting in the largest tank weight and size, etc. and was, therefore, conservative for initial design work.

From the outset of this study, Spar has been aware of the possibility of saving weight and other performance enhancement through the use of bipropellant engines (of greatest interest is the TIROC thruster built by Dr. L. Kayser of Technologieforschung, Stuttgart, see Ref. 5, catalytic or

R677/57

electrothermal plenum with warm gas thrusters, electrothermal hydrazine thrusters with and without superheating and/or colloid, pulsed plasma or ion engines. As a result, the design has evolved so as not to preclude any of these options (except colloid, pulsed plasma and ion engines). As will be discussed in Volume II, Section 2.1.2.6 Spar does not recommend the use of these electric propulsion thrusters for application on an operational satellite to be launched in 1979.

A discussion of these engine types and their applicability to GPB is presented in Volume II, Section 2.1.

The RCS design has been modularized with engines in rocket engine modules (REMs), etc. to ensure ease of integration, minimum envelope, minimize thermal control complexity and to permit efficient subassembly testing to minimize program costs and schedule. Emphasis has been placed in the RCS Preliminary Specification, SPAR-SG.350, on the desirability of a modularized RCS design.

5.2.2.2 RCS Performance

The RCS delta velocity and attitude control performance requirements were derived from the requirements presented in Section 5.2.1 and are tabulated in the GPB RCS Preliminary Requirements Specification, SPAR-SG.350, Table 1, see Section 5.2.3 of this report. It is important to note that vehicle effective performance has been defined as the subsystem requirement and that inefficiencies due to plume impingement, thruster canting, rotational losses, etc. would be taken into account by the candidate RCS supplier. These requirements are independent of location of thrusters. Spar considers it essential, at this point in the spacecraft evolution, to design the RCS (8 year tankage and 6 year fuel) to accommodate an increased fuel requirement of 5% by weight. This could be considered, by some, to be a weight contingency in the design. Worst case launch date delta velocities have been used in deriving the requirements for north/south stationkeeping. Other firing performance requirements were derived from CTS experience to ensure predictable operation over the life of the mission, to minimize unwanted misalignment torques and delta velocity uncertainties and to define ground test margins with respect to flight life. See Section 5.2.3 of of this report.

The high thrust engine (HTE) to be used for spinning mode precession manoeuvres has been sized at between 1.0 and 5.5 lbf steady state thrust to permit several candidate suppliers to bid flight qualified engines. It is not acceptable to thrust below 1.0 lbf SS because of the time required for performing these manoeuvres and it is not acceptable to thrust above 5.5 lbf SS because of the lack of resolution in the precession angle resulting from delivery of 1 pulse. The duty cycle range chosen for these manoeuvres is consistent with the spacecraft spin rate and maximizing of overall manoeuvre efficiency.

The low thrust engine (LTE) minimum thrust level was set at .01 lbm because of the very long burn times required for north/south station acquisition and keeping manoeuvres and resulting orbit inefficiencies if thrust levels are below this value.

5.2.2.3 Tankage Size And Location

Most of the candidate propulsion subsystems, including the preliminary design, would employ a blowdown mass expulsion propellant management design requiring pressurant ullage volume in the main tanks and utilizing approximately a 3 to 1 blowdown ratio. This ratio must be tailored to maximize engine performance in both thrust regimes.

An estimate of approximately 240 lbm was made for the monopropellant hydrazine fuel required for the worst case (1984 launch) 8 year mission using CTS technology catalytic hydrazine thrusters for all functions. This included a weight penalty for providing offset thruster linear I_{BIT} of 4×10^{-3} lbf. sec. at the beginning of life (consistent with CTS capabilities) because engine moment arm was reduced to 1 foot to maintain torque I_{BIT} of 4×10^{-3} ft. lbf.sec. maximum. The resultant tankage volume required at 3 to 1 blowdown is approximately 10,000 in^3. The decision was made to recommend mounting the RCS directly to the spacecraft primary structure only (i.e. thrust tube and bulkheads) thereby facilitating RCS integration and test, see Spar drawing 31138J, Sheets 1 & 2, Structure & RCS General Purpose Satellite Bus. The RCS project implementation plan is presented in Volume III Section 2.4 of this report. The basic structure, described in Section 4.4 has been designed for minimum weight and maximum payload capacity. A four spherical RCS tank design was chosen and the tanks are located as shown in Spar 31138J for the following reasons:

a) 4 spherical tanks of approximately 16.5 to 16.8 inches ID could contain the maximum fuel load including the 5% growth propellant. 16.5" ID elastomeric diaphragm tank has been previously qualified and flown. 16.8" ID surface tension tank has been qualified for Thor Delta 3914, geogynchronous communications satellite, SATCOM, to be launched December, 1975. Therefore, for whichever of these two propellant management device types we would choose, qualified designs exist in the size we would require for a 4 tank subsystem. (Note that forgings are normally the most significant tank non-recurring cost and, as long as tank diameter required is the same as a qualified design, the cost of modifying the forgings for new mounting arrangements is usually less expensive than the cost of a new size of tank).

b) The 4 tanks are mounted basically to the east west bulkheads and are outboard (east/west) as far as is allowable by shroud dimensions and mounting and thermal considerations in order to maximize spinning mode moment of inertia ratio and to maximize useful payload area on the inside of the north/south panels.

By placing the geometric centre of the tanks at station 43.0 inches along Z-Z axis, the centre of mass of the fully loaded tanks will be at a location which maximizes spinning mode moment of inertia ratio.

c) The tank mounting proposed is polar bosses, forward and aft, with a rigid 4 strut mounting at the aft boss (one strut connected directly to the thrust tube at the apogee motor ring, the other 3 mounted to the east/west bulkheads) and a two strut A-frame pin jointed at the forward boss to provide radial rigidity but allow axial tank growth during pressurization. This configuration is especially compatible with the spacecraft structural design and affords a low weight installation within minimum envelope. With the propellant port at the aft boss, the lowest point on the tank in the launch configuration, the ground handling complexity of fluids within the RCS is minimized. Spacecraft centre of mass shift along the Z-Z axis (forward and aft) due to propellant depletion with this spacecraft axial expulsion configuration would be less than .3 inches maximum over the life of the mission. See Figure 5-5. It does not appear beneficial to provide radial propellant feed tankage (propellant expelled radially outwards from Z-Z, spin axis) in order to reduce C of M excursions because the shift to be expected with the axial feed (<±.15 inches) is much smaller than the overall Z-Z axis C of M uncertainty (±1.5 inches) which is due to solar array bending, ABM residuals uncertainty, initial balancing uncertainty, etc. With surface tension tankage it is expected that positive axial feed could be provided in the spinning mode for the 6 or 8 year load of propellant. The diameter of the tanks shown in Spar drawing 31138J are oversized due to a conservative fuel estimate presented at the time the design was being committed to paper.

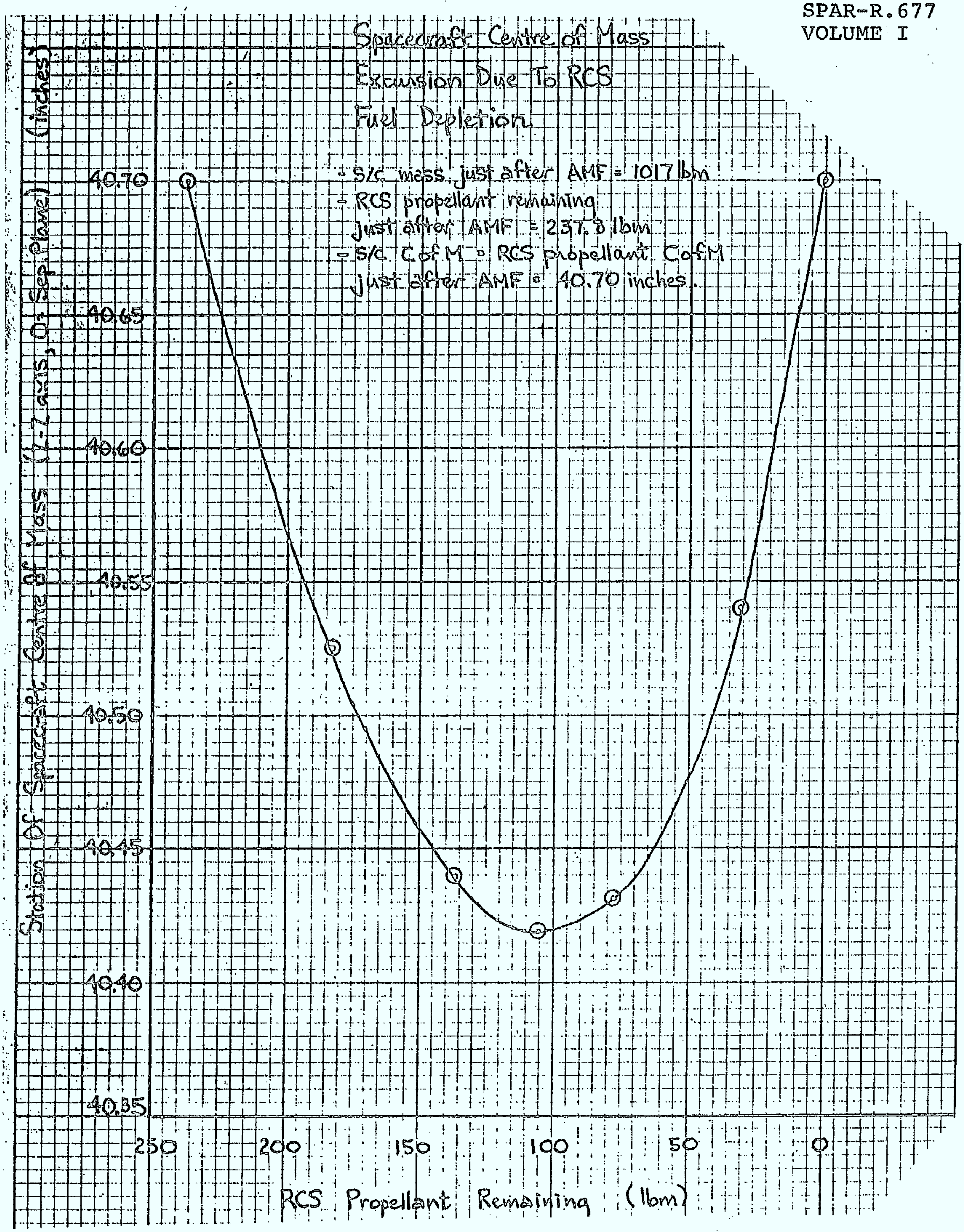

Spacecraft Centre of Mass
Excursion Due To RCS
Fuel Depletion
- S/C mass just after AMF = 1017 lbm
- RCS propellant remaining
just after AMF = 237.8 lbm
- S/C C of M = RCS propellant C of M
just after AMF = 40.70 inches.
Station Of Spacecraft Centre Of Mass (Y-Z axis, 0 = Sep. Plane) (inches)
40.70
40.65
40.60
40.55
40.50
40.45
40.40
40.35
250
200
150
100
50
0
RCS Propellant Remaining (lbm)

FIGURE 5-5

5.2.2.4 Thruster Plume Impingement Analysis

RCS engines should be located to minimize or eliminate heating and/or thrust deflections and unwanted torques due to impingement of their exhaust plumes on spacecraft surfaces. In addition, it must be ensured that no surface degrading contaminants, if present in the plume, are deposited on solar cells, second surface mirrors, attitude sensor optics, etc. A general rule of thumb for hydrazine thrusters is to keep spacecraft surfaces outside a 40 degree half angle cone about each nozzle centreline. On GPB, with arrays pointing north/south, it is a difficult task to position north/south stationkeeping thrusters (which must provide the majority of the mission fuel) so that they are efficiently performing the manoeuvre without creating unacceptable external forces and torques on the spacecraft which would result in a need for additional fuel expenditure. Figure 5-6 shows the results of a study of the forces imposed on the spacecraft by the impingement of the plume of a .1 to .3 lbf catalytic hydrazine (CTS technology) thruster on the GPB array. With reference to the inset in this figure, the solar array was assumed to be 50" wide by 220" long, beginning 50" north of the north panel. For the analysis, the thruster was positioned a distance ℓ from the array centreline and the array plane was considered perpendicular to the line drawn between the thruster and the array centreline. At first, $\beta=0°$ was examined for a thruster pointing directly north/south. Forces on the array were resolved as normal and tangential axial (drag loss) and expressed as a percentage of the non impeded engine steady state thrust. As can be seen from the figure, over a typical spacecraft dimension range of $0 \leq \ell \leq 60$ inches, the forces are very significant except as ℓ approaches 0 inches. For example, with a 6 year north/south stationkeeping nominal fuel requirement of approximately 132 lbm, if ℓ were chosen to be 40 inches, the extra fuel requirement due to plume impingement, and taking into account doing north/south stationkeeping at different times of the day (and therefore different array aspect angles) throughout the

THRUSTER PLUME IMPINGEMENT

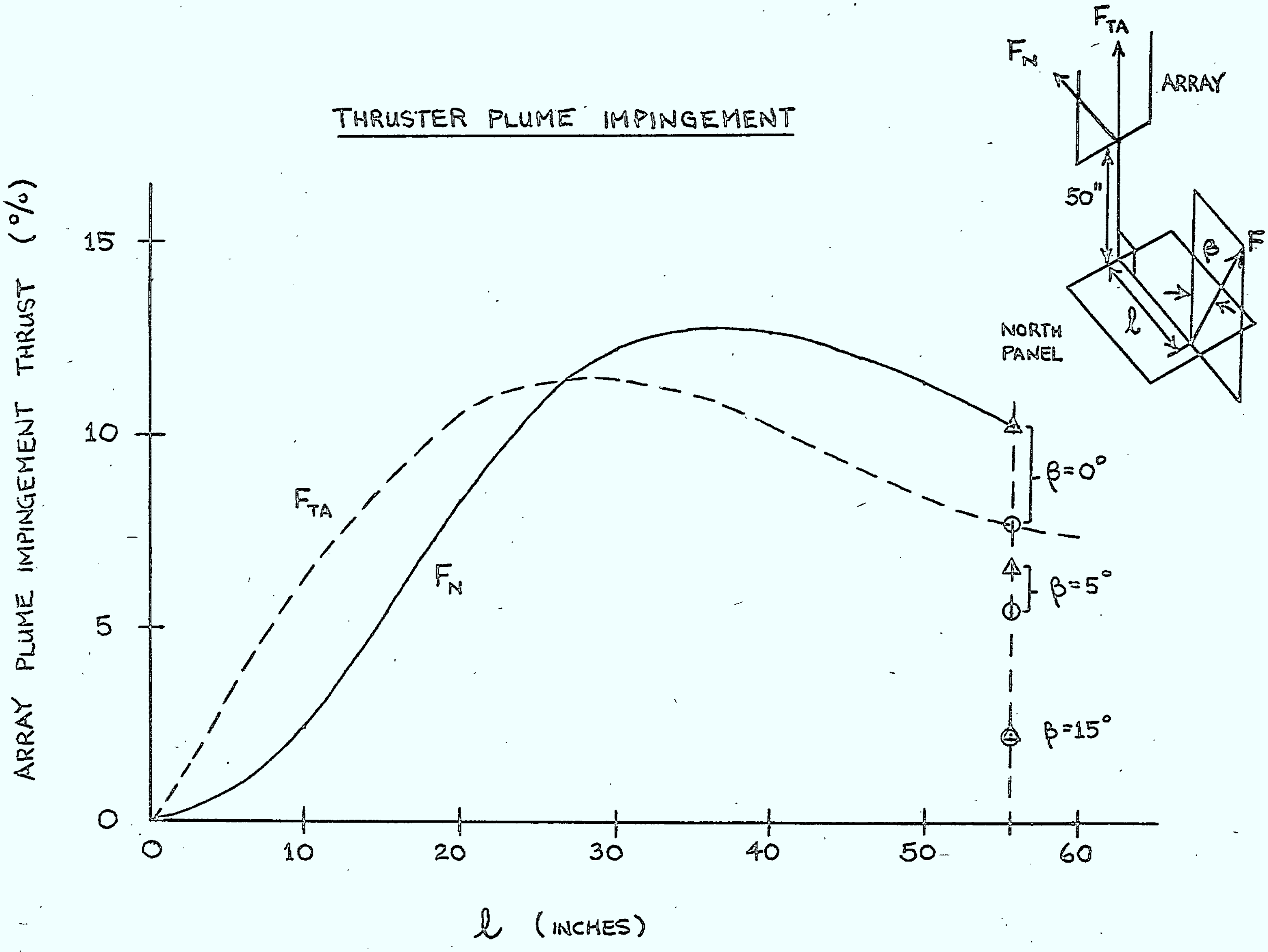


FIGURE 5-6

year, would be approximately 15 lbm. In addition, the external torques would exceed .1 ft.lbf. during the manoeuvre (roughly 4 orders of magnitude higher than solar torques) and attitude would be very difficult to control.

Figure 5-6 implies that the solution would be to place thrusters as close as possible to $\ell = 0$ inches. Unfortunately, this figure does not show the effect of impingement on the array yoke and elevation arm assembly if thrusters are located close to $\ell = 0$ inches. This impingement torque would be unacceptably high. In addition, all north/south thrusters cannot be located close to $\ell = 0"$, with the array offset forward of the spacecraft C of M by 17", because net spacecraft torques must be zero during delta velocity manoeuvres. A unique solution, which has been adopted in the RCA SATCOM design, also with offset array, is to place 4 thrusters pointing directly north clustered around the C of M and rotate the plane of array at each north/south stationkeeping manoeuvre to be perpendicular to the spacecraft forward platform so that ℓ is effectively brought close to 0 inches and the yoke and elevator arm assembly are out of the thruster plume. This imposes an operational constraint on the mission. This solution is not feasible for GPB because the north south panel real estate, especially at the centre of mass and forward of it, is required exclusively for mounting high heat dissipating communications equipment. SATCOM, fortunately, has much lower power dissipation.

The effect of canting the thruster away from the array, $\beta \neq 0$ degrees, is shown to dramatically reduce the impingement forces. However, a direct cosine loss in effective thrust accompanies such a canting.

Spar has now developed a computer program to investigate the impingement phenomenum for any array aspect angle and thruster geometry and with more than one thruster firing at a time. The greatest uncertainty is in modelling the physics of the exhaust gas - array surface interactions.

There will be no detectable heating of the solar array due to RCS thruster firing because of the 50 inches minimum separation between the two. Plume impingement analysis would represent a significant design and systems analysis effort during PDP and Phase C.

5.2.2.5 RCS Thruster Complement and Configuration

The need to perform north/south stationkeeping efficiently governs the placement of RCS engines. Figure 4 of Spar Specification SPAR-SG.350 and Spar Drawing 31138J show the configuration finally chosen and Table IV of that document explains the function of each thruster. Some features of the design are noted below:

a) all thrusters are mounted from spacecraft primary structure.

b) HTEs for spinning mode precession, engines #17 and #18, are located to maximize their moment arm with respect to the C of M. They must be positioned on the west or east bulkhead and not on the north or south bulkhead in order to avoid being covered by the stowed array.

c) North station acquisition and keeping Engines #1, #6 and #11 are the fewest possible, consistent with the need to design for a Z-Z C of M uncertainty of ± 1.5 inches. Engines #1 and #6 are positioned far enough forward of the nominal C of M (at this point in the design, 1.5 inches) to ensure that when fired together they will provide zero or negative roll torque on the spacecraft. They would be "off-pulse width" modulated during north station acquisition and keeping to account for unwanted yaw (Z-Z) torques they might produce because of thrust, and/or moment arm mismatch due to X-X axis C of M shifts. Engine #11 would be onpulsed as necessary to counter any negative roll torque build-up due to the other two engines. All three engines

R677/64

would be canted approximately 15 degrees (with associated 3.4% cosine loss) to avoid plume impingement with the solar arrays and the spacecraft main body. Even with this canting and with the CTS technology LTEs, an overall impingement loss of approximately 2.5 percent can be expected during north and south station acquisition and keeping which must be accounted for in the propellant budget. Note that if these engines were not located to enable them to take out the torques they create due to C of M shifts, plume impingement, etc., efficiently while performing the manoeuvre (i.e., if thrusters not pointed north/south were required for this function) approximately 10 lbm of extra fuel could be required for the worst case six year mission. Similar comments apply for the thrusters performing south stationkeeping, #2, #7 and #12. By positioning thrusters to perform north and south stationkeeping, we are maintaining the greatest flexibility in the subsystem. Should electrothermal LTEs be chosen at a thrust level of approximately 20 mlbf thrust and with the design goal of stationkeeping no more frequently than every 21 days, orbit inefficiencies for the manoeuvre become non-negligible and are minimized by thrusting at both nodes. (i.e. there are two points in the orbit at which stationkeeping can efficiently be performed). These thrusters are also used for yaw and roll control. It was possible to position Engines #11, #12, #13, #14, #15, and #16 where they could be partially covered by the stowed array because none of these engines are required prior to array deployment.

d) Thrusters #3 and #8, mounted through the nominal C of M, would be the primary west and east station acquisition and keeping engines but could be efficiently backed up by #4 and/or #5 and #9 and/or #10 in off-pulse width mode and inefficiently by #1 and #2 or #6 and #7.

e) Pitch function would be nominally performed using Engines #4 and #5 with relatively small moment arms of 14 inches and would be backed up by Engines #9 and/or #10.

f) The "offset" engines for Whecon operation are mounted in a similar orientation to those on CTS except that no outboard canting to avoid plume impingement is required in this case. As with CTS, these engines would provide back-up roll control in the event of a failure of Engine #11 or #12. In this case, a tolerable and correctable unwanted yaw torque would have to be removed.

g) Two HTEs (one for back-up) are provided because the only LTEs capable of performing the precession manoeuvres, as a back-up, are engines #4, #5, #9 or #10 which have a very inefficient moment arm for this function.

h) Engines #1, #2, #3, #6, #7 and #8 are mounted on brackets as far outboard (east-west) as is possible within the shroud dimensions to avoid plume impingement and to provide as large a moment arm as possible for despin.

i) Roll and pitch attitude engines are on axes and yaw, because of their use for north/south stationkeeping are slightly offset. As a result pure couples are not needed, as would be the case with an off-axis design, to avoid attitude torque coupling.

This complement of engines appears to comprise the minimum number of thrusters and optimum location for performing the intended manoeuvres with at least single point failure protection (with minimal unwanted torques) and with minimum propellant expenditure.

R677/66

5.2.2.6 RCS Mechanical Schematic

For the purpose of expeditiously issuing an RFP for budgetary estimates to candidate suppliers, the mechanical schematic was based on a CTS technology, blowdown mass expulsion, monopropellant RCS. The covering letter to that RFP, see Volume II, Section 1.3, encouraged bidders of pressure regulated RCS to propose a design to accommodate their hardware.

Figure 1 of SPAR-SG.350 is the preliminary mechanical schematic diagram for the RCS for GPB. The tanks would be diagonally cross-strapped and latched in pairs, see also Spar Drawing 31138J, to minimize spacecraft cross products of inertia and/or C of M shifts in the event of need to close off one side of the propellant management system. Because of severe weight limitations on the spacecraft, it is not possible to carry enough propellant to enable completion of the standard mission in the event of failure of one tank at the beginning of life. Considering the probability of tank or latching valve failure to be low, the trade-off was made to pair the tanks upstream of a common latching valve to afford some back-up mission in the event of non-catastrophic failure of one propellant management branch, but not to add two extra latching valves, pressure transducers, fill and drain valves and liquid filters to provide redundancy for each tank separately because of the weight implications.

Separate pressurant fill and vent valves are required for each tank with a propellant fill and drain valve for each pair of tanks to ensure loading of equal quantities of propellant in each tank during loading operations. Since tankage will undoubtedly be 6Al4V titanium and since stainless steel tubing is preferable downstream of the tanks to maximize thermal conductivity, transition joints have been called up in the schematic. A pressure transducer will be placed upstream of each tank latching valve in order to be able to monitor independently each half of the propellant management system.

Propellant isolation from the engines was proposed to be accomplished by use of six latching valves with engines grouped to provide single point failure protection for a failed open thruster while minimizing the effect of such a failure on the operation of other engines. All of these latching valves would be fed from a common tubing manifold. An alternate configuration is the half system approach where the engines are isolated in two groups only by the tank latching valves and the cross connect manifolding is latched. This configuration usually employs dual series seat engine valves because of the serious implications, for this design, of a valve open failure. However, since a dual seat valve can still fail open, it is recommended that the increased latching proposed here be adopted to increase reliability and operational flexibility. The use of dual versus single seats is another question to be addressed by the candidate suppliers in providing the most reliable, qualified hardware to meet the requirements (min I_{BIT}, subsystem weight, etc). It is not felt worthwhile to latch the manifold between LV5 and LV6 in order to avoid a catastrophic manifold failure. This is because of the negligible probability of such a failure and the extra operational complexity of normally operating one side of the system at a time with this latching valve closed. The probability of catastrophically clogging a liquid system filter is also very low and cross strapping to protect against this failure does not appear justified. All latching valves will be provided with positive position indication.

5.2.2.7 RCS Electrical Schematic

For the purpose of expeditiously issuing an RFP for budgetary estimates to candidate suppliers, the electrical schematic was based on a CTS technology, blowdown mass expulsion, monopropellant RCS. The covering letter to that RFP encouraged bidders of other acceptable RCS to propose a design to accommodate their hardware. Figure 2 of SPAR-SG.350 is the preliminary electrical schematic diagram for the RCS for GPB. Command and telemetry requirements are presented in Table VI of the same document. All electronics would be processed through the RCS electrical control unit (ECU). Telemetry includes temperature sensors on all thrust chambers (18), pressure transducers (2), position indicators for all latching valves (8), flags for all heater groups (if required) and total RCS 27.5 VDC input current. Valve and heater driver commands were assumed to be latched, if necessary, outside the ECU. The RCS would be supplied with a 27.5 VDC $\pm$ 0.5 VDC regulation rail by ACS to provide all RCS functions with power conditioning within the ECU for signal conditioning voltage. Figure 2 of SPAR-SG.350 requires that the RCS power switch be situated within the ECU. All valve drivers and signal conditioning required for the RCS would be provided within the ECU. In addition, a test connector would be provided which would enable ground measurements of all valve currents and heater group currents if required.

Since the issuance of this specification, it has become evident that for the large current draw required by the superheated electrothermal LTEs, if chosen, see Section 5.2.1.4, it would be beneficial to operate these thrusters directly from the unregulated main spacecraft voltage rail which, in sunlight array operation, would produce 35 VDC $\pm$ 1%. This would result in two voltage rails being distributed to the RCS.

5.2.2.8 RCS Thermal Design

RCS has been given thermal responsibility for only the engine thrust chambers, from the mounting flange outboard, with the remainder of the subsystem including closure blankets thermally the responsibility of the thermal subsystem. This group will guarantee RCS component temperatures of between 40 and 130°F except for tanks where 40 to 120°F will be maintained. This approach is not necessarily compatible with engine packaging techniques employed by TRW and others. It may be a cleaner interface if RCS were to be responsible for valve temperatures as well as thrust chambers. If this were the case, valve heaters and perhaps temperature sensors would become RCS responsibility and the electrical design would need to be so modified. This is a most complicated interface due to the complex nature of engine geometries and their thermal sensitivities and the fact that thrusters poke holes in spacecraft which have to be sealed. Further work would be necessary to completely specify this interface prior to contract award, before and after the RCS contractor has been chosen. In any event, the thermal subsystem would be responsible for thermal conditioning tanks, lines and latching valves.

5.2.2.9 RCS Weight (See Table 5.2.2.9-1)

Initial estimates of fuel and hardware weight budgets required for the six and eight year life for various types of subsystems were derived based on preliminary vendor information. This was done to obtain weight design goals for the specification, to investigate the viability of the baseline design, to investigate the potential weight savings of advanced propulsion designs and to prepare a preliminary trade-off for the April 18, 1975 verbal presentation to the Department of Communications, see Reference 3. This information will not be presented again in this report because it has been superseded by the information presented in the vendor budgetary estimates received in May by Spar. Instead, the present design status and trade-off of types of RCS based on these proposals is contained in Volume II, Section 2.1 of this report.

5.2.3 RCS Specification And Statements of Work

Using the design and interface information which has been presented in Sections 5.2.1 and 5.2.2, SPAR-SG.350, Multi-Purpose Bus Study, Specification, Requirements, Reaction Control Subsystem, Preliminary, For Budgetary Estimates, April 15, 1975 and SPAR-SOW.71, Multi-Purpose Bus Study, Statement of Work, Reaction Control Subsystem, Preliminary, For Budgetary Estimates, April 15, 1975 were produced to define the RCS GPB requirements, preliminary design, requested deliverables and subsystem test requirements. These documents are contained in Volume II, Section 1.3 of this report.

R677/70

GENERAL PURPOSE BUS

RCS WEIGHT SUMMARY

FUEL FOR 6 YEARS

COMPONENT	NUMBER REQUIRED	NOMINAL WEIGHT (lb_m)	Catalytic N_2H_4 System Fuel Weight	Super Heat Electrothermal System Fuel Weight
Propellant Tank	4	22.92		
Fill and Drain/Vent Valve	6	0.78		
Filter	2	0.44		
Pressure Sensor	2	0.36		
Latching Valve	6	3.12		
HTE - TCA	2	0.76		
HTE Valve	2	0.57		
HTE Chamber Heater	2	0.22		
HTE Thermal Shield	2	0.14		
Chamber Temperature Sensor	18	1.12		
LTE - TCA	16	1.75		
LTE Valve	16	2.59		
LTE Chamber Heater	16	0.29		
LTE Thermal Shield	16	0.20		
Electrical Control Unit	1	7.96		
Tubing	AR	1.93		
Wiring	AR	2.85		
Structure	AR	11.95		
Fasteners and Miscellaneous Hardware	AR	1.80		
Nominal Estimated Dry Weight		61.75		
± 20 Allowance		2.32		
Maximum Estimated Dry Weight		64.07		
			194.64	154
Catalytic N_2H_4 System Total Weight Including Hardware			258.69	
Super Heat Electrothermal System Total Weight Including Hardware				218

TABLE 5.2.2.9-1

5.2.4 List of References

1. DOC Statement of Work to Spar for General Purpose Bus Study, dated March, 1975.

2. Mission Study Report, submitted to Spar Aerospace Products Ltd., by Canadian Astronautics Inc., included in this report as an appendix.

3. Spar presentation to DOC/DSS on Status of GPB Design, held at Spar, 18 April, 1975.

4. CTS Paper. AIAA Paper Number 73-1268, Monopropellant Hydrazine Reaction Control Subsystem for the Communications Technology Satellite by Vincent Sansevero and Carl Arvidson, Hamilton Standard Division, W.D. Boyce, Communications Research Centre, DOC, and S. Archer, Spar Aerospace/CRC, DOC presented at the AIAA/SAE 9th Propulsion Conference, Las Vegas, Nevada, November 5-7, 1973.

5. AIAA Paper No. 74-1181, TIROC-Attitude and Orbit Control Propulsion for Satellites, by L.T. Kayser, Technologie for Schung GmbH, Stuttgart, Germany and Capt. R.C. Kelso, AFRPL, Edwards California, presented at AIAA/SAE 10th Propulsion Conference, San Diego, California, October 21-23, 1974.

6. AIAA Paper No. 74-1179, Development of a Five-Pound Thrust Bipropellant Engine, by R.C. Schindler and L. Schoenman, Aerojet Liquid Rocket Company, Sacremento California, presented at AIAA/SAE 10th Propulsion Conference, San Diego, California, October 21-23, 1974.

7. TRW Document 20266-6099-RO-00, Monopropellant Hydrazine Resistojet, Preliminary Design Task Summary Report, Prepared for GSFC, by C.K. Muych, February, 1972.

8. TRW Document 20266-6024-RO-00, Monopropellant Hydrazine Resistojet, Engineering Model Fabrication and Test Task Summary Report, Prepared for GSFC, by C.K. Muych, March, 1973.

9. TRW Document 20266-6025-RO-00, Monopropellant Hydrazine Resistojet, Data Correlation Task Summary Report, Prepared for GSFC, by C.K. Muych, March, 1973.

10. AIAA Paper No. 71-60, Electrothermal Hydrazine Engine Performance, by T.K. Pugmire, T. O'Connor, R. Shaw, W. Davis AVCO Corporation, presented at AIAA/SAE 7th Propulsion Joint Specialist Conference, Salt Lake City, Utah, June 14-18, 1971.

5.3 Deployable Solar Array Subsystem

5.3.1 Introduction

This section of the report presents the results of work carried out during this Feasibility Study in arriving at a preliminary definition of a deployable solar array subsystem to meet the requirements of the General Purpose Satellite Bus.

The preliminary requirements as defined in the D.O.C. specificaton and statement of work are presented, followed by a section on design alternatives and trade-offs considered. These were either conducted during this study or previously.

A section on sizing of the array concept chosen includes a preliminary cell layout based on solar cell performance suggested by D.O.C., transfer orbit considerations, adaptability to alternative payloads and potential improvements.

A section on the system weight estimate is followed by conclusions and recommendations.

5.3.2 Requirements

The D.O.C. requirements are summarized in Table 5.3-1. As a result of this study, these requirements have been expanded to form a Performance Specification which is included in Section 1.4 of Volume II of this report. The D.O.C. specification was used as a guideline along with the power budget established in conducting the study.

The power requirements for Payloads A to D are shown in Table 5.3-2. Due to its uncertainty nothing has been included for payload E. However, if the initially specified 24 channel 4-6 GHz communications system is used for this payload, the total spacecraft power requirement will lie between Payloads A and C. If the 12 channel 12-14 GHz communications system is used, the total power requirement will be slightly more than that of Payload B (800 watts) and if the 8 channel 12-

TABLE 5.3-1

REQUIREMENTS - DEPLOYABLE

SOLAR ARRAY SUBSYSTEM - G.P. SAT. BUS

1. Provide 500-800 watts, end of life 6-8 years, 35 to 40 V.
2. Minimum weight, tailor to particular mission requirements.
3. Survive Launch Vibration - Thor Delta 3914.
4. Provide sufficient exposed solar cells in launch configuration for 70 watts transfer and drift orbit power.
5. Minimized thermal deformations and fabrication tolerances to reduce solar torquing effects.
6. Deploy in a controlled manner to the fully deployed condition.
7. Deployed array stiffness consistent with requirements of ACS.
8. Stowed array must provide thermal protection to the S/C North and South Panels.
9. Stowed array must clear those RCS thrusters which operate in transfer and drift orbit.
10. Lower end of array must be elevated to minimize shadowing by the antenna and to achieve the desired thermal field-of-view.
11. Deployed arrays should track the sun with an accuracy of $\pm 1^{\circ}$.

TABLE 5.3-2 - POWER REQUIREMENTS - G.P. SATELLITE BUS

Subsystem	Payload A Synch. Power-W	Payload B Synch. (w)	Payload C Synch. (w)	Payload D Synch. (w)	Payload E Synch. (w)
Communications	415	625	505	513	
Telemetry and Command	30	30	30	30	
Power	20	20	20	20	
Battery Charging	30	30	30	30	
Harness	5	5	5	5	
DSA (Tracking, including Electronics)	10	10	10	10	
ACS	25	25	25	25	
RCS	15	15	15	15	
Thermal Heaters	20	20	20	20	
Total	570	780	660	668	
Contingency	20	20	20	20	
Design Total	590	800	680	688	

14 GHz system using four 20 watt TWTs and four 50 watt TWTs is used the spacecraft power requirement will exceed the maximum specified (800w) by over 100 watts.

A review of Table 5.3-2 will show that three basic solar array sizes are required. These are 600 watts for Payload A, 700 watts for Payloads C and D and 800 watts for Payload B. Paragraph 5.3.4.3 of this report shows how the preliminary design can be adapted to these payloads.

5.3.3 Design Alternatives and Trade-offs

5.3.3.1 Deployable Array

5.3.3.1.1 Alternative Concepts

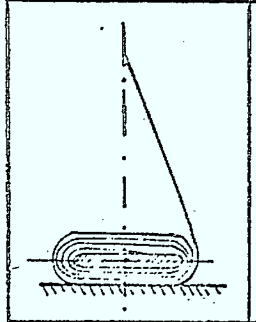
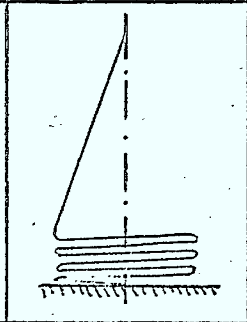
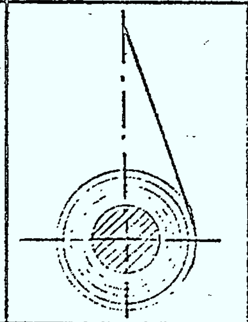
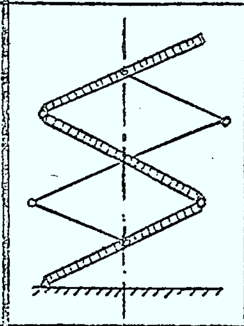
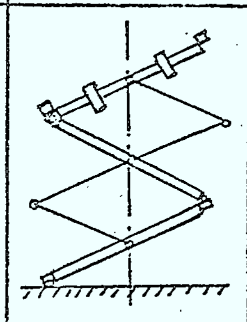
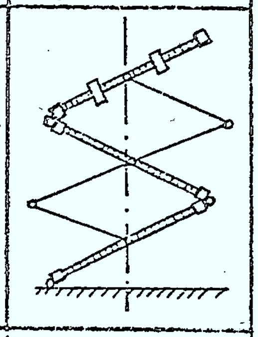
Table 5.3-3 shows in a summary form the alternate concepts that have been considered in the past and during this study. Generally, deployable solar array systems can be classed as either "flexible" or "rigid".

Flexible arrays are generally constructed using a thin plastic substrate - usually Kapton or Kapton reinforced with fibreglass - on which solar cells are mounted. The array is deployed and supported by deployable structures such as pantograph mechanisms or storable tubular extendible members (STEMs). Stiffness of the deployed system is achieved by placing the flexible array in tension and the deployable support structure in compression. Usually such systems have lower natural frequencies and hence greater potential of interaction with the spacecraft's attitude control system, than do rigid arrays. On the other hand such systems offer significant weight advantages over conventional rigid systems in systems larger than about 1 kilowatt.

So far, three basic stowage techniques have been considered for flexible array systems.

CD/83

TABLE 5.3-3

ALTERNATE DSA STOWAGE & DEPLOYMENT CONCEPTS					
FLEXIBLE			RIGID		
FLAT SPINDLE	FLAT PACK	ROLL UP	RIGID SUBSTRATE	RIGID FRAME/ FLEX. SUBSTRATE	RIGID FRAME/ 'RIGID' SUBSTRATE
					
o UNUSED CORNERS VOLUME > FLAT PACK o TRANSLATION OF PIVOT CENTER REQD. o ADDITIONAL SPINDLE DRIVE REQD. o ROTARY POWER TRANSFER REQD. o CELL PROTECTION BY INTERLEAFS REQD. o IRREGULAR EXTENSION o LOW DEPLOYED f_n o WEIGHT > FLAT PACK	o WEIGHT EFFECTIVE SMALLEST VOLUME o RETRACTION IS DIFFICULT o SIMPLE DEPLOYMENT o NO ROTARY POWER TRANSFER REQD. o CELL PROTECTION BY INTERLEAFS REQD o IRREGULAR EXTENSION o LOW DEPLOYED f_n o LOWEST WEIGHT	o UNUSED DRUM VOLUME >> FLAT PACK o EXTENSION & RETRACTION POSSIBLE o ADDITIONAL DRUM DRIVE REQUIRED o ROTARY POWER TRANSFER REQD. o CELL PROTECTION BY INTERLEAFS REQD. o REGULAR EXTENSION o LOW DEPLOYED f_n o WEIGHT > FLAT PACK	o HIGH SOLAR CELL DENSITY POSSIBLE VOLUME > FLEX FLATPACK o RETRACTION IS DIFFICULT. o SIMPLE DEPLOYMENT PANTOGRAPH OR CABLE o NO ROTARY POWER TRANSFER REQD. o NO CELL PROTECTION REQD. o REGULAR EXTENSION o DEPLOYED f_n > OTHERS o WEIGHT > OTHERS	o SOLAR CELL LAYOUT SLIGHTLY AFFECTED BY CROSS BEAMS o RETRACTION IS DIFFICULT. o SIMPLE DEPLOYMENT PANTOGRAPH OR CABLE o NO ROTARY POWER TRANSFER REQD o INTER PANEL 'BUTTONS' PREVENT CELL DAMAGE. o REGULAR EXTENSION o DEPLOYED $f_n \leq$ RIGID SUBSTRATE o WEIGHT < RIGID SUBSTRATE & FLEX FOR	o SOLAR CELL LAYOUT SLIGHTLY AFFECTED BY CROSS BEAMS. o RETRACTION IS DIFFICULT. o SIMPLE DEPLOYMENT PANTOGRAPH OR CABLE o NO ROTARY POWER TRANSFER REQD. o NO CELL PROTECTION REQD. o REGULAR EXTENSION o DEPLOYED $f_n \leq$ RIGID FRAME/FLEX S/ST. o WEIGHT > RIGID FRAME/FLEX S/STRATE.

These are:

a) Flat Spindle
b) Flat Pack
c) Roll-Up

Of these, the last two have been developed by various aerospace companies. The relative advantages and disadvantages of each concept are summarized in Table 5.3-3. The only large flexible solar array system that has been flown is a roll-up array -FRUSA- designed and manufactured by Hughes Aircraft Co. (Ref. 1). This was a low earth, polar orbit flight. The first flexible flat pack solar array system that will fly in earth synchronous orbit will be the CTS Deployable Solar Array System developed jointly by Spar Aerospace Products Ltd. and AEG-Telefunken under contracts to the Canadian Government and ESRO respectively. (Ref. 2 and 3). The CTS is to be launched in December, 1975.

Other flexible array systems have been developed by Lockheed Missiles and Space Company, General Electric, RAE, Farnborough, U.K. (Ref. 11), AEG-Telefunken, MBB and S.A.T./Aerospatiale (Ref. 4). A comparison of the most promising flexible solar array designs is presented in Reference 5.

Rigid arrays have so far been constructed using conventional honeycomb substrates to support the solar cell arrays. The systems are constructed in panels which are hinged together and folded for stowage on the spacecraft and deployed by spring loaded hinges controlled either by pantographs or closed cable systems. Other lighter weight "rigid" systems are now being developed which are weight competitive with flexible arrays in the 1 to 5 kilowatt range. These are the rigid frame/flexible substrate systems and rigid frame/"rigid" substrate systems.

A description of the construction techniques of such designs is summarized in Table 5.3-4 and the relative advantages and disadvantages of all three "rigid" concepts in Table 5.3-3.

Amongst new developments of the rigid frame/flexible substrate design are the MBB, Germany Ultra Light-weight Panel design (Ref. 6) which has a flexible substrate tensioned within a carbon fibre composite framework and the Spar Aerospace Products Ltd. design (Ref. 7) developed under a D.O.C. contract. This latter design also uses a flexible substrate stretched within a carbon fibre composite frame-work. Deployment is controlled by means of a pantograph mechanism. A vibration model of this system has been built and testing is imminent.

The rigid frame/"rigid" substrate concept has been developed by TRW Systems (Ref. 8 and 9) using a Fleet Satcom configuration. The adaptability of this design to smaller configurations is questionable.

In the area of rigid substrates other than conventional aluminum honeycomb core/aluminum or fibreglass skin designs some newer designs have been developed. McDonnell Douglas have experimented with a machined magnesium iso-grid substrate. MBB, Germany have developed a carbon fibre composite skin/aluminum honeycomb core system (Ref. 10) which will be flown on the European Orbital Test Satellite (OTS) and the European Maritime Orbital Test Satellite (MAROTS). Engins MATRA, France, developed a lightweight aluminum honeycomb core/ fibreglass skin substrate a few years ago (Ref. 12) but have not carried on their development.

5.3.3.1.2 Trade-Offs

Under a separate contract with the Department of Communications, a trade-off study conducted at Spar (Ref. 7) showed that the MBB ULP carbon composite frame/tensioned flexible substrate and the Spar Rigid Panel Solar Array (RPSA) - also of similar construction are weight competitive with flexible array systems even up to 5 kilowatt power levels.

For purposes of selection of a design concept for the General Purpose Satellite Bus the following guidelines were established:

CD/85

a) The chosen concept should be of a reasonably well-proven type with little new development work required.

b) The natural frequency of one wing should exceed 0.3 Hz in the deployed state to avoid interaction with ACS.

c) The design should be light-weight within the above constraints.

Using available CTS information for solar cell performance (see paragraph 5.3.4 of this report) and cell packing density and radiation degradation models provided by the D.O.C., three 800 watt end of life arrays were sized. These were for the following concepts:

(i) Rigid honeycomb substrate using aluminum face sheets and aluminum honeycomb core.

(ii) Rigid frame/flexible substrate of the RPSA type (Ref. 7).

(iii) Rigid frame/semi-rigid substrate of the TRW type (Ref. 8 and 9).

The equivalent 1 MeV electron radiation dosage was doubled for concept (ii) to account for backside radiation. This produced array sizes shown in Table 5.3-5. A detailed sizing for concept (iii) was not done. It is anticipated that the higher calculated operating temperature may be compensated for by the additional backside radiation protection in comparison to concept (ii) to eventually produce the same size array as in concept (ii).

The structural elements for the three concepts were sized to produce a greater than 0.3 Hz natural frequency, allowing for hinge degradation. It ws found that concept (i) required a 1 inch deep 1 lb/cu-ft. aluminum honeycomb core with a .0035" face sheet on the solar cell side and .014" face sheet with 50% holes on the rear. The weight for this concept would be 50.2 lb per side or 100.4 lb

TABLE 5.3-4

CONSTRUCTION TECHNIQUES - RIGID ARRAYS		
1) RIGID SUBSTRATE	2) RIGID FRAME/FLEXIBLE SUBSTRATE	3) RIGID FRAME/'RIGID' SUBSTRATE
1. Aluminum honeycomb with facesheets of any one of the following: - fibreglass - aluminum - carbon fibre Depth and density of honeycomb core can be varied to achieve desired stiffness. Rear face sheets can have holes to reduce operating temperature. 2. Light metal (magnesium or aluminum alloy) milled in triangular or rectangular grid with bonded aluminum or kapton skin.	Kapton or kapton/fibreglass substrate typically about .003" (.08 mm) thick, stretched between a framework of rectangular tubular construction. The materials used for the framework can be any one or a combination of the following: - aluminum - carbon fibre - beryllium Depth and thickness of tubes can be varied to achieve desired stiffness. Substrate/cells on adjacent panels are prevented from contacting each other during stowed vibration by the use of short columns termed buttons distributed across the panels.	Aluminum honeycomb with kapton facesheets typically .0015" (.04 mm) thick on the solar cell side and .003" (.08 mm) thick with 50% holes on the rear, supported between a framework of rectangular tubular construction similar to 2). This design may not need as much support within the panel as 2) to prevent contact during vibration.

TABLE 5.3-5

MAJOR TRADE-OFF PARAMETERS - RIGID ARRAYS			
PARAMETER	1) RIGID SUBSTRATE	2) RIGID FRAME/FLEXIBLE SUBSTRATE	3) RIGID FRAME/'RIGID' SUBSTRATE
1. Size - Deployed	Typically 4 panels per side, each panel 59.25" (150.5 cm) long and 50" (127 cm) wide, for 800 watt EOL, assuming $1x10^{15}$ eq. 1 mev electrons total deployed length including elevation yoke = 301.25" (765 cm).	Typically 4 panels per side, each panel 72" (183 cm) long and 50" (127 cm) wide, for 800 watt EOL, assuming $2x10^{15}$ eq. 1 mev electrons to allow for backside radiation. Total length = 365" (927 cm).	Same as (2). Increased operating temperature may be compensated by additional rearside radiation shielding. Detail study not done.
2. Size - Stowed	60.5" x 51" x 7" (154 cm x 130 cm x 18cm)	73" x 51" x 7.75" (185 cm x 130 cm x 20 cm)	Same as (2).
3. Weight	About 50.2 lb. per side or 100.4 lb. per spacecraft.	About 40.7 lb. per side or 81.4 lb. per spacecraft.	About 45.3 lb. per side or 90.6 lb. per spacecraft.
4. Natural frequency - deployed	0.31 Hz using 1", 1 lb/cu.ft. Al. H/C core, .0035" al. skin - cell side, .014" al. skin - rear, 50% holes.	0.46 Hz using beryllium tubes, 0.32 Hz using carbon fibre tubes.	0.44 Hz using beryllium tubes, 0.30 Hz using carbon fibre tubes.
5. Operating temperature -			
a) summer solstic	52.2°C	~52°C end panels, 46°C intermediate	~61.4°C end panels, 55.3°C intermediate
b) equinox	64.4°C	~64°C end panels, 57.7°C intermediate	~75.7°C end panels, 68.2°C intermediate
6. Development Status	Honeycomb rigid substrate construction most commonly used - well proven.	This concept is in preliminary design stage at SPAR. Vibration test to prove the concept is planned.	Preliminary static load and acoustic noise tests have been carried out by TRW of this concept.

per spacecraft. (Note: These weights do not include the Orientation and Power Transfer System, but do include the solar cells and elevation yoke).

Concepts (ii) and (iii) would weigh 40.7 lb per side and 45.3 lb per side respectively with natural frequencies of 0.46 Hz and 0.44 Hz using beryllium frames and 0.32 Hz and 0.30 Hz respectively for an equi-weight carbon composite frame structure. Each concept assumed using a beryllium yoke.

The above trade-off indicated that designs could be produced with all three concepts to meet conservative natural frequency requirements. However, the development status of concepts (ii) and (iii) is not regarded as sufficiently advanced to recommend them for this application. Subsequent to this trade-off it was learned that concept (iii) may be limited in its applicability to smaller frame sizes than those used on the TRW Fleet Satcom spacecraft and also in a concertina design, due to large substrate vibration amplitudes anticipated.

Also a cost analysis performed during the previous trade-off study mentioned above showed a significant cost increase with the use of carbon fibre composites and beryllium.

5.3.3.1.3 Description of Design

On the basis of the above, it was decided to pursue the conventional honeycomb substrate design as the baseline design concept for the General Purpose Satellite Bus DSA. A layout of this concept is shown in drawing 31138J2 - Layout - Solar Array General Purpose Bus - Figure 3-5.

The system basically consists of four panels on each side of the spacecraft - North and South, plus an elevation yoke. The elevation yoke is hinged down at the interface with the forward deck mounted orientation and power transfer system. A deployment drive will be mounted at this joint.

This drive will control rotation of the elevation yoke and by means of a belt drive the rotation of the pantograph which starts at the mid-point of the yoke. There will be three inter-panel hinges which will feature deployment springs and locks with an anti-backlash feature. The pantographs links will be pivoted at mid-panel points and will interface with each other at the same line as the inter-panel hinges. The pantograph/deployment mechanism will merely control deployment, the deployment energy itself being provided by the hinge springs.

For stowage, it is anticipated, that a four-point hold down system will be used. Release will be by a central redundant pyrotechnically actuated system.

Details of the tie-down/release system and a more detailed tradeoff between a pantograph and a closed cable deployment control system yet need to be carried out. These, it is anticipated could be done during a preliminary design phase.

In order to provide the versatility required of a General Purpose Bus, it is anticipated that a special hinge/lock design will be required that will allow the use of either 4 or 3 panels per side and still retain the transfer orbit power feature. Also the provision of zero backlash will require innovative design.

5.3.3.2 Array Orientation and Power Transfer

Table 5.3-6 shows three basic alternate configurations that the array orientation and power transfer might take. These are:

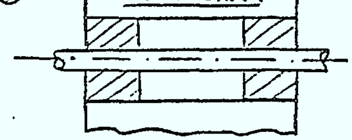
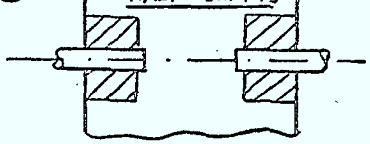
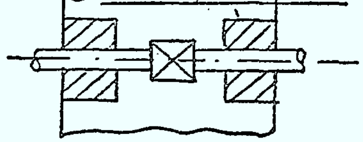
a) The through shaft configuration i.e. both arrays driven together.

b) The half shaft, or split configuration, i.e. each array driven independently.

c) The half shaft with flexible connector.

CD/88

TABLE 5.3-6

ARRAY ORIENTATION & POWER TRANSFER - ALTERNATE CONFIGURATIONS			
CONFIGURATION	① THRU SHAFT 	② HALF SHAFTS 	③ ② WITH FLEX CONNECTOR 
ADVANTAGES	∘ COMPLETE SYNCHRONIZATION OF ARRAYS ∘ ENTIRE DEVICE CAN BE BUILT & TESTED AS AN ASSY. ∘ BEARING PRELOAD CAN BE ACCURATELY ADJUSTED AT ASSY. ∘ GIVES BEST ARRAY ROOT STIFFNESS ∘ PERMITS MAX. POWER TRANSFER VOLUME ∘ PROBABLY MOST RELIABLE	∘ TEMPERATURE COMPENSATION MINIMUM. ∘ SEIZURE OF ONE END DOES NOT AFFECT THE OTHER. ∘ EACH ASSY. OF SMALLER VOLUME THAN ①. ∘ POTENTIALLY LIGHTEST	∘ COMPLETE SYNCHRONIZATION ∘ FLEX - CONNECTOR PERMITS FLEXURE DUE TO TEMPERATURE EFFECTS. ∘ ONE DRIVE COULD BE OPERATIVE IF OTHER FAILED.
DISADVANTAGES	∘ GREATEST VOLUME ∘ SEIZURE OF MAIN BEARINGS AT EITHER END CAUSES SYSTEM FAILURE ∘ INDEPENDENT OPERATION OF ARRAYS NOT POSSIBLE. ∘ PROBABLY HEAVIEST ∘ CENTRE OF ROTATION OFFSET FROM S/C C. OF.M CAUSES SOLAR TORQUES.	∘ SYNCHRONIZATION BETWEEN ARRAYS MORE DIFFICULT. ∘ GIVES LESS ARRAY ROOT STIFFNESS THAN ①. ∘ MAY REQUIRE HANDED CONFIGURATIONS. ∘ EACH MODULE HAS SIMILAR RELIABILITY TO ① BUT OVERALL RELIABILITY MAY BE LOWER.	∘ POWER TRANSFER SIZE LIMITED. ∘ LESS ARRAY ROOT STIFFNESS THAN ①. ∘ MAY REQUIRE HANDED CONFIGURATIONS ∘ SLIGHTLY HEAVIER THAN ② ∘ RELIABILITY SIMILAR TO ② ∘ OFFSET CENTRE OF ROTATION - SIMILAR TO ①

In previous trade-off studies carried out at Spar (Ref. 13 and 14), the through shaft configuration was chosen as the preferred one for reasons of reliability, stiffness and power transfer capability. Arguments forwarded in this trade-off are summarized in Table 5.3-6.

On the basis of these arguments, a through shaft configuration has been chosen as the preferred approach for the General Purpose Satellite Bus. No detailed weight trade-off has been made.

In a separate study contract with the Department of Communications, a survey of existing orientation and power transfer systems was carried out (ref. 15). This survey showed a variety of design philosophies used by manufacturers of such systems. The report recommends the conduct of a detail trade-off study to select between direct drive brushless torquer and geared stepper motor drive systems. However, in the absence of a detail trade-off, on the basis of information available, a recommendation for a geared stepper motor driven mechanism is made.

For the baseline design concept of the GPB DSA an existing design of such a device has been assumed. The space available allows the use of many of the surveyed devices or adaptations thereof. A detailed selection will have to be made at a later stage.

It has been assumed that control of the GPB orientation system will be carried out by the electronics within the bus attitude control system. The mechanism will be capable of providing two basic rates - 1 revolution per day and 15 degrees/minute. An analog sun sensor - also part of the ACS - will be mounted on the array.

The orientation and power transfer system is shown conceptually blocked in drawing number 31138J2 - Figure 3-5.

CD/89

5.3.4 Array Sizing

5.3.4.1 Solar Cell Layout and Sizing

5.3.4.1.1 Solar Cell Performance

Based on the experience of the Communications Research Centre during the design and fabrication of the Communications Technology Satellite, a recommendation was made to use solar cell performance parameters typical of the AEG-Telefunken cells used on the CTS DSA Housekeeping Section. These cells were subjected to selection and performance matching and produced an average of 65 milliwatts per cell at 25°C, A.M.O. The following open-circuit voltage, short-circuit current and maximum power point voltage and current characteristics were suggested for B.O.L. at 25°C:

$$V_{oc} = 0.5981V$$
$$I_{sc} = 0.1377A$$
$$V_{mp} = 0.4993V$$
$$I_{mp} = 0.1288A$$

The following models were suggested for calculating radiation degradation:

$$\frac{I_{sc}}{I_{sc\phi}} = 1.0 - 2.147 \times 10^{-9} \text{ (Dose)}^{.5334}$$

$$\frac{I_{mp}}{I_{mp\phi}} = 1.0 - 2.277 \times 10^{-9} \text{(Dose)}^{.5334}$$

$$\frac{V_{oc}}{V_{oc\phi}} = 1.0 - 1.552 \times 10^{-10} \text{ (Dose)}^{.5855}$$

$$\frac{V_{mp}}{V_{mp\phi}} = 1.0 - 2.871 \times 10^{-9} \text{ (Dose)}^{.5003}$$

The suffix ϕ stands for original or beginning of life. "Dose" stands for the equivalent 1 MeV electron flux. For a rigid substrate design, with adequate rear-side shielding (equivalent to the front 6 mil cover glass) this is 1×10^{15} electrons per cm^2 of 1 MeV electrons and for a flexible substrate design, this is 2×10^{15} electrons/cm^2 of 1 Mev electrons.

The following models were suggested for the cell temperature coefficient:

$$I_{sc} = 0.815 \times 10^{-4} \text{ per } °C$$
$$I_{mp} = 0.074 \times 10^{-4} \text{ per } °C$$
$$V_{oc} = -2.137 \times 10^{-3} \text{ per } °C$$
$$V_{mp} = -1.944 \times 10^{-3} \text{ per } °C$$

The following additional degradation factors other than radiation and temperature were suggested for array sizing:

Cell Mismatch	0.98 or 2%
Cover Glass Darkening	0.99 or 1%
Cell Breakage	0.98 or 2%
Calibration Uncertainty	0.99 or 1%
Summer Solstice sun intensity	0.96 or 4%
Solstice Angle effect	0.917 or 8.3% (cos 23.5°)

Using the above models and factors the performance of the typical AEG cell was derived for end-of-life for both the rigid substrate design and flexible substrate design, using calculated summer solstice operating temperature based on the CTS AEG array thermo-optical properties. The results are summarized in Table 5.3-7. This shows a cell performance, taking into account radiation degradation and summer solstice operating temperature, of 40.37 mw for the rigid substrate and 34.07 mw for the flexible substrate. When the other degradation factors are applied, the normalized cell performance for sizing purposes becomes 33.45 mw and 28.23 mw respectively.

5.3.4.1.2 Array Sizing

Using the end of life cell performance calculated above, the minimum number of solar cells required for an 800 W E.O.L. array can be obtained - see Table 5.3-8. The CTS flexible solar cell array is a good example of solar cell layout and has been used to arrive at a packing density factor for preliminary sizing. A typical CTS Solar Panel Assembly (SPA) is of 1756 sq. inch area and has 1944 cells. This gives a factor of 0.9 sq. inch

TABLE 5.3-7

TYPICAL AEG - CTS TYPE SOLAR CELL PERFORMANCE - mW/Cell -				
NO.	STATUS	FOR RIGID SUBSTRATE	FOR FLEXIBLE SUBSTRATE	REMARKS
1	Beginning of Life @ 25°C, A.M.O, Max. Power Point	65.00	65.00	Based on CTS Housekeeping Array - Cell selection used
2	After radiation dosage	45.48	37.90	1 x 10^{15} eq. 1 MeV electrons for rigid 2 x 10^{15} eq. 1 MeV electrons for flex.
3	At summer solstice temperature	40.37	34.07	52°C for rigid substrate 48°C average for flexible
4	Cell mismatch	39.56	33.39	2 % loss
5	Cover glass darkening	39.17	33.05	1 % loss
6	Cell breakage	38.38	32.39	2 % loss
7	Calibration uncertainty	38.00	32.07	1 % loss
8	Summer solstice sun intensity	36.48	30.79	4 % loss
9	Solstice angle effect	33.45	28.23	8.3% loss (cos 23.5 = 0.917)
	End of Life power/cell - normalized due to effects 4 to 9	33.45 mW	28.23 mW	Max. power point occurs at approx. 0.401V, for rigid at E.O.L. and at approx. 0.390 V for flex, E.O.L. (This applies at Status 3)

per cell. Using a 10% factor above this to allow for tie-down points and hinges for the rigid substrate design and 12.5% to allow for frames for the rigid frame design the total required area has been calculated.

This typically gives, for a 50" wide array (a practical width considering the launch vehicle fairing envelope, spacecraft size and substrate depth), 4 panels per side of lengths shown in Table 5.3-8.

Assuming an average solar cell string wiring length per wing of 220 inches and limiting the current density to 4 Amps/mm^2 a voltage drop due to wiring of 0.4V is calculated per wing. Voltage drop across blocking diodes will typically be 0.8V. In order to operate the array at 40V, the string voltage at the cells will need to be 41.2V. In order to operate the array on the stable side of the solar cell maximum power point at all times (V_{mp} = 0.401V), the minimum number of solar cells per series string

$$= \frac{41.2}{0.401} = 103 \text{ cells}$$

for the rigid substrate and

$$\frac{41.2}{0.390} = 106 \text{ cells for the}$$

flexible substrate (V_{mp} = 0.390V)

5.3.4.1.3 Cell Layout

A preliminary solar cell layout for the rigid substrate design is shown in Table 5.3-9. Connecting 9 modules of cells - a module comprising 12 cells in series, 3 in parallel - in series provides larger than the minimum required number of cells in series. Connecting 10 of these matrices in parallel as shown gives a panel that will produce 108.4 Watts at 43.3 Volts, end of life.

The central ground return feature gives a zero magnetic movement.

R677/92

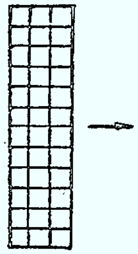
TABLE 5.3-8

PRELIMINARY ARRAY SIZING			
NO.	FACTOR	RIGID SUBSTRATE	FLEXIBLE SUBSTRATE
1	Minimum No. of cells required for 800 W end of life	23,917 cells	28,339 cells
2	Area required - using 1.1 x CTS packing factor for rigid 1.125 x CTS packing factor for rigid frame/flex. substrate to allow for frames	23,678 sq.in.	28,693 sq.in.
3	Length of each panel for a 4 panel per side, 50" wide array	59.2 in.	71.7 in.
4	Minimum No. of cells required per panel	2,990 cells	3,543 cells
5	For 40 volt array operation, minimum No. of cells in a series string (allowing for average 0.4V drop in wiring and 0.8V drop in diodes)	103 cells	106 cells

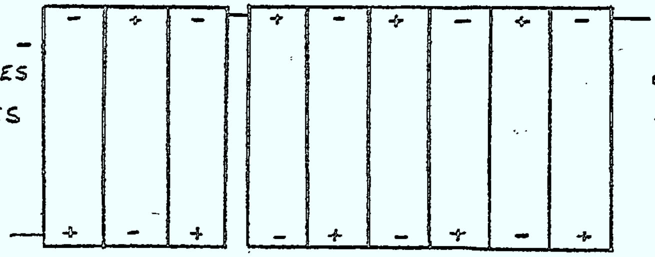
TABLE 5.3-9

PRELIMINARY CELL LAYOUT - RIGID SUBSTRATE

MODULE - 12 CELLS IN SERIES, 3 CELLS IN PARALLEL

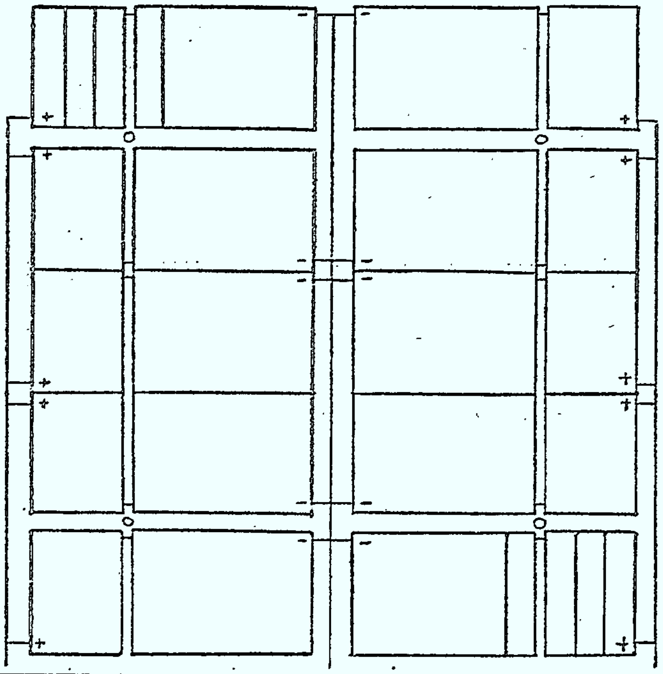


MATRIX - 9 MODULES IN SERIES



PROVIDES - 108 CELLS IN SERIES, 10.84 WATTS AT 43.3 VOLTS AT E.O.L

PANEL - 10 MATRICES IN PARALLEL



FEATURES -

- 108.4 WATTS AT 43.3 VOLTS, E.O.L
- CENTRAL GROUND RETURN FEATURE GIVES ZERO MAGNETIC MOMENT.
- ALLOWS 6" WIDTH FOR FCC'S
- ALLOWS 6" IN LENGTH DIRECTION FOR TIE DOWN POINTS, HINGES.
- 8 PANELS PER SPACECRAFT, PARALLEL CONNECTED PROVIDE 867 WATTS AT 43.3 VOLTS AT CELL STRINGS, OR 835 WATTS AT 40 VOLTS ON SPACECRAFT SIDE OF DIODES.
- TO COMPLETELY AVOID PARABOLIC UHF ANTENNA SHADOWING, INBOARD-MOST STRING CAN BE 18, 6x3 MODULES IN SERIES, GIVING 1.25% POWER REDUCTION OR 825 WATTS AT 40 VOLTS.

Eight of these panels per spacecraft connected in parallel will produce 867 Watts at 43.3 Volts at the cell strings or 835 watts at 40 volts on the spacecraft side of the diodes, allowing for wiring and diode losses.

This layout allows 6 inches in the width direction for flat conductor cable wiring and 6 inches per panel in the length direction for tie-down points and hinges.

With a 61.25 inch elevation as shown in the layout 31138J2, the bottom 4.75 inch of the inboard panel will temporarily be shadowed by the UHF antenna around midnight during solstice seasons.

This shadowing will tend to reduce the array power by two strings or 20 watts intermittently. If this is not acceptable the bottom most string can be made of 18, 6 x 3 modules connected in series across the width of panel giving a constant reduction of 10 watts.

5.3.4.2 Transfer Orbit Considerations

Table 5.3-10 shows how the array sized above will meet the transfer orbit requirement of 80 watts.

The 65mW, B.O.L. at 25°C cell will operate at 45.75mW after 10 transfer orbits and other degradation factors applied as shown. The reason for the low performance is that the array will be operated considerably below its maximum power point with the power subsystem being regulated at 40 Volts.

The peak power per panel will be 148W, varying in a cosine fashion as shown. The average power output will be 94W at 40V.

5.3.4.3 Adaptability to Payloads

Table 5.3-11 shows how the design arrived at can be adapted for payloads A to D.

Utilizing all 8 panels will provide 835 Watt and satisfy Payload B.

TABLE 5.3-10

TRANSFER ORBIT CONSIDERATIONS.

BASED ON CTS. THE OPERATING TEMPERATURE RANGE WILL BE 3°C TO 24°C TAKING INTO ACCOUNT SEASONAL VARIATIONS AND SUN ANGLE TO NORMAL (OF S/C SPIN AXIS) OF ± 25°

POWER OUTPUT OF THE 65 mW/B.O.L CELL @ 25°C WILL BE AFFECTED AS FOLLOWS:

PARAMETER	DEGRADATION FACTOR
MISMATCH & ASSEMBLY	0.98
BREAKAGE & U-V DEGRADATION	0.98
CALIBRATION UNCERTAINTY	0.99
SOLSTICE INTENSITY	0.96
RADIATION DEGRADATION (3.5×10^{13} ER. 1 MEV ELECTRONS)	0.97
SUN ANGLE EFFECT ($\cos 25°$)	0.906
TEMPERATURE EFFECT (3°C)	1.088
OVERALL DEGRADATION	0.873

POWER PER CELL = 56.75 mW @ THE MAX. POWER POINT, AT 0.491 VOLTS (53 V. STRING)

TO OPERATE AT A VOLTAGE OF 40 V

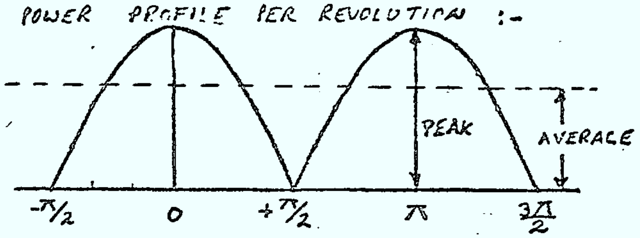
POWER PER CELL ≃ 52.4 mW, B.O.L.

≃ 45.75 mW, E.O.L

NUMBER OF CELLS = 3240 PER PANEL

PEAK POWER/PANEL = 148 W

POWER PROFILE PER REVOLUTION :-



AREA UNDER EACH CURVE $= \int_{-\pi/2}^{\pi/2} \cos\theta \, d\theta = 2$

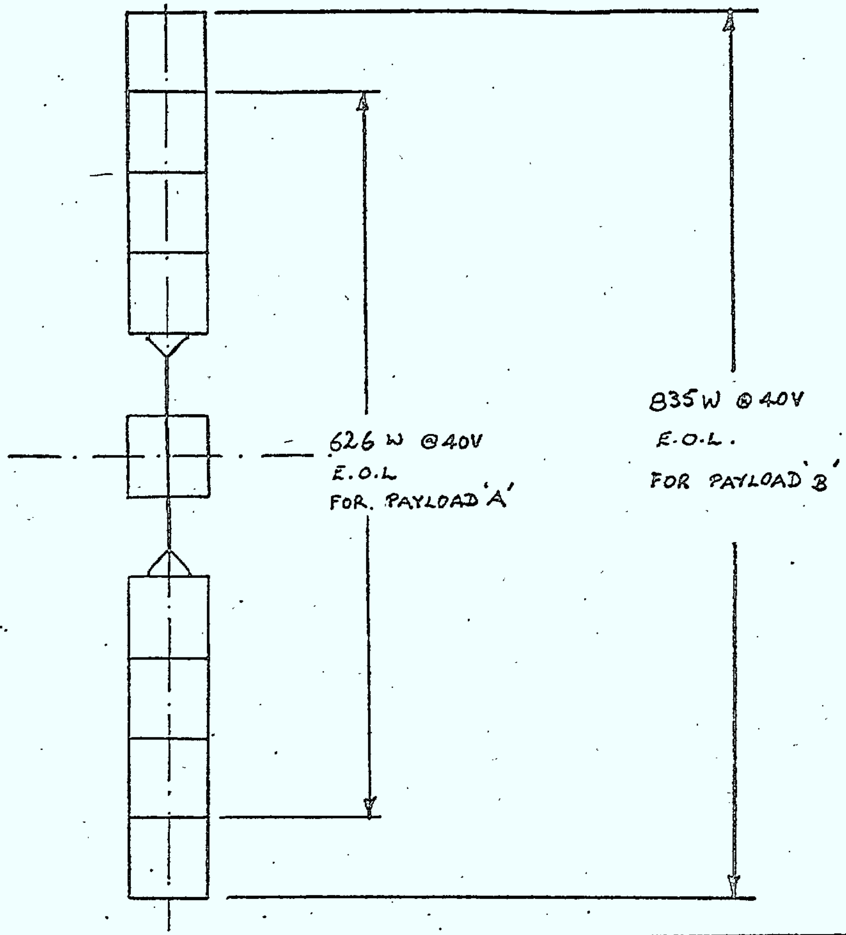
AVERAGE HEIGHT $= 2/\pi$

$\therefore$ AVERAGE POWER $= 148 \times \frac{2}{\pi} = 94$ W @ 40 V

SPAR

TABLE 5.3-11

ADAPTABILITY TO PAYLOADS



FOR PAYLOADS C & D :-

OPTION 1 – UTILIZE ONLY 3 ½ PANELS / SIDE GIVING:- 731 W E.O.L

OPTION 2 – REDUCE PANEL LENGTHS BY 12.5% TO 52" EACH. NEW CELL LAYOUT & CELL MODULES REQUIRED. TRANSFER ORBIT POWER WILL BE REDUCED TO 82 WATTS.

Using only 6 panels will provide 626 Watts to satisfy payload A. This will require the hinges to be of such a design that panels can be inverted to retain a cell covered side facing outward in the stowed condition to provide transfer orbit power.

For payloads C&D (700W), two options exist. One is to not provide a full complement of solar cells on the inboard most panels and the other is to produce a new design with 12.5% less area. The latter approach is not economical.

5.3.4.4 Potential Improvements

There is an indication now that Violet Cells in the U.S. and High Efficiency Blue Cells in Europe are becoming viable on a production basis.

Both S.A.T. in France and AEG Telefunken in Germany are in the process of having their production cells qualified by ESTEC. These cells will produce 50 mW at 25°C after a dosage of 1 x 10^{15} 1 MeV electrons/cm^2.

This compares with the 45.48 mW/cell for the baseline cell assumed in this tudy - a 10% improvement. Also, the begnning of life power is lower - 61mW compared to 65mW for the baseline cell - giving a smaller variation in power, B.O.L. to E.O.L.

Although a cell layout and preliminary design has not been preformed, using these high efficiency cells, at least a 5% reduction in size and weight of the system can be expected.

Further, another 5% weight reduction may be achieved by using the rigid frame/flexible substrate array if sufficient development work is done.

R677/94

5.3.5 System Weight

Although the original trade-offs were carried out assuming the need for a natural frequency of 0.3 Hz, this requirement has subsequently been reduced to 0.15 Hz by the ACS. It is estimated, therefore, that a weight saving can be affected in the stowage system. On this basis, the following weight estimate is made of the General Purpose Bus, DSA subsystem:

Solar Cell Array and Substrate using violet cells (including hinges, pantographs)	91.0 lbs
Deployment Mechanism	2.0 lb.
Wiring, Tie-Down/Release System Miscellaneous Hardware	5.0 lb.
Orientation & Power Transfer System	22.0 lb.
	120.0 lb.

5.3.6 Conclusions and Recommendations

A preliminary design concept for the General Purpose Satellite Bus Deployable Solar Array subsystem has been created. This has been based on an evaluation of presently available alternate concepts of stowage and deployment. Use has been made of available literature, past trade-offs and trade-off conducted during this study to arrive at a recommended design approach.

Preliminary sizing and performance parameters of this concept have been arrived at.

Areas of potential improvement are pointed out. It is recommended that more work, if possible, be done in the following areas:

a) Establishing confidence in Violet or Blue Cells and sizing a system for this improved performance.

b) Conducting vibration tests on the rigid frame/flexible substrate design to establish confidence in its viability.

CD/95

c) Conducting a detailed trade-off on pantograph versus closed cable deployment systems.

d) Establishing a tie-down/release system approach based on stowage system parameters and spacecraft interaction.

e) Conducting a detail trade-off study between brushless torquer and geared stepper motor drive systems.

5.3.7 References

1. The Flight of the FRUSA, by George Wolff and Alois Wittmann, Hughes Aircraft Company, El Segundo, Calif. AIAA 9th Electric Propulsion Conference, Bethesda, M.D. April 17-19, 1972.

2. The Communications Technology Satellite Deployable Solar Array Subsystem, by S.S. Sachdev, E. Quittner, J.D. Graham, Spar Aerospace Products Ltd., Toronto, Canada. International Photo Voltaic Power Generation Conference, Hamburg, West Germany, September, 1974.

3. Design and Qualification of the CTS Solar Cell Blanket, by P. Paulsen and W. Woodcock, AEG-Telefunken, Hamburg, Germany, P. Sachs, ESTEC, Noordwijk, Holland. International Photo Voltaic Power Generation Conference, Hamburg, West Germany, September, 1974.

4. Development of a Flexible, Fold-Out Solar Array, by J. Fremy and D. Forans, Societe Anonyme de Telecommunications, Paris, France. International Photo Voltaic Power Generation Conference, Hamburg, West Germany, September, 1974.

5. Flexible Solar Array Applications in Communications Satellites, by W.J. Billerbeck and D.J. Curtin, Comsat Laboratories, Clarksburg, M.D., 9th Intersociety Energy Conversion Engineering Conference, San Francisco, California, August 26-30, 1974.

CD/96

6. Requirements and Design of an Ultra Light Solar Array (ULP), by H.V. Bassewitz, H. Schweig, MBB, Munich, International PhotoVoltaic Power Generation Conference, Hamburg, September, 1974.

7. SPAR-R.662, Final Report Rigid Panel Solar Array Development Program, March, 1975.

8. Lightweight Rigid Solar Array Development", by W. Luft and R.E. Patterson, TRW Systems Group, Redondo Beach California, 11th IEEE PhotoVoltaic Specialists Conference, Phoenix, Arizona, May 6-8, 1975.

9. Lightweight Rigid Solar Array - Structural Considerations, D.A. Corbett and W.J. Dean, TRW Systems. 11th IEEE PhotoVoltaic Specialists Conference, May 6-8, 1975.

10. Development of an Advanced Lightweight Rigid Solar Array by R.L. Crabb, ESTEC and Karl Schneider, MBB, Germany, 10th IEEE PhotoVoltaic Specialists Conference, Palo Alto, California, November, 1973.

11. Power Generation for the X4 Spacecraft - A Step in the Development of a High Power/mass Ratio, Hybrid Solar Array for Applications Spacecraft, by B. Collins, Hawker Siddeley Dynamics Ltd., Stevenege, U.K., International PhotoVoltaic Power Generation Conference, Hambur, September, 1974.

12. Problems of Development and Test of Large Lightweight Solar Arrays, by H. Larsson, Engins MATRA, Velizy, France, International Congress - The Sun in Service of Mankind, Paris, 2-6 July, 1973.

13. SPAR-TM.774 - Early Version Intelsat V - Technology Evaluation, May, 1972.

14. SPAR-TM.774/1 - Early Version Intelsat V Solar Array Assembly Trade-Offs, June, 1972.

CD/97

15. SPAR-R.669 - Report on Technology Survey of Solar Array Orientation and Power Transfer Mechanisms, 30 April, 1975.

5.4 Structure Subsystem

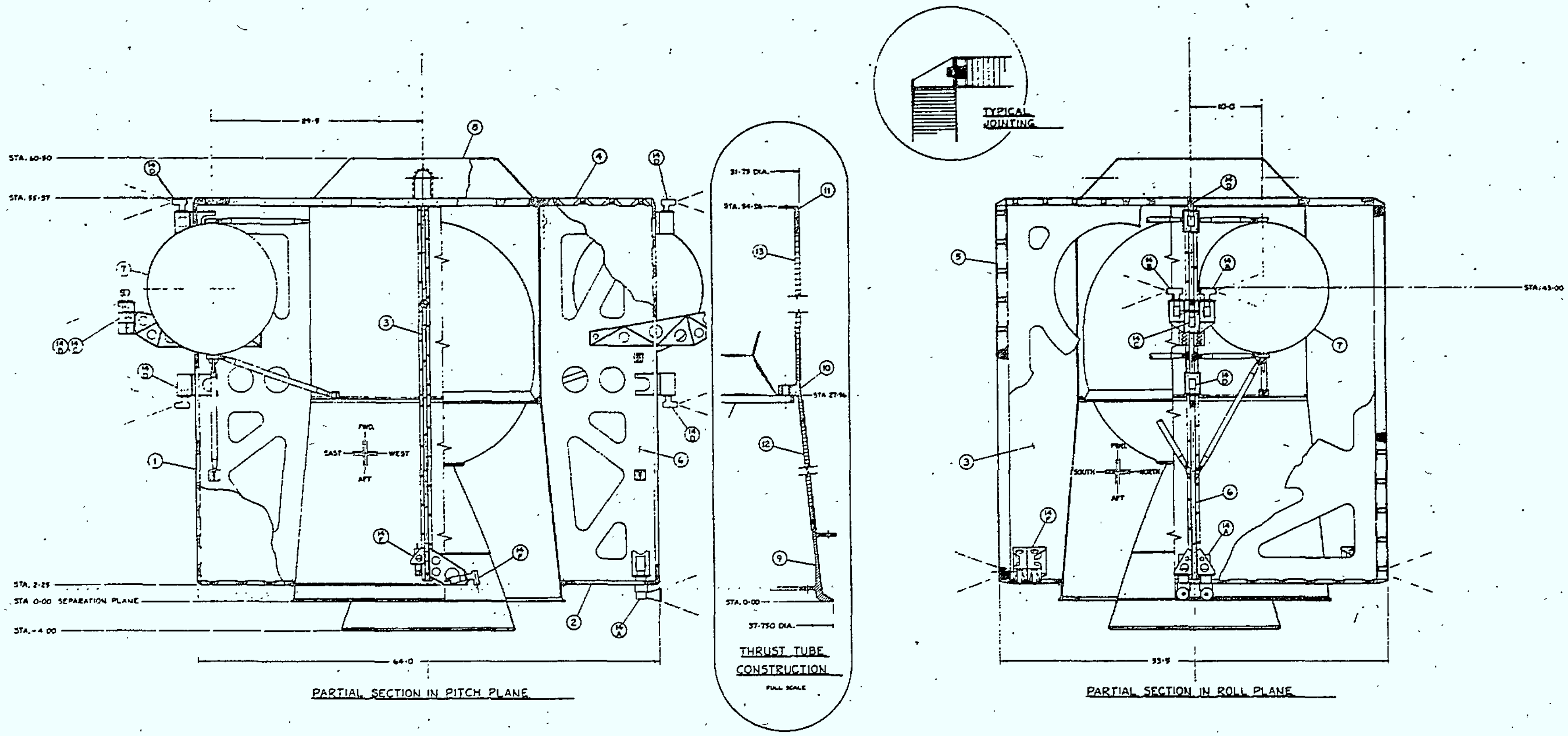
5.4.1 Structure Description (See Figure 5-7)

For this study the General Purpose Satellite Bus structure has been designed to comply with the requirements of minimum weight, maximum payload mounting areas and compatibility with the 3914 launch vehicle, utilizing in the main conventional materials and fabrication techniques. In addition, consideration has been given to those requirements peculiar to this application, i.e., ready adaptability to varying payload arrangements and freedom to locate IMU's within the payload areas to achieve balance with a minimum of impedance by structural bracing or support members.

The central thrust member consists of a thin wall tube, partly a cylinder, and partly a frustum, large enough to accommodate and mount the apogee motor and insulation within. It is fabricated from three machined rings; an aluminum separation ring at the aft end, an aluminum intermediate ring at the cylinder/frustum interface used also to mount the apogee motor, and a magnesium ring at the forward termination. The rings are connected by overlapped monoply half-shells of magnesium and fastened by close-pitched rivetting. An alternate and lighter structure would be to connect the rings with bonded aluminum honeycomb substrate as illustrated on Drawing No. 31138J1, Figure 5-7. This method, as yet untried, would require a development program to establish techniques and integrity.

Fastened directly to the thrust tube and running almost its entire length are four bulkhead panels lying external and on the plane of the spacecraft's east/west and north/south axes. These bulkheads, in cruciform arrangement, act as the main load servers into the thrust tube, and are utilized to support the reaction control system and peripheral external panels of the spacecraft. They are fabricated from aluminum honeycomb core and face-sheets and considerably weight relieved by a systematic removal of panel in areas of neutral stress.

CD/100



COMPONENTS :

(1) EAST AND WEST CLOSURE PANELS
(2) AFT LATERAL RESTRAINT PANEL } 0·5 THICK 2·0LB/FT³ ALUM. HONEYCOMB, ·006 ALUM. FACESHEETS
(3) NORTH AND SOUTH BULKHEADS
(4) FORWARD EQUIPMENT PLATFORM 1·0 THICK " " " " " "
(5) NORTH AND SOUTH EQUIP'T. PANELS 1·5 THICK " " " " " "
(6) EAST AND WEST BULKHEADS 0·62 THICK " " " " " "
(7) HYDRAZINE FUEL TANKS TITANIUM
(8) DISH ANTENNA MOUNT 0·04 THICK ALUM. SPINNING

(9) SEPARATION RING
(10) APOGEE MOTOR RING } MACHINED ALUMINUM
(11) FORWARD RING " MAGNESIUM
(12) AFT THRUST TUBE 0·125 THICK 3·1 LB/FT³ ALUM. HONEYCOMB, ·007 ALUM. FACESHEETS
(13) FWD. THRUST TUBE " " 2·0LB/FT³ " " ,·005 " "
(14) THRUSTERS : A = HIGH THRUST PRECESSION, B = NORTH/SOUTH STATION AND YAW ATTITUDE,
C = EAST/WEST STATION, D = EAST/WEST STATION AND PITCH ATTITUDE,
E = NORTH/SOUTH STATION AND ROLL ATTITUDE, F = OFFSET, ROLL/YAW ATTITUDE.

SPAR AEROSPACE PRODUCTS LTD.
STRUCTURE AND R.C.S.
GENERAL PURPOSE SATELLITE BUS
34480 B 31138J1

FIGURE 5-7

The bulkheads are supported in a radial attitude by an aft lateral restraint panel and a forward platform. The aft restraint panel is designed similarly to the bulkheads, with extensive weight removal, and serves to convey lateral forces from peripheral panels directly into the separation ring of the thrust tube. The forward platform is fabricated from 1.0 inch thick aluminum honeycomb panel, and in addition to bulkhead constraint, serves as a major payload mounting facility.

The structure thus far may be considered as a primary or basic assembly, at which time all RCS components can be integrated. As can be noted on Drawing No. 31138J1, all REM's are mounted via bracketry directly to the bulkheads and hydrazine fuel tanks are mounted via support struts to the east/west bulkheads and the thrust tube. The primary structure could be further abbreviated by applying an interim constraint fixture to the bulkheads instead of the forward platform at and during the period immediately following RCS integration. All peripheral panels are designed to be attachable and removeable without disturbing the RCS assembly.

Peripheral panels form the exterior of the spacecraft body, and lie in planes normal to the north/south and east/west axes. The north and south panels are constructed of 1.5 inch thick aluminum honeycomb and are the principal payload carriers of the vehicle, affording a large internal surface for the mounting of electronic equipment and a large uninterrupted surface externally for launch stowage of the deployable solar arrays. East/west panels are 0.5 inch thick aluminum honeycomb, weight relieved, and are designed as support members for north/south panels. In addition to this function, they are considered to be the last on/first off access panels to the spacecraft interior. These panels could also provide mounting locations for light-weight miscellaneous items of payload.

All bulkheads, panels and platforms are designed to include integrally a contoured and thinly machined edging which is bonded to both core and facesheets. These edge members provide a facility for frequent attachment of panel to panel, panel to thrust tube, etc., utilizing small machine screws and replaceable locking inserts. By this method, load paths are diffused over many points, thus avoiding concentrated areas of stress which would require localized reinforcement in the form of core densification and facesheet doubling. This form of structural coupling also affords a complete absence of payload obstruction right up to the field limits of the panels, and allows for continuous placement of second surface mirrors, etc.

The structure, while staying within the realm of established manufacturing methods and processes, offers a versatile light-weight general purpose enclosure for the conveyance and environmental protection of payloads assuming wide differences in configuration.

5.4.2 Structural Analysis

The structure was optimized for strength and stiffness using the constraints and criteria given in Section 5.4.2.1.

Design assumptions were used as stated in Section 5.4.2.3.

The major spacecraft vibration modes and associated frequencies are given in Section 5.4.2.4, the computer model used to derive these is briefly described in a footnote.

A light-weight design has been achieved while meeting all strength and stiffness criteria. The Structure Summary Table 5.4-1 gives the pertinent details.

During this study the honeycomb type thrust tube was optimized from strength, stiffness and weight point of view. It is considered that the .006 inch thick facesheet version meets both the strength and the natural frequency requirements as indicated in Para. 5.4.2.4.2.

TABLE 5.4-1

SUMMARY TABLE - STRUCTURE

Item	Structural Characteristics	Weight (lb)	Structural Function	First Natural Frequency (Hz)	Critical Loading Case	Type of Critical Loading or Stress	Ultimate Stress or Load	Allowable Stress or Load	M.S. (Interaction Formula)
Forward Thrust Tube	Honeycomb Cylinder Face Sheets t = .004 in. Al Alloy, Core: 2 lb/ft^3 Al Alloy, Core Depth = .125 in.	5.3	To resist Fwd. Platform vertical & lateral loads. (own, or induced by N&S panels). Combined axial load, and bending, & shear	Not Applicable	Case 1	Intra-Cell Buckling	P_c = 12,460 #in. M_c = 14,200 #in.	P_{allow} = 34,000 # in.	2.1
					Case 3	Intra-Cell Buckling	P_c = Small M_c = 120,000 # in.	M_{allow} = 270,000 # in.	1.2
	Alternate: Mag. Alloy Sheet, t = .040 in.	7.0				Local Buckling			>0
Forward Platform to Thrust Tube, Ring	Magnesium Alloy	1.0	To transfer fwd. platform loads into thrust tube	Not Applicable	Case 1	Local loads on flange	Satisfactory by comparison with corresponding CTS spacecraft ring		
Apogee Motor Ring	Al Alloy	5.8	To transfer apogee motor loads to thrust tube	Not Applicable	Case 1	Local loads on flanges	Satisfactory by comparison with CTS spacecraft ring		
Aft Thrust Tube	Honeycomb, Conical Section. Face Sheets t = .006" Al Alloy, Core: 3.1 lb/ft^3 Al Alloy Core Depth: 1/8 in.	6.9	To transfer axial, shear & bending loads from the spacecraft into the adapter	Not Applicable	Case 1	Face plate wrinkling	P_c = 44,000 #in. M_c = 90,000 #in.	P_{allow} = 65,000 # in.	0.21
					Case 3	Face plate wrinkling	P_c = Small M_c = 328,000 # in.	M_{allow} = 600,000 # in.	0.85
	Alternate: Mag. Alloy Sheet, t = .080 in.	14.2				Local Buckling			>0
Separation Ring		8.0	To carry the structural loads into the adapter	Not Applicable			By comparison with CTS, also designed by GSFC stiffness criteria		

TABLE 5.4-1 Cont'd

Item	Structural Characteristics	Weight (lb)	Structural Function	First Natural Frequency (Hz)	Critical Loading Case	Type of Critical Loading or Stress	Ultimate Stress or Load	Allowable Stress or Load	M.S. (Interaction Formula)
Forward Platform	Al Alloy Honeycomb Face Sheets t = .006" Core: 2 lb/ft^3 Core Depth = 1 inch	9	To carry loads from the equipment on platform into the thrust tube				Satisfactory by comparison with North and South Panels		
North Panel	Al Alloy Honeycomb Face Sheets t = .006" Core: 2.0 lb/ft^3 Core Depth = 1.5 in.	13	To carry the equipment loads into the thrust via the shear web and forward and aft platform	35 Hz 1st Panel Bending	N/S assumed panel lateral vibration response Levels: 30g Peak 20g Mean (Qual.)	Plate bending (Face plate wrinkling)	30 KSI	39.7 KSI	0.32
South Panel	Same as North Panel	13	Same as North Panel	Same as North Panel			Same as North Panel		
East/West Panels	Al Alloy Honeycomb Face Sheets t = .006" Core: 2.0 lb/ft^3 Core Depth = .50" thick	9	Edge support of N/S panels, and box closure	Not Applicable	Case 1 N/S panel Lateral Response	Local bending and direct stresses	9.0 KSI 25.3 KSI	39.7 KSI Face Plate Wrinkling	1.0 0.57
North/ South Rib	Al Alloy Honeycomb (see E/W Panels)	4	To carry N/S panel loads into thrust tube	Not Applicable	Case 1 N/S panel Lateral Response	Local bend. and direct stresses	7.2 KSI 5.6 KSI	39.7 KSI Face Plate Wrinkling	High High
East/West Rib	Al Alloy Honeycomb (see E/W Panels)	5	To carry axial hydrazine tank loads and to support the E/W panels	Not Applicable	Case 1	Local bending and direct stresses	11.1 KSI	39.7 KSI Face Plate Wrinkling	Satisfactory

SPAR

TABLE 5.4-1 Cont'd

Page 3

Item	Structural Characteristics	Weight (lb)	Structural Function	First Natural Frequency (Hz)	Critical Loading Case	Type of Critical Loading or Stress	Ultimate Stress or Load	Allowable Stress or Load	M.S. (Interaction Formula)
Hydrazine Tank Support Struts	5/8 Dia .035" Alum. Alloy Round Tubes	4	To carry axial loads and N/S and E/W vibratory response loads into structure	Not Calculated	Assumed 50g Ult. Lateral 2 Case 1	Strut Compression Loads	640 # 480 #	1,100 # 800 #	.72 .67
Miscellaneous Hardware	Structural Attachments only	4					This weight does not include hardware required to attach payload or any non-structure item and the associated inserts		
Aft Closure Panel	Al Alloy Honeycomb (see E/W Panels)	5	To provide closure panel, and also to provide support for N/S panels at aft edge				As per East/West Panels		
Complete Structure		93	To provide stable platform for all equipment	16.1 Hz 50.9 Hz	1st S/C Lateral 1st S/C Axial		This weight is based on using: i) Pre-bonded core instead of FM123 bond on panels ii) Honeycomb sandwich construction thrust tube		
		97		As Above			This weight is based on using: i) FM123 face sheet-to-core bond on panels ii) Honeycomb sandwich construction thrust tube		
		102		25.5 Hz 63.9 Hz	1st S/C Lateral 1st S/C Axial		This weight is based on using: i) Pre-bonded core instead of FM123 bond on panels ii) Magnesium skin thrust tube		
		106		As Above			This weight is based on using: i) FM123 face sheet-to-core bond on panels ii) Magnesium skin thrust tube		

Further discussions with the NASA Delta Office will be required to confirm that these requirements are still acceptable for the 3914 launch vehicle.

A trade-off study will be required to identify the most cost/weight effective version of the structure.

5.4.2.1 Major Design Constraints for Spacecraft Structure

5.4.2.1.1 Launch Environment

a) Quasi-Static Loads

POGO + MECO	:	16 g qual. axial (thrust)
		± 1 g qual. lateral (X or Y axis)
Maximum Lift-Off	:	3.9 g qual. axial (thrust)
		± 2.8 g qual. laterial (X or Y axis)

b) Sinusoidal Vibration Loads

Qualification level inputs at base of space-craft adapter:

Input Axis	Frequency (Hz)	Input Acceleration (g's)
	5 - 10	2.3
	10 - 15	2.3
Thrust	15 - 21	6.8
	21 - 250	2.3
	250 - 400	4.5
	400 - 2000	7.5
	5 - 10	2.0
	10 - 14	2.0

CD/103

Lateral (X or Y)	14 - 250	1.5
	250 - 400	4.5
	400 - 2000	7.5

- Notching to Quasi-Static levels allowed at:

  1st spacecraft laterial mode (3 g at Centre of Mass)
  1st spacecraft axial mode (16 g maximum response)

  The above must be overall spacecraft modes

5.4.2.1.2 Stiffness Requirements

When spacecraft hard-mounted (no adapter):

1st spacecraft lateral resonance	15 Hz
1st spacecraft axial resonance	35 Hz

The above restraints are for overall spacecraft modes

5.4.2.1.3 Total Spacecraft Weight

Structure designed for 2,200 lb total spacecraft weight, excluding spacecraft adapter

5.4.2.2 Design Requirements for Structure

5.4.2.2.1 Safety Factors

1.10 for yield

1.25 for failure (ultimate level loads)

5.4.2.2.2 Margins of Safety (MS)

Must be positive

$$MS = \frac{\text{Allowable Load or Stress}}{\text{Applied Load or Stress}} - 1$$

CD/104

5.4.2.3 Major Assumptions made for Stress Analysis of Structure

- For lateral spacecraft sinusoidal response, the design case was taken as:

  3 g qual. at the spacecraft centre of mass

  0.5 g qual. at the base of the spacecraft adapter linear distribution of acceleration over the length of the spacecraft

- A uniform distribution of equipment was assumed for the North and South panels

- No moment transfer assumed between connected panels; all joints have been treated as simple supports

- A maximum North-South panel vibration response of 30 g qual. has been assumed in a normal to panel direction

- The aft thrust tube was analyzed as a cylinder with its diameter equal to the thrust tube diameter near the separation ring

- The axial load and bending moment in the thrust tube near the separation ring were conservatively used to evaluate the Margin of Safety with respect to aft thrust tube buckling

- A total apogee motor weight of 1,065 lb was assumed with its Centre of Mass 36.3 inches above the separation plane

5.4.2.4 Major Spacecraft Vibration Modes*

*All modes without spacecraft adapter; modes and frequencies were obtained using a multi-mass computer model of the spacecraft; the thrust tube was represented by beam-elements, whereas panels were modelled using plate-elements.

5.4.2.4.1 Magnesium Alloy Thrust Tubes

.040 inch thick forward thrust tube
.080 inch thick aft thrust tube

1st spacecraft lateral mode : 25.5 Hz
1st spacecraft axial mode : 63.9 Hz

5.4.2.4.2 Honeycomb Thrust Tubes

1/8 inch thick Al Alloy core, density of 3.1 lb/ft^3

a) .006 Inch Thick Facesheets (Al Alloy)

1st spacecraft lateral mode : 16.1 Hz
1st spacecraft axial mode : 50.9 Hz

b) .008 Inch Thick Facesheets on Forward Thrust Tube) Al Alloy
.010 Inch Thick Facesheets on Aft Thrust Tube)

1st spacecraft lateral mode : 19.0 Hz
1st spacecraft axial mode : 56.0 Hz

5.4.2.5 Loading Cases - Qualification Levels

Case	Type	S/C Axial G-Level	S/C Lateral G-Level
1	Quasi-Static	16g - uniform	1g - uniform
2	Quasi-Static	3.9g - uniform	2.8g - uniform
3	Vibration	1g - uniform	3g - 1st S/C Bending Mode

CD/106

5.5 Thermal Subsystem

5.5.1 Design Requirements

Design requirements for the General Purpose Bus thermal subsystem are identified in the study contract Statement Of Work Issue 2, March, 1975 (Section 4.0 and Attachment B).

Where information on thermal parameters of payload, power and/or telemetry tacking and command subsystems is not contained in the statement of work, assumptions have been made based on CTS payload data.

Table 5.5-1 summarizes the power and weight requirements of the five payload options. Table 5.5-2 presents a breakdown of the power dissipations and assumed temperature limits for the various transponder options and for the critical housekeeping components.

5.5.2 Major Thermal Design Features

The following are the main design features of the general purpose bus thermal subsystem, arrived at after detailed examination of component requirements (radiating area, temperature limits) and spacecraft structural configuration.

5.5.2.1 Transponder high power dissipation components mounted directly on the internal face of the north and south panels. These are the most favourable locations as far as radiating area requirements are concerned (and hence minimum thermal subsystem weight).

5.5.2.2 Solar arrays, in the deployed configuration, separated from the north and south panels such that the minimum new factor to space from the north/south panel is greater than .93.

5.5.2.3 Batteries mounted on the north panel and separated from the power dissipating components by at least 8 inches (in order to maintain battery temperature below 10°C).

5/CGL/1

TABLE 5.5-1

GP BUS PAYLOAD CHARACTERISTICS

PAYLOAD OPTIONS	ANTENNAE			TRANSPONDER			DC POWER
	TYPE	NO. & SIZE	WEIGHT	SIZE	WEIGHT lbs.	DISSIPATION watts	
a) UHF/4-6 GHz Transponder (12 channel) (5-6W TWT)	Dep. parabola or parabola + quad helix	13 ft. dia. or 82" dia. x 100" long	≈ 55 lbs.	TBD	218.4	S 314.9 E 264.3	S 414.9 E 339.3
b) UHF/12-14 GHz Transponder (20W TWT)	Dep. parabola or parabola + quad helix	13 ft. dia. or 82" dia. x 100" long	≈ 55 lbs.	TBD	180.5	S 456.6 E 351.5	S 624.6 E 463.5
c) UHF/7-8 GHz Transponder with auxilary experimental payload	Dep. parabola	13 ft. dia.	≈ 55 lbs.	TBD	221.9	S 404.2 E 252.3	S 505.2 E 305.3
d) UHF/SHF/L Band Transponder	Dep. parabola	13 ft. dia.	≈ 60 lbs.	TBD	190.4	S 397.2 E 305.3	S 513.2 E 373.3
e_1) 12-14 GHz Transponder (12 channels) 30W TWT						?	S 6.36
e_2) 12-14 GHz Transponder 4-20 watt TWT 4-50 watt TWT + 50% redundancy						?	S E 732

TABLE 5.5-2

GENERAL PURPOSE BUS - COMPONENT THERMAL DATA

1. TRANSPONDER OPTIONS

Component	Pwr Dissipn. (watts) Max	Min	Temp Limits (°C) Operating		Non-Op.	# Read	Wt (lbs)	Dimensions (Ins)	Reqd. Thermal Mounting Area(Ins^2)
(1) UHF TRANSPONDER (PAYLOADS (A), (B), (C), (D))									
EPC	29.1		45	-5	(1) -40	1			150
UHF Driver	69.6		70	-5	-40	1			244
UHF HPA Module	75.0		70	-5	-40	1			263
Isolator	5.7		45	-5	-40	1			29
UHF Switch	4.4		45	-5	-40	1			23
UHF O/P Filter	9.8		45	-5	-40	1			50
Cabling to Ant.	8.2		45	-5	-40	1			42
(2) SHF 4-6 ghz TRANSPONDER (PAYLOAD (A))									
5 watt TWT	14.25	9.25	55	-15	-40	12			63
EPC	3.7		45	-5	-30	12			19
Comm. Recvr	9.0		35	20		1			
(3) SHF 12-Hghz TRANSPONDER (PAYLOAD (6))									
20 watt TWT	49		70	-5	-40	4			172
EPC	10.5		45	-5	-40	4			54
PWR Adaptor	2.35		45	-5	-40	4			12
Receiver	3.4		40	0	-40	1			18

TABLE 5.5-2 Cont'd

2. HOUSEKEEPING COMPONENTS

Component	Pwr Dissipn. (watts) Max	Min	Temp Limits (°C) Operating		Non-Op.	# Read	Wt (lbs)	Dimensions (Ins)	Reqd. Thermal Mounting Area (sq.ins)
POWER SUBSYSTEM									
Battery	8.3	0	10[2]	0		3			
Power Supply Electronics	21	0	50	5	-20	1			
ACS Sec Con-verter			55	0	-20	1			
DSA Sec. Con-verter			55	0	-20	1			
RCS Sec. Con-verter			55	0	-20	1			
TT&C SUBSYSTEM									
Typical Compo-nent	5		50	5	-20	6	6.7		
ATTITUDE CONTROL SUBSYSTEM									
Momentum Wheel			55	10	-10	2			
ACEA			50	5	-25	1			
Earth Sensor (N/S)			40	5	-25	2			
Earth Sensor Elec.			45	0	-25	1			
Earth Sensor (Spin)			45	0	-25	2			
Sun Sensor (Spin)			65	-50	-65	2			
Sun Sensor Elect.			50	-5	-20	2			
Sun Sensor (N/S)			65	-50	-65				
REACTION CONTROL SUBSYSTEM									
EJB			50	5	N/A	1			
Latching Valve Module			50	5	N/A	1			

5.5.2.4 TT&C components and the momentum wheels located on remaining available mounting locations on the north and south panels, again because of radiating area requirements.

5.5.2.5 UHF and/or SHF antennas have, of necessity, to be mounted on the forward or earth facing platform. To minimize alignment errors between antennas and attitude control sensors, caused by thermal distortion of the spacecraft structure, the latter are also mounted on the forward platform.

5.5.2.6 No restriction has been placed on the location of the RCS tanks. Multi-layer insulation and minimal heater power provide adequate thermal protection for these.

5.5.2.7 Location of the shunt regulator resistors is dependent on the amount of power to be dissipated and the allowable temperature limits of the resistors. For minimum design temperature limits less than -100°C location on the exterior of the east/west panels is acceptable. For maximum total power dissipation of less than 100 watts location on the forward thrust tube is acceptable.

5.5.2.8 All non-radiating areas of the spacecraft are covered with multi-layer insulation of various thicknesses.

5.5.2.9 During the spin phase, all radiating areas of the spacecraft are thermally insulated to maximize spacecraft internal temperatures during this low power phase. The minimum internal power dissipation requirement during this phase, to maintain adequate spacecraft temperatures, is 80 watts.

5.5.3 Detailed Design

Figures 3-5a, b, c, detail; as shown in Section 3.3.

a) The spacecraft configuration.

b) The equipment platform layout for:

- the UHF transponder for payload option models a, b, c or d (Figure 3-5).
- the 4-6 GHz transponder for payload option model a (Figure 3-5).
- the 12-14 GHz transponder for payload option model a (Figure 3-5).

The above three are considered the most critical from volume/thermal control requirements.

Payload options, models e1 and e2, (Telesat requirements see Table 5.5-1) were not considered as their characteristics were not identified until late in the study program and inclusion of them would have meant resizing the thermal subsystem.

The above configurations/layouts have evolved based mainly on thermal considerations regarding radiating area requirements.

These radiating requirements depend on:

- component dissipation
- component design temperature limits
- view factor from radiator (north/south panels) to space.

A detailed analysis was performed to establish the latter. It was found that with the present separation of deployed solar array from the north/south panels the minimum view factor to space from any point on the north/south panels is greater than 0.93.

Hence, with the above information and using quartz second surface mirrors on the exterior of the panel, the radiating area requirements and hence payload locations were established.

5/CGL/3

5.5.3.1 UHF Transponder Layout (Models a), b), c), d))

The UHF transponder layout is shown in Figure 3-5 for the maximum power, and hence maximum thermal hardware weight, payload model b). Due to lack of definition of size/shape of each component the mounting area of each has been assumed to be equal to its radiating area requirements. (With the exception of the backhaul 20 watt TWT). This assumption is not unrealistic in light of the multiplcity of dissipating elements in each UHF component, which can be used to achieve an essentially isothermal baseplate with optimum packaging.

The three batteries are located on the same equipment platform and are located as far aft as mass balancing requirements will allow. Separation between batteries and other components is required because of the vastly different design temperature limits of batteries and typical transponder or housekeeping components. In the present location the temperature limits of 0 to 10°C can be met, with use of heater power in non-battery dissipation and/or non-solar load cases.

5.5.3.2 <u>4-6 GHz SHF Transponder Layout</u> (Model a))

The above is shown in Figure 3-5. No attempt has been made to electrically optimize this layout, but merely to show that volume/thermal radiating area requirements can be met. The housekeeping components allocated to this platform have been located as far forward as possible for mass balancing considerations.

5.5.3.3 <u>12-14 GHz Transponder Layout</u> (Model b))

The above is shown in Figure 3-5. Initially, in determining the weight of thermal control hardware for payload model b) (maximum power and maximum weight of thermal subsystem hardware) material required for laterally distributing the dissipated power away from the maximum power density locations (eg. TWT collectors, etc.) was assumed to be conventional thermal doubler material (6061-T6Al). However, the weight penalty associated with this technique was excessive (approximately 17 pounds). Hence, it was felt that although the SOW called for a passive thermal design, if possible without the use of heat pipes, the potential weight savings of the latter, together with the relatively simple

5/CGL/4

construction and space worthiness of this type of pipe required for the above application, necesitated their use. The payload configuration in Figure 3-5, hence reflects the presence of heat pipes.

5.5.3.4 General Spacecraft Configuration

The general spacecraft configuration is shown in Figure 3-5 and thermal hardware used summarized in the weight breakdown Table 5.5-3. Almost all internal surfaces will be anodized to achieve a high IR emittance and maximum thermal radiative coupling between them.

Radiating areas on the north/south panels will have second surface mirrors to minimize solar heat input. A five layer insulation blanket will cover non-radiating surfaces on the north/south panels.

The five layer blanket will cover the external surface of the east/west panels except for the area occupied by the shunt regulator resistors.

To minimize heat loss from two platforms not irradiated with solar energy during the spin phase, namely the forward and aft platforms, a 20 layer blanket will cover the external surfaces of both of these panels. An additional two layers of 1/2 mil crinkled stainless steel foil will be used to provide thermal protection for the aft platform blanket (and hence spacecraft) during apogee motor firing.

The apogee motor case and nozzle exteriors will be covered with seven layer insulation blankets, (five layers kapton; two layers of crinkled stainless next to the motor), to minimize heat loss from the motor prior to firing and to protect the spacecraft from apogee motor "soakback" heating. The apogee motor case will be mounted to the spacecraft structure via a thermal insulating material, again to minimize soakback effects.

5/CGL/5

TABLE 5.5-3 GENERAL PURPOSE BUS

THERMAL SUBSYSTEM WEIGHT BREAKDOWN

A.	Multilayer Insulation Blankets		
	Forward Platform	- 20 layer	1.96 lbs.
	Aft Face	- 22 layer	2.55 lbs.
	East & West Panels	- 5 layer	1.04 lbs.
	North & South Panels	- 5 layer	0.34 lbs.
	Apogee Motor	- 7 layer	2.26 lbs.
			8.15 lbs.
B.	Panel Surface Finishes		
	External Face of N/S Panels	- S.S.M.	4.42 lbs.
	Internal Face of Equipment Platforms	- Anodized	-
C.	Thermal Conductors		
	Heat Pipes and Doublers		6.50 lbs.
	(Thermal Doublers alone as alternative		17.50 lbs.)
D.	10% for Miscellaneous		1.91 lbs.
			20.98 lbs.

The antenna farm will be thermally decoupled from the forward platform as much as possible and will require its own thermal control system to maintain antenna temperatures within design absolute temperature limits and to minimize antenna thermal gradients.

Electric resistance heaters (thermostatically controlled with ground command override) will be located in critical areas of the spacecraft to:

a) compensate for "component-off" conditions, (i.e. act as substitute heaters).

b) compensate for inadequate dissipation levels (low component temperatues) in various regions of the spacecraft.

Use of space-proven thermostatically controlled heaters eliminates the sensitivity of the spacecraft to heater nominal power levels, eliminates complex ground monitoring/control of heaters, and does not significantly affect system reliability.

Only certain RCS heaters will be required during eclipse.

For the spin phase configuration, the stowed solar arrays will provide thermal protection for the spacecraft by covering the radiating areas of the north/south panels during this low power phase. Areas of the north/south panels not covered by the arrays will require either permanent multi-layer insulation or a deployable or jettisonable cover (deployed or jettisoned with deployment of the arrays).

The weight breakdown of the thermal system hardware is given in Table 5.5.-3.

5/CGL/6

5.6 Apogee Kick Motor

5.6.1 Description (See Figure 5-8)

The apogee motor will be capable of providing the velocity increments sufficient for placing the spacecraft weighing 1925 to 2125 lb into synchronous orbit. The motor size had been optimized for an orbit inclination plane of 28.3°.

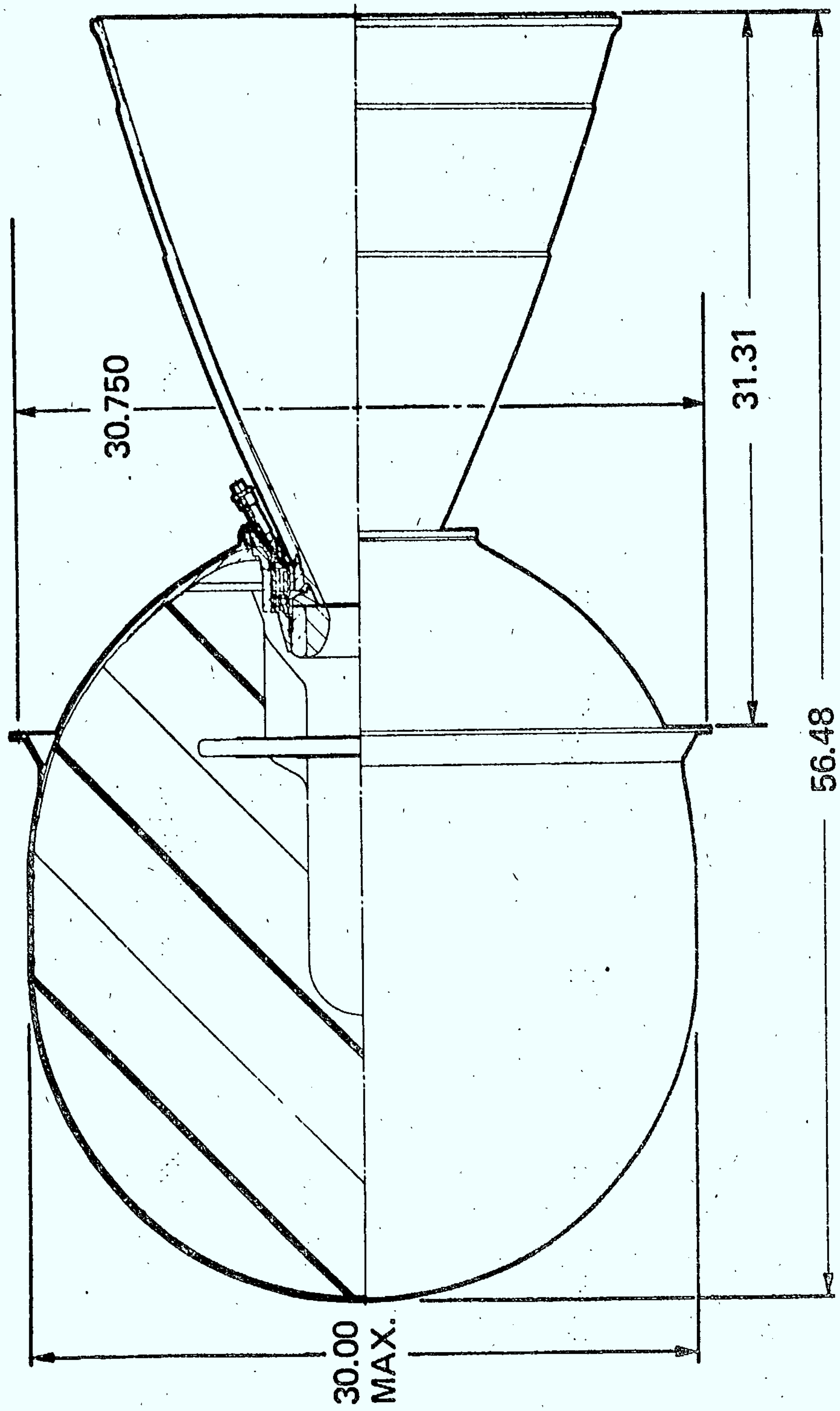
The apogee motor for this satellite will be selected from motor specialists (Bristol, Thiokol, United Technology or Aerojet General), and integrated with the structure at the launch facility. Interface details and interface tooling are available, consequently no integration problems are anticipated. Sea level qualification tests on this motor will be conducted, followed with high altitude firing tests. These details are discussed in Volume III Implementation Plan.

The motor comprises the following major components:

- Case and insulation chamber
- Expansion Nozzle
- Ignition system consisting of igniter, pyrotechnic train and safe arm device
- Motor attachment flange

It should be noted that motors currently designed can be overloaded by 10% additional propellant so without any major redesign a 2125 lb satellite can be injected into geosynchronous orbit with this motor.

The data that follows is what has been provided by Thiokol for the study and has been used to establish the Bus baseline design. Overall dimensions are given in the attached configuration.



APOGEE KICK MOTOR

5.6.2 Typical Apogee Motor Performance

5.6.2.1 Characteristics at 65°F

Total Impulse	301,200 lbf-sec
Average Thrust	5,930 lbf
Maximum Thrust	7,000 lbf
Burn Time	50.2 sec
Propellant Specific Impulse	297 lbf-sec/lbm
Effective Specific Impulse	295 lbf-sec/lbm

5.6.2.2 Weight Summary

Item	Nominal Wt. lbs
Propellant	895.3
Inerts	62.6
Loaded Motor	957.9
Fired Motor	56.4

5.6.2.3 Design Conditions

External Temperature	Case-700°F Max Soak Nozzle-1,000°F Max Soak
Operational Temperature	+20°F to +110°F
Storage Temperature	+60°F to +80°F
Aging	Five year shelf life goal
Spin Rate	60 - 110 rmp

5.6.2.4 Nozzle Details

Design:
Carbon-Carbon Exit Cone
Carbon Felt External Insulation
6Al-4V Titanium Closure
G-90 Graphite Throat

Experience:	Carbon-Carbon	- 6 static firings
	Closure	- 8 star motors (Star 27 - Star 37)
	G-90	- All Star Motors
Attributes:	Lightweight	
	Symmetrical Erosion	
	Excellent Alignment	

5.6.2.5 Temperature

Operating	20°F to 100°F
Flange Satellite Interface During Burn	150°F maximum
Motor Casing (after burn)	500°F maximum

5.6.2.6 Alignment

Perpendicularity of Thrust Vector to a Mounting Flange	0.002"
Concentricity of Nozzle Throat to Mounting Flange	0.020"
Flatness of Mounting Flange Mounting Flange O/D	0.003"

5.6.2.7 Electrical (Squib Data)

No Fire Current	1 amp DC
All Fire Current	4.5 amps at 28 VDC

5/CD/109

APPENDIX A

PRE-STATIONKEEPING SEQUENCE AND FUEL BUDGET

A.1.0 PRE-STATIONKEEPING SEQUENCE AND FUEL BUDGET

The following is a rough outline of a baseline sequence for achieving geostationary orbit. It has been generated only as a framework for Multi-Purpose Bus studies.

Elapsed Time	Event/Task	Duration
00h 00m	Liftoff	
00 26	Injection into transfer orbit	
00 30	Preliminary NASA Orbit Data received	
01 05	Eastern Hemisphere Tracking Station acquires	
02 00	Preliminary Orbit of Attitude Determination	
05 40	First Apogee	
10 00	Loss of signal at Eastern Hemisphere station	
11h 50m	Western Hemisphere Tracking Station acquires	
16 00	First half of reorientation maneuver	45 min
16 10	Second Apogee	
17 15	Second half of reorientation maneuver	45 min
20 55	Loss of signal at Western Hemisphere Station	

	During succeeding passes, attitude and orbit determinations are performed and spacecraft health is monitored. Attitude touch up maneuvers are performed as required	
	3rd Orbit visible from Eastern Hemisphere Station 4th Orbit visible from Western Hemisphere Station 5th Orbit visible for Eastern Hemisphere Station 6th Orbit - poor visibility 7th Orbit visible from Western Hemisphere Station	
68 40	Seventh Apogee - Apogee Motor Firing	-1 min
73 00	Determine Orbit and Attitude	
80 00	Drift Orbit Correction Maneuver	1/2 to 1 hour
AMF + 1 day*	Drift Orbit Correction Maneuver	1/2 to 1 hour
(see Att. Acq.)*	Attitude Acquisition Sequence	
AMF +10 days*	Drift Rate Reduction	
AMF +10-1/2 days*	" " "	
AMF +15 days*	" " "	
+15-1/2 days*	" " "	
AMF +18-1/2 days*	Station Rate Acquisition	
AMF +19 days*	Initial Station Acquisition	

*These times are subject to large variations due to launch vehicle and Apogee Motor injection errors.

CFJ/2

The reorientation to AMF attitude is through an angle of 129.5 degrees.

Fuel is required for the attitude acquisition sequence (see ACS section).

An allowance of 160 ft/sec is required to correct (at 99% level), the launch vehicle, apogee motor, orbit determination, attitude determination and RCS maneuver errors, and to allow for drift reversal and station acquisition maneuvers.

The errors assumed for the 3914 launch vehicle with Multi-Purpose Bus are:

- Apogee Height $\pm$ 300 n.mi
- Perigee Height $\pm$ 3.3 n.mi
- Inclination $\pm$ 0.25 degrees

For the apogee motor firing:

- Total impulse $\pm$ 0.5 percent
- Mean Thrust Direction (Including Attitude Determination Errors) 0.4 degrees

The drift reversal and station acquisition allowances are 1 degree per day drift.

A.2.0 BASELINE STATIONKEEPING PLAN

The following is a rough cut at a stationkeeping cycle for maintaining $\pm 0.05°$ or $\pm 0.1°$ latitude and longitude. The purpose is to determine numbers for the fuel budget, and to examine compatibility of the satellite system design with the specified stationkeeping requirements.

The baseline cycle is three weeks long (a requirement of Telesat Canada). The reference starting point is the east-west correction manoeuvre.

CFJ/3

	Approximate Time of Manoeuvre		Type of Manoeuvre
1.	0.0 days	-	East-West manoeuvre (first burn) to correct the orbit for effects of triaxiality, solar pressure and coupling, of inclination manoeuvres in the east-west direction
2.	0.5 days	-	Second burn of the east-west manoeuvre
3.	0.5 days (very approximate)	-	Optional east-west touch up correction manoeuvre to correct errors in (1) and (2). This manoeuvre may also be performed in two parts
4.	18 days	-	Inclination (North-South) manoeuvre
5.	21 days	-	End of cycle, manoeuvre (1) begins again

This manoeuvre cycle appears to be feasible for stationkeeping the multi-purpose satellite, based upon the preliminary design information. Further optimization is possible although this is more appropriately performed after satellite system design information and station longitude are better defined.

A.2.1 Solar Radiation Pressure

The preliminary estimate of solar radiation pressure force was 1.0 x 10^{-5} pounds, and a later estimate was 2.0 x 10^{-5} pounds. The spacecraft, near end of life is assumed to weigh approximately 800 pounds. The resulting acceleration on the satellite is then:

Case 1 $a = 4.02 \times 10^{-7}$ ft/sec (1.0×10^{-5} lbs force)

Case 2 $a = 8.04 \times 10^{-7}$ ft/sec (2.0×10^{-5} lbs force)

A geostationary orbit which is initially circular, will become slightly elliptical due to the solar radiation pressure force. The change in mean

CFJ/4

eccentricity is approximately 0.52 x 10^{-5}/day (Case 1) or 1.04 x 10^{-5}/day (Case 2). This eccentricity in turn causes a buildup in the diurnal east-west motion of the satellite (but does not significantly effect drift rate). If the mean eccentricity must be minimized over the stationkeeping cycle, the cost in fuel is calculated to be about 9.6 ft/sec per year (Case 1) or 19.2 ft/sec per year (Case 2).

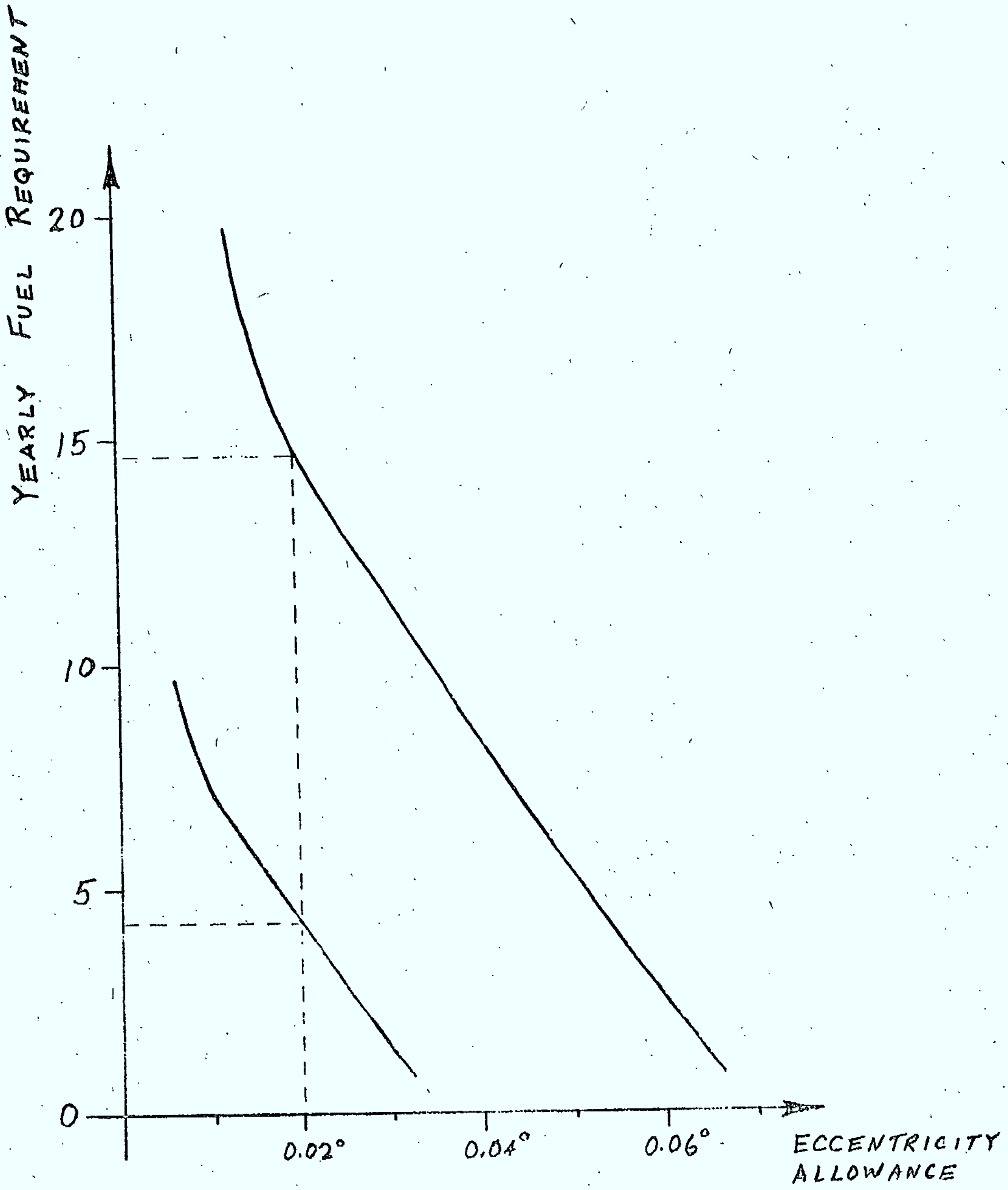
If however, over the 21 day cycle, the mean eccentricity is always maintained below some specified value without an attempt at minimization, some fuel can be saved. The specified maximum eccentricity represents some acceptable portion of the east-west stationkeeping error budget. Figure A-1 shows the fuel cost per year for various eccentricity allowances in the east-west budget (Note that for a 21 day cycle, there will always be a minimum of ± 0.0063; east-west variation (Case 1) or ± 0.0126 (Case 2) at the beginning or end of each cycle).

As well as reducing the fuel budget for solar radiation pressure corrections, the size of each of these manoeuvres is reduced, simplifying the east-west manoeuvre cycle.

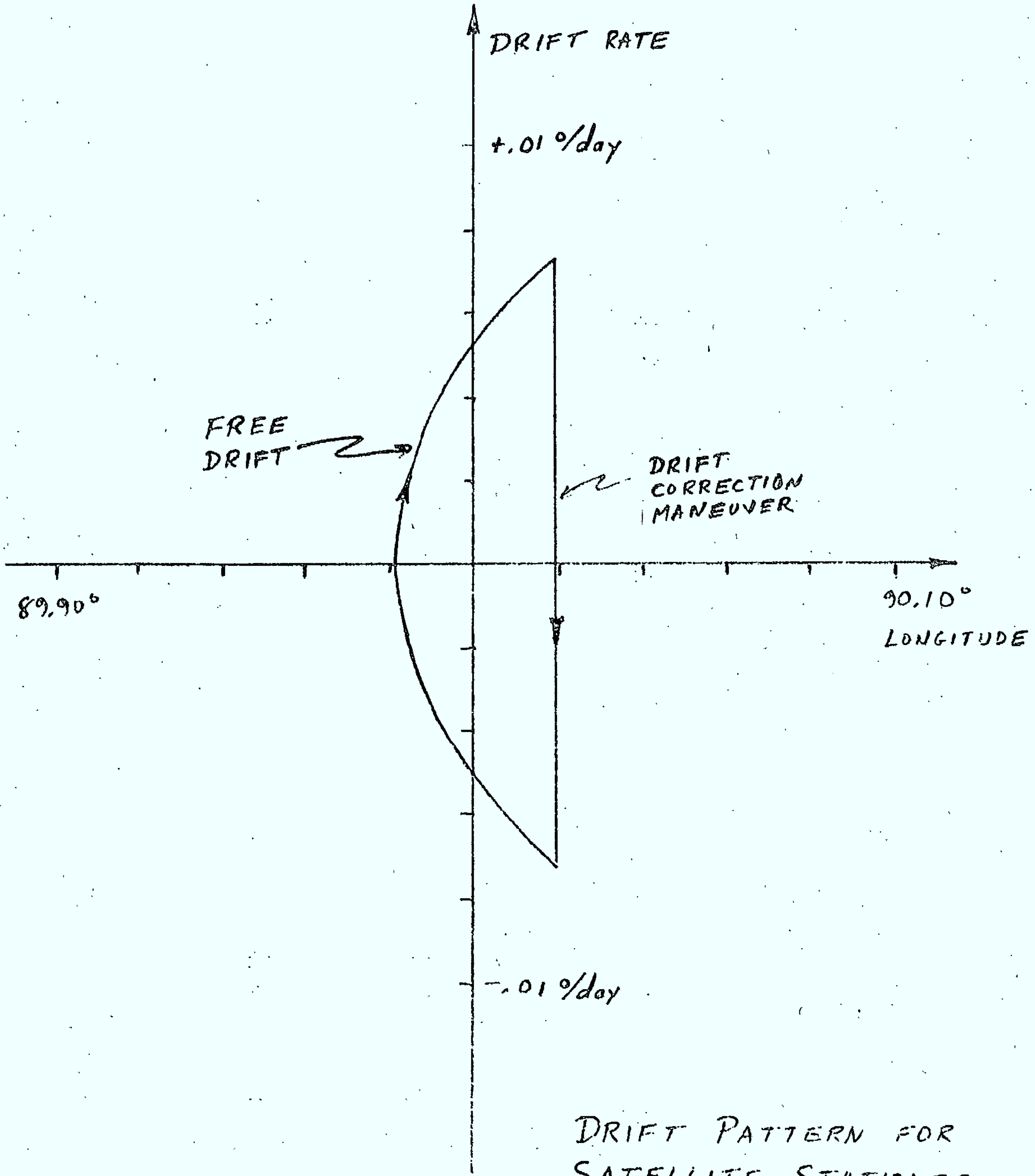
A.2.2 Triaxiality Effect

For the purposes of this study, it is assumed that the satellite station will be anywhere in the range of 90°W to 120°W longitude. For triaxiality corrections the largest accelerations are at the ends of the allowable range. As a worst case, therefore, a stationkeeping longitude of 90°W was chosen, at which the longitudinal acceleration is +0.0007 deg/day/day. A satellite at rest (no change in mean longitude) will be drifting at +0.00735 deg/day (Westward) after 10-1/2 days (one half of the stationkeeping cycle). Therefore to start the east-west cycle (assuming only triaziality perturbations), a drift rate of -0.00735 deg/day is initiated. Figure A-2 shows the nominal mean longitude versus mean longitude drift rate. The

CFJ/5



FUEL REQUIRED FOR CORRECTION OF ORBITAL ECCENTRICITY EFFECTS DUE TO SOLAR RADIATION PRESSURE



DRIFT PATTERN FOR SATELLITE STATIONED AT 90° W (21 DAY CYCLE)

FIGURE A-2

satellite remains within ± 0.02 degrees of nominal (n.b. the east-west manoeuvre was shown here, for simplicity as a single burn).

The ΔV required for triaxiality is that required to change the drift rate from +0.00735 deg/day to -0.00735 deg/day or about 0.14 ft/sec every 21 days (total of 2.4 ft/sec per year). Normally however, if solar radiation pressure is large and the stationkeeping box is small, the corrections for solar radiation pressure can be arranged to correct the triaxiality effects "for free" (see section A.2.4).

A.2.3 Inclination (North-South)

The inclination manoeuvres will be planned for the purposes of this analysis, to take out only the secular inclination drift due to solar and lunar gravity. Figure A-3 shows the total ΔV required for 6, 8, and 10 years as a function of launch date. The peak requirements for 6 years is for a launch in 1985, and the fuel required is 999 ft/sec. For 8 years, the peak is 1320 ft/sec for a 1984 launch, and for 10 years, the peak is 1630 ft/sec for a 1983 launch.

In addition, as a comparison with the RCA Domsat launch, a launch in 1975 requires 829 ft/sec for 6 years, 1124 ft/sec for 8 years, and 1440 ft/sec for 10 years.

A 21 day cycle, correcting only luni-solar secular drift will have a maximum drift during the cycle of

$$\frac{21}{365.2425} \times 0.96 \text{ deg/yr} = 0.055 \text{ degrees, or}$$

$$\pm 0.028 \text{ degrees latitude variation}$$

The ΔV required for this manoeuvre is 9.7 ft/sec maximum.

CFJ/6

Typically, if the engine is 0.1 pounds thrust and four are used to perform the manoeuvre, and if the spacecraft mass is 31.5 slugs, then the manoeuvre is 12.7 minutes long. If only two engines are used, the manoeuvre will take 25.4 minutes typically. During this time the satellite varies $\pm$ 1.6 degrees (4 engines) in its orbit from the desired firing point for an impulsive manoeuvre.

For a two engine manoeuvre, the corresponding variation is $\pm$3.2 degrees. The inefficiency of this manoeuvre is therefore very slight, even for the two thruster case, amounting to a loss of about 0.05% or 1/2 ft/sec over a six year lifetime.

Since it is assumed here that only the secular inclination drift is corrected, the actual variation in inclination will be slightly larger than 0.028 degrees, but well within the 0.05 degrees specified. The inclination deviation can be reduced slightly in practice with negligible increase in fuel.

A.2.4 Combined Manoeuvres

A.2.4.1 North-South, East-West Tradeoff

Since the north-south excursions can probably be kept down to approximately $\pm$ 0.03° compared to the $\pm$ 0.05° and $\pm$ 0.1° requirements, the east-west variation may be extended slightly while still remaining within the same maximum angular error from nominal station location (total angular error allowed is: $\sqrt{2} \times 0.05° = 0.0707°$ or $\sqrt{2} \times 0.1° = 0.1414°$). If the north-south excursion is contained to within 0.03° then the allowable east-west variation is $\pm\ (0.0707^2 - 0.03^2)^{1/2} = \pm\ 0.064°$ or $\pm(0.1414^2 - 0.03^2)^{1/2} = \pm\ 0.138°$.

A.2.4.2 Combining Triaxiality and Solar Radiation Pressure Corrections

As mentioned previously, triaxiality corrections can be given a "free ride" on solar radiation pressure corrections. In this case, if say $\pm$ 0.02°

CFJ/7

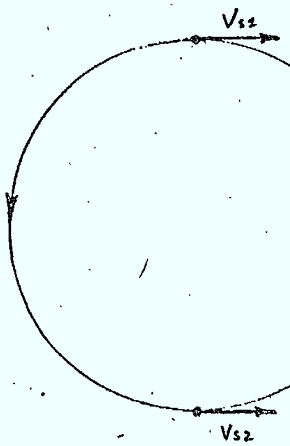
of the east-west budget is allotted to solar radiation pressure incuded eccentricity variations, the mean eccentricity can be maintained at about 0.000175 and the manoeuvres required every cycle for this maintenance are two - each one being 0.121 ft/sec (Case 1) or 0.417 ft/sec (Case 2) maximum. The first manoeuvre is performed in a direction opposite to the velocity vector. The second is performed in the same direction as the velocity vector (Figure A-4a). The drift (tri-axiality) correction is also broken into two parts, each 0.069 ft/sec, in a direction opposite to the velocity vector, (and each simultaneously with the eccentricity corrections (Figure A-4b)). The result of the combination is (Figure A-4c) one manoeuvre equal to the sum of the velocity magnitudes - 0.190 ft/sec (Case 1), 0.486 (Case 2) and another equal to the difference - 0.052 ft/sec (Case 1), 0.348 ft/sec (Case 2). The total magnitude is the same as for the solar radiation pressure manoeuvres alone*. A plot of longitude versus drift rate for Case 1 is shown in Figure A-5. The satellite always remains well within limits. The fuel required is 4.2 ft/sec per year or 25.2 ft/sec for six years. Figure A-8 shows a comparison between drift patterns for Case 1 and Case 2. For Case 2, the satellite also remains well within bounds, although the margin has been reduced, and there is a period of time after the first east-west manoeuvre of the cycle when the satellite is drifting very rapidly. If the second manoeuvre cannot be performed as scheduled, the satellite will be rapidly approaching the other side of the stationkeeping box, and a new twenty-one day cycle must be set up.

A.2.4.3 Effect of Inclination Manoeuvre Coupling

Assuming a maximum manoeuvre size of 9.7 ft/sec and a deviation of 0.2° in the mean thrust vector from nominal, the ΔV coupling can be 0.034 ft/sec.

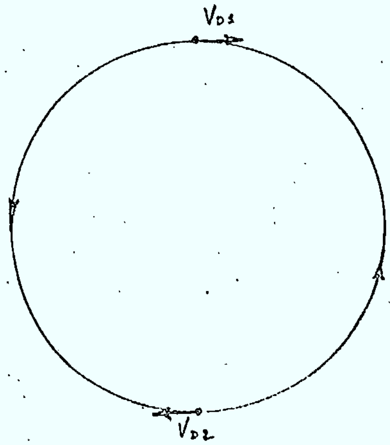
*Because of the twelve hour drift between east-west manoeuvres, the manoeuvre magnitudes will not be exactly as stated above (see drift pattern figures). The total delta velocity will however be very close to the total given above.

CFJ/8



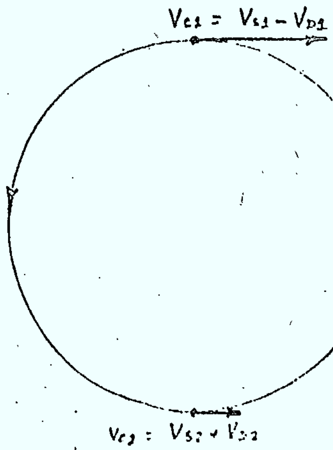
Solar Pressure correction ΔV_S

Figure A-4a



Drift correction ΔV_S

Figure A-4b

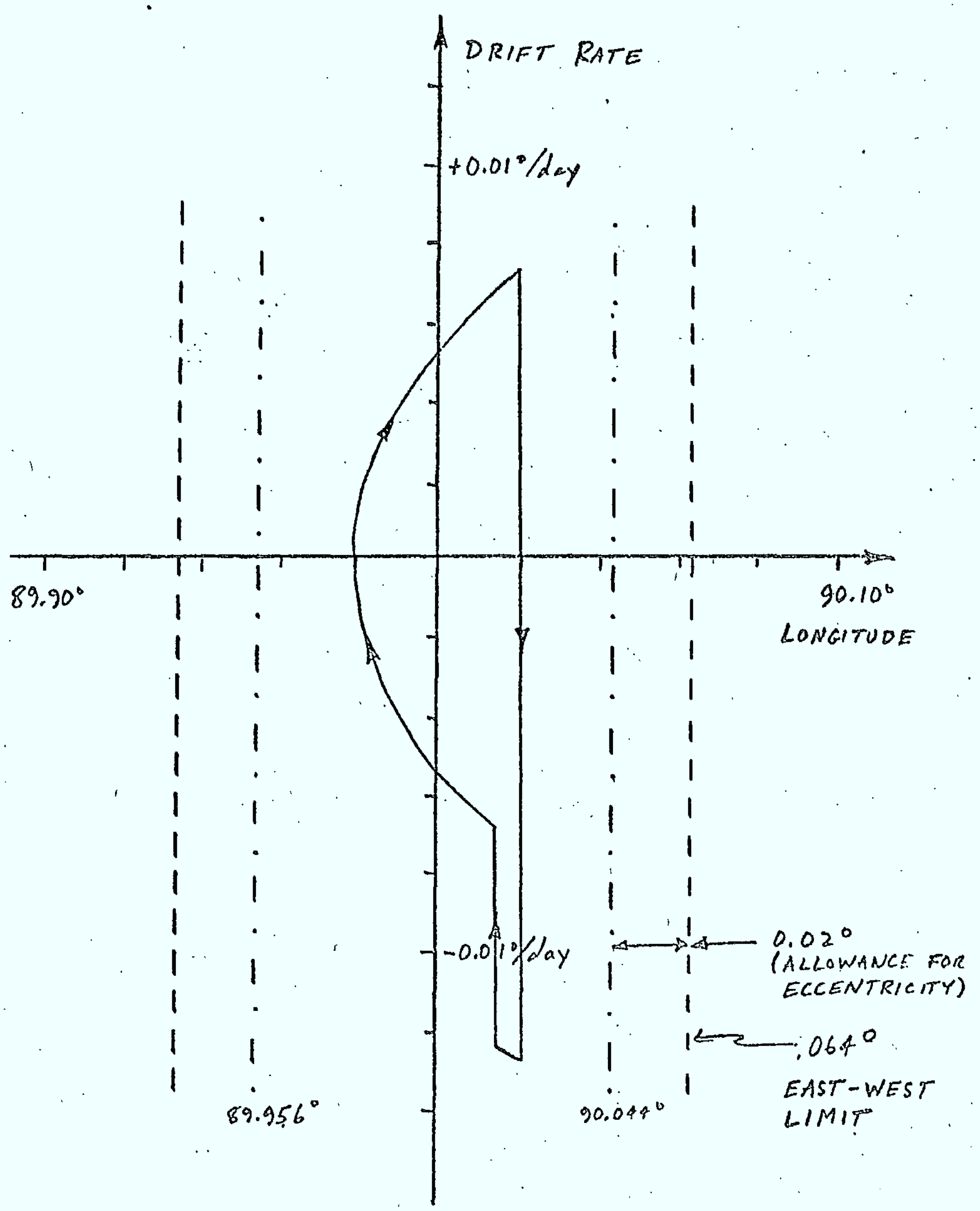


Combined correction ΔV_S

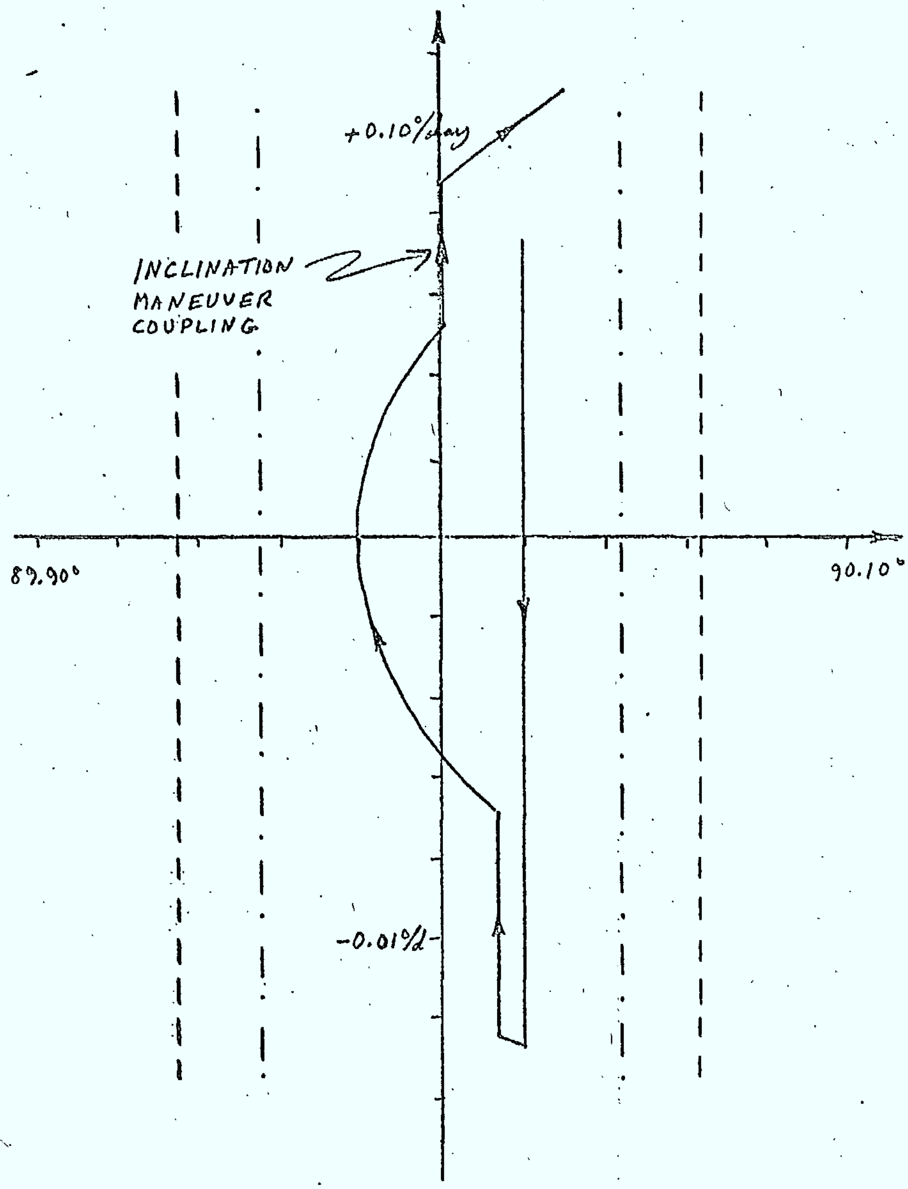
$V_{C1} + V_{C2} = V_{S1} + V_{S2}$

Figure A-4c

Combined Solar Pressure and Drift Maneuvers



DRIFT PATTERN FOR COMBINED DRIFT AND ECCENTRICITY CORRECTIONS



DRIFT PATTERN FOR WORST CASE INCLINATION COUPLING

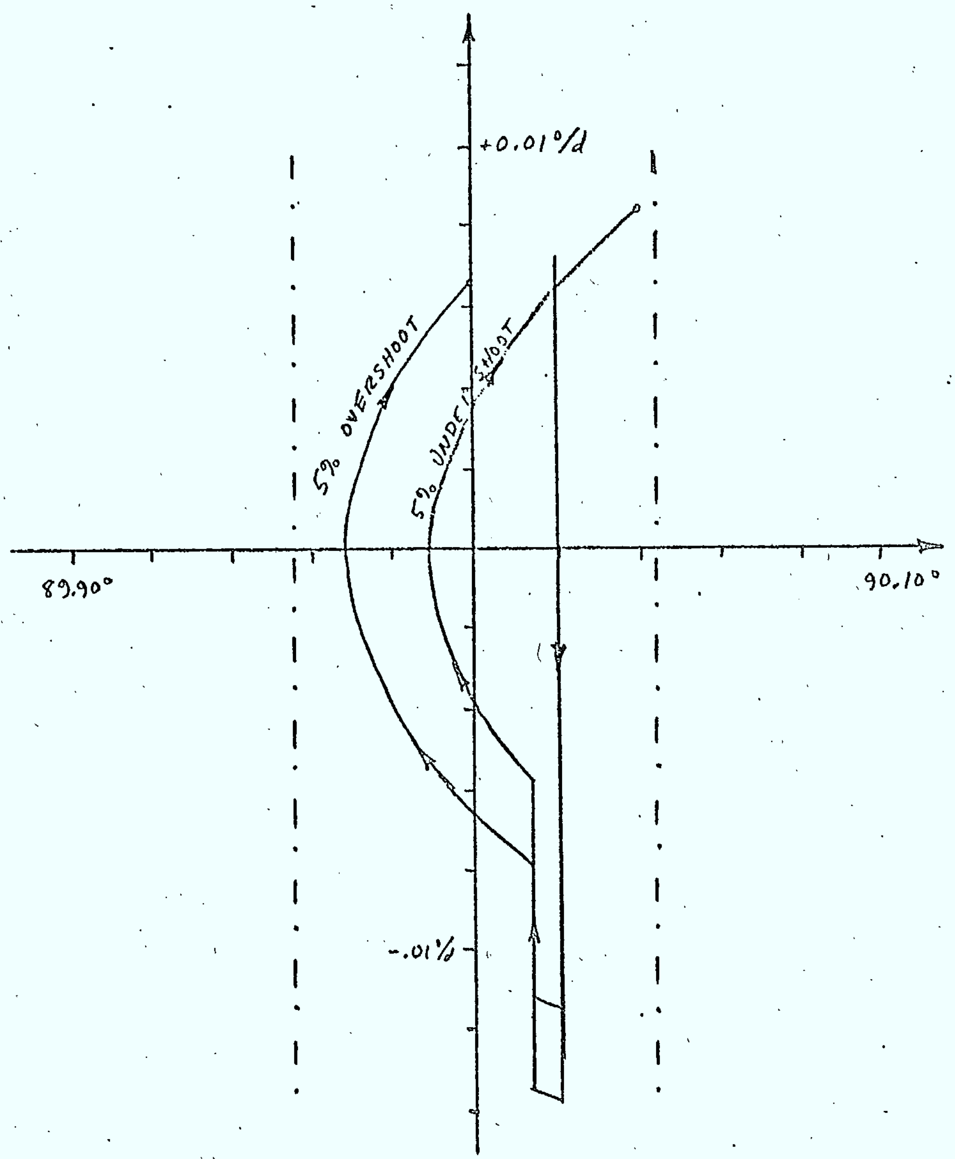
FIGURE A-6

This can result in a change in the drift rate of up to ± 0.0036 deg/day if colinear with the satellite velocity vector. Figure A-6 shows the result of this drift change for Case 1. The satellite remains well within bounds. Figure A-8 shows the end point of the drift pattern for Case 2 if the same adverse 0.2° coupling occurs. The margin for further error is now fairly small (less than 0.01°). It may be desirable in this case to begin the next manoeuvre cycle one day early. If the inclination coupling is entirely in the radial direction, the result is a negligible change in mean longitude of 0.0004 degrees.

A.2.4.4 Effect of East-West Errors

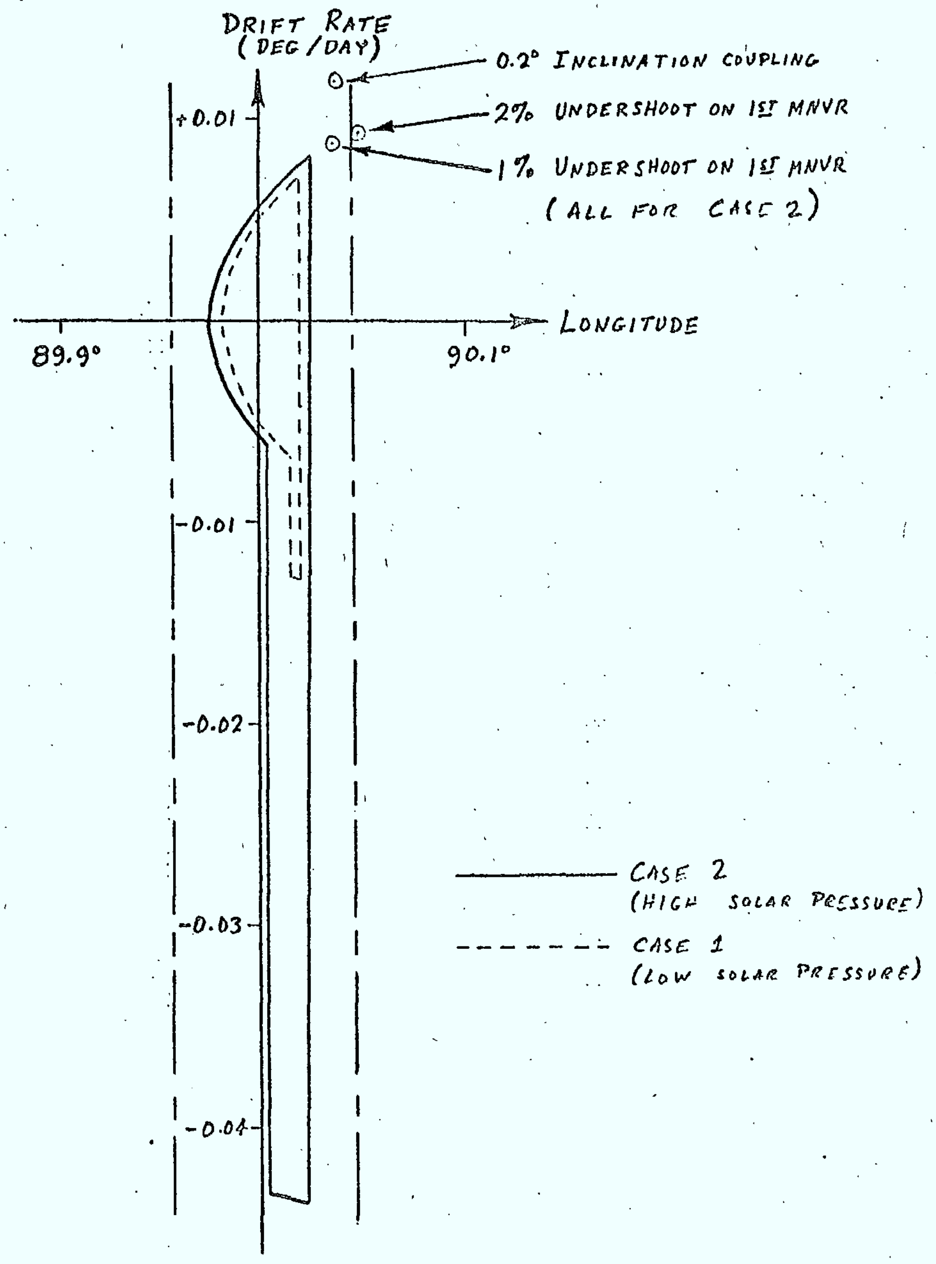
Assuming a worst case 5% undershoot or overshoot on an east-west manoeuvre, the drift rate of the first manoeuvre could vary by ± 0.001 deg/day (Case 1). This overly severe error results in the drift pattern shown in Figure A-7. The limit is not exceeded for either undershoot or overshoot, although for ± 0.05° stationkeeping, the mean longitude comes close to the limit allowing little room for other error sources such as those resulting from other manoeuvres or from orbit determination. In this case the optional mid cycle manoeuvre could be used to reduce excursions. These manoeuvres are small relative to the main east-west manoeuvres. The total ΔV requirement over one year would probably not exceed 0.3 ft/sec per year, making the total east-west fuel budget about 4 1/2 ft/sec per year or 27 ft/sec for a six year mission. A more realistic error, after on-station calibration of the reaction control system, would be one to two percent. Applying this error to the more realistic Case 2 solar radiation pressure model drift pattern results in the end points shown in Figure A-8. The two percent undershoot manoeuvre definitely results in the satellite going out of bounds and a mid cycle manoeuvre is definitely required.

CFJ/9



EFFECT OF DRIFT
RATE ERRORS

FIGURE A-7



DRIFT PATTERN FOR
COMBINED DRIFT AND
ECCENTRICITY CORRECTIONS
(COMPARISON BETWEEN CASES 1 AND 2)

An error in the direction of the manoeuvre delta velocity of up to five degrees results in a negligible shift in mean longitude on the same order of magnitude as the inclination manoeuvre coupling error.

A.2.4.5 Stationkeeping Summary

A three week stationkeeping cycle appears to be feasible for the UHF multi-purpose bus satellite. As more parameters such as solar pressure and station longitude are better defined, the actual strategy may be developed. The feasibility of this cycle for the +0.05° stationkeeping box goal is particularly sensitive to the assumptions for solar radiation pressure, as is the east-west fuel budget. The nominal ± 0.1° stationkeeping box appears to be no problem. As well, a preliminary study placing the satellite at 85°W and with solar pressure twenty-five percent higher than Case 2 showed that stationkeeping to ± 0.1° is feasible.

CFJ/10

FUEL BUDGET SUMMARY PAGE

(MULTIPURPOSE BUS)

MANOEUVRES	MISSION		
	6 Years	8 Years	10 Years
Pre-AMF Reor	129.5°	129.5°	129.5°
Spin Down (from 50 RPM)	250 lb.sec.	250 lb.sec.	250 lb.sec.
Reor Post-AMF	68.3°	68.3°	68.3°
Attitude Acquisition	70 lb.sec.	70 lb.sec.	70 lb.sec.
Station Acquisition and Launch Vehicle - AMF Dispersion Correction	160 ft./sec.	160 ft./sec.	160 ft./sec.
East-West Stationkeeping	87 ft./sec.	116 ft./sec.	145 ft./sec.
North-South Stationkeeping	999 ft./sec.	1,320 ft./sec.	1,630 ft./sec.

A.3.0 LAUNCH WINDOW

The spacecraft design, thermal and electrical, will tolerate solar aspect angles (angle between the spin axis and the sun) of ϕ = 65° to 115° prior to attitude acquisition. The resulting launch window is shown in Figure A-9. The window is always longer than a half hour. The window shown is the midnight window only. A noon window is also available, although near the equinoxes, this window is severely restricted due to shadowing of the satellite rear apogee.

If it should be considered desirable for any reason to maintain sun angles closer to 90°, this can be done at the expense of window length (see Figure A-10). At the low end ϕ = 65°, the window is limited in winter to about 1 3/4 hours, dropping to 1/2 hour at approximately ϕ = 74°. Similarly, at the high end ϕ = 115°, the window is limited to 2 1/4 hours in summer, dropping to 1/2 hours at approximately ϕ = 104° (Figure A-10).

A.4.0 T&C PATTERN VISIBILITY

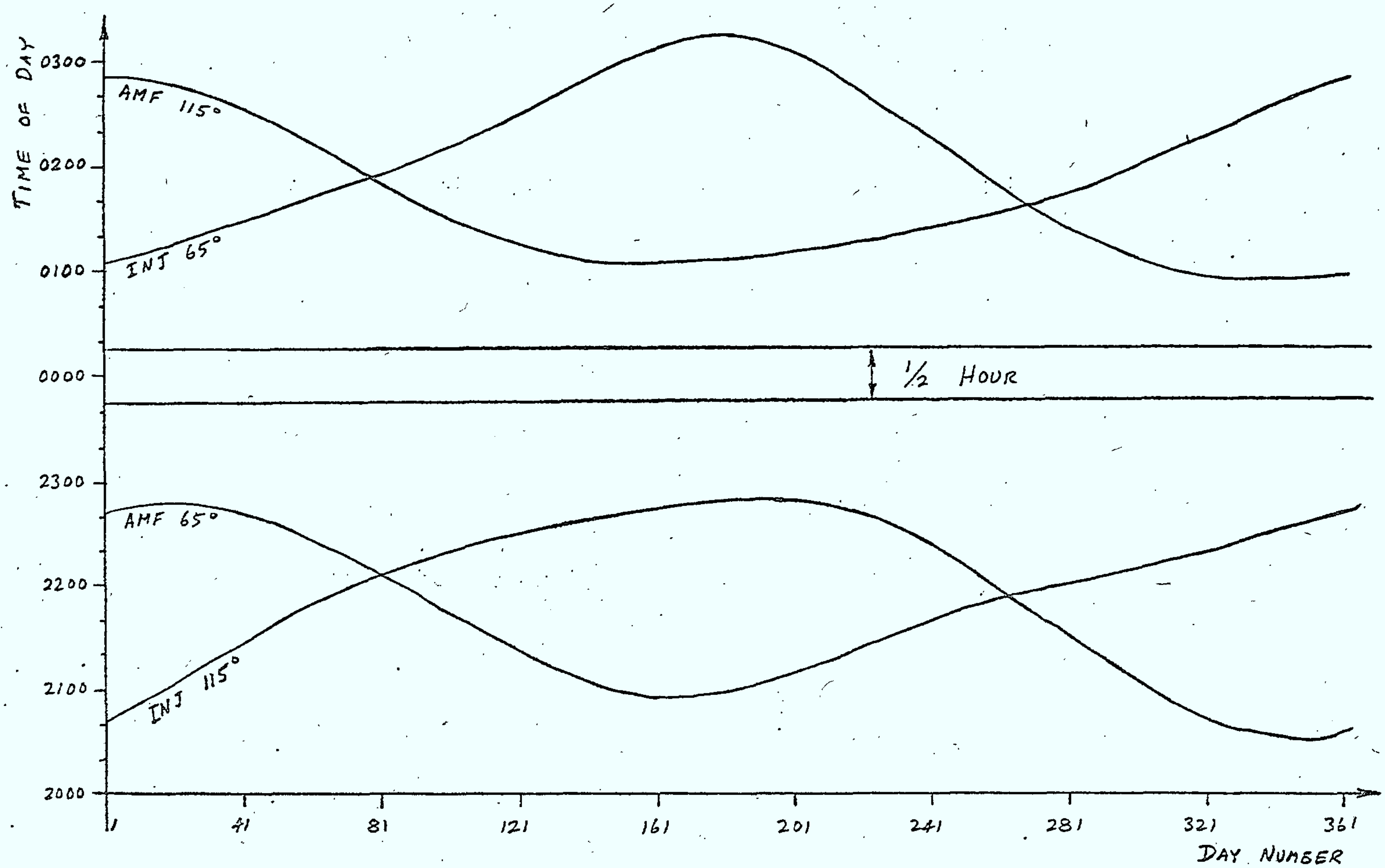
The T&C toroidal coverage patterns should present little trouble in transfer orbit. Visibility will be good for a ground network with one station in Ontario and one in the eastern hemisphere (e.g. Guam). For a seventh or ninth apogee AMF mission, only the sixth and eighth orbits will have poor visibility.

In drift orbit, the visibility will "drop out" twice per orbit while in the AMF attitude (see Figure A-11). This visibility will become 100% if the spin axis is brought to orbit normal or if the spacecraft three axis attitude is acquired.

A.5.0 POINTING REQUIREMENTS FOR THE MULTIPURPOSE BUS

A brief analysis of the impact of specified spacecraft pointing errors on the antenna position on earth has been performed which indicates that the requirement for yaw errors can be relaxed somewhat from an original specification of $\pm 0.3°$.

CFJ/12



LAUNCH WINDOW FOR $\emptyset = 65°$ TO $115°$ ($90° \pm 25°$)

FIGURE A-9

LENGTH OF WINDOW FOR
VARIOUS SUN ANGLE CONSTRAINTS

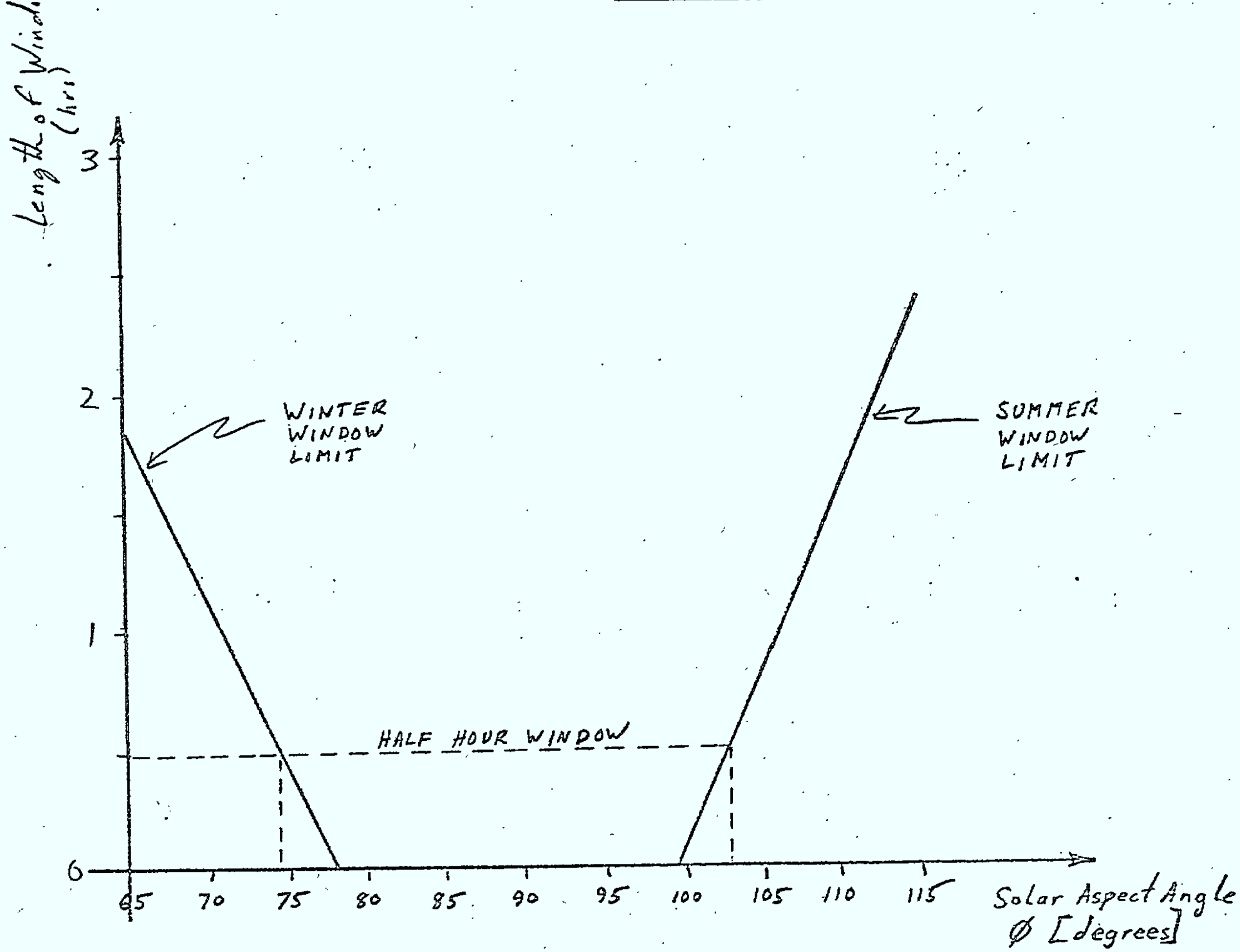


FIGURE A-10

DRIFT ORBIT T&C VISIBILITY WHILE IN AMF ATTITUDE

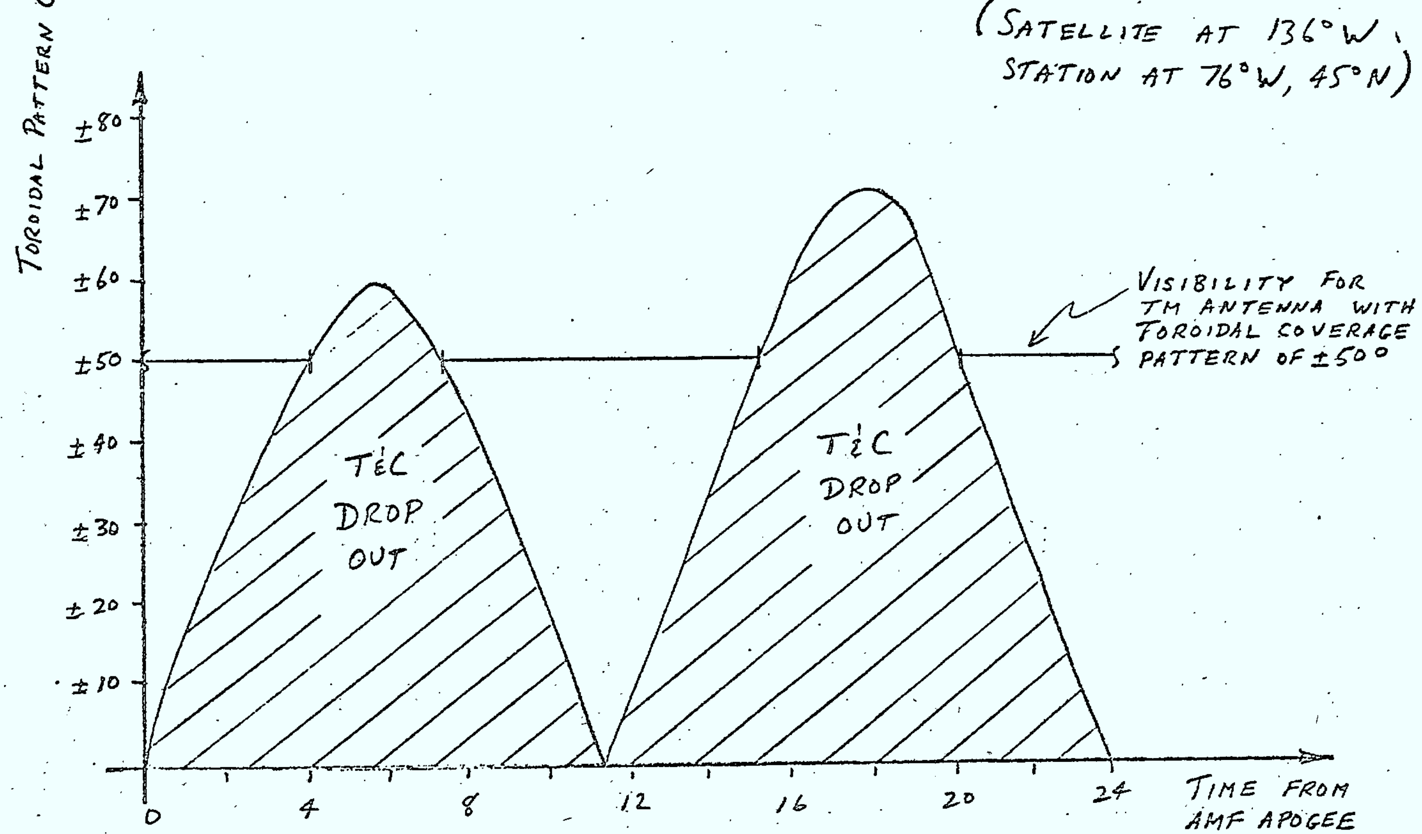


FIGURE A-11

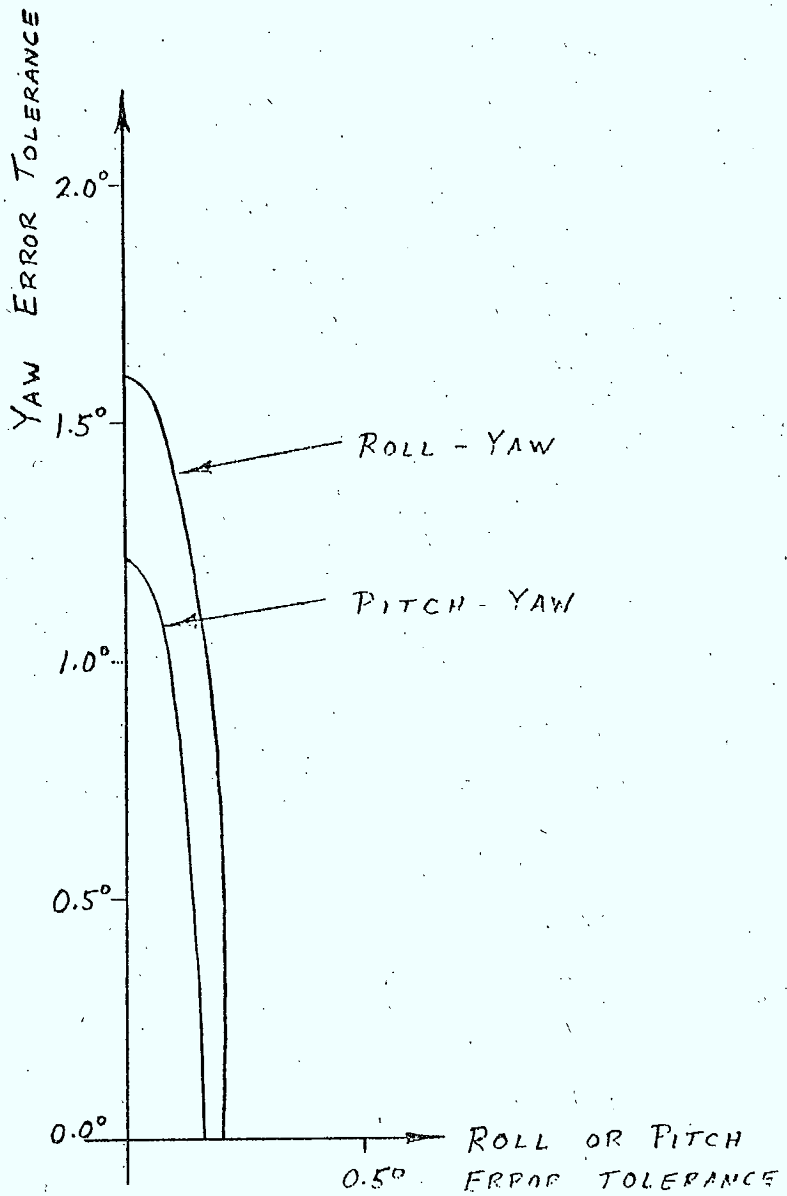
In the worst case, for a point on the earth having an elevation angle of 0° to the satellite, a ± 0.3° yaw error results in a physical position error of = 18.0 n.mi which subtends an angle at the satellite of ± 0.046°. This is a factor of 3.3 smaller than the allowable pitch error (0.15°) or a factor of 4.3 smaller than the roll error (0.2°). Due to the relative difficulty in sensing yaw (with no rate gyro), it would be desirable to loosen up the yaw error requirements, (even if at the expense of pitch and roll).

The sensitivity of the tradeoff between allowable yaw error and allowable roll error was calculated for a simplified case. For this case, the effect of pitch, roll and yaw errors on the position of the boresight of a spot beam was calculated. The boresight at the earth was assumed to be at a latitude of 50° and at the subsatellite longitude. The resulting graph (Figure A-12) shows the high sensitivity of yaw error to roll error near the specified limits. If decreased roll error may be traded off for increased yaw error, substantial relief on yaw pointing results. For example, if the roll error is restricted to ± 0.15° rather than ± 0.2°, then allowable yaw error increases from ± 0.3° to ± 1.1° (16 degrees yaw per degree of roll).

Similarly, if the pitch error tolerance was decreased by 25% to 0.1125°, the allowable yaw error increases from 0.3° to 0.84° (14.4 degrees yaw per degree pitch). If both roll and pitch errors are reduced to 0.15° and 0.1125° respectively, the allowable yaw error increases to 1.34°. Note that the yaw error is still very sensitive to changes in allowable pitch and roll errors.

It is therefore recommended that, for the purposes of this feasibility study, the yaw error tolerance be increased to at least 1 degree. Preferably, a new error budget should be calculated based on the expected gain slopes at the edge of the spot and Canadian coverage beams and the maximum permissible gain variations. The budget should be in a form that allows trade-offs with attitude control system design and stationkeeping.

CFJ/13

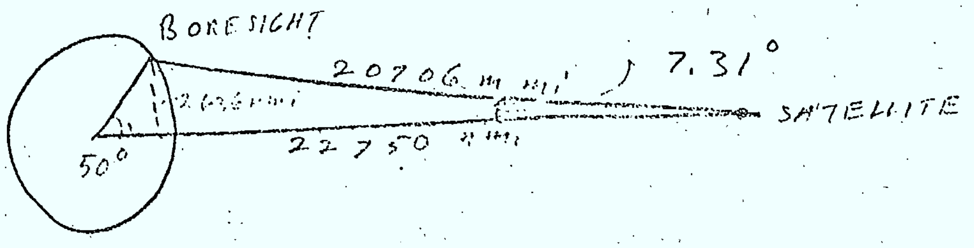


ROLL - YAW AND PITCH - YAW
ERROR TOLERANCE TRADEOFFS

FIGURE A-12

A.5.1 Beam Pointing Geometry

Station (boresight) is at LAT 50° and subsatellite longitude.



$\pm$ 0.3° yaw translates to a ground displacement of

$$\left(\frac{0.3°}{57.3} \times 2636\right) \text{ n.mi} = \pm 13.8 \text{ n.mi (east-west)}$$

This subtends an angle of $\pm$ 0.038° at the satellite, = 0.127 x yaw error.

Assume the total angular displacement at the earth is governed by the RSS angular east-west displacements and the RSS north-south displacement.

East-West = $\pm\ (0.038°^2 + 0.150°^2)^{1/2} = 0.155°$
North-South = $\pm$ 0.2°
Total allowable angular displacement is:

$$\pm\ (0.155^2 + 0.2^2)^{1/2} = 0.253°$$

- The error equation is:

$$(0.127Y)^2 + P^2 + R^2 = 0.253^2$$

- Holding P constant at 0.15°

CFJ/14

Y	R
1.60°	0.00°
1.55	0.05
1.40	0.10
1.09	0.15
0.75	0.18
0.58	0.19
0.30	0.20
0.00	0.204

- Holding R constant at 0.2°

Y	P
1.22°	0.00
1.15°	0.05
0.93°	0.10
0.66°	0.13
0.52°	0.14
0.30°	0.15
0.00°	0.155

A.6.0 STATIONKEEPING PLAN SUMMARY

- 3 week cycle, ±0.05° nominal maximum north-south and east-west

- East-West Manoeuvre Pair on Day 0 - at 90° W longitude, manoeuvres are approximately 0.49 and 0.35 ft./sec.

- Optional East-West Manoeuvre Pair in mid-cycle to correct errors, manoeuvres are less than 0.01 ft./sec.

- Inclination Manoeuvre on Day 18 - maximum manoeuvre size is 9.7 ft./sec.

- Yearly east-west fuel is 14 1/2 ft./sec. maximum

- 8 year north-south fuel is 1,320 ft./sec. maximum (for 1975 launch - 1,124 ft./sec.)

CFJ/15

APPENDIX B

ACS FLEXIBLE INTERACTION ANALYSIS

B.0 ACS FLEXIBLE INTERACTION ANALYSIS

The attitude control system the for stationkeeping mode of operation has been studied and the performance evaluated. The non-linear control system has been simulated on a digital computer that includes models which represent the sensors, compensation network, pseudo-rate modulator, thrusters, rigid body dynamics, solar panel flexible body dynamics, and a random noise generator to generate sensor noise. A sun sensor and two thrusters are assumed for the baseline configuration to provide sufficient yaw control torque for large disturbances. The thruster are modulated on-off by a dual-time-constant pseudo rate modulator which also provide sufficient damping and establish a pre-selected minimum pulse on-time that minimizes limit cycle frequency due to disturbance torques.

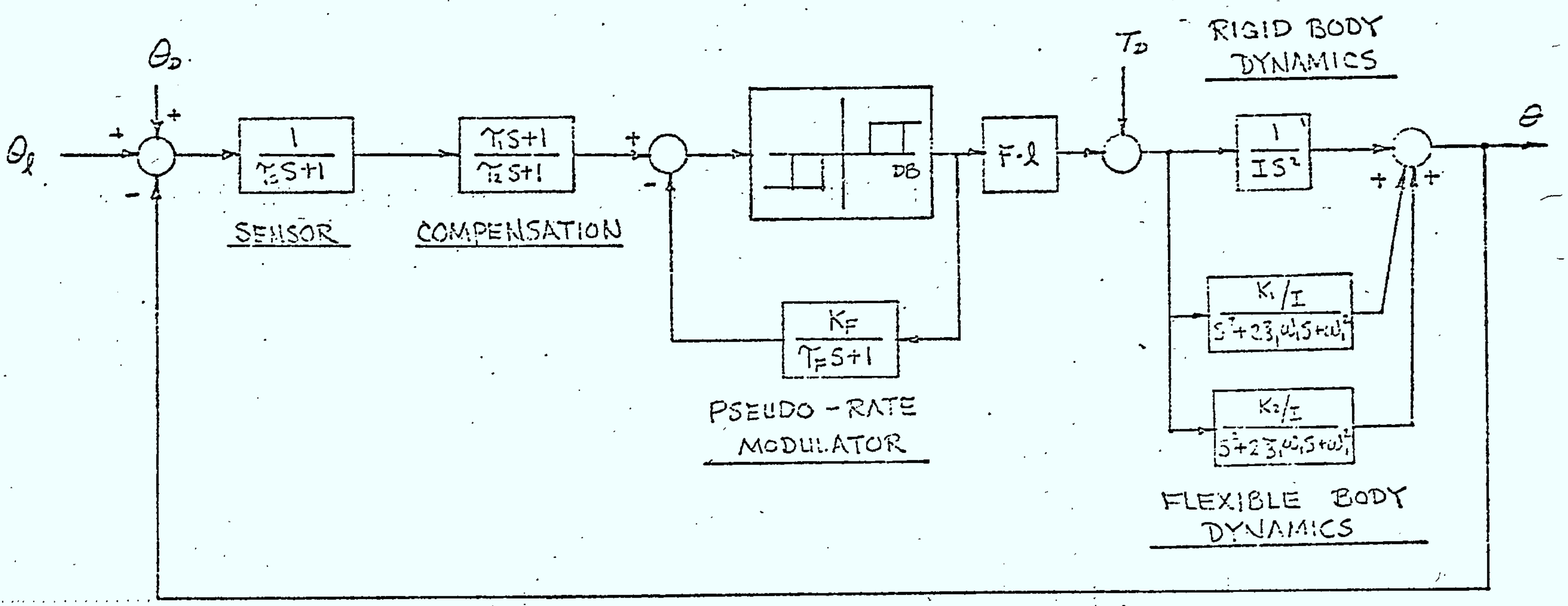
A pointing accuracy of 0.1 degree in both roll and pitch and 0.2 degree in yaw were assumed.

This study shows a maximum pointing error of 0.06 degree in both roll and yaw and 0.085 degree in pitch may be experienced.

B.1 Control System Design

The analytical model for the attitude control system during stationkeeping is depicted in Figure B-1. The dynamic response of the attitude sensors (earth and sun) were represented by a first-order lag with a 1.5 second time constant. A lead-lag compensation network is introduced in order to meet performance requirement. The lead time constant is chosen to compensate the lag effect of the sensor. The lag tends to stabilize the control system but will amplify the sensor noise transmission.

CFJ/16



ATTITUDE CONTROL SYSTEM FOR STATIONKEEPING

FIGURE B-1

B.1.1 Pseudo Rate Modulator (PRM)

The pseudo rate modulator is an important part of the overall control loop and is identified in Figure B-2. It is basically comprised of three-state switch with hysteresis in the forward path and a gain with first-order lag in the feedback loop. A deadband is established, consistent with pointing requirements, to minimize thruster pulsing.

Assuming $\varepsilon \geq 1$ the Schmitt trigger output can be either 0 or 1 depending on the value of the feedback signal X. When C = 1, the feedback lag decreases until $\varepsilon_i > 1$ and C jumps to 1. $\varepsilon i = 1-H$, where the Schmitt trigger turns off. When C = 0, the feedback signal decays until εi 1 and C jumps to 1.

In order to use the full torque capability and to meet the pointing requirement, the dual-time-constant PRM was chosen as the baseline. The time constant with the Schmitt trigger on is T_N with it off is T_F, where $T_F > T_N$. This feature permits an adjustment that is very useful with a saturated attitude sensor; i.e. the pseudo rate modulator can be adjusted to saturate before the attitude signal saturates, so as to use the full torque capability of the thrusters. Furthermore, this saturation feature permits the control system to handle a higher rate than is possible with a controller with a single-time-constant, i.e. equal on and off time constants.

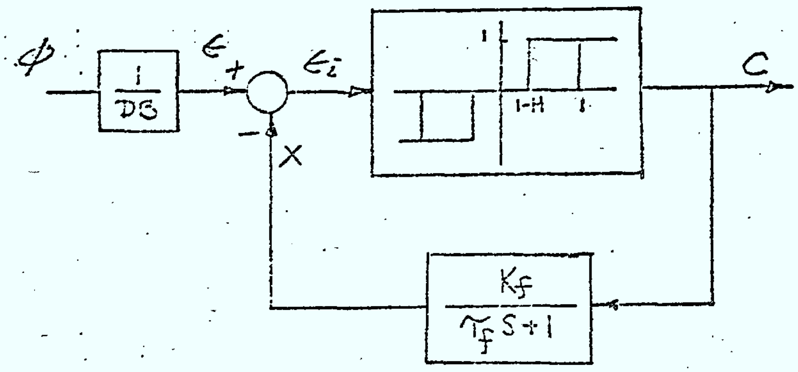
Since the dual-time-constant PRM is highly non-linear, an approximate "equivalent" single-time-constant PRM is adopted for linear analysis such as root locus analysis to investigate the stability margins.

The governing equations for the PRM are:

$$\ln\left(1 + \frac{H}{\frac{\varepsilon}{DB} - 1 - K_F}\right) = - \frac{T_F/T_N}{T/PW} \ln\left(1 + \frac{H}{\frac{\varepsilon}{DB} - 1}\right) \quad (1)$$

CFJ/17

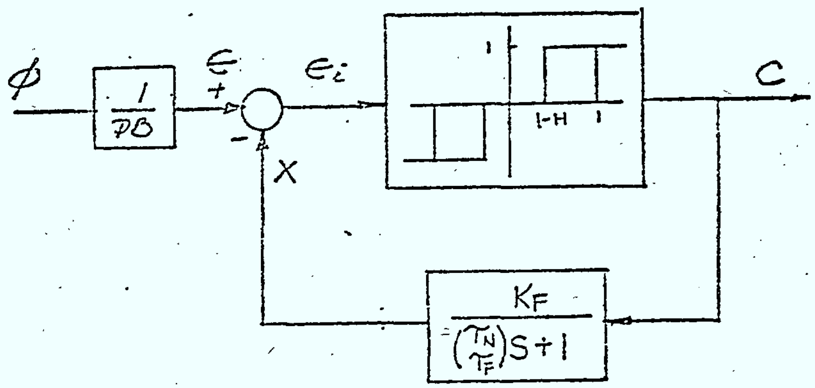
(A) SINGLE-TIME-CONSTANT PRM



K_f = feedback gain

τ_F = feedback time constant.

(B) DUAL-TIME-CONSTANT PRM



K_F = feedback gain

τ_N = "ON" time constant

τ_F = "OFF" time constant

$\tau_F \geq \tau_N$

PSEUDO-RATE MODULATOR

FIGURE B-2

$$PW_{min} = - T_N \ln \left(1 - \frac{H}{K_F}\right) \qquad (2)$$

Where the deadband (DB) depends on the pointing requirement, the hysteresis (H) is picked based on feasibility of mechanization and minimum pulse width (PW_{min}), ε the steady-state offset angle, may be sized about half of the pointing requirement. For a single-time-constant PRM, $T_F = T_N$ i.e., $T_F/T_N = 1$. T is the pulsing period. The ratio of pulsing width (PW) to pulsing period (T) should be greater than the ratio of disturbance torque (T_D) to control torque (Tc) in order to have an effective control, i.e.

$$\frac{PW}{T} \geq \frac{T_D}{Tc} \qquad (3)$$

Table 1 shows the parameter values of pseudo rate modulator.

CFJ/18

TABLE B1

PSEUDO RATE MODULATOR PARAMETERS

Parameter	Pitch	Roll	Yaw
Deadband (DB) in deg.	.03	.03	.03
Hysteresis (H) in % of DB	10%	10%	10%
Minimum Pulse Width (PW_{min}) in sec.	.01	.01	.01
o Dual-Time-Constant PRM			
Feedback Gain (K_F)	2.76	1.72	1.72
On Time Constant (T_N)	.276	.172	.172
Off Time Constant (T_F)	8.28	5.15	5.15
o "Equivalent" Single-Time-Constant PRM			
Feedback Gain (K_F)	52.5	21	21
Time Constant (T_F)	5.25	2.1	2.1

CFJ/19

B.1.2 Spacecraft Dynamics

The body nutation frequency (≈.003 Hz) is an order of magnitude smaller than the limit cycle frequency (≈.05 Hz) and the wheel control torque is much smaller than the torque generated by the thruster during stationkeeping. A simple uncoupled rigid body dynamics of:

$$\frac{1}{IS^2}$$

was assumed in all three axes.

The effect of the non-rigid solar panel is modelled according to Figure 1. The corresponding transfer function representing spacecraft dynamics is:

$$\frac{\Theta}{T} = \frac{\text{pointing angle}}{\text{effective control torque}}$$

$$\frac{\Theta}{T} = \underbrace{\frac{1}{IS^2}}_{\text{rigid body}} + \underbrace{\frac{K_1/I}{S^2 + 2S_1W_1S + W_1^2}}_{\text{flexible body, 1st unconstrained mode}}$$

or

$$\frac{\Theta}{T} = \frac{(K_1+1)\ S^2+2S_1W_1S+W_1^2}{IS^2\ (S^2 + 2S_1W_1S + W_1^2} \qquad (4)$$

Note that the solar panel flexible dynamics introduce a pair of complex zeros and poles in the s-plane, the separation of the complex poles and zeroes is a function of K_1/I, and it affects the stability and performance of the control loop.

B.1.3 Stability Consideration

Although the root locus analysis does not explicitly represent the non-linear control loop we experience here, some insight into the relative stability of

CFJ/20

the system may be obtained. As mentioned in Section 7.1.2.1 an approximate "equivalent" single-time-constant, instead of dual-time-constant, PRM has been used for the root locus analysis. The PRM network can be modelled by the following linear transfer function.

$$\text{PRM T.F.} \simeq \frac{T_f S + 1}{(HK_f)\left(\dfrac{T_f}{1 + K_f} S + 1\right)} \qquad (5)$$

A detail state space representation of the over all control loop can be found in Appendix A. The root locus plot for the roll/yaw and pitch axes including flexible modes is shown in Figure B-3 and B-4.

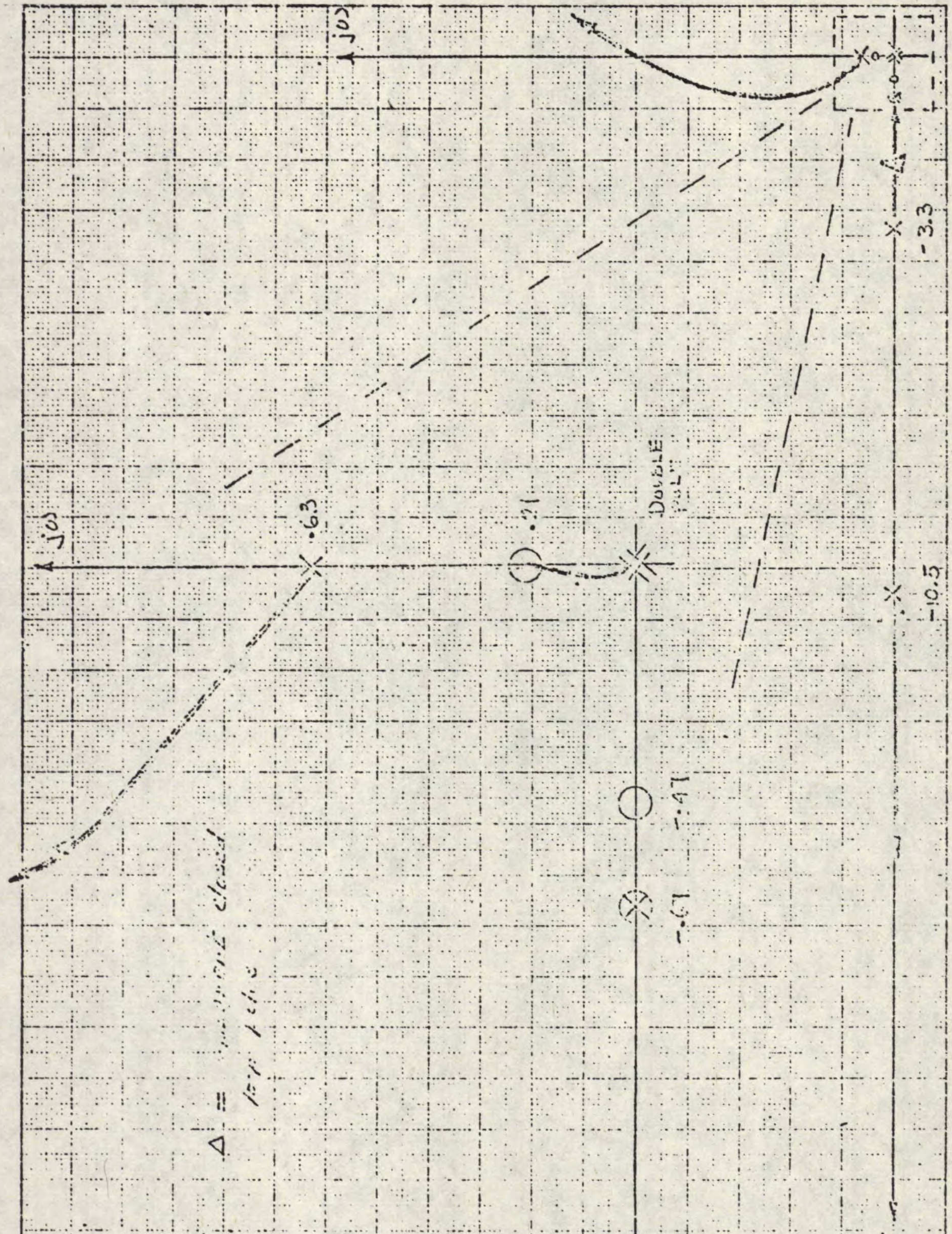
It is important to know that reducing the modal gain K_1 tends to move the complex zeros ($W_1/ 1 + K_1$) toward the complex poles ($\approx W_1$); this means reducing the flexible body effect on the control loop. If a high modal gain such as $K_1 = 8$ in the roll axis is expected, a low limit cycle frequency and a smaller steady state offset pointing error should be designed in order to have enough error budget for the flexible displacement. According to Figure B-5, this calls for a small disturbance torque. Since the disturbance torque during stationkeeping is primarily due to center-of-mass offset during jet firing, it can be biased out periodically by ground command.

B.2 Results

Table B2 summarizes the attitude control loop parameters that result in desired performance during stationkeeping. The roll/yaw pointing error with different values of disturbance torque and modal gain are tabulated in Table B3.

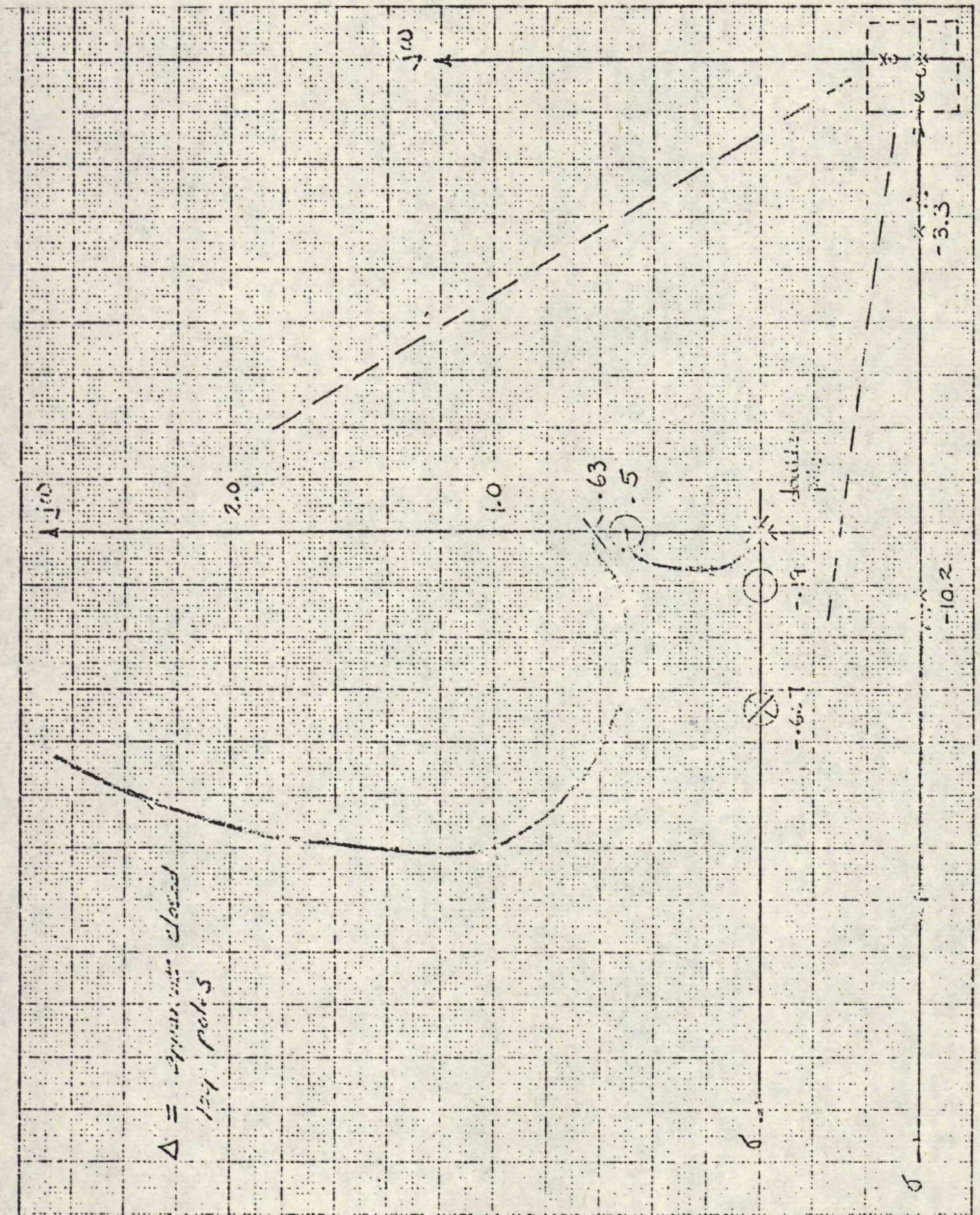
The time responses in roll/yaw and pitch axes are shown in Figures B-6 and B-7. Those responses include the flexible body effect as well as random sensor noise input.

CFJ/21



ROLL/YAW ROOT LOCUS PLOT INCLUDING FLEXIBLE MODE

FIGURE B-3



PITCH AXIS ROOT LOCUS PLOT INCLUDING FLEXIBLE MODE

FIGURE B-4

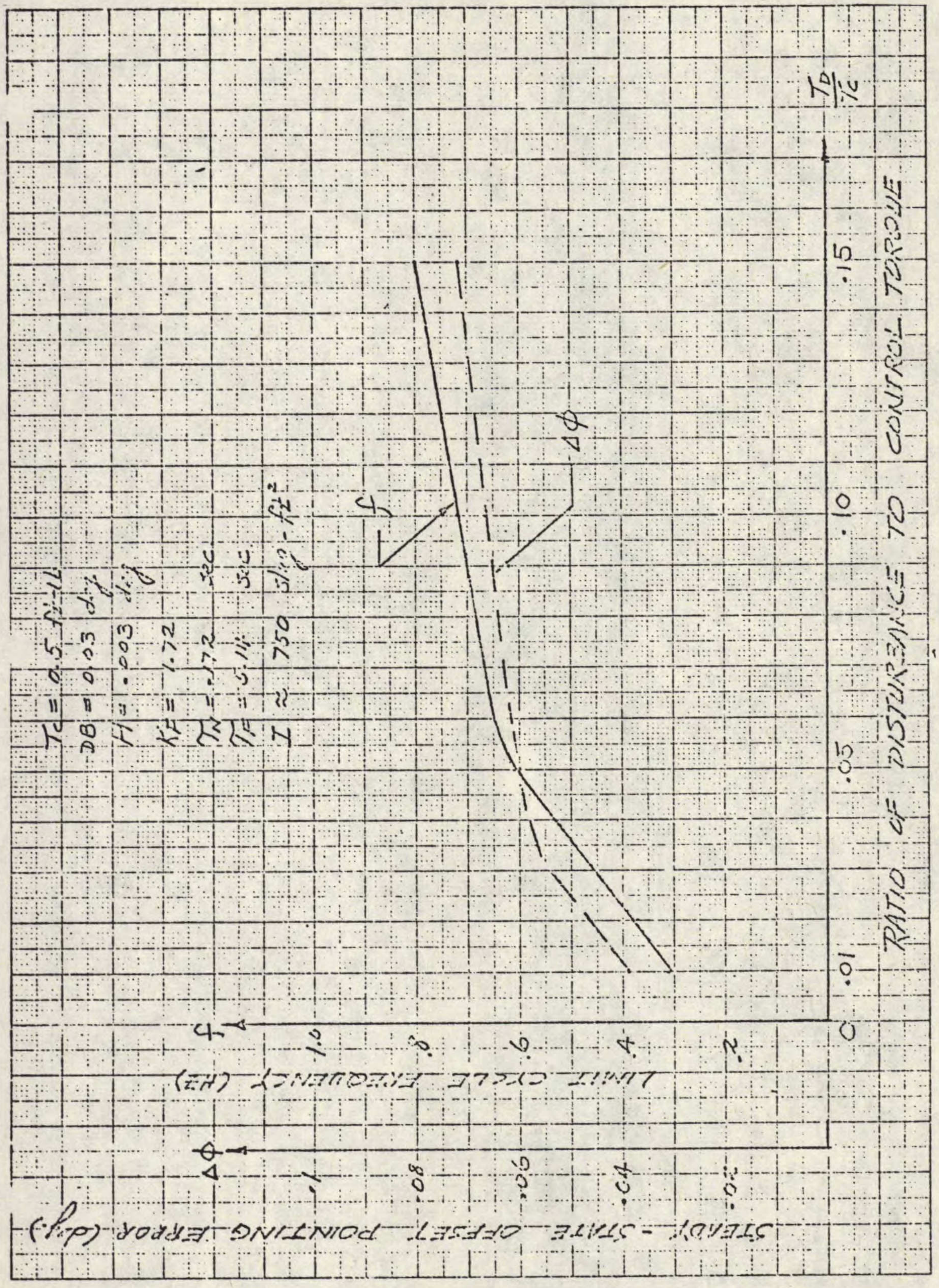

RATIO OF DISTURBANCE TO CONTROL TORQUE
T_D/T_C
0
.01
.05
.10
.15
LIMIT CYCLE FREQUENCY (HZ)
f
1.0
.8
.6
.4
.2
STEADY - STATE OFFSET POINTING ERROR (deg.)
Δφ
.1
.08
.06
.04
.02
f
Δφ

TABLE B2

ATTITUDE CONTROL LOOP PARAMETERS

	Pitch	Roll	Yaw
Sensor Lag (T_S) in sec.	1.5	1.5	1.5
Compensation			
o Lead Time Constant (T_1) in sec.	1.5	1.5	1.5
o Lag Time Constant (T_2) in sec.	.3	.3	.3
Pseudo Rate Modulator			
o Deadband (DB) in deg.	.03	.03	.03
o Hysteresis (H) in % of DB	10%	10%	10%
o Feedback Loop Gain (K_F)	2.76	1.72	1.72
o On Time Constant (T_N) in sec.	.276	.172	.172
o Off Time Constant (T_F) in sec.	8.28	5.15	5.15
o Min. Pulse Width (PW_{min}) in sec.	.01	.01	.01
Dynamics			
o Moment of Inertia (I) in slug-ft^2	68.	730.	770.
o Unconstrain 1st Modal Gain (K_1)	0.6	8.	5.6
o Unconstrain 1st Mode Frequency (W_1) in Hz	0.1	0.1	0.1
o Damping Ratio (S_1)	0.001	0.001	0.001
Control Torque (T_C) in ft-lbs.	0.25	0.5	0.5

TABLE B3

ROLL/YAW POINTING VERSUS T_D AND K

T_O (ft-lb) Disturbance Torque	T_O (ft-lb) Control Torque	3_S (deg.) Sensor Noise	W_1 (Hz) 1st Mode Frequency	$_1$ Damping Ratio	K_1 1st Modal Gain	Max. (deg.) Pointing Error
.025	.5	.025	.1	.001	8.	Unstable
.025	.5	.025	.1	.001	6.	Unstable
.025	.5	.025	.1	.001	5.6	.085
.025	.5	.025	.1	.001	1.	.071
.01	.5	.025	.1	.001	8.	Unstable
.01	.5	.025	.1	.001	5.6	.063
.005	.5	.025	.1	.001	8.	.058
.005	.5	.025	.1	.001	5.6	.055

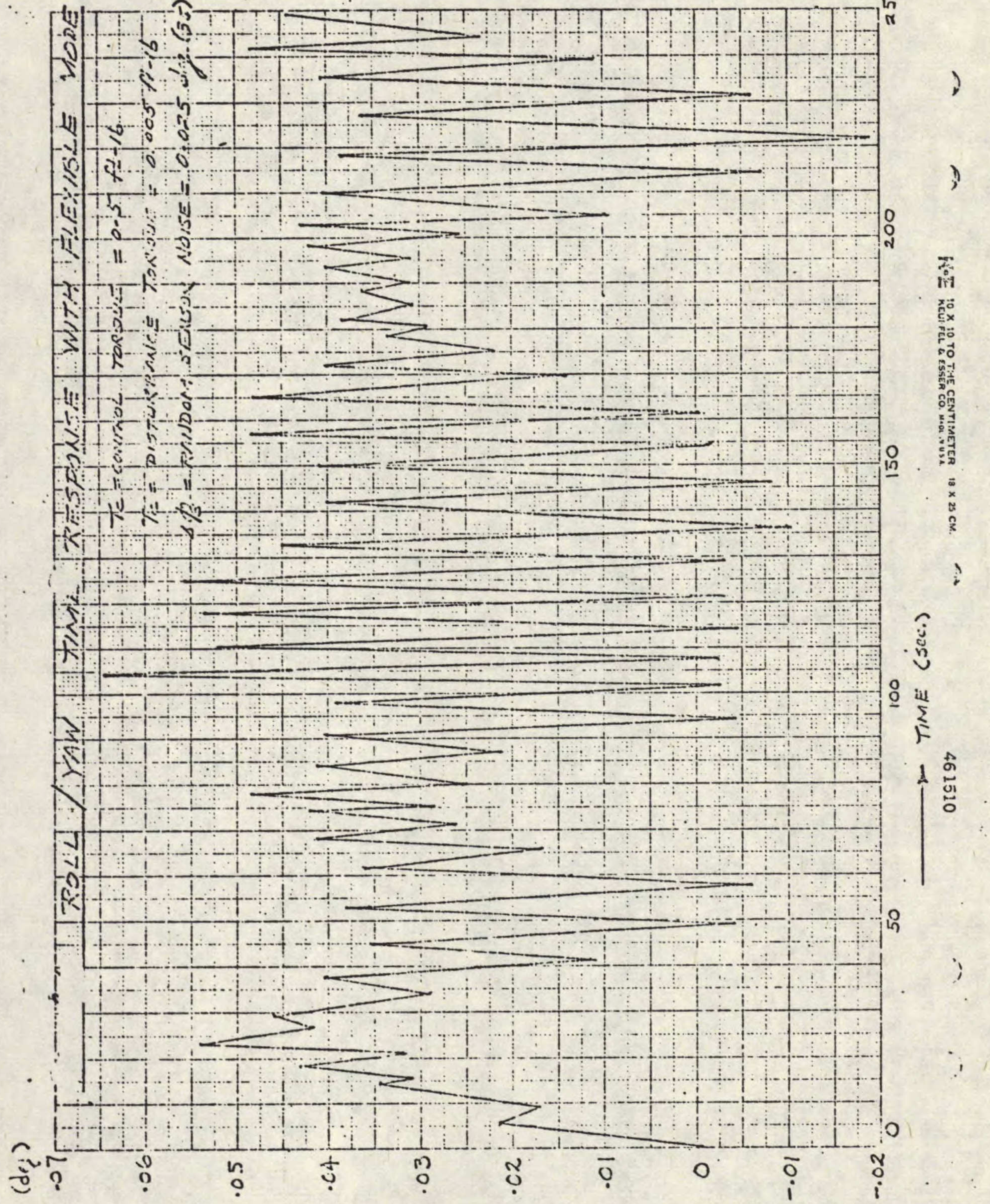

ROLL / YAW TIME RESPONSE WITH FLEXIBLE MODE
TIME (sec.)
0
50
100
150
200
250
(deg)
.07
.06
.05
.04
.03
.02
.01
0
-.01
-.02
10 X 10 TO THE CENTIMETER 18 X 25 CM.
KEUFFEL & ESSER CO. MADE IN U.S.A.
46 1510

FIGURE B-6

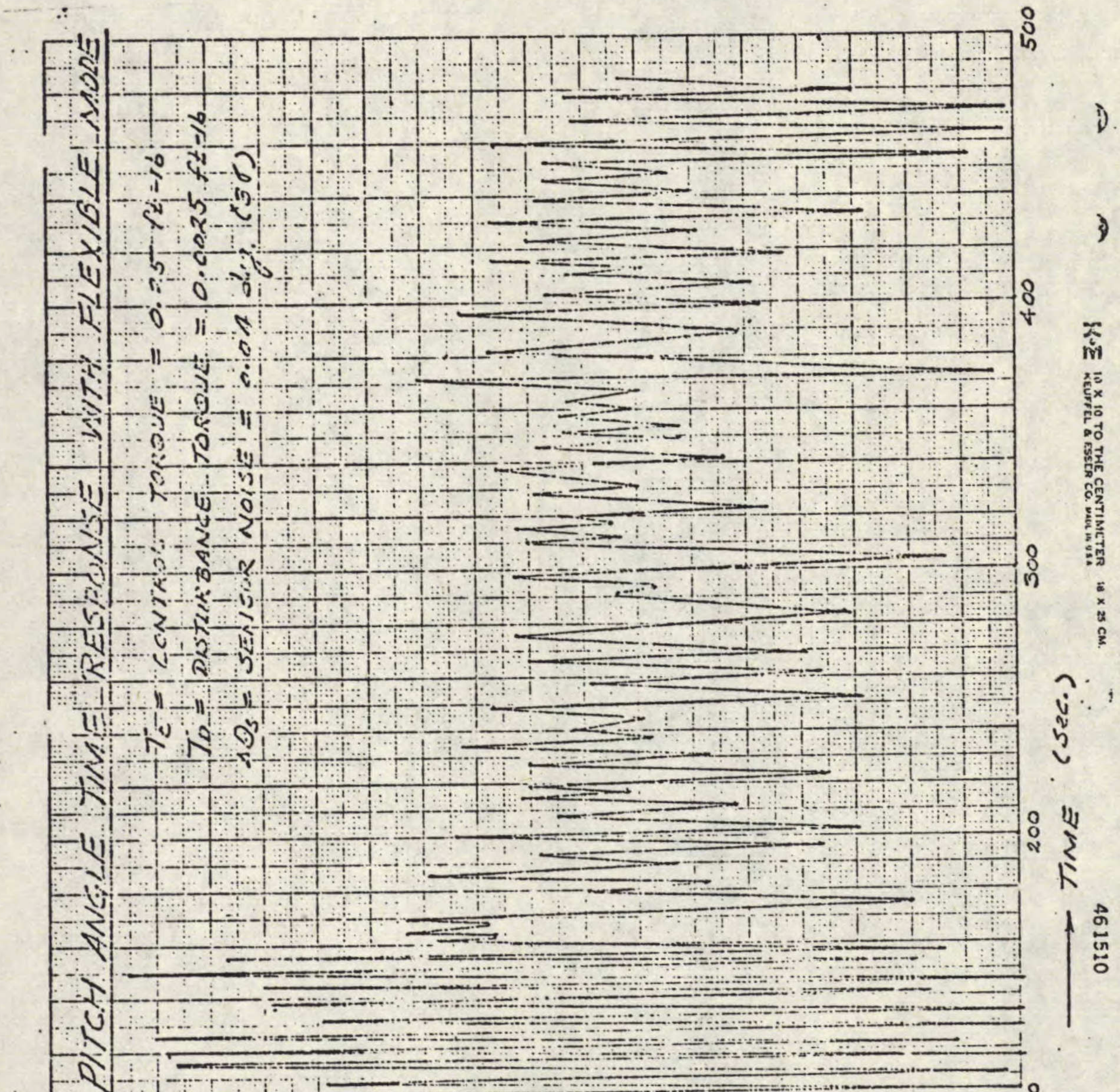

PITCH ANGLE TIME RESPONSE WITH FLEXIBLE MODE
T_C = CONTROL TORQUE = 0.25 ft-lb
T_D = DISTURBANCE TORQUE = 0.0025 ft-lb
$\Delta\theta_S$ = SENSOR NOISE = 0.04 deg. (3σ)
(deg.)
.09
.08
.07
.06
.05
.04
.03
.02
.01
0
100
200
300
400
500
TIME (sec.)
461510
10 X 10 TO THE CENTIMETER 18 X 25 CM.
KEUFFEL & ESSER CO. MADE IN U.S.A.

FIGURE B-7

Appendix A shows a detailed state space representation of the overall control system. The computer programs for root locus analysis and time response can be found in Appendix B.

CFJ/22

B.3 STATE VARIABLE REPRESENTATION

Fig B-8 shows the state variable representation of attitude control system shown in Fig B-1. The state equations are

$$\dot{X}_1 = X_2$$
$$\dot{X}_2 = \frac{FL}{I}\left[X_5 + \tau_F \dot{X}_5\right]$$
$$\dot{X}_3 = X_4$$
$$\dot{X}_4 = \frac{K_1 FL}{I}\left[X_5 + \tau_F \dot{X}_5\right] - 2\zeta_1\omega_1 X_4 - \omega_1^2 X_3$$
$$\dot{X}_5 = \frac{1}{(DB)\tau_F}\left(X_6 + \tau_1 \dot{X}_6\right) - \left(\frac{1+K_F}{\tau_F}\right) X_5$$
$$\dot{X}_6 = \frac{1}{\tau_2} X_7 - \frac{1}{\tau_2} X_6$$
$$\dot{X}_7 = -\frac{57.3}{\tau_3}(X_1 + X_3) - \frac{1}{\tau_3} X_7$$

and

$$\dot{X}_5 = \frac{1}{(DB)\tau_F} X_6 + \frac{\tau_1}{(DB)\tau_F}\left[\frac{1}{\tau_2}X_7 - \frac{1}{\tau_2}X_6\right] - \frac{1+K_F}{\tau_F} X_5$$

$$\therefore \quad \dot{X}_5 = -\frac{1+K_F}{\tau_F} X_5 + \frac{1}{(DB)\tau_F}\left(1 - \frac{\tau_1}{\tau_2}\right) X_6 + \frac{1}{(DB)\tau_F}\frac{\tau_1}{\tau_2} X_7$$

$$\dot{X}_2 = \frac{FL}{I}\left\{-K_F X_5 + \frac{1}{(DB)}\left(1 - \frac{\tau_1}{\tau_2}\right) X_6 + \frac{1}{(DB)}\frac{\tau_1}{\tau_2} X_7\right\}$$

$$\therefore \quad \dot{X}_2 = -\frac{FL\,K_F}{I} X_5 + \frac{FL}{I}\frac{1}{DB}\left(1 - \frac{\tau_1}{\tau_2}\right) X_6 + \frac{FL}{I}\frac{1}{(DB)}\frac{\tau_1}{\tau_2} X_7$$

$$\dot{X}_4 = -\frac{K_1 FL\,K_F}{I} X_5 + \frac{K_1 FL}{I}\frac{1}{DB}\left(1 - \frac{\tau_1}{\tau_2}\right) X_6 + K_1\frac{FL}{I}\frac{1}{DB}\frac{\tau_1}{\tau_2} X_7$$
$$- 2\zeta_1\omega_1 X_4 - \omega_1^2 X_3$$

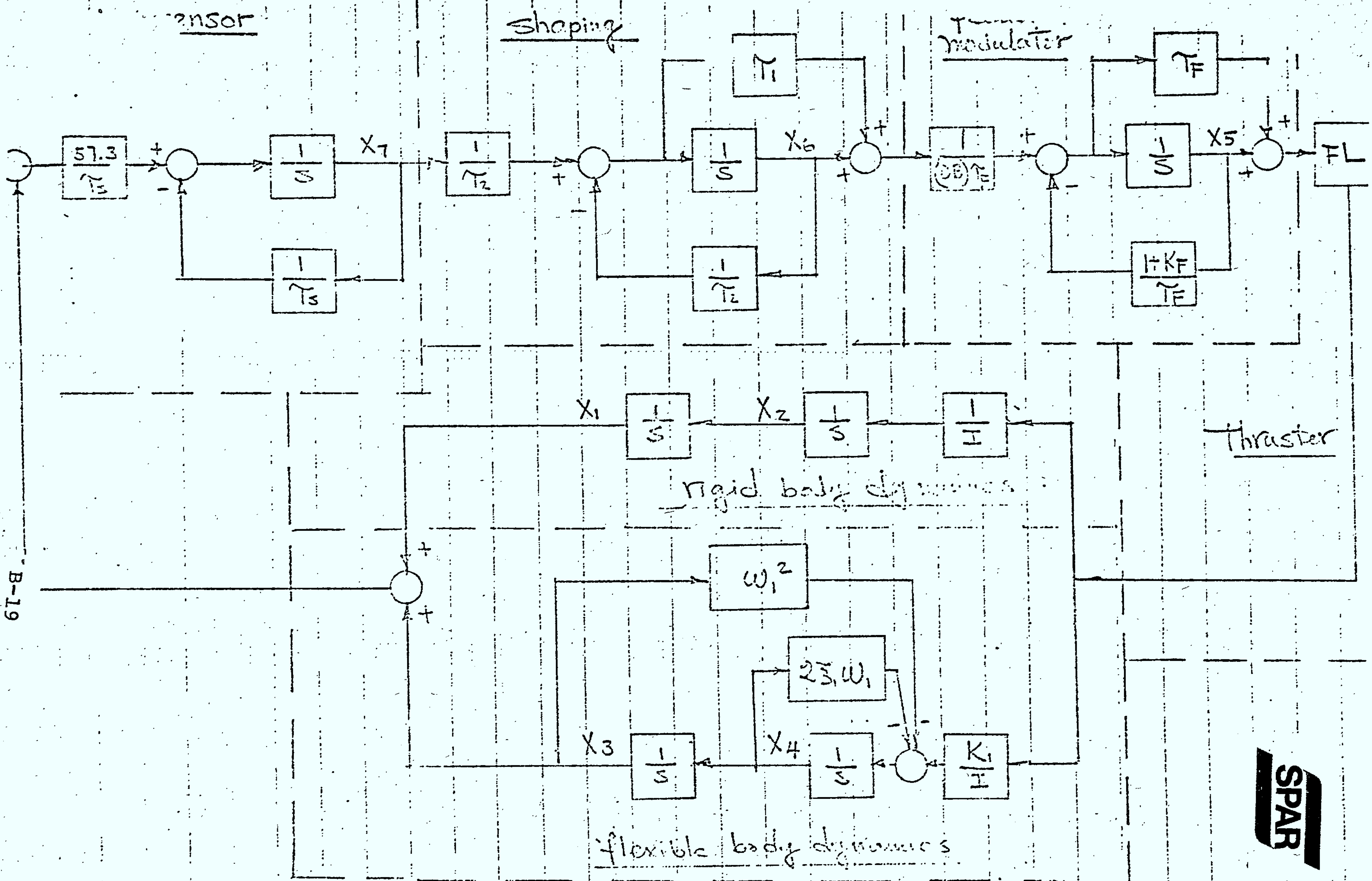


FIG. B-8 STATE VARIABLE REPRESENTION FOR CONTROL SYSTEM

SPAR

$$\begin{Bmatrix} \dot{x}_1 \\ \dot{x}_2 \\ \dot{x}_3 \\ \dot{x}_4 \\ \dot{x}_5 \\ \dot{x}_6 \\ \dot{x}_7 \end{Bmatrix} = \begin{bmatrix} 0 & 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & -\frac{FLK_F}{I} & \frac{FL}{I}\frac{1}{DB}(1-\frac{T_1}{T_2}) & \frac{FL}{I}\frac{1}{DB}\frac{T_1}{T_2} \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & -\omega_1^2 & -2\zeta_1\omega_1 & -\frac{K_1FLK_F}{I} & \frac{K_1FL}{I}\frac{1}{DB}(1-\frac{T_1}{T_2}) & \frac{K_1FL}{I}\frac{1}{DB}\frac{T_1}{T_2} \\ 0 & 0 & 0 & 0 & -\frac{1+K_F}{T_F} & \frac{1}{(DB)}\frac{1}{T_F}(1-\frac{T_1}{T_2}) & \frac{1}{(DB)}\frac{1}{T_F}\frac{T_1}{T_2} \\ 0 & 0 & 0 & 0 & 0 & -\frac{1}{T_2} & \frac{1}{T_2} \\ -\frac{57.3}{T_S} & 0 & -\frac{57.3}{T_S} & 0 & 0 & 0 & -\frac{1}{T_S} \end{bmatrix} \begin{Bmatrix} X_1 \\ X_2 \\ X_3 \\ X_4 \\ X_5 \\ X_6 \\ X_7 \end{Bmatrix}$$

$$\dot{\underline{X}} = [A]\ \underline{X}$$

where

$a(1,2) = 1.0$

$a(2,5) = -\frac{(FL)K_F}{I}$

$a(2,6) = \frac{(FL)}{I}\frac{1}{DB}\left(1-\frac{T_1}{T_2}\right)$

$a(2,7) = \frac{(FL)}{I}\frac{1}{DB}\frac{T_1}{T_2}$

$a(3,4) = 1.0$

$a(4,3) = -\omega_1^2 \times (2\pi)^2$

$a(4,4) = -2\zeta_1\omega_1(2\pi)$

$a(4,5) = -\frac{K_1(FL)K_F}{I}$

$a(4,6) = \frac{K_1(FL)}{I}\frac{1}{DB}\left(1-\frac{T_1}{T_2}\right)$

$a(4,7) = \frac{K_1(FL)}{I}\frac{1}{DB}\frac{T_1}{T_2}$

$a(5,5) = -\frac{(1+K_F)}{T_F}$

$a(5,6) = \frac{1}{DB}\frac{1}{T_F}\left(1-\frac{T_1}{T_2}\right)$

$a(5,7) = \frac{1}{DB}\frac{1}{T_F}\frac{T_1}{T_2}$

$a(6,6) = -\frac{1}{T_2}$

$a(6,7) = \frac{1}{T_2}$

$a(7,1) = -\frac{57.3}{T_S}$

$a(7,3) = -\frac{57.3}{T_S}$

$a(7,7) = -\frac{1}{T_S}$

B.4 DIGITAL COMPUTER PROGRAMS

ROOT LOCUS PROGRAM

SPAR

```
-OP,JCL
  >OU
JOB,NTARCHENUHF,70001,A325,CM100000.
DISPOSE,OUTPUT,*LP,GT.
FTN.
ATTACH,AAA,LISTAN1,PW=SPAR.
COPYBF,AAA,DUM.
BKSP,AAA.
REWIND,LGO.
COPYBR,LGO,AAA,4.
ATTACH,LIB,AUXFORTRAN.
SELECT,D,I=AAA,L=LIB.
AAA.
7/8/9
  >CL
JCL  CLOSED
  >
```

```
 OP,SYSEQ
  >OU
C=======================================================================
      SUBROUTINE SYSEQ(KEY)
C=======================================================================
C
C
      COMMON/SET/   A(40,40)  ,X(40)  ,B(40,40)  ,R(40)
     *                        ,Y(40)  ,H(40,40)
      COMMON/INP/  V(1000)
      GO TO (100,399,300),KEY
C
C***********************************************************************
C******  SET UP CONSTANT ELEMENTS OF MATRIX (KEY=1)   *****************
C***********************************************************************
C
  100 A(1,2)=1.
      A(3,4)=1.
C
C***********************************************************************
C*******  SET UP VARIABLE ELEMENTS OF MATRIX (KEY=3)
C***********************************************************************
C
  300 A(2,5)=-(V(60)/V(54))*V(52)
       A(2,6)=(V(60)/V(54))*(1./V(55))*(1.-V(57)/V(58))
      A(2,7)=(V(60)/V(54))*(1./V(55))*(V(57)/V(58))
      A(4,3)=-(6.2832*V(51))**2.
      A(4,4)=-2.*6.2832*V(59)*V(51)
      A(4,5)=-V(50)*(V(60)/V(54))*V(52)
      A(4,6)=V(50)*(V(60)/V(54))*(1./V(55))*(1.-V(57)/V(58))
      A(4,7)=V(50)*(V(60)/V(54))*(1./V(55))*(V(57)/V(58))
      A(5,5)=-(1.+V(52))/V(53)
      A(5,6)=(1./(V(55)*V(53)))*(1.-V(57)/V(58))
       A(5,7)=V(57)/(V(55)*V(53)*V(58))
      A(6,6)=-1./V(58)
      A(6,7)=1./V(58)
      A(7,1)=-57.296/V(56)
      A(7,3)=-57.296/V(56)
      A(7,7)=-1./V(56)
```

```
000400          END
   >CL
SYSEQ  CLOSED
   >



    OP.LDATA
   >OU
000100  7/8/9 NS
000150    CARLS STATIONKEEPER
000200      1     7.
000300      2    1.
000400      3    60.
000500      4    0.0
000600      5    1.0
000700      6    10.
000800      7    59.
000900      8    0.
001000      9    4.
001100     10    2.
001200     50    5.0
001300     51    .1
001400     52    21.
001500     53    2.1
001600     54    770.
001700     55    .03
001800     56    1.5
001900     57    1.5
 02000     58    .3
002100     59    .001
002110     60    0.1
002200  7/8/9 NS
002300
   >CL
LDATA  CLOSED
   >
```

```
*TY1-200
C      THIS PROGRAM IS WRITTEN BY CARL L. CHEN ON MARCH 25,
C      1975 FOR SPAR AEROSPACE PRODUCTS, LTD., TORONTO, CANADA.
C      THE PURPOSE OF THE PROGRAM IS TO STUDY THE PERFORMANCE
C      OF A GENERAL PURPOSE 3-AXIS SATELLITE DURING STATION
C      KEEPING
C      ********** THE MAIN PROGRAM **********
C
       DIMENSION DV(6),V(6),X(5)
       COMMON DV,V,X,TS,TC,XK,XIX,XH,XH1,DB,TF,DT,NV
       COMMON TD,IX,XNOISE,SI,WI
       SI=.001
       WI=.62832
       XK1=5.6
       IX=173159
       TC=.5
       NV=6
       FT=600.
       TS=.3
       XKF=1.7157
       XK=XKF
       XIX=730.
       XH=0.1
       DB=0.03
       XH1=1.-XH
       TF=5.147
       TD=.005
       T=0.0
       N=0
   100 FORMAT(6X,'    T        X(1)    X(2)        X(3)       V(4)
V(6)
      1       X(5)          XNOISE            V(1)             TF')
   110 FORMAT(1X,E12.5,3(F8.3),5(E12.4),F8.3,E12.4)
   120 FORMAT(1X,13(E10.2))
       X(1)=0.0
       X(2)=0.
       X(3)=0.
       X(4)=0.
       X(5)=0.
       V(1)=0.0
       V(2)=0.
       V(3)=0.
       V(4)=0.
       V(5)=0.
       V(6)=0.
       XNOISE=0.
C      PRINT 120,T,X(1),X(2),X(3),X(4),X(5),V(1),V(2),V(3),
C      1V(4),V(5),V(6),XNOISE
       PRINT 100
    10 X(1)=V(1)/DB-V(2)
       IF(ABS(X(1))  .LE.  XH1) GO TO 20
       IF(ABS(X(1))  .GE.  1.) GO TO 30
       IF (X(2) .EQ. 1.) GO TO 35
       DT=0.1
       TF=5.147
       GO TO 40
    35 DT=.002
       TF=.17157
       GO TO 40
    20 X(2)=0.
       DT=0.1
       TF=5.147
```

```
   40 X(3)=X(2)*TC-TD
      XNOISE=.0083*GAUSS(IX)
      X(4)=X(3)*XK1/XIX-2.*SI*W1*V(5)-W1*W1*V(6)
      X(5)=(V(4)+V(6))*57.296
C
      CALL RKGIL
      TPRINT=T-N*0.5
      IF(TPRINT .GE. 0.5) GO TO 42
      GO TO 44
C
C
   42 PRINT 110,T,X(1),X(2),X(3),V(4),V(6),X(5),XNOISE,V(1),TF,
X(4)
      N=N+1
C
   44 T=T+DT
      IF(T .GE. FT) GO TO 50
      GO TO 10
   50 STOP
      END
C     *****************
C
      SUBROUTINE RKGIL
C
C
C
      DIMENSION AA(4),BB(4),CC(4),Z(30)
C
      COMMON DV(6),V(6),X(5),TS,TC,XK,XIX,XH,XH1,DB,TF,DT,NV
      COMMON TD,IX,XNOISE,SI,W1
      DATA AA/.5,.2928931,1.707107,.1666667/, BB/2.,1.,1.,2./
     1 , CC/.5,.2928931,1.707107,.5/, Z/30*0.0/
      J=1
C
C
   10 AJ=AA(J)
      BJ=BB(J)
      CJ=CC(J)
      DO 11 I=1,NV
      R1=DT*DV(I)
      R2=AJ*(R1-BJ*Z(I))
      V(I)=V(I)+R2
      R2=R2+R2+R2
   11 Z(I)=Z(I)+R2-CJ*R1
      IF(J-4) 12,15,15
C
C
C
C
C
   12 J=J+1
      IF(J-3) 13,14,13
   13 CONTINUE
   14 CALL DER
      GO TO 10
   15 CALL DER
      RETURN
      END
C
       SUBROUTINE DER
C
      COMMON DV(6),V(6),X(5),TS,TC,XK,XIX,XH,XH1,DB,TF,DT,NV
```

```
      DV(1)=-(X(5)+XNOISE+V(1))/TS
      DV(2)=(XK*X(2)-V(2))/TF
      DV(3)=X(3)/XIX
      DV(4)=V(3)
      DV(5)=X(4)
      DV(6)=V(5)
      RETURN
      END
--EOF HIT AFTER  168.
*
!COPY GAUSS
      FUNCTION GAUSS(IX)
      A=0.0
      DO 50 I=1, 12
      IY=IX*65539
      IF(IY)5,6,6
    5 IY=IY+2147483647+1
    6 Y=IY
      Y=Y*.4656613E-9
      IX=IY
   50 A=A+Y
      GAUSS=A-6.0
      RETURN
      END

!
```

FEASIBILITY STUDY OF A GENERAL
PURPOSE SPACECRAFT BUS.

DATE DUE
DATE DE RETOUR
