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FEASIBILITY STUDY OF A GENERAL PURPOSE SPACECRAFT BUS

VOLUME II
SPECIFICATIONS AND RESPONSES FROM VENDORS

SPAR

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

For Department of Communications

MAY, 1975

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SPAR AEROSPACE PRODUCTS LTD

SPAR-R.677
VOLUME II

FEASIBILITY STUDY OF A GENERAL PURPOSE SPACECRAFT BUS

SPECIFICATIONS AND RESPONSES FROM VENDORS

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CFX/10

May, 1975

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.

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1.0 SPACECRAFT BUS AND SUBSYSTEM SPECIFICATIONS

This volume of the report presents preliminary performance specifications that have been created for budgetary pricing purposes for the spacecraft Bus and each of the mechanical subsystems.

For the Attitude Control Subsystem, a specification prepared during a previous study has been included. This is fairly representative of the requirements for the General Purpose Satellite Bus. A Statement of Work for the Reaction Control Subsystem has also been included to indicate a typical document.

Also included in this volume are listings of potential vendors and assessments for the Reaction Control Subsystem (RCS), Deployable Solar Array Subsystem (DSA), and Apogee Motor Subsystem.

RCS and Apogee Motor vendor technical proposals are appended. Also appended is a report of the status of Electric Propulsion (Ion) engines reviewed during this study.

The report contains the tradeoff studies that were conducted on RCS and which have been used in selecting super heated electro-thermal engine systems as the baseline. Further details on this tradeoff can be found in Section 2.1.2.

1.1 Spacecraft Bus Performance Specification

SPAR-SG.359

PERFORMANCE SPECIFICATION

FOR

SPACECRAFT GENERAL PURPOSE BUS

(FOR BUDGETARY PRICING PURPOSES)

Prepared by: Staff

Approved by: P.A. McIntyre
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Program Manager

May, 1975

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1.0 SCOPE

This specification is a general requirement for the spacecraft BUS for future Canadian communication satellites flying geostationary missions in the next decade.

Most probable communication payloads are among the following:

- UHF 4-6 GHz transponder with 12 channels
- UHF/SHF transponder
- UHF/SHF transponder with auxilary experimental payload
- UHF/SHF/L-Band transponder
- 4-6 GHz transponder with 24 channels

2.0 ASSUMPTIONS AND DESIGN INFORMATION

As a design guide the following assumptions will apply:

- Launch lift off spacecraft weight including apogee motor, (however, excluding the launch vehicle attach fitting) is 1,925 lbs., (i.e. 3914 Thor Delta launch vehicle capability.

- Spin stabilized during transfer orbit and 3-axis stabilized spacecraft in synchronous orbit, with a synchronous orbit attitude acquisition sequence to be defined by the contractor.

- On-orbit solar arrays to be deployable and sun-oriented with a single axis. The same arrays, when stowed, should provide power for transfer orbit.

- UHF payload will require a deployable dish antenna of 10 to 16 feet diameter.

- Operational life-time is six to eight years.

Other design data relating to communications and other subsystems requirements are given in Attachment I, (Design Requirements). The contractor should make reference to CTS technology in conducting this study but should not feel constrained to base the BUS design on that of CTS.

The contractor should consider the costs of development and production as a prime trade-off parameter.

3.0 BUS SUBSYSTEM DESIGN REQUIREMENTS

3.1 Structure

The payload components as described in Attachment I, (Design Requirements), shall be supported by the structure.

The design of the structure should facilitate a change of communications payloads, (e.g. antennae) for the various options listed in paragraph 1.0 above.

The forward (earth facing) deck should be designed to minimize angular distortion during launch and during the various on-orbit thermal environments.

The payload arrangements should be designed for modularized subsystems so as to accommodate mission-to-mission variations in subsystems design such as battery size, solar array area and RCS fuel capacity. Ease of access, for integration and component substitution, should be a design goal. Consideration should be given to ground handling requirements.

Minimum structural weight shall be a design goal.

Payload envelope in the launch vehicle, attach fitting configuration and launch environment shall be compatible with the 3914 Thor Delta launch vehicle.

3.2 Thermal Control

Thermal control of the spacecraft shall maintain all subsystems within their specified temperature limits for all phases of the mission, including pre-launch operations. The spacecraft temperatures shall be maintained by passive temperature control techniques such as the following:

a) Selection of unit locations/baseplate to structure and conductance and surface emittance depending on the magnitude of internal heat dissipation.

b) Choice of solar absorptance and infrared emittance materials for externally mounted equipment, heat shields and closures.

c) Use of ground command or thermostatically controlled heater elements.

d) See Attachment I, (Design Requirements), for dissipations and temperature limits for payload components. A passive thermal design philosophy should be utilized, if possible, without the use of heat pipes or louvres.

e) The subsystem will also consist of insulating blankets, paints and finishes, isolations, heat shield and spacecraft closures to provide adequate thermal control.

3.3 Solar Array

The solar array power requirements shall not be less than 800 watts (EOL). A minimum weight deployable array is required which can be tailored to particular mission requirements within the range 600-800 watts (EOL). The array shall be capable of surviving launch vibration and shall provide sufficient exposed cell array in the launch configuration to meet spacecraft needs in transfer and drift orbit, (see Design Requirements). Thermal deformations and fabrication tolerances shall be minimized to reduce in-orbit solar torquing effects.

Reference should be made to the Design Authority with regard to selection of type of solar cells, degradation factors, etc.

3.4 Reaction Control Subsystem (RCS)

The Reaction Control Subsystem shall be capable of supplying the thrust required to position and orient the spacecraft and maintain station and orientation as specified during the spin stabilized and three axis stabilized phases of the mission.

a) Fuel - hydrazine

b) Thrusters - TBD

c) Feed System - blowdown from pressurized fuel storage tanks

d) High Level Thrusters

i) Precess spacecraft 180° with a spin rate of 60-100 rpm.

ii) Velocity increment for station acquisition of 170 fps.

e) Low Level Thrusters

i) Despin to 2 rpm,
ii) East/West stationkeeping,
iii) Attitude control,
iv) Singly redundant.

f) Fuel Weight - sufficient for six years.

g) Reliability - 0.95 over two years.

h) Control Method - valve drive electronics via telemetry.

j) TM Data - fuel blowdown pressure and temperature.

3.5 Attitude Control Subsystem (ACS)

The Attitude Control Subsystem shall be designed such that throughout the spin stabilized and three axis stabilized phase of the mission, the spacecraft attitude is completely determined and controlled. Attitude of the spacecraft, it is assumed, will be determined from data derived from on-board sensors and the attitude is controlled through ground command and on-board closed-loop control of spacecraft momentum. The spacecraft shall have spinning and non-spinning attitude sensors, rate sensors, sufficient momentum transfer between the spacecraft and either the external environment or internal rotating wheels to meet the requirements as specified.

a) Spin Stabilized Phase

During the spin stabilized phase of the mission, the attitude control system shall provide for determination of spin axis attitude through ground processing of data from spinning earth and sun sensors. The ACS shall provide, (in conjunction with the RCS), control of the spin axis precession manoeuvres, prior to and after apogee motor firing, providing an inclination accuracy of $\pm 0.2^{\circ}$ after apogee motor firing and control of the despin manoeuvre to 2 rpm which occurs during three axis acquisition.

b) Three Axis Stabilized Phase

During the 3-axis stabilized phase of the mission, the ACS provides an attitude acquisition mode of operation and a "normal pointing" mode of operation.

1. Attitude Acquisition Mode

i) The attitude control subsystem shall eliminate rotation rates of up to 0.1 rpm about all body axes.

ii) The ACS shall provide autonomously controlled rotation manoeuvres to orient the spacecraft into its normal earth pointing attitude.

2. Normal Pointing Mode

In the normal pointing mode the attitude control shall provide the following:

i) $\pm 0.15^{\circ}$ in roll and pitch control, ($\pm 0.1^{\circ}$ design goal).

ii) $\pm 1.0^{\circ}$ in yaw control.

This is provided during all phases of spacecraft operation, including eclipse, momentum dumping, and stationkeeping.

3.6 Apogee Motor Subsystem

General

The apogee motor subsystem shall include a solid propellant motor, a high expansion ratio nozzle, a pyrogenic ignitor and an electromechanical safe and arm device. The subsystem shall be capable of providing the required impulse to inject a 1,925 lb. spacecraft into synchronous orbit. The normal transfer orbit perigee altitude and inclination is 100 nautical miles and 28.5° respectively.

Initiation of the apogee motor will be by ground command at the specific time in the mission sequence.

Thermal, Structural and Safety Details

a) Motor plume and soak-back shall meet spacecraft thermal requirements.

b) Acceleration levels shall not exceed those specified for structural integrity of the spacecraft.

c) Provision shall be made for safe/arm ignitor inspection while the spacecraft is readied for launch.

d) The weight of the motor case and propellant is TBD.

e) Maximum pressurized diameter of the motor case is TBD.

3.7 Power Subsystem

Apart from the solar array, as described in paragraph 3.3 above, no design work is required on the power subsystem. Battery data and power conditioning details are provided in Subsystem Requirements as given in Attachment I, (Design Requirements).

3.8 Telemetry, Tracking and Command Subsystem (TT&C)

The TT&C subsystem is not part of this specification; however, subsystem requirements are given in Attachment I, (Design Requirements).

3.9 Communications Subsystems

The subsystem requirements given in Design Requirements for various antenna configurations will impact the spacecraft layout and the design of solar array, ACS and RCS, and close liaison with the Design Authority is required.

3.10 Mass Properties

Weight estimates shall be prepared for the spacecraft BUS subsystems, and moments of inertia calculated to ensure that inertia ratios are in a suitable range for stability during transfer and drift orbit spinning phases, i.e. MOI ratio $\geq$ 1.1.

3.11 Other Requirements

3.11.1 Structural Design Load

The structure shall meet the following design load requirements:

a) Steady State

1.25 x loads as calculated from launch vehicle boost accelerations, or combined apogee motor and spacecraft spin accelerations.

b) Vibration

Qualification vibration loads as defined in the Delta Restraints Manual for the 3914 vehicle, and as established by analysis.

3.11.2 Structural Safety Factors

The structural components of a spacecraft shall possess safety factors defined as follows:

a) Non-critical components: Components whose failure will not lead to catastrophic spacecraft failure.

Yield safety factor = 1.20
Ultimate safety factor = 1.50

b) Critical components: Components whose failure may lead to catastrophic spacecraft failure.

Yield safety factor = 1.20
Ultimate safety factor = 1.50

c) Special components: Components whose dimentions are critical for spacecraft alignment.

Yield safety factor = 1.20
Ultimate safety factor = 1.50

3.11.3 Maximum Static Loads

The maximum quasi-static loads/accelerations occur during main engine cut-off and qualification levels related to this are provided in the table below:

	Thrust Axis g Levels	Lateral Axis g Levels
Main Engine Cut-Off (MECO) accelerations	16 g	1 g
Lift-Off accelerations	3.91 g	3.81 g

In addition to the accelerations mentioned, spin-up at 60 rpm ±10% will induce a radial component of acceleration given by:

$$g \text{ radial} = \frac{\left(R \frac{\pi}{30} \times \text{rpm}\right)^2}{386}$$

which, for a 60 rpm spin rate, becomes:

$$g \text{ radial} = .102 \times R \text{ g where R is in inches}$$

Margins of safety when applied to the above loads and calculated on the basis of combined stresses, including any stress concentrations that cannot be dissipated by local yielding, are equal to or greater than 0.

4.0 SPACECRAFT ENVIRONMENTAL DESIGN LEVELS

This section describes a history of each environment throughout the mission periods and certain environments applicable during testing as listed below:

a) Thermal and thermal vacuum,
b) Pressure altitude,
c) Shock,
d) Vibration,
e) Acoustics,
f) Acceleration,
g) Radio frequency interference,
h) Magnetics,
j) Radiation,
k) Meteoroid damage,
l) Humidity,
m) Contaminates.

The flight level environments have been established from data attained from Delta Restraints Manual which has been derived from previous Delta flights. Where possible, three sigma statistical values based on a standard distribution can be made available. However, due to lack of data, certain environments have been considered as being only to the two sigma level. These are shock, random vibration and acoustics. All other environments in this section are to the three sigma probability level.

Figure 2.1 shows a typical time history starting at launch of the flight level shock vibration acceleration and acoustics environment. It is presented to give an overall view of the dynamics and other events conveyed into actions. Details provided later in figures 2.3 to 2.5 are used to determine when the analysis dynamic load should be superimposed to compute design load factors.

4.1 Thermal Environment (Temperature, Radiation & Humidity)

a) Ground Period, Ground Transportation

The spacecraft and its equipment is protected from any exposure beyond that called for in the relevant test requirements. This is achieved by controlling the surrounding ambient air to 23°C ±5°C in relative humidity to less than 50% throughout all ground period operations.

b) Launch Period

The spacecraft will be subjected to radiant heating from the fairing during ascent, details are provided in Figure 2.2.

c) Apogee Motor Burning Period

Maximum motor case temperature shall not be greater than 700°F.

d) Drift Period

The spacecraft is designed to withstand the thermal vacuum requirements listed below:

Thermal Vacuum Requirements

During transfer orbit the spacecraft is subjected to the following:

i)	Eclipse period:	30 minutes
ii)	Vacuum level :	greater than 10^{-7} torr
iii)	Heat flux :	same as synchronous orbit

During drift and synchronous orbit the spacecraft is subjected to:

i)	Eclipse period:	72 minutes
ii)	Vacuum level :	greater than 10^{-7} torr
iii)	Solar flux :	Max. 450 BTU/hr.ft.^{2}
		Min. 408 BTU/hr.ft.^{2}

4.2 Shock Environment

a) Ground and Pre-Launch Periods

Designed to withstand handling shocks in accordance with MIL-STD-810B.

b) Launch Period

The spin table and third stage motor act to filter out any high frequency shocks from fairing separation and other major launch flight events.

c) Spin-up Separation and Transfer Orbit Periods

Maximum shock impulse 1400 g, 0.3 ms duration, (see Fig. 2.3).

4.3 Vibration

The spacecraft shall be designed to withstand the following vibration spectra and levels.

a) Ground and Pre-Launch Period

MIL-810B.

b) Launch Period

During the power launch period the spacecraft will be subjected to sinusoidal, random, acoustic and shock excitations as follows:

i) Sinusoidal Vibration

a) Z (Thrust Axis)

Frequency	g Levels	Duration Secs.
5-15	1.5	2-3
15-21	4.5	20
21-100 Hz	1.5	20

b) X-Y (Lateral Axis)

Frequency	g Levels	Duration Secs.
5-14	1.3	2-3
14-100	1.0	
250-400 Hz	4.5	
400-2000 Hz	7.5	

ii) Random Vibration Spectra

A duration of one minute at levels given in the paragraph below has been used to simulate the flight excitation, at the levels shown in Figure 2.4:

Frequency in Hz	Power Spectral Density G^2 per Hz	g RMS Approx.
20-300	+ 3db per octave	6.1
300-2000	.02	6.1

iii) Acoustic Environment

This is shown in Figure 2.5 and the table below:

Acoustic Flight Levels

Frequency in Hz	Sound Pressure Levels in db (Reference .02 dynes c^2)
37.5-75	126
75-150	131
150-300	134
300-600	136
600-1200	137
1200-2400	134
2400-4800	130
4800-9600	125

The overall summation, 142 db.

4.4 Natural Radiation Environment

This section describes the radiation environment dissipated and the information on high energy particle fluxes corresponds to the 1975-1977 time period. The average fluxes specified will be significantly different for longer periods or for missions with a different start date. Details associated with the above are contained in Tables 4.1, 4.2, 4.3, 4.4, and 4.5, and cover:

a) Solar Radiation Intensity Levels,
b) Earth Albedo Radiation,
c) Spectral Distribution,
d) Earth Emitted Radiation Intensity Levels,
e) Earth Emitted Radiation Spectral Distribution.

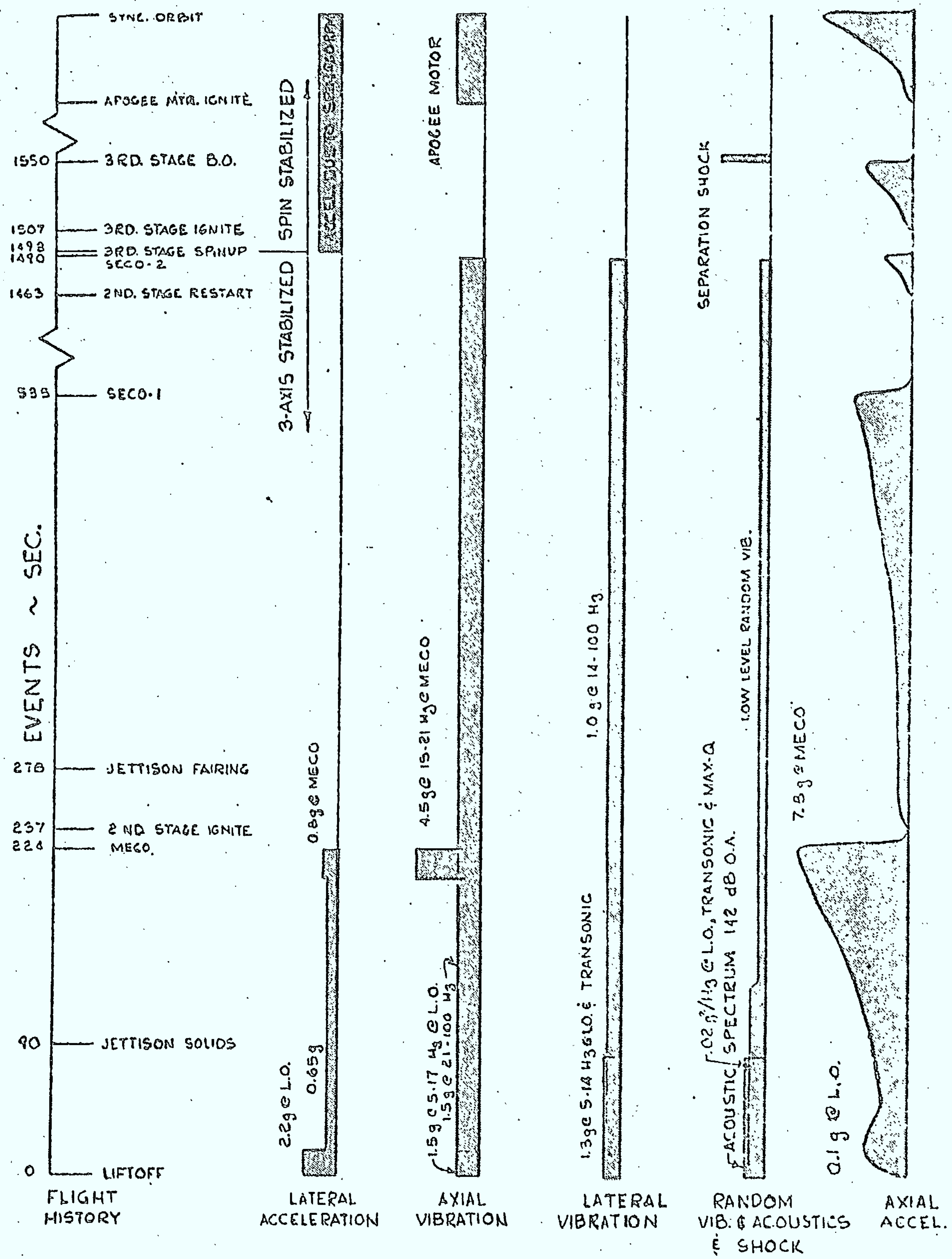
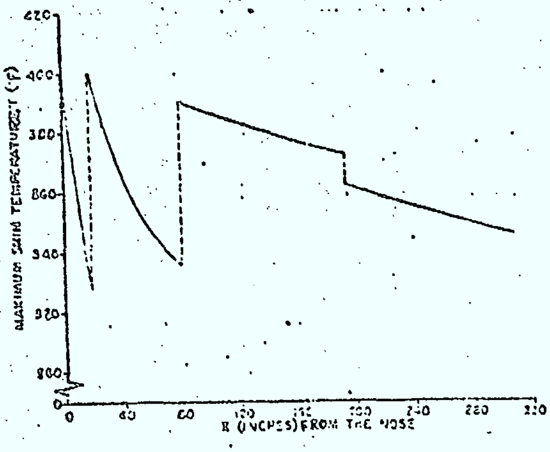
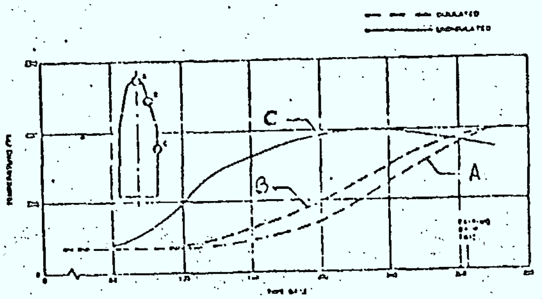
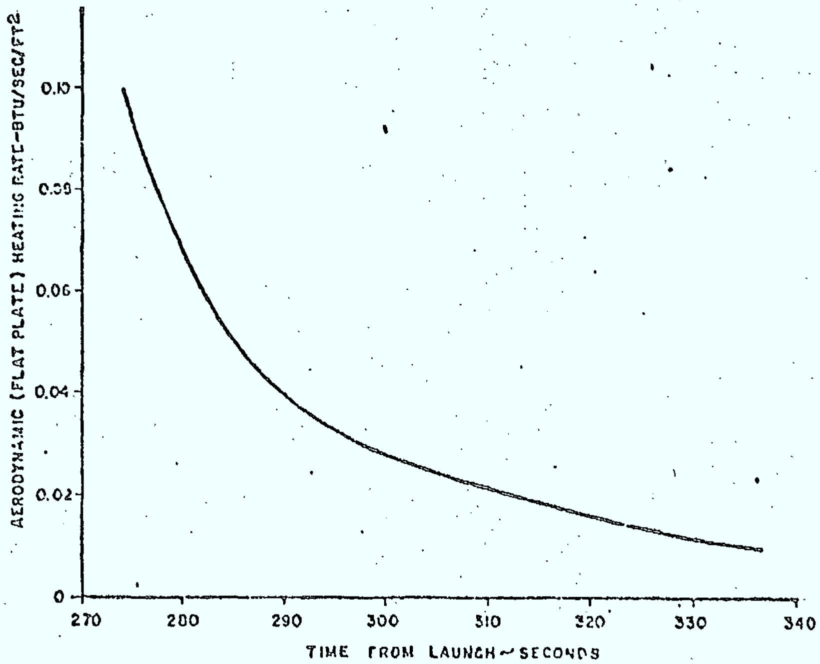


FIGURE 2.1 MECHANICAL ENVIRONMENT TIME HISTORY



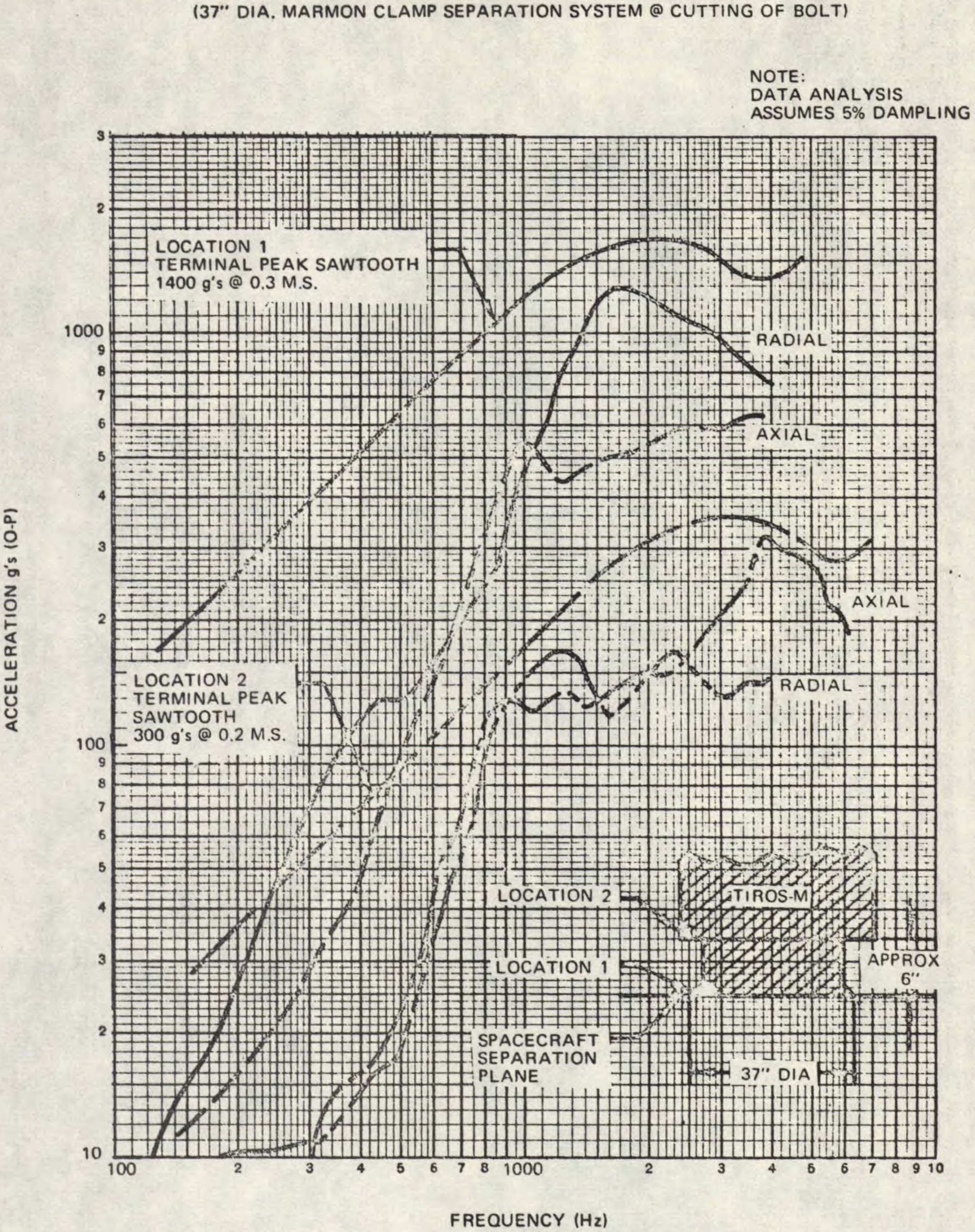


Spacecraft Fairing Internal Temperature vs Time From Liftoff



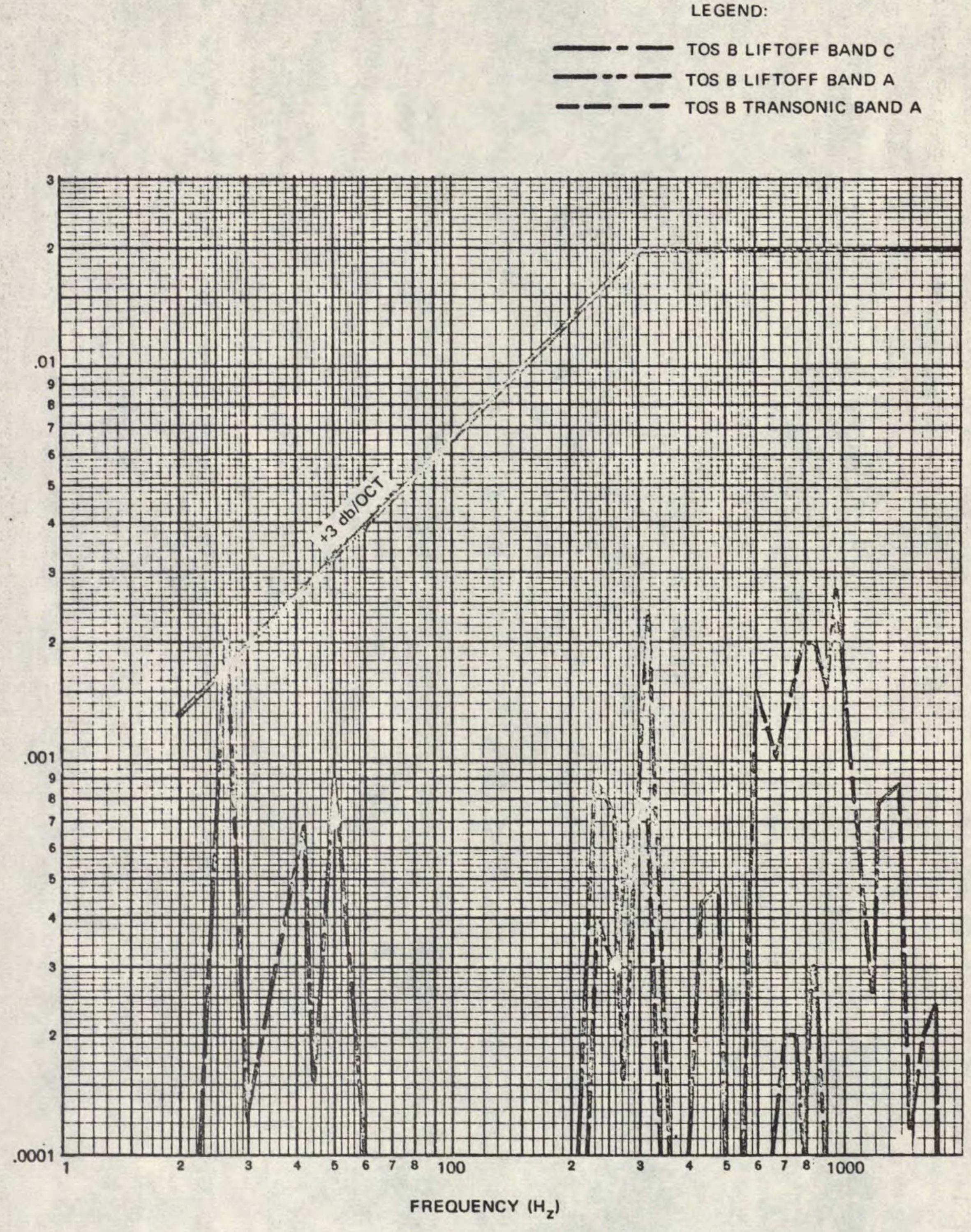
AERODYNAMIC HEATING RATE DECAY WITH TIME, DELTA 2914 LAUNCH VEHICLE

FIGURE 2.2



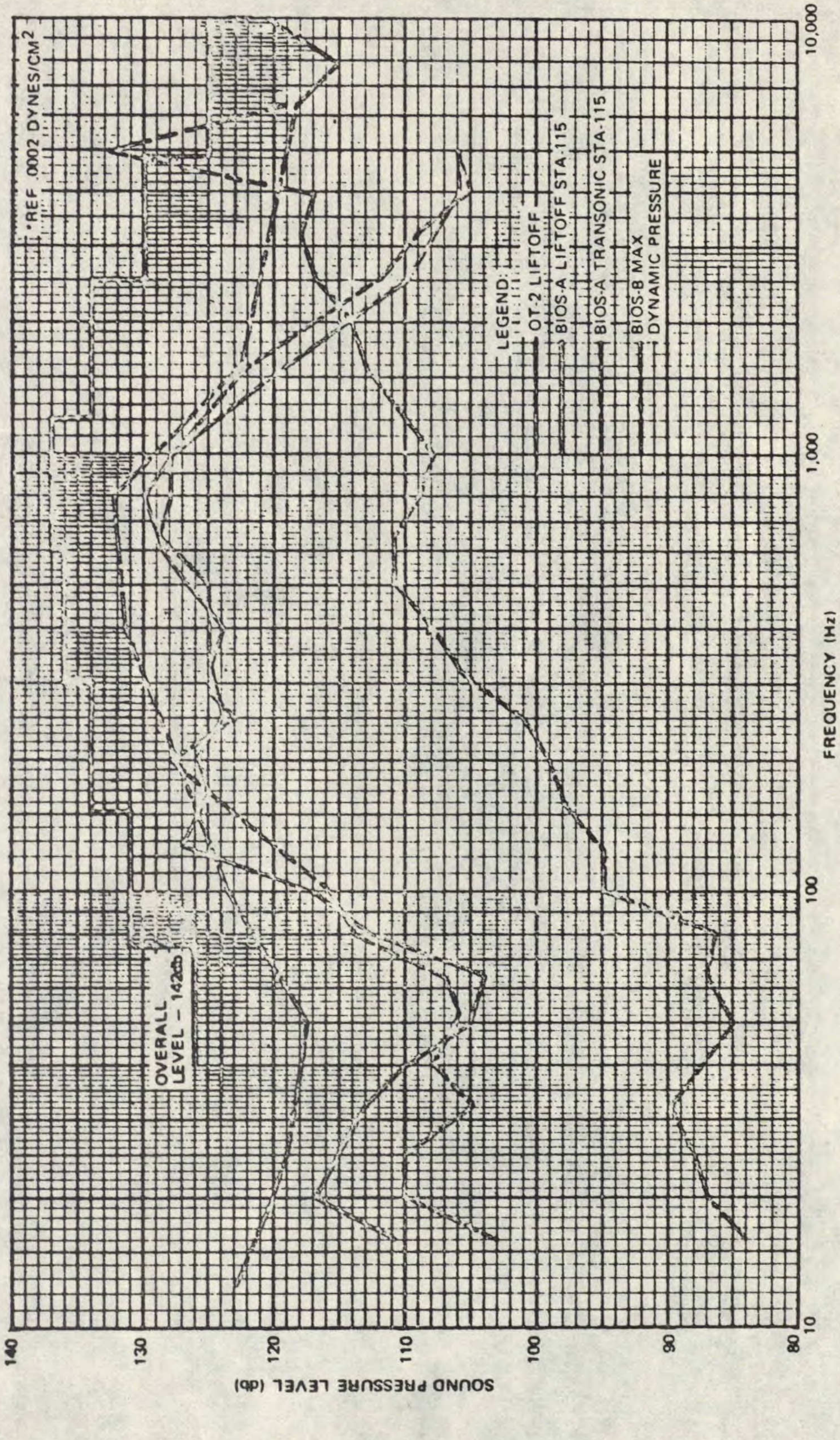
Shock Spectrum – TIROS-M Shock Test

FIGURE 2.3



Thrust & Lateral Axis Random Flight Levels – Delta Three Stage Vehicles

FIGURE 2.4



Delta Vehicles – Acoustic Noise Flight Levels

FIGURE 2.5

TABLE 4.1

SOLAR RADIATION INTENSITY LEVELS W/cm^2

Case	Boost Phase to Transfer Orbit	Synchronous Orbit & Drift		
		Winter	Summer	Equinox
Maximum	.1420	.1420	.1327	.1373
Minimum	.1288	.1378	.1288	.1333
Nominal	.1351	.1407	.1305	.1351

TABLE 4.2

EARTH ALBEDO RADIATION W/cm^2

Case	From Earth Surface with Normal Illumination	Transfer Orbit	Synchronous Orbit
Maximum	0.1148	0.0714	0.0075
Minimum	0.0122	0	0
Nominal	0.0486	0	0

TABLE 4.3

SPECTRAL DISTRIBUTION

Wavelength Band (Micron)	% of Total Energy	Energy in Band (W/cm^2)
0.3 - 0.4	6.1	0.003
0.4 - 0.7	42.8	0.021
0.7 - 1.3	36.8	0.018
1.3 - 2.7	14.3	0.007

TABLE 4.4

EARTH EMITTED RADIATION INTENSITY LEVELS W/cm^2

Case	100 Naut. M Park	Transfer Orbit	Synchronous Orbit
Maximum	0.0215	0.0215	0.00052
Minimum	0.0195	0.0047	0.00047
Nominal	0.0204	0.0070	0.00049

TABLE 4.5

EARTH EMITTED RADIATION SPECTRAL DISTRIBUTION

Wavelength Band (Micron)	% of Total Energy	Energy in Band (W/cm^2)
5.0 - 8.3	8.0	0.0018
8.3 - 12.5	27.5	0.0062
12.5 - 20.0	31.0	0.0071
20.0 - 60.0	28.0	0.0064

4.5 Time-Averaged Integral Flux of High Energy Protons and Electrons Encountered During Transfer Orbit

The high energy particle flux for protons and electrons encountered in transfer orbit are listed in the table below:

TYPE OF PARTICLE	ENERGY RANGE	AVERAGE INTEGRAL PARTICLE FLUX	
TRAPPED	4.0-E-14.5	(E) = 1.87 x 10 exp (-.302E)	
PROTONS	14.5-E-50	(E) = 6.90 x 10 exp (-.0742E)	protons
	50-E-100	(E) = 4.30 x 10 exp (-.0194E)	cm^{2}orbit
TRAPPED	0.3-E-1.35	(E) = 1.06 x 10 exp (-2.64E)	electrons
ELECTRONS	1.35-E-40	(E) = 4.00 x 10 exp (-1.94E)	cm^{2}orbit

4.6 Electromagnetic Compatibility Specification for Subsystems

SPAR-SG.253, EMC Specification for Solar Array Subsystem Components, may be used as a good definition for all sub-systems on this spacecraft. Also forming part of this specification are the following documents:

MIL-STD-462A, August 1, 1968	Electromagnetic Interference Characteristics, Measurement of
MSFC-SPEC-279, Nov. 15, 1967	George C. Marshall Space Flight Center National
MIL-STD-831, August 28, 1963	Test Reports, Preparation of

4.7 Testing Requirements

The spacecraft will be tested to the following environments and to the qualification levels listed in the following tables:

- Sinusoidal Vibration
- Random Vibration
- Shock
- Acceleration
- Thermal Vacuum
- EMC
- RFI (as defined by the experimenter)

4.7.1 Sinusoidal Qualification Vibration Levels

a) Z (Thrust Axis):

Frequency	g Levels
5-10 Hz	.4
10-15 Hz	2.3
15-21 Hz	6.8
21-40 Hz	2.3
40-70 Hz	.83 (notched)
70-250 Hz	2.3
250-400 Hz	4.5
400-2000 Hz	7.5

b) X-Y (Lateral Axis)

Frequency	g Levels
5-10 Hz	.4
10-12 Hz	2.0
12-25 Hz	.35 (notched)
25-250 Hz	1.5
250-400 Hz	4.5
400-2000 Hz	7.5

It should be noted that the 40-70 Hz region in the thrust axis, the spacecraft is notched down to a .83 level and this has been accepted by the NASA Delta office.

4.7.2 Random Vibrations

Test to levels defined in Section 2.5.4 (ii) and Figure 2.4.

4.7.3 Shock

Shock levels shall be in accordance with Section 2.5.3 and Figure 2.3.2.

Alternative (i) to the above is to carry out shock test by firing the third stage/spacecraft attach clamp separation bolts.

Alternative (ii) for the complete spacecraft is to perform shock vibration to the following spectrum:

Frequency Hz	Accelerations g's (O-P)	Sweep Rate
100-250	(1.5 lateral) (2.3 thrust)	
250-400	4.5	2 oct/ cm
400-2000	7.5	

ATTACHMENT A

DESIGN REQUIREMENTS

A.1 COMMUNICATIONS PAYLOAD

The following sections describe the major design characteristics of the five payload options.

A.1.1 Model (a) Payload Parameters: UHF/4-6 GHz Transponder

i) Capacity

Number of 4-6 GHZ channels	
in sunlight	12 (UHF Backhaul)
in eclipse	10 (UHF Backhaul)
Target number of UHF EIRP	
units in sunlight	50
in eclipse	25

ii) Antenna

EOC UHF Antenna Gain	19 db
4-6 GHz Coverage	4 X 8° beam ≁ spot beam
Antenna Type	Deployable Parabola
Antenna Size	13 ft. diameter
Antenna Weight	55 lb.

iii) Weight

Weight of UHF Transponder ≁ 4-6 GHz Backhaul	138.0 lb.
Weight of 11 4-6 GHZ channels	80.4 lb.
Total	218.4 lb.

D4/CK/1

iv) Power DC (assuming UHF HPA efficiency of 56%) watts

	Sunlight	Eclipse
DC Power to UHF / 4-6 GHz backhaul	224.0	165.7
DC Power to 4-6 GHZ channels	190.9	173.6
Total	414.9	339.3

v) Power Dissipated watts

In Sunlight	314.9
In Eclipse	264.3

A.1.2 Model (b) Payload Parameters: UHF/12-14 GHz Transponder

i) Capacity

Number of 12-14 GHZ Channels	
in Sunlight	4 (UHF backhaul)
in Eclipse	3 (UHF backhaul)
Target Number of UHF EIRP	
Units in Sunlight	85
in Eclipse	40

ii) Antenna

EOC UHF Antenna Gain	19 db
12-14 GHz Coverage	Spot Beam 4 X 8° Beam
Antenna Type	Deployable Parabola
Antenna Size	13 ft. diameter
Antenna Weight	55 lb.

iii) Weight

Weight of UHF Transponder 12-14 GHz backhaul	142.5 lb.
Weight of Four 12-14 GHz Channels	38.0 lb.
Total	180.5 lb.

D4/CK/2

iv) DC Power (assuming UHF PA efficiency of 56%) watts

	Sunlight	Eclipse
DC Power to UHF 12-14 GHz backhaul	376.6	277.5
DC Power to 12-14 GHz Channels	248.0	186.0
Total	624.5	462.5

v) Dissipation watts

In Sunlight	456.6
In Eclipse	351.5

A.1.3 Model (c) Payload Parameters: UHF/7-8 Transponder ≠ Experiments

i) Capacity

Experimental payload (operated only in sun-light)

Target UHF EIRP Units

In Sunlight	120
In Eclipse	60

One 7-8 GHz channel for UHF backhaul (operating in both sunlight and eclipse).

ii) Antenna

EOC UHF Antenna Gain	19 db
7-8 GHZ Coverage	4 X 8° spot beam
Antenna Type	Deployable Parabola
Antenna Size	13 ft. diameter
Antenna Weight	55 lb.

D4/CK/3

iii) Weight

Weight of UHF Transponder 7-8 GHz backhaul	145.9 lb.
Magnetometer (on 10 ft. boom)	15.0 lb.
Cosmic Ray Detector	15.0 lb.
Atmospheric Constituent Monitor	30.0 lb.
Contingency for Experimental Payload	15.0 lb.
Total	221.9 lb.

iv) DC Power (assuming UHF HPA efficiency of 56%) watts

	Sunlight	Eclipse
DC power to UHF ≠ 7-8 GHz backhaul	445.20	305.3
DC power to experimental payload (10 ≠ 10 ≠ 30 ≠ 10 contingency)	60.0	-
Total	505.20	305.3

v) Dissipation

In Sunlight	404.2
In Eclipse	252.3

A.1.4 Model (d) Payload Parameters: UHF/7-8 GHz/L-Band

i) Capacity

One Aerosat compatible channel (both sunlight and eclipse)

Four Marisat compatible channels (sunlight and eclipse)

D4/CK/4

One 7-8 GHz backhaul channel for UHF (sunlight and eclipse)

Target UHF Capacity EIRP	
Units: Sunlight	120
Eclipse	60

ii) Antenna

EOC UHF Antenna Gain	19 db
7-8 GHz Coverage	4 X 8° Beam Spot Beam
L-Band Coverage	4 X 8°
Antenna Type	Deployable Parabola
Antenna Weight	60 lb.

iii) Weight

Weight ~~of~~ UHF/7-8 GHz Transponder	146.9 lb.
Weight of L-Band Transponder	43.5 lb.
Total	190.4 lb.

iv) DC Power (assuming UHF HPA efficiency of 56%) watts

	Sunlight	Eclipse
DC Power to UHF/7-8 GHz	445.2	305.3
DC Power to L-Band Transponder	68.0	68.0
Total	513.2	373.3

v) Dissipation watts

In Sunlight	397.20
In Eclipse	305.30

D4/CK/5

ATTACHMENT B

B.1 Model (a) PAYLOAD: UHF PAYLOAD DISSIPATIONS

Weights

i)	Total UHF/4-6 GHz transponder weight (including antennas)	218.4 lb.
ii)	Weight of UHF HPA	6.4 lb.

Dissipations (watts)

		Sunlight	Eclipse
i)	DC power to complete transponder	414.9	339.3
ii)	DC power 4-6 GHz portion	190.9	173.6
iii)	DC power to VHF portion	224.0	165.7
iv)	DC power to UHF HPA	158.0	99.7
v)	Power radiated at UHF	39.5	19.8
vi)	Power dissipated in UHF PA and related components	118.5	79.9
vii)	Breakdown of Dissipations in UHF HPA Related Components		
	EPC Units	17.1	10.8
	UHF Driver	41.0	25.8
	UHF HPA Module	44.1	35.1
	Isolator	3.7	1.6
	UHF Switch	2.6	1.3
	UHF Output Filter	5.7	2.9
	Cabling to Antenna	4.8	2.4
	Totals	118.5	79.9

D4/CK/6

B.2 MODEL (b) PAYLOAD: UHF PAYLOAD DISSIPATIONS

Weights

i)	Total transponder weight (including antenna)	180.5 lb.
ii)	Total UHF HPA Weight	19.9 lb.

Dissipations (watts)

		Sunlight	Eclipse
i)	DC power to complete transponder	624.6	463.5
ii)	DC power to 12-14 GHz portion	248.0	186.0
iii)	DC power to UHF portion (including backhaul)	376.6	277.5
iv)	DC power to UHF HPA	268.6	169.5
v)	Power radiated at UHF	67.2	33.6
vi)	Breakdown of Dissipation in UHF HPA Related Components		
	EPC Unit	29.1	18.3
	UHF Driver	69.6	43.8
	UHF HPA Module	75.0	59.9
	Isolator	5.3	2.7
	UHF Switch	4.4	2.2
	UHF Output Filter	9.8	4.9
	Cabling to Antenna	8.2	4.1
	Total	201.4	135.9

B.3 MODELS (c) AND (d) PAYLOAD: UHF PAYLOAD DISSIPATIONS

Weights

i)	Total of UHF/7-8 GHZ transponder (including antenna)	146.9 lb.
ii)	Weight of UHF HPA	15.3 lb.

D$/CK/7

Dissipations (watts)

		Sunlight	Eclipse
i)	DC Power to Transponder	445.2	305.3
ii)	DC Power Supplied to Transponder Components Except UHF HPA	66.0	66.0
iii)	Power Supplied to UHF HPA	379.2	239.0
iv)	Power Radiated at UHF	94.8	47.4
v)	Power Dissipated in UHF PA and Related Components	284.4	191.6
vi)	Breakdown of Dissipation in UHF PA Related Components:		
	EPC Unit	41.0	25.9
	UHF Driver	98.3	61.9
	UHF PA Module	106.0	84.3
	Isolator	7.5	3.7
	UHF Switch	6.3	3.1
	UHF Output Filter	13.7	6.9
	Cabling to Antenna	11.6	5.8
	Total	284.4	191.6

D4/CK/8

1.2 Attitude Control Subsystem Performance Specification

APPENDIX B

PERFORMANCE SPECIFICATION

FOR AN

ATTITUDE CONTROL SUBSYSTEM

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1.0 SCOPE

1.1 General

This Specification establishes the requirements for the performance of a satellite Attitude Control Subsystem, hereinafter referred to as the ACS. The ACS will form a subsystem of the Multi-Purpose Bus (MPB) Satellite to be used in a geostationary equatorial orbit with several possible compliments of communication payloads.

1.2 Function

The ACS shall provide functions required on-board for the stabilization and control of spacecraft attitude throughout the transfer orbit, apogee injection, attitude acquisition, station acquisition, stationkeeping and on-orbit operation phases. During these phases two basic spacecraft configurations are employed. Following separation from the launch vehicle third stage, the spacecraft will be spin stabilized, with favourable moment of inertia ratios, until the start of the attitude acquisition phase. During the attitude acquisition phase the spacecraft will be transferred from this spinning configuration to a non-spinning three axis stabilized configuration, which will be maintained for the duration of on-orbit operation.

The primary functions of the ACS during these various phases are:

a) To provide sufficient attitude information and access to attitude control actuation so that ground control can implement attitude stabilization and control of the spacecraft during the transfer orbit and apogee injection.

b) To provide autonomous stabilization and control during the attitude acquisition phase with a minimum of ground control support to enable the spacecraft to be transferred from the spin stabilized configuration to a three axis stabilized configuration with the solar arrays deployed.

c) To provide autonomous three-axis attitude stabilization and control during the on-orbit operation phase, with a minimum of ground support.

d) To provide autonomous three-axis attitude stabilization and control during orbit adjust operations.

e) To provide autonomous sun pointing control for the solar array subsystem.

2.0 APPLICABLE DOCUMENTS

TBD.

3.0 REQUIREMENTS

3.1 Performance

3.1.1 Functional Characteristics

3.1.1.1 Spinning Phase

3.1.1.1.1 Spinning Phase Operation

During the transfer orbit and apogee injection, the spacecraft will be spinning about the yaw axis. The spacecraft angular momentum vector is nominally in the direction of the positive yaw axis. Following injection into transfer orbit, the spin axis attitude will be determined and the spacecraft precessed into the appropriate apogee motor firing attitude.

3.1.1.1.2 Spinning Phase Support Function

In order to enable appropriate attitude stabilization and control during the spinning phases, the ACS will provide ground control with the following support functions:

a) A sun reference pulse when the sun passes through the spacecraft roll-yaw plane, for spin rate monitoring.

b) The sun line position relative to the spacecraft spin axis.

c) The earth line position relative to the spacecraft spin axis.

d) Timing and execution of trains of axial high thrust engine pulses (of specified number and duty cycle) on command for spin axis orientation control. The sun reference pulse will be used for timing individual pulses.

e) Passive damping of spin axis nutation.

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3.1.1.1.3 Spin Rate

The ACS shall meet the performance requirement specified herein with a spacecraft spin rate of 60 $\pm$ 6 RPM about the yaw axis.

3.1.1.1.4 Attitude and Earth Radius

The ACS shall meet the performance requirements specified herein at an altitude of 19,323 ±10 nautical miles above the earth's surface whose radius is 3,444 nautical miles.

3.1.1.1.5 Sun Reference Pulse Accuracy

The three sigma angular error between the sun reference pulse and the sun trailing edge shall be $\pm$ 0.12° when the sun line lies in the roll-pitch plane.

3.1.1.1.6 Sun Elevation Accuracy

The three sigma sun elevation accuracy shall be less than $\pm$ 0.27 degrees.

3.1.1.1.7 Earth Chord Length Accuracy

The three sigma earth chord length accuracy shall be $\pm$ 0.12 degrees.

3.1.1.1.8 Axial Thruster Pulse Timing Accuracy

Thruster electrical pulse centroid and pulse width accuracy shall be equivalent to $\pm$ 1.0° three sigma or less.

3.1.1.1.9 Nutation Damper Time Constant

The nutation damper time constant shall not exceed 30 minutes.

3.1.1.2 Attitude Acquisition Phase

The attitude acquisition phase extends from the initiation of despin to the transfer of control to the on-orbit three axis control system.

3.1.1.2.1 Attitude Acquisition Operation

The attitude acquisition operation will include a sequence of manoeuvres performed to obtain the following orientation of the reference axes:

a) Yaw axis pointing at the geometric centre of the earth.

b) Pitch axis pointing along the southerly orbit normal.

c) Roll axis pointing along the tangential orbit velocity vector.

The attitude acquisition operations will include the following manoeuvres:

a) Despin
b) Sun acquisition
c) Solar array deployment
d) Earth acquisition
e) Transfer to on-orbit three-axis control.

The attitude acquisition manoeuvres will be performed autonomously upon initiation by ground command.

3.1.1.2.2 Acquisition Duration

The total time to complete the acquisition manoeuvres shall not exceed 3.5 hours.

3.1.1.2.3 Launch Window Constraint

There shall be no time of year limitation on launch (launch on any day of the year) due to the chosen attitude acquisition control system.

3.1.1.3 Station Acquisition

The ACS shall provide three-axis attitude control during longitude station acquisition where east-west stationkeeping thrusters are used.

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3.1.1.4 Sun Hold Station Acquisition

In the event that earth acquisition cannot occur from the initial longitude station, the ACS will maintain control of the sun line along the spacecraft roll axis to allow a station change with east-west stationkeeping thrusters.

3.1.1.5 On-Orbit Three Axis Attitude Control

Following station acquisition the ACS will provide autonomous three-axis attitude control.

3.1.1.5.1 Attitude Error Budget

The spacecraft boresight error budget will consist of the following three sigma error contribution:

a) Roll and Pitch

i)	Antenna boresight alignment	$\pm 0.05°$
ii)	Spacecraft thermal and vibration induced distortion	$\pm 0.02°$
iii)	Attitude sensor alignment	$\pm 0.005°$
iv)	ACS control error	$\pm 0.14°$
	RSS Total	$\pm 0.15°$

b) Yaw

i)	Antenna boresight alignment	$\pm 0.15°$
ii)	Spacecraft thermal and vibration induced distortion	$\pm 0.1°$
iii)	ACS component alignment	$\pm 0.1°$
iv)	ACS Control Error	$\pm 1.08°$
	RSS Total	1.1°

3.1.1.5.2 Attitude Control System Pointing Error

The attitude control system pointing error shall be ± 0.14° in roll and pitch and ± 1.08° in yaw, where these errors include sensor noise, bias and dynamic control errors, but exclude alignment errors, thermal and vibration induced distortions and antenna boresight alignment.

As a design goal, during periods of no orbit adjust thruster activity, the ACS pointing error in roll and pitch shall be ± 0.1°.

3.1.1.5.3 Attitude Rate Error

In order to accommodate attitude rate sensitive payloads, the attitude rates shall not exceed the following three sigma limits during periods of no-orbit adjust thrusting.

a) Roll and pitch TBD deg/sec
b) Yaw TBD deg/sec

3.1.1.6 Station Keeping

3.1.1.6.1 Station Keeping Operation

The ACS shall provide autonomous three-axis control during orbit adjust thrusting.

3.1.1.6.2 Station Keeping Support Function

The ACS shall provide timing and execution of station keeping low thrust engines (of specified thrust duration) upon initiation by ground command.

3.1.1.6.3 Station Keeping Limits

The RSS sum of the east-west and north/south stationkeeping limits will be:

$$\sqrt{(0.1)^2 + (0.1)^2} = 0.1414 \quad \text{degrees.}$$

As a design goal, the RSS sum will be

$$\sqrt{(0.05)^2 + (0.05)^2} = 0.0707 \quad \text{degrees}$$

3.1.1.6.4 Station Keeping Cycle

As a design goal, the ACS shall provide three-axis control with a 21 day stationkeeping cycle.

3.1.1.7 Solar Array Tracking

The ACS shall provide solar array sun pointing control to within an azimuth error of $\pm$ 1.0 degree.

3.1.1.8 Station Change Requirement

The ACS shall be capable of performing a pitch slew manoeuvre, on command, in order to maintain three-axis attitude control during and following a change in longitudinal station.

3.1.1.9 Spacecraft Dynamic Environment

The ACS shall be capable of meeting the performance requirements of this Specification with the dynamic environments defined in the subparagraphs below.

3.1.1.9.1 Mass Properties

3.1.1.9.1.1 Spin Phase Mass Properties

During the mission spinning phase the roll, pitch and yaw moments of inertia will be respectively:

$I_{xx} = 155$ slug ft^2

$I_{yy} = 175$ slug ft^2

$I_{zz} = 182$ slug ft^2

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3.1.1.9.1.2 On-Orbit Orbit Phase Mass Properties

In the deployed configuration the roll, pitch and yaw moments of inertia will be respectively:

$I_{xx} = 800$ slug ft^2

$I_{yy} = 250$ slug ft^2

$I_{zz} = 830$ slug ft^2

3.1.1.9.1.3 Solar Array Mass Properties

The total weight of the deployable portion of the solar arrays will be 90 lb. The distance between the solar array centre of mass, which will lie on the yaw axis, will be 15 inches. The moments of inertia of the deployable portion of the solar array subsystem about its centre of mass will be:

$I_{xx} = 650$ slug ft^2

$I_{yy} = 4$ slug ft^2

$I_{zz} = 650$ slug ft^2

3.1.1.9.2 Flexible Appendage Properties

The spacecraft will include flexible solar arrays mounted in a north-south configuration, and flexible antennas with a boresight contained within the yaw pitch plane and canted towards the north by up to 5.0 degrees.

3.1.1.9.2.1 Analytic Modelling of Flexible Appendages

An "unconstrained" modal decomposition of the flexible appendages may be assumed, including equivalent viscous damping. Only the first three flexible modes need be considered for analysis.

3.1.1.9.2.2 Solar Array Flexible Parameter

The solar array natural frequencies and gains will be as follows:

TABLE 1

SOLAR ARRAY FREQUENCIES AND GAINS

Mode	Range of Frequencies (Hz)	Range of Gains
First out-of-plane	0.1 - 0.4	4.0 - 8.0
Second out-of-plane	TBD	TBD
Third out-of-plane	TBD	TBD
First Torsional	0.1 - 0.4	0.01 - 0.02
Second Torsional	TBD	TBD
Third Torsional	TBD	TBD

3.1.1.9.2.3 Antenna Flexible Parameters

The antenna natural frequencies and gains will be as follows:

TABLE 2

ANTENNA FREQUENCIES AND GAINS

Mode	Range of Frequencies (Hz)	Range of Gains
First Bending	0.5 - 1.0	TBD
Second Bending	TBD	TBD
Third Bending	TBD	TBD
First Torsional	TBD	TBD
Second Torsional	TBD	TBD
Third Torsional	TBD	TBD

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3.1.1.9.2.4 Flexible Damping

The equivalent viscous damping ratio for all flexible modes will be greater than 0.05 percent.

3.1.1.9.3 Solar Perturbation Torque

The major perturbation torque will be due to solar radiation pressure and will have the following form:

a) Pitch Torque (micro ft. lb.)

$$T_p = 22 \cos^2\theta_D \cos(\omega_o t)$$

b) Roll Torque (micro ft.lb.)

$$T_R = 8\sin(2\theta_D) + [3\cos^2\theta_D + 3\sin(2\theta_D)]\cos(\omega_o t)$$

c) Yaw Torque (micro ft.lb.)

$$T_y = -[3\cos^2\theta_D + 3\sin(2\theta_D)]\sin(\omega_o t)$$

The variables θ_D, ω_o and t are defined as:

θ_D = sun declination angle

ω_o = orbit rate (7.29×10^{-5} rad/sec)

t = time (seconds)

3.1.1.10 Reaction Control Subsystem Interface

3.1.1.10.1 General

The Reaction Control Subsystem (RCS) High Thrust Engines (HTE) and Low Thrust Engines (LTE) will provide the means whereby external torques may be applied to the spacecraft as required by the ACS.

3.1.1.10.2 RCS LTE Configuration

The RCS LTE are grouped into two sets - primary and redundant. Each LTE is identified by resultant torque produced in the case of the attitude control LTE and by thrust direction in the case of the stationkeeping LTE (summarized in Table 3).

3.1.1.10.3 Moment Arms and Torque Polarity

The moment arm and torque polarity associated with each LTE is listed in Table 3.

TABLE 3

LTE THRUSTER IDENTIFICATION, MOMENT ARM AND TORQUE POLARITY

TBD

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3.1.1.10.4 RCS Performance Parameters

The RCS LTE performance is defined as follows:

a) LTE Force - Steady State

The range of steady state thrust levels is TBD.

b) LTE Impulse Bit

The impulse bit per LTE during pulse operation will be TBD.

c) LTE Impulse Bit Repeatability

The repeatability from pulse to pulse for a given electrical on-time will be TBD for any LTE. Predictability between LTE's will be TBD.

3.1.2 Operability

3.1.2.1 Reliability

The subsystem will be designed to achieve a mission reliability not less than 0.9999 for 110 hours in transfer orbit, and as a design goal 0.89 during on-orbit operation with the environmental and life requirements detailed in this Specification. In order to achieve the on-orbit reliability, a high level of semi-conductor screening will be required.

3.1.2.2 Useful Life

The ACS will be capable of performing as specified for not less than six years in space following a maximum two years of prelaunch test and storage.

It will be a design goal to provide hardware capable of eight years life in space.

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1.3 Reaction Control Subsystem Performance Specification and Statement of Work

April 17, 1975

REFERENCE : SPAR RFP #5427
REACTION CONTROL SUBSYSTEM

Gentlemen,

You are invited to submit a budgetary proposal (and delivery date ARO) for a Reaction Control Sub-System (RCS) for a proposed Multi Purpose Satellite. This satellite is a 3 axis stabilized geo-synchronous vehicle with an eight year orbital life.

In preparing your proposal (of which 3 copies are required), you are to respond to the requirements stated in this letter and in the applicable documents which are listed below :

The quantities required are as follows :

ONE	DYNAMIC AND THERMAL MODEL (DTM)
ONE	ENGINEERING MODEL WHICH WILL BECOME A QUALIFICATION MODEL FOLLOWING NECESSARY REWORK
TWO	FLIGHT QUALITY MODELS
ONE	FLIGHT QUALITY SPARE MODEL. Please refer to the SOW. for a more detailed description of each model.

Documents applicable to this proposal are enclosed and consist of :

1. SPAR SOW. 071, Multipurpose Bus Study, Statement of Work, Reaction Control Subsystem, Preliminary, For Budgetary Estimates, 15 April, 1975.

2. SPAR-SG.350, Multipurpose Bus Study Specification,

Requirements, Reaction Control Subsystem, Preliminary, For Budgetary Estimates, 15 April, 1975.

3. SPAR Drawing 31138J1, Sheets 1 and 2, Structure and RCS, General Purpose Satellite Bus.

The Specification has been written to allow either a monopropellant hydrazine or bipropellant MMH/N_2O_4 Subsystem. Although the suggested mechanical schematic shows a blowdown monopropellant system, bipropellant subsystem Suppliers should modify it as necessary to meet the requirements of their hardware and should not consider this modification as a lac k of compliance. Although enclosure (3) shows right angled configuration thrusters and a preliminary design for tank and thruster bracketry, subsystem suppliers should consider the bracketry necessary to accommodate their own hardware. Subsystem suppliers who wish to respond at this time should first respond to the thruster complement and location suggested. They are encourage, however, to recommend alternate configurations should reliability or their particular component designs so dictate.

The candidate supplier should concentrate on submitting the following :

(i) A dry weight estimate, and mechanical schematic if it differs from Spar SG-350.

(ii) A wet weight estimate for six year and eight year mission.
N.B. Any weight below design goal in the Spec. would be very advantageous.

(iii) A reliability assessment for an eight year mission.

(iv) A pressure blowdown vs manoeuvre curve, if applicable.

(v) Proposed thruster operating ranges in I_{BIT}, thrust and cyclelife, approximate.

(vi) A budgetary fixed price and schedule estimate assuming a contract start date of 1 July, 1976. This date is given as a possible start date only to ensure uniformity of responses and does not represent a commitment to procure.

(vii) A presentation of the qualification status of proposed tankage and low and high thrust engines to meet the MPB requirements.

(viii) The proposed component test programs, especially for tankage and high and low thrust engines, to meet the MPB requirements.

(ix) The power requirements for latching valves, engine valves, chamber heaters if required, telemetry, drivers, and power conditioner if required.

(x) A preliminary presentation of where, on the primary structure, the plumbing, latching valves and electrical control unit should be located and approximate envelope required for these components. Assume they can be mounted anywhere on the bulkheads.

(xi) A preliminary presentation of feasibility of thruster mounting in required locations.

(xii) Approximate worst case heat flux into S/C due to thruster operation, and

(xiii) A plume density profile for each thruster proposed, if available.

In calculating the wet weight, the candidate supplier shall take into consideration the 15° canting of the NS Stationkeeping thrusters, unless his design changes this configuration, and shall assume there is a further degradation of NS thrust of 2.5% due to plume impingement, unless he can show that there will not be such a degradation.

Candidate suppliers should discuss the possibilities for Canadian content in the program and the impact on cost, if any, of providing this Canadian participation.

Responses to this request for budgetary estimate are to be mailed bearing a postmark date not later than midnight, May 1, 1975, and addressed to :

SPAR AEROSPACE PRODUCTS LIMITED,
825 CALEDONIA ROAD,
TORONTO,
ONTARIO M6B 3X8,
CANADA.

PHONE : (416) 781-1571
TELEX : 02-2054
TWX : 610-491-1503

and marked for the attention of the undersigned.

Should you have any questions regarding this request for budgetary estimate, please contact the undersigned.

Thank you for considering this request.

Yours very truly,

WILLIAM N. OWER
SUBCONTRACTS ADMINISTRATOR.
/hhm

AVCO SD WIL — ALSO SENT TO ROCKET RESEARCH; TCW; HAM-STANDARD[illegible]

SPAR TORONTO MARQUARDT; & BELL AERO (BUFFALO)

APR. 30/75 12:00 PR

ATTN: MR. KENT PUGMIRE 617-657-5111 Loc 2845

MSG. NO: PD2026

PLEASE CANCEL MSG. NO. PD2026 SENT APRIL 29/75 9:45 AND REPLACE WITH FOLLOWING MSGE. SORRY FOR ANY INCONVENIENCE THIS MAY HAVE CAUSED.

SUBJ: SPAR RFP #5427 - CHANGES IN RFP

1. RESPONSE DATE FOR THIS RFP IS HEREBY CHANGED: UR PROPOSAL IS NOW TO BE MAILED BEARING A POST-MARK DATE NOT LATER THAN MIDNIGHT 5 MAY/75 IN LIEU OF 1 MAY/75.

2. REVISE SOW .071 PARA 4.4 TO READ FLIGHT SPARE COMPONENTS. THE TOTAL QUANTITIES OF SPARE COMPONENTS REQUIRED SHALL BE ONE (1) TANK; TWO (2) HTE ASSEMBLIES; FOUR (4) LTE ASSEMBLIES; TWO (2) LV'S; TWO (2) FILTERS; TWO (2) PRESSURE TRANSDUCERS; TWO (2) SPARES FOR EACH PRESSURE MANAGEMENT DEVICE, IF REQUIRED FOR PRESSURE REGULATED SYSTEM, EXCEPT FOR 1 TANK ONLY; AND THREE (3) SPARES OF EACH UNIQUE ELECTRIC CIRCUIT BOARD USED IN THE ECU.

3. A DELIVERY SCHEDULE TO DELIVER ALL HARDWARE IN AS COST-EFFECTIVE A TIME FRAME AS POSSIBLE SHOULD BE ASSUMED. HOWEVER A GUIDELINE SCHEDULE OF DELIVERABLES IS AS FOLLOWS:

CONTRACT GO-AHEAD - 1 JULY/76 (AS STATED IN RFP LETTER)

DTM - LATE '76 OR EARLY '77

EM RCS - THIRD QUARTER '77 (QUALIFICATION S/S DELIVERED AS EM RCS)

FLIGHT, F1 - EARLY '78

FLIGHT, F2 - MID '78

FLIGHT, F3 - EARLY '79

PLEASE QUOTE PRICES IN 1975 DLRS

4. THREE (3) COPIES OF UR PROPOSAL ARE TO BE PREPARED AS PER RFP LETTER. TWO COPIES ARE NOW TO BE SENT TO SPAR AND ONE COPY TO:

STEPHEN F. ARCHER,
192 MALBOROUGH AVE.,
OTTAWA, ONTARIO, CANADA
K1N 8G4

W.N. OWER
SUBCONTRACTS ADMINISTRATOR

AVCO SD WIL

ISSUE

MULTIPURPOSE BUS STUDY

STATEMENT OF WORK-REACTION CONTROL SUBSYSTEM PRELIMINARY, FOR BUDGETARY ESTIMATES

PREPARED BY Kshnaneserinhan
FV
S.F. Archer
Propulsion/Systems Engineer

APPROVED BY D.A. Staley
D.A. Staley
Group Leader, Control Systems

APPROVED BY
P.A. McIntyre
Program Manager

APPROVED BY

REASON FOR REISSUE

DATE

AMENDMENTS			
NUMBER	REASON FOR AMENDMENT	SIGNATURE	DATE

SPAR FORM 2249 A

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

SPAR AEROSPACE PRODUCTS LTD

SPAR-SOW.071

MULTIPURPOSE BUS STUDY

STATEMENT OF WORK

REACTION CONTROL SUBSYSTEM

PRELIMINARY, FOR BUDGETARY ESTIMATES

Prepared by:........................
for S.F. Archer
Propulsion/Systems
Engineer

Approved by: D. A. Staley
D. A. Staley
Group Leader,
Control Systems

Approved by: P. A. McIntyre
P. A. McIntyre
Program Manager

5/CDP/1

APRIL 15, 1975

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1.0 INTRODUCTION

This document outlines the tasks required to design, fabricate, test and deliver the Reaction Control Subsystem (RCS) for the Multipurpose Bus (MPB) Satellite. The requirements of this document shall be applicable to the work performed by the contractor.

2.0 SCOPE

The contractor shall, in accordance with the terms and conditions set forth herein, furnish the necessary management, personnel, labour, services, documentation, materials, equipment, tools and facilities to design, develop, fabricate, test and deliver items as described below.

3.0 APPLICABLE DOCUMENTS

3.1 The following documents define the subsystem requirements:

MIL-D-1000	Military Specification for Engineering Drawings.
AFE-TRM-127-1	Eastern Test Range Safety Requirements.
D.A.C.-61687	Delta Spacecraft Design Restraints.
G-SFC-S320-G-1	General Environment Test Specification for Spacecraft Components.
FED.STD.209	Clean Room Requirements.
NHB 5300,4(1B)	Quality Assurance Requirements for Space Systems.
SPAR-SG.350	Multipurpose Bus Study Specification, Requirements, Reaction Control Subsystem Preliminary, for Budgetary Estimates.

The contractor shall direct his design, fabrication and test activities to meet these requirements within the most cost effective framework.

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3.2 Priority

TBD

4.0 HARDWARE DELIVERABLES

The RCS contractor shall furnish the following hardware items:

4.1 Dynamic/Thermal Model (DTM)

The Dynamic/Thermal Model shall consist of: one set of four (4) flight configuration propellant tanks capable of being pressurized to the maximum flight operating pressure and tank mounting provisions, at least simulations of the thrusters, valves, filters, sensors, electrical control unit and wiring harness, and flight configuration bracketry for mounting these components to the spacecraft primary structure. These components and simulations shall accurately simulate the mass, geometry, centre of mass, and vibrational, structural and thermal characteristics of the final design. In addition, plumbing and plumbing mounting provisions and routing shall be provided which shall, as nearly as possible, be identical with the proposed flight model design. The subsystem shall provide the capability to store the flight mass of water in the tanks and in the plumbing down to the simulated engines as Dynamic Model referee fluid when pressurized by gaseous nitrogen at maximum flight operating pressure.

4.2 Engineering Model (EM)

The Engineering Model shall be the Subsystem Qualification Model, and shall be the flight configuration. This model shall have undergone component acceptance testing and component and subsystem qualification to prove compliance with all of the requirements of the RCS Specification, SPAR-SG.350, and in accordance with the approved Subsystem Test Plans/Procedures for the RCS.

4.3 Flight Model (FM)

The Flight Model shall be identical with the item described by the RCS Specification, SPAR-SG.350.

4.4 Flight Spares

The quantities of these items are to be the following flight model and acceptance tested components: one (1) tank, one (1) high thrust engine (HTE) assembly including chamber heater, if required, and temperature sensor, two (2) low thrust engine (LTE) assemblies including chamber heaters, if required, and temperature sensors, one (1) latching valve, one (1) system filter, one (1) pressure transducer, one (1) spare each of pressurant management devices, if required, and one (1) spare of each unique electrical circuit board (ECB) used in the electrical control unit (ECU).

4.5 Electrical Servicing Cart

An electrical servicing cart shall be supplied to interface with the ECU of the RCS. It shall be capable of testing the RCS by supplying flight commands and/or flight power and/or monitoring telemetry through the ECU as required to:

a) Actuate thruster valves one at a time with programmable inputs of

   on time - 5 msec to steady state
   off time - 100 msec. to steady state
   number of pulses - 1 to 1000

   and monitor thruster valve currents.

b) Actuate latching valves and ascertain latched position.

c) Actuate heater circuits and monitor current loads.

d) Readout of temperature sensors.

e) Readout of pressure transducers.

The design shall protect against operator error damaging the RCS or spacecraft where feasible.

This cart shall be a four wheeled vehicle and designed for use in normal pad environments at Kennedy Space Center, Florida. The unit shall require only 115 volt, 60 cycle, electrical single phase power.

4.6 Alignment Mandrels

Two HTE and two LTE alignment mandrels shall be provided by the RCS contractor to meet the requirements of the RCS Specification, SPAR-SG.350, paragraph 3.3.1.4.

5.0 DOCUMENTATION DELIVERABLES

The RCS contractor shall furnish the following software items:

5.1 Analyses

The RCS contractor shall forward copies of analyses carried out on any aspect of the subsystem design. Such submissions may be in longhand but must be legible and presented on standard 8 1/2" x 11" size numbered sheets. The analyses shall contain the following:

- Heading page with title and section number; there shall preferably be one section per subject.
- Discussion of objectives and assumptions.
- Discussion of special methods of analyses.
- References.
- Calculations or computer input and output printouts. There shall be clear and concise explanations providing for rapid understanding of the subject matters, with the results of calculations clearly indicated.
- Summary of results and analyses.

The following analyses shall be carried out by the RCS contractor at the appropriate time in the design and be delivered as specified in Table I.

a) Mass properties including mass, centre of mass and moment of inertia of all components and the RCS as a unit.

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b) Engine thermal analysis including heat flux to the spacecraft.

c) Structural analysis.

d) Reliability analysis.

e) Plume impingement analysis.

f) Leakage correlation liquid to gas analysis.

g) Mission profile including pressure schedule and engine duty cycles and cycle life analysis.

h) Data reduction of all engine firing tests and preparation of flight performance prediction tables.

5.2 Program Controls

A progress report will be submitted by the RCS contractor monthly and shall include the following information.

5.2.1 Subsystem Technical Progress Narrative

The subsystem technical progress narrative shall comply with the following format:

Section 1 - A description of overall progress made during the reporting period and current weight status.

Section 2 - A description of current problems that may impede progress along with the proposed corrective action to meet milestone schedules.

Section 3 - A description of work to be performed during the next reporting period.

Section 4 - Product Assurance progress report.

Section 5 - A summary on the status of all Problem/Failure Reports and remedial and preventive actions during the reporting period.

5.2.2 Subsystem Cost Report

The subsystem cost report shall contain a breakdown of cost by category similar in format to the following:

Category	Planned Cost (Original) Estimate	Actual Cost to Date	% Of Task Completed	Estimated Cost to Completion	Variance From Total Planned Cost
Engineering					
Drafting					
Manufacturing					
Materials					
Services (or direct charges)					
Subcontracts					

The subsystem Cost Report will also provide an explanation of cost variances greater than 5%.

5.3 Subsystem Design Review Data Packages

The data to be supplied by the RCS contractor for the subsystem Design Reviews shall be that defined in Table I. Design Evaluation drawings shall be Category A, Form 1, in accordance with Military Specification MIL-D-1000. Interface control drawings shall be Category B, Form 3, in accordance with Military specification MIL-D-1000.

There shall be a preliminary design review (PDR) after subsystem design has been completed and

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prior to fabrication of the DTM and EM hardware. There shall be a critical design review (CDR) prior to EM final assembly and qualification test. There shall be a final design review (FDR) after system qualification is complete and prior to FM fabrication.

The RCS contractor shall notify Spar in writing 14 days in advance of the Critical Design Review. The subsystem definition resulting from this review shall constitute the first configuration baseline.

N.B. The RCS contractor may schedule such conceptual or preliminary design reviews as are considered necessary. Spar shall be notified in writing 14 days prior to such reviews to enable attendance to be arranged.

5.4 Drawings

Three complete sets of production drawings issued concurrently with their release to manufacturing. Updated issues of these drawings shall be sent as they are released.

5.5 Component Specifications

All RCS component specifications shall be submitted to Spar for review as delineated in Table I.

5.6 End Item Data Package

An end item data package shall be delivered with each of the EM and FM systems and with each flight spare and shall contain the data defined in Table I-F.

5.7 Quality Assurance, Configuration Management and Reliability

Quality assurance, configuration management and reliability plans shall be sumitted to Spar with the final proposal, requirements TBD.

5.8 Test Plans, Procedures and Reports

All development, acceptance and qualification tests on RCS hardware will be performed only after test plans and procedures have been written and submitted as delineated in Table I. Formal reports will be written after each test is completed and data is reduced and shall be submitted as delineated in Table I.

5.9 Interface Documentation

The RCS contractor shall deliver the following interface control documentation:

a) Connector wiring detail.
b) Mechanical interface detail drawings.
c) Overall subsystem electrical schematic drawings.

These drawings shall be prepared in accordance with Military Specification MIL-D-1000, Category B, Form 3. Preliminary data shall be delivered within 30 days of contract award, and shall be updated as required to reflect design status.

5.10 Manuals

Manuals will be supplied with each model and also with the Electrical Servicing Cart to delineate packing, repacking, installation, safety and test check list instructions.

5.11 Computer Programs

The RCS contractor shall deliver computer programs which have been developed for MPB work. The programs shall include at least the following documentation:

- Program listing printout with sufficient explanation to follow logic and operations to reconstruct the program if required, on a different installation, using different subroutines.

- Input and output printouts of program test cases (if any) and critical design cases with symbols and headings clearly identified for meaning.

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- Written purpose of program and listing of equations that are being solved by the program.

6.0 SCHEDULE AND DELIVERY

6.1 Schedule

The candidate RCS contractor shall assume a contract award date of July 1, 1976.

6.2 Delivery

Contract document requirements and delivery shall be according to Table I.

TABLE I

DOCUMENTATION REQUIREMENTS

Item No.	Item	Submittal Category A - For Approval I - For Information R - For Review	Schedule
A.	PRODUCT ASSURANCE DOCUMENTATION		
	TBD		
B.	PRELIMINARY DESIGN REVIEW DATA PACKAGE		
1.	Contractor/Supplier Identification List	R	
2.	Component and Subsystem Design Specifications	R	
3.	Functional Block Diagrams	R	Items 1 to 9 to be delivered with notification of Preliminary Design Review date.

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4.	Engineering Drawings:		
	4.1 Assembly, Parts and Materials List	R	
	4.2 Block Diagrams	R	
	4.3 Circuit Schematics	R	
	4.4 Main Mechanical	R	
	4.5 Main Assembly	R	
	4.6 Subassembly	R	
5.	Test Reports on Bread-board Experiments	R	
6.	Acceptance Test Plans	A	
7.	Qualification Test Plans	A	
8.	Reliability Analysis Data		
	8.1 Reliability Predictions and Models	R	
	8.2 Fail Safe and Redundancy	R	
	8.3 Failure Mode, Effect, and Criticality	R	
	8.4 Trade-Offs	R	
	8.5 Interface Compatibility	R	
	8.6 Parts and Material Application	R	
9.	Processes List	R	
10.	Mass Properties, Thermal, Structural, Plume Impingement, Leakage and Mission Profile Analyses	R	
11.	Design Review Reports	A	Within 14 days of end of preliminary design review.

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C. <u>CRITICAL DESIGN REVIEW DATA PACKAGE</u>

1.	Final Component and Subsystem Design Specifications	R	
2.	Final Functional Block Diagrams	R	
3.	Final Engineering Drawings		Items 1 to 8 to be delivered 14 days prior to Design Review Meeting.
	3.1 Assembly, Parts and Lists	R	
	3.2 Block Diagrams	R	
	3.3 Circuit Schematics	R	
	3.4 Main Assembly, Mechanical	R	
	3.5 Main Assembly, Electrical	R	
	3.6 Sub-Assembly, Mechanical	R	
	3.7 Sub-Assembly, Electrical	R	
4.	Development Test Results and Report	R	
5.	Acceptance Test Specifications and Procedures	R	
6.	Qualification Test Specifications and Procedures	R	
7.	Final Reliability Analysis Data	R	
8.	Final Processes List	R	
9.	Final Mass Properties, Thermal, Structural, Plume Impingement, Leakage and Mission Profile Analyses	R	
10.	Design Review Reports	A	Within 14 days of end of critical design review.

D.	FINAL DESIGN REVIEW DATA PACKAGE		
1.	EM Acceptance and Qualification Test Results Report	R	Items 1 and 2 to be delivered 14 days before design review.
2.	Revisions to the following:		
	2.1 Final Design Specifications	R	
	2.2 Final Functional Block Diagram	R	
	2.3 Final Engineering Drawings	R	
	2.4 Acceptance Test Specifications and Procedures	R	
	2.5 Final Reliability Analysis Data	R	
	2.6 Final Processes List	R	
	2.7 Final Mass Properties, Thermal, Structural, Plume Impingement, Leakage and Mission Profile Analyses	R	
3.	Final Design Review Report	A	Within 14 days of end of final design review.
E.	CONFIGURATION DATA		
1.	Configuration Control Plan	A	With Proposal
2.	Engineering Change Proposals	A	After Critical Design Review and prior to implementation.
3.	Engineering Change Notice	R	Within 7 days of Contractor change action.

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F.	**END ITEM DATA PACKAGE** (Engineering Model, and Protoflight Subsystems, Spares)		
1.	Subsystem and Component Up-dated drawings	A	
2.	Equipment Logs and Completed Check Lists	A	All items to be delivered with each subsystem and spares.
3.	End Item Acceptance Test Reports	A	
4.	End Item Inspection Summary Reports	A	
G.	Engine Flight Performance Prediction Tables	A	
H.	Safety Plan	A	

One reproducible and three copies of each software line item shall be delivered to Spar. All other correspondence addressed to Spar may be single submission but of reproducible quality.

6.3 Final Acceptance

Final acceptance of the product will be made at contractor's plant, and shall demonstrate:

a) That the equipment as manufactured and assembled is exactly the equipment described by released engineering documentation.

b) The new equipment is exactly the configuration of the equipment identified for production.

c) That the validity of the acceptance test data and testing methods exactly meet the requirements as specified in the procurement specification.

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7.0 SUBSYSTEM NON-DELIVERABLES

The RCS contractor shall fabricate, as necessary, development test subsystem assemblies to determine performance capability. This category of hardware shall include breadboards, mock-ups, and experimental modules. Generally, these are not deliverable items.

They shall be retained for the duration of the project by the RCS contractor, and the work on them will be subject to monitoring by Spar. All documents pertaining to their design, evaluation and testing shall be made available on request to Spar.

8.0 SAFETY

The RCS contractor shall prepare a safety plan which defines the Contractor's approach in implementing a safety program which meets the requirements of the AFETRM-127-1 Eastern Test Range Safety Requirements.

9.0 PERIOD OF DOCUMENTS RETENTION

The RCA contractor shall retain all non-deliverable records, including records of manufacturing, test and inspection, graphs, photographs, reports and film of non-destructive testing, for a period of three years after the contract is completed. The documents to be retained are those that apply to the parts, components, and assemblies qualified for flight.

ISSUE [ ]

MULTIPURPOSE BUS STUDY

SPECIFICATION REQUIREMENTS,

REACTION CONTROL SUBSYSTEM

PRELIMINARY, FOR BUDGETARY ESTIMATES

PREPARED BY [signature] for
S.F. Archer
Propulsion/Systems Engineer

APPROVED BY D.A. Staley
D.A. Staley
Group Leader, Control Systems

APPROVED BY P.A. McIntyre
P.A. McIntyre
Program Manager

APPROVED BY ______

REASON FOR REISSUE ______ DATE ______

AMENDMENTS			
NUMBER	REASON FOR AMENDMENT	SIGNATURE	DATE

SPAR FORM 2249 A

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MULTIPURPOSE BUS STUDY SPECIFICATION

REQUIREMENTS, REACTION CONTROL SUBSYSTEM

PRELIMINARY, FOR BUDGETARY ESTIMATES

Prepared by:..........................
S.F. Archer
Propulsion/Systems
Engineer

Approved by: D. A. Staley ....
D. A. Staley
Group Leader,
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Approved by: P. A. McIntyre ....
P. A. McIntyre
Program Manager

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APRIL 15, 1975

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ERRATA

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1. Column 1 - For Pre-Apogee precession, total impulse should read as 2,807 ft. lbf. sec instead of 2200 ft.lbf. sec

2. Column 2 - For post apogee precession, total impulse should read as 1337 ft. lbf. sec instead of 1137 ft. lbf. sec.

1.0 SCOPE

This specification establishes the requirements for performance, design, qualification and acceptance of a subsystem identified as the Reaction Control Subsystem (RCS). It is proposed that the RCS shall fly as a subsystem of the proposed Multipurpose bus (MPB) to be used for transporting to and maintaining in geostationary equatorial orbit several possible complements of communications payloads.

2.0 APPLICABLE DOCUMENTS

The documents listed in paragraph 3 of the Statement of Work SPAR-SOW.071 shall form a part of this specification.

3.0 REQUIREMENTS

The RCS shall be a monopropellant hydrazine or bipropellant, MMH/N_2O_4 (monomethyl hydrazine/ Nitrogen tetraoxide), mass expulsion propulsion system composed of the following components:

a) Four propellant tanks capable of liquid feed with spacecraft in either spin or three-axis stabilized modes, and pressurant storage and pressure regulation equipment, if required.

b) Thrusters capable of providing spinning and three-axis stabilized mode impulse as specified in Section 3.1 below.

c) Fill and drain valves as necessary to load and unload fuels and pressurants in the ground handling orientation.

d) Fuel latching valves with position indication located so as to maximize subsystem redundancy.

e) Plumbing for fuel and pressurant feed.

f) Liquid filters to ensure that no performance degradation occurs as a result of contamination.

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g) Tank pressure transducers.

h) Engine thermal sensors on all thrust chambers.

i) Electrical control unit (ECU).

j) Subsystem wiring harness.

k) Structural bracketry for all RCS components.

l) Propellant and pressurant.

3.1 Mission Requirements

The purpose of the RCS is to perform the following functions:

a) Spacecraft in Spinning Mode at 60 $\pm$6 RPM (Around the Z-Z Axis)

   i) Precession of spin axis prior to and after apogee motor firing.
   ii) Despin, and spin-up if required.

b) Attitude Acquisition Including Three-Axis Limit Cycle, Wheel Spin-Up and On-Board Capture Operation

c) Spacecraft in Three-Axis Stabilized Mode

   i) Station acquisition in plane/out of plane velocity change (X-X and Y-Y axes).
   ii) North or south stationkeeping, velocity change (Y-Y axis).
   iii) East and west stationkeeping, velocity change (X-X axis).
   iv) Momentum wheel unloading around pitch axis (Y-Y axis).
   v) Whecon control about roll and yaw axes (X-X and Z-Z axis) with 10° offset angle, firing positive roll and negative yaw with one offset engine and negative roll and positive yaw with a second offset engine. Total impulse for this manoeuvre is split equally between the two engines.

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3.1.1 Mission Sequence

The mission sequence is shown in Table I. Individual thrusters when installed in the RCS must be capable of performing all their prescribed functions as noted in the table with any single thruster failure and with a 25% margin on total impulse.

3.1.2 High Thrust Engine (HTE)

The High Thrust Engine shall have a steady state thrust level between 1.0 and 5.5 lbf over the range of anticipated supply pressures required to perform the manoeuvres defined in Table I. At a given supply pressure and propellant inlet temperature the HTE steady state thrust shall be predictable within $\pm 3\%$ for a given engine and shall be within $\pm 5\%$ from engine to engine.

3.1.2.1 Cumulative Angular Centroid and Cumulative Effective Impulse Predictability

The cumulative angular centroid and the cumulative effective impulse as defined below shall be predictable at each point in the matrix of variables of supply pressure, initial bed temperature and propellant temperature over their nominal operating ranges as a function of pulse train length as specified in Table II when firing at the duty cycles as specified in Table I.

$$1.\quad t_d = \frac{1}{w} \arctan \left\{ \frac{\sum_{n=1}^{PTL} \int_{t_n}^{t_{n+1}} F \sin(wt)\, dt}{\sum_{n=1}^{PTL} \int_{t_n}^{t_{n+1}} F \cos(wt)\, dt} \right\}$$

$$2.\quad I_{eff} = \sqrt{\left(\sum_{n=1}^{PTL} \int_{t_n}^{t_{n+1}} F \sin(wt)\, dt\right)^2 + \left(\sum_{n=1}^{PTL} \int_{t_n}^{t_{n+1}} F \cos(wt)\, dt\right)^2}$$

where:

t_d = cumulative angular centroid time (seconds)

w = vehicle spin rate (rad/sec)

t = time (seconds) from start of electrical pulse

t_n = start time of n^{th} pulse (seconds)

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t_{n+1} = start time of $(n+1)^{th}$ pulse (seconds)

I_{EFF} = cumulative effective impulse

F = engine instantaneous thrust (pounds)

PTL = pulse train length

3.1.2.2 Cumulative Rotational Efficiency

The high thrust mission average cumulative rotational efficiency for the high thrust manoeuvres shall be greater than .93, at the spacecraft nominal spin rate of 60 rpm plus 10 percent. Cumulative rotational is defined below.

1) $$\eta_{PTL} = \frac{I_{eff}}{I_{Engine}}$$

2) $$\eta_{mission} = \frac{\sum_{n=1}^{M} I_{Eff\,n}}{\sum_{n=1}^{M} I_{Engine\,n}}$$

where: η_{PTL} = cumulative rotational efficiency

I_{Eff} = cumulative effective impulse (lbf.sec)

I_{Efn} = cumulative effective impulse of the n^{th} firing train (lbf. sec.)

I_{Engine} = engine delivered total impulse (lbf.sec.)

$I_{Engine\,n}$ = engine delivered total impulse of the n^{th} firing train (lbf.sec.)

$\eta_{Mission}$ = average mission cumulative rotational efficiency

M = number of H.T.E. pulse trains

3.1.3 Low Thrust Engine (LTE)

The low thrust engines shall have a steady state thrust level between .3 lbf and .01 lbf over the

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range of anticipated supply pressures required to perform the manoeuvres defined in Table I. At a given supply pressure and propellant inlet temperature the steady state thrust shall be predictable to within ±3% for a given engine and within ±5% from engine to engine.

3.1.3.1 Impulse Predictability

The total impulse and/or impulse bit predictability shall be as defined below:

a) North, south, east or west stationkeeping and acquisition cumulative total impulse shall be predictable at each point in the matrix of variables of supply pressure, initial bed or thrust chamber temperature and propellant temperature over their nominal operating ranges to within ± five (5) percent.

b) Pitch momentum dumping cumulative total impulse shall be predictable at each point in the matrix of variables of supply pressure, initial bed or thrust chamber temperature, and propellant temperature over their nominal operating ranges as a function of the pulse train length required, when firing at the nominal duty cycle as specified in Table I, to the accuracy delineated in Table III.

c) The linear impulse bit for very low duty cycle, single pulses required for the offset engines and misalignment torque removal, shall be predictable at each point in the matrix of variables of supply pressure, initial bed or thrust chamber temperature, and propellant temperature over their nominal operating ranges to within ± ten (10) percent. At each point in the same matrix 50% and 90% of this impulse bit shall be delivered within .1 and 5 seconds respectively after initiation of the electrical command.

3.2 System Definition

The RCS shall be designed to supply propellant to each thruster at pressure levels which allow the

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thruster to meet the requirements of paragraph 3.1 above.

3.2.1 Subsystem Mechanical Schematic

The suggested mechanical schematic for a monopropellant RCS system is shown in Figure 1. Modification to this schematic to accommodate specific engine designs or a bipropellant system will be considered.

3.2.2 Subsystem Electrical Schematic

The suggested electrical schematic for the RCS is shown in Figure 2. Modification to this schematic to accommodate specific engine designs or a bipropellant system will be considered. The Electrical Control Unit shown in this schematic shall be supplied by the RCS contractor and shall provide the following electronics:

a) Valve drivers for all RCS valves including valve suppression.

b) Drivers for all engine performance heaters, if required, grouped to minimize commands but at the same time provide redundancy needed to meet subsystem reliability requirements.

c) Signal conditioning for; RCS pressure transducers, engine chamber temperature sensors, latching valve position indicators, total RCS driver current and chamber heater group flags, if heaters are required.

d) Power conditioning if required.

e) Test points to measure thruster valve currents and heater group currents, if heaters required.

3.2.3 RCS Mechanical Configuration

The spacecraft principal axes (three-axis stabilized mode) and suggested engine locations and

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directions are shown in Figures 3 and 4. Tables IV and V define the suggested function and moment arm of each thruster. Figures 5 and 6, enclosed, Spar drawing 31138J1 sheets 1 and 2 entitled, Structure and RCS General Purpose Satellite Bus show the suggested spacecraft and RCS mechanical layout.

3.3 Subsystem Interface

3.3.1 Mechanical Interface

3.3.1.1 Subsystem Integration

The RCS components shall be mounted on the spacecraft east, west, north and south bulkheads either directly or using struts supplied by the RCS contractor. The spacecraft primary structure including these bulkheads mounted on the central thrust tube shall be supplied to the RCS contractor for each subsystem final assembly, test and shipment. The packaging and structural design shall be modularized as much as possible.

3.3.1.2 Envelope

Final envelopes for RCS components are TBD. Reference Figures 5 and 6 for preliminary spacecraft configuration.

3.3.1.3 Engine Plume Impingement

No RCS engine plume shall cause heating of any spacecraft surface of greater than 900 BTU/hr.ft^2. It is recognized that there shall be significant thrust degradation for N-S engines. The propellant mass budget shall be based upon engine net thrust. Net thrust is defined as the engine thrust in the desired direction less any vectoring losses less the resultant plume drag on spacecraft surfaces parallel and normal to the desired thrust direction.

3.3.1.4 Engine Alignment

The actual thrust vector of each thruster shall subtend an angle of $\pm .15$ degrees, 3σ variation,

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with the nozzle geometric centreline under all the operating conditions specified herein. The alignment fixture tolerance, (i.e. the uncertainty in knowledge of the nozzle geometric centreline angle with respect to the alignment mandrel mirror reference), shall be $\pm$.1 degrees, 3σ variation. An adjustment range allowing $\pm$2 degrees rotation of the nozzle geometric centreline about two (2) mutually perpendicular directions with an adjustment resolution of $\pm$.05 degrees shall be provided in the design of each engine mount. This alignment shall be possible with the thruster sealed in the subsystem. The nozzle geometric centreline null shall be coincident with the axes as defined in Figure 4. Yielding of the propellant inlet tube upstream of each thruster valve during alignment shall be allowed if integrity can be maintained.

3.3.2 Thermal Interface

The RCS contractor shall be responsible for thermal control of engine thrust chambers to meet performance requirements specified herein. All other thermal control of RCS components shall be the responsibility of Spar Aerospace Products Ltd.

3.3.3 Electrical Interface

3.3.3.1 Power

The spacecraft shall supply the RCS ECU with 27.5 and/or 50 VDC power regulated to $\pm$1%. If the RCS requires electrical power at voltages other than 27.5 VDC or 50 VDC $\pm$1% the required power conditioning shall be provided within the ECU. Power allotment for RCS components is TBD.

3.3.3.2 Telemetry and Command

Table VI defines the telemetry and command requirements for the RCS. Input and output voltages shall be 0-5 VDC, tolerances and impedances TBD.

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3.4 Subsystem Requirements

3.4.1 Proof and Burst Pressure

The components and integrated subsystem shall have a proof pressure of 1.5 times and a burst pressure of not less than 4.0 times the maximum operating pressure, except the propellant tank, which shall have a burst to operating pressure ratio of 2 to 1 minimum at maximum operating pressure and maximum temperature, normalized for minimum membrane thickness and tank material physical properties.

3.4.2 Subsystem Leakage

The subsystem gas side leakage rate including external, and internal leakage across the gas/ liquid interface and out through the capped fill and drain valves and other pressurant management devices, if required, shall result in a total loss of pressurant after eight (8) years in space of not more than the equivalent of two (2) percent of the maximum operating pressure. The liquid leakage rate from the liquid side of the subsystem shall result in a loss of less than 1.0 lbm of propellant after eight (8) years in space. Analyses shall be presented, for Spar approval, to justify the test gas leakage rates comparable to these allowable propellant and pressurant leakages.

3.4.3 Servicing

The RCS shall incorporate provision for convenient servicing with fuels and pressurant, drainage and flushing, vacuum drying the system and thrusters, leak checks and flow checks from a RCS ground support cart while mounted in the spacecraft in the launch orientation. The RCS contractor is not responsible for fabricating this ground support cart.

3.4.4 Subsystem and Component Cleanliness

To ensure proper performance of the subsystem, all components, and the subsystem itself, shall meet the cleanliness requirements of Table VII. In addition, no metal particles shall be allowed which are over 50 microns.

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In addition, the RCS contractor shall, in his proposal, submit for Spar approval the cleanliness requirements for prevention of non-volatile residues in the propellant.

3.4.5 Pressurants

The pressurant shall be gaseous nitrogen per MIL-P-27401B which shall contain a tracer gas for leak detection usage loaded into the subsystem through a five (5) micron absolute (or less) filter.

3.4.6 Useful Life

The RCS shall be capable of performing as specified for not less than 8 years in space following a maximum of two year storage after delivery to Spar unfuelled.

3.4.7 Reliability

The RCS shall have a minimum probability for successful operation of .95 for the mission life of eight (8) years.

3.4.8 Weight

The weight of each assembly/component of the RCS shall be the minimum consistent with the intended function and other design requirements. The RCS dry weight design goal shall be 70.0 lbm, maximum, with a design goal wet weight of 260 lbm for 6 years on orbit life and 310 lbm for 8 years on orbit life, including all of the leakage, contingency, loading tolerance residual and 5% growth propellant. No design changes resulting in a change in components or assembly weights shall be implemented without the prior approval of the customer who shall be notified in writing denoting the change, reason thereof and subject weight increase or decrease. The accuracy of assembly/component dry weight measurement shall be $\pm$ 0.01 lbm or $\pm$ 0.02% of the total dry weight (whichever is greater).

3.4.9 Safety

The RCS shall be designed to limit hazards to personnel and equipment. Explosive and toxic hazards shall be defined and procedures for limiting their effect on personnel and equipment shall be formulated and enforced. The requirements of AFETRM - 127 - 1 Eastern Test Range Safety Requirements shall apply.

3.4.10 Balance

Static and dynamic balancing of the RCS mounted to the spacecraft aft platform will be the responsibility of Spar. The wet system imbalance shall be minimized by control of tank diameter and location. Tanks shall be centred within $\pm$.050 inches of nominal position.

3.5 System Environment

3.5.1 Structural Environment

The RCS shall be designed, analyzed and tested to meet the mechanical environments specified in D.A.C.-6187 Delta Spacecraft Design Restraints.

3.5.2 Thermal Environment

All RCS components other than engine thrust chambers will be maintained by the spacecraft thermal control between 40 and 130°F except for the propellant tanks which will be maintained between 40 and 120°F. The engine thrust chambers shall be designed to perform as specified herein when exposed to steady state full solar flux of 450 BTU/hr/ft^2 and when exposed to steady state cold space with no solar flux. The engine mounting flange interface temperature will be 40 to 150°F in the case of no net heat flux across the interface. Maximum heat leak to space through a HTE shall be TBD BTU/hr and through a LTE shall be TBD BTU/hr.

3.5.3 Electromagnetic Interference

TBD

4.0 QUALITY ASSURANCE

Unless otherwise stated, all tests, analyses and inspections specified below shall be performed by the RCS contractor.

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The RCS program shall comply with the following documents:

NHB 5300.4(1B)	Quality Assurance Requirements for Space Systems
G-SFC-S320-G-1	General Environment Test Specification for Spacecraft Components.
FED.STD.209	Clean Room Requirements.

Table VIII defines the method to be used by the RCS contractor to demonstrate compliance to the requirements of Section 3.0 of this specification.

4.1 Development

4.1.1 Component Development

It is required that all components proposed for use in the RCS be fully developed prior to contract award.

4.1.2 Subsystem Development

The following development tests shall be conducted on the dynamic thermal model RCS at the RCS contractor's facility prior to shipment to Spar Aerospace Products Ltd.

a) examination of product
b) dry weight
c) proof and leakage test of tankage and plumbing
d) final examination of product.

4.2 Qualification

4.2.1 Component Qualification

It is preferred that all components be currently flight qualified to meet the requirements of this specification. Any exception to this preference shall be noted by the candidate RCS contractor in his proposals to Spar along with proposed component qualification test program descriptions.

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4.2.2 Subsystem Qualification Tests

The following tests shall be conducted on the engineering model RCS:

a) Examination of product.
b) Subsystem weight.
c) Functional and electrical check (all components).
d) Electromagnetic interference test (EMI test).
e) Proof and leakage test (external and internal leakage).
f) Engine gas flow tests (all engines).
g) Vibration test (random and sine, qualification).
h) Functional and electrical check (all components).
i) Acceleration test.
j) Functional and electrical check (all components).
k) Leakage and gas flow test (external and internal leakage).
l) Thermal vacuum test (40°F and 130°F, stabilize and run functional and electrical tests at extremes).
m) Fuel compatibility test.
n) Contamination check.
o) Basepoint firing test all engines (assume 4 hrs. firing sequence per engine, engines may be fired in groups).
p) Mission simulation firing test (assume 168 hrs. firing sequence).
q) Basepoint firing test all engines.
r) Functional and electrical check (all components).
s) Leakage and gas flow test (external and internal leakage).
t) Post test inspection.

4.3 Acceptance

4.3.1 Component Acceptance Test

All engines used in qualification tests, in the flight subsystem and for flight spares shall be tested according to the outlines subsequently listed and shall meet their performance requirements specified herein.

a) Examination of product.
b) Proof and leakage test (external and internal leakage).

c) Vibration (spares only).
d) Basepoint test (assume 4 hrs. firing sequence).
e) Leakage and gas flow (internal and external leakage).
f) Post test inspection.

All other component acceptance tests TBD.

4.3.2 <u>System Acceptance Test</u>

The following tests shall be conducted on the flight model RCS.

a) Examination of product.
b) Dry weight.
c) Functional and electrical check (all components).
d) Proof and leakage (internal and external leakage).
e) Engine gas flow tests (all engines).
f) Vibration test (random and sine, acceptance).
g) Contamination check.
h) Functional and electrical check (all components).
i) Leakage and gas flow test (external and internal leakage).
j) Post test inspection.

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Manoeuvre	Total Impulse Or Delta Velocity	Maximum Thrust/ Torque Bit	Minimum Thrust/ Torque Bit	Duty Cycle	Maximum Starts	Maximum Pulses
1. Pre Apogee Precession	2,200 ft.lbf.sec.	5.5 lbf	1.0 lbf	.135 sec. on, .835 sec. off $\pm 10\%$	10	Function of Thrust Level
2. Post Apogee Precession	1,137 ft.lbf.sec.	5.5 lbf	1.0 lbf	.135 sec. on, .835 sec. off $\pm 10\%$	10	Function of Thrust Level
3. Despin	954 ft.lbf.sec.	.3 lbf	.01 lbf	Continuous Burn	3 per engine	3 per engine
4. Attitude Acquisition						
a) pitch wheel spinup	60 ft.lbf.sec.	$7.5\text{x}10^{-3}$ ft.lbf.sec.	TBD ft.lbf.sec	Average Torque during manoeuvre = $6\text{x}10^{-3}$ ft.lbf.	1	Function of Torque Bit Level
b) 3 axis limit cycle	80 ft.lbf.sec.		Very Low Duty Cycle - TBD			
c) on-board capture with offset engines	40 ft.lbf.sec.		Very Low Duty Cycle - TBD			

5. Station Acquisition - in plane - out of plane	 80 ft./sec. 80 ft./sec.	 .3 lbf.	 .01 lbf.	 Continuous Burn (inverse pulse width modulation)	 40 Orbit Manoeuvres per engine	 TBD
6. On-Board Roll-Yaw Attitude Control	1885 ft.lbf.sec. (6 years) 2515 ft.lbf.sec. (8 years)	8.0×10^{-3} ft.lbf.sec.	5×10^{-4} ft.lbf.sec.	Continuous pulsing, 20 sec. to .5 day off time between pulses	Function of IBIT Level	Function of Torque Bit Level
7. Pitch Momentum Dumping	104 ft.lbf.sec. (6 years) 139 ft.lbf.sec. (8 years)	7.5×10^{-3} ft.lbf.sec.	TBD ft.lbf.sec.	Manoeuvre every 21 days, average torque bit during manoeuvre = 6×10^{-3} ft.lbf.	150	Function of Torque bit level
8. East-West Stationkeeping	84 ft./sec. (6 years) 112 ft./sec. (8 years) Split between E&W TBD	.3 lbf.	.01 lbf.	Manoeuvre every 21 days, continuous burn (inverse pulse width modulation)	150	TBD

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9. North-South Stationkeeping (North or South only permissible operating mode)	999 ft.sec. (6 years) 1,320 ft. sec. (8 years)	.3 lbf.	.01 lbf.	Manoeuvre every 21 days, continuous burn (inverse pulse width modulation)	150	TBD

NOTES: 1. Manoeuvres 6, 7, 8 and 9 comprise the on-orbit mission and are interspersed over the mission life.

2. The spacecraft mass prior to commencing manoeuvre 2 shall be 1,017 lbm.

3. Vehicle effective total impulse stated for manoeuvres 1 and 2.

4. Figure 4 shows offset engines 13, 14, 15 and 16 lying in a projection of the roll yaw plane.

5. Vehicle effective delta velocity stated for manoeuvre 9.

6. Although manoeuvre 6 allows a torque bit of 8×10^{-3} ft.lbf.sec. there will be a spacecraft weight penalty (momentum wheel) for values $\geq 4 \times 10^{-3}$ ft.lbf.sec. assume that this penalty is linear to a maximum of 10 lbm. at 8.0×10^{-3} ft.lbf.sec.

7. Tankage shall be sized and fuel shall be allotted for 5% growth propellant.

TABLE II

HIGH THRUST ENGINE PREDICTABILITY

Pulse Train Length	Cumulative Angular Centroid Predictability (Three Sigma Variation)	Cumulative Effective Impulse Predictability (Three Sigma Variation)
5-9	± .040 sec.	± 30%
10-49	± .025 sec.	± 10%
50-99	± .020 sec.	± 8%
100-149	± .015 sec.	± 6%
150-249	± .010 sec.	± 5%
250	± .005 sec.	± 5%

TABLE III

PITCH MOMENTUM DUMPING CUMULATIVE

IMPULSE PREDICTABILITY

Pulse Train Length	Cumulative Total Impulse Predictability (Three Sigma Variation)
5-9	± 30%
10-24	± 20%
25-49	± 10%
50-74	± 8%
75-99	± 7%
100-149	± 6%
150	± 5%

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TABLE IV - PRELIMINARY

RCS THRUSTER FUNCTIONS - NOMINAL C OF M LOCATION

THRUSTER	+P	-P	+R	-R	+Y	-Y	N	S	E	W	+O	-O	PR
1	△			△		(II)	(III)			[VII]			
2	△		△		(I)			(IV)		[VII]			
3										◇P			
4	◇P									(VIII)			□
5		◇P								(VIII)			□
6		△		△	(I)		(III)		[V]				
7		△	△			(II)		(IV)	[V]				
8									◇P				
9		◇S							(VI)				□
10	◇S								(VI)				□
11			◇P				(III)						
12				◇P				(IV)					
13				□								◇P	
14			□								◇P		
15				□								◇S	
16			□								◇S		
17													◇P
18													◇S

KEY

◇A PERFORMS INTENDED FUNCTION BY ITSELF: "A"=P, PRIMARY; "A"=S, SECONDARY

(A) PERFORMS INTENDED FUNCTION WITH OTHER ENGINE(S) NUMBERED "A"

□ PERFORMS SECONDARY FUNCTION WITH PROPELLANT INEFFICIENCY

△ HAS UNWANTED SIDE EFFECT WHEN FIRED BY ITSELF

NOTE: THRUSTERS 11 & 12 CANNOT PERFORM PRECESSION BECAUSE THEY ARE COVERED UNTIL THE ARRAYS ARE DEPLOYED DURING ATTITUDE ACQUISITION

TABLE V - PRELIMINARY

RCS THRUSTER LOCATIONS AND DIRECTIONS - NOMINAL C OF M LOCATION

THRUSTER	EXIT PLANE DISTANCE FROM C OF M			DIRECTION
	X-X	Y-Y	Z-Z	
1	- 42.0 in.	-3.4 in.	+ 1.5 in.	NORTH 15° WEST [1]
2	- 42.0 in.	+ 3.4 in.	+ 1.5 in.	SOUTH 15° WEST [1]
3	- 42.5 in.	0.0 in.	0.0 in.	DIRECTLY WEST
4	- 35.0 in.	0.0 in.	+ 14.0 in.	DIRECTLY WEST
5	- 35.0 in.	0.0 in.	- 14.0 in.	DIRECTLY WEST
6	+ 42.0 in	- 3.4 in.	+ 1.5 in.	NORTH 15° EAST [1]
7	+ 42.0 in	+ 3.4 in.	+ 1.5 in.	SOUTH 15° EAST [1]
8	+ 42.5 in.	0.0 in.	0.0 in.	DIRECTLY EAST
9	+ 35.0 in	0.0 in.	+ 14.0 in.	DIRECTLY EAST
10	+ 35.0 in.	0.0 in.	- 14.0 in.	DIRECTLY EAST
11	+ 1.3 in.	- 26.8 in.	- 37.0 in.	NORTH 15° AFT, VECTORED THROUGH PITCH AXIS [2]
12	+ 1.3 in.	+ 26.8 in.	- 37.0 in.	SOUTH 15° AFT, VECTORED THROUGH PITCH AXIS [2]
13	- 6.8 in.	- 24.0 in.	- 38.0 in.	AFT 10° WEST, VECTORED NORTH IF NECESSARY
14	- 6.8 in.	+ 24.0 in.	- 38.0 in.	AFT 10° WEST, VECTORED SOUTH IF NECESSARY
15	- 6.8 in.	- 22.0 in.	- 38.0 in.	AFT 10° WEST, VECTORED NORTH IF NECESSARY
16	- 6.8 in.	+ 22.0 in.	- 38.0 in.	AFT 10° WEST, VECTORED SOUTH IF NECESSARY
17	- 32.0 in.	- 1.6 in.	- 40.0 in.	WEST 2.3° NORTH [2]
18	- 32.0 in.	+ 1.6 in.	- 40.0 in.	WEST 2.3° SOUTH [2]

NOTES:
1. YAW MOMENT ARM ± 39.7 in.
2. NECESSARY OFFSET TO MOUNT ON BULKHEADS.
3. ENGINES MAY NOT ENTER AREA BELOW SEPARATION PLANE (FIGURE 5) WITHIN 25 INCHES OF Z-Z AXIS.

TABLE VI

RCS COMMAND AND TELEMETRY REQUIREMENTS

CHANNEL NUMBER	COMMAND CHANNEL	COMMAND DURATION	TELEMETRY CHANNEL
1	THRUSTER 1 VALVE	AS LONG AS VALVE OPEN	THRUSTER 1 CHAMBER TEMP.
2	" 2 "	"	" 2 "
3	" 3 "	"	" 3 "
4	" 4 "	"	" 4 "
5	" 5 "	"	" 5 "
6	" 6 "	"	" 6 "
7	" 7 "	"	" 7 "
8	" 8 "	"	" 8 "
9	" 9 "	"	" 9 "
10	" 10 "	"	" 10 "
11	" 11 "	"	" 11 "
12	" 12 "	"	" 12 "
13	" 13 "	"	" 13 "
14	" 14 "	"	" 14 "
15	" 15 "	"	" 15 "
16	" 16 "	"	" 16 "
17	" 17 "	"	" 17 "
18	" 18 "	"	" 18 "
19	LV1 OPEN	50 msec	TANK PRESSURE 1
20	LV1 CLOSE	"	TANK PRESSURE 2
21	LV2 OPEN	"	LV1 POSITION
22	LV2 CLOSE	"	LV2 "
23	LV3 OPEN	"	LV3 "
24	LV3 CLOSE	"	LV4 "
25	LV4 OPEN	"	LV5 "
26	LV4 CLOSE	"	LV6 "
27	LV5 OPEN	"	LV7 "
28	LV5 CLOSE	"	LV8 "
29	LV6 OPEN	"	RCS INPUT CURRENT (ANALOG)
30	LV6 CLOSE	"	CHAMBER HEATER GROUP ON/OFF FLAGS, IF HEATERS REQUIRED.
31	LV7 OPEN	"	
32	LV7 CLOSE	"	
33	LV8 OPEN	"	
34	LV8 CLOSE	"	
35	RCS POWER ON		
36	RCS POWER OFF		
37 ETC	CHAMBER HEATER GROUPS, IF REQUIRED, ON/OFF, TBD	AS LONG AS HEATER GROUP TO BE ACTIVATED	

TABLE VII

Propellant Particulate Size Range (Microns)	Cleanliness Requirements Maximum Particles Allowed Per 100 Millilitre Sample
5-10	1200
10-25	200
25-50	50
50-100	5
over 100	0

TABLE VIII

METHODS OF COMPLIANCE WITH REQUIREMENTS

SPECIFICATION PARAGRAPH NUMBER AND SUBJECT	PROOF OF COMPLIANCE BY		
	TEST	ANALYSIS	INSPECTION
3.1.1 Mission Sequence	X	X	
3.1.2 HTE Thrust	X	X	
3.1.2.1 HTE Pulsed Performance	X	X	
3.1.2.2 HTE Pulsed Performance	X	X	
3.1.3 LTE Thrust	X	X	
3.1.3.1 LTE Impulse Predictability	X	X	
3.2.1 Mechanical Schematic			X
3.2.2 Electrical Schematic			X
3.2.3 RCS Mechanical Configuration			X
3.3.1.1 Subsystem Integration		X	X
3.3.1.2 Envelope		X	X
3.3.1.3 Plume Impingement		X	
3.3.1.4 Engine Alignment		X	X
3.3.2 Thermal Interface		X	
3.3.3.1 Power	X	X	
3.3.3.2 Telemetry & Command	X	X	
3.4.1 Proof & Burst Pressure	X	X	
3.4.2 Subsystem Leakage	X	X	
3.4.3 Servicing	X		X
3.4.4 Cleanliness	X		X
3.4.5 Pressurants	X		X
3.4.6 Useful Life	X	X	
3.4.7 Reliability	X	X	

TABLE VIII (CONTINUED)

METHODS OF COMPLIANCE WITH REQUIREMENTS

SPECIFICATION PARAGRAPH NUMBER AND SUBJECT	PROOF OF COMPLIANCE BY		
	TEST	ANALYSIS	INSPECTION
3.4.8 Weight	x	x	
3.4.9 Safety			x
3.4.10 Balance		x	x
3.5.1 Structural Environment	x	x	
3.5.2 Thermal Environment	x	x	
3.5.3 Electromagnetic Interference	x	x	

NOTE: ALL TESTING SHALL BE PERFORMED WITH QUALITY INSPECTION

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REACTION CONTROL SUBSYSTEM

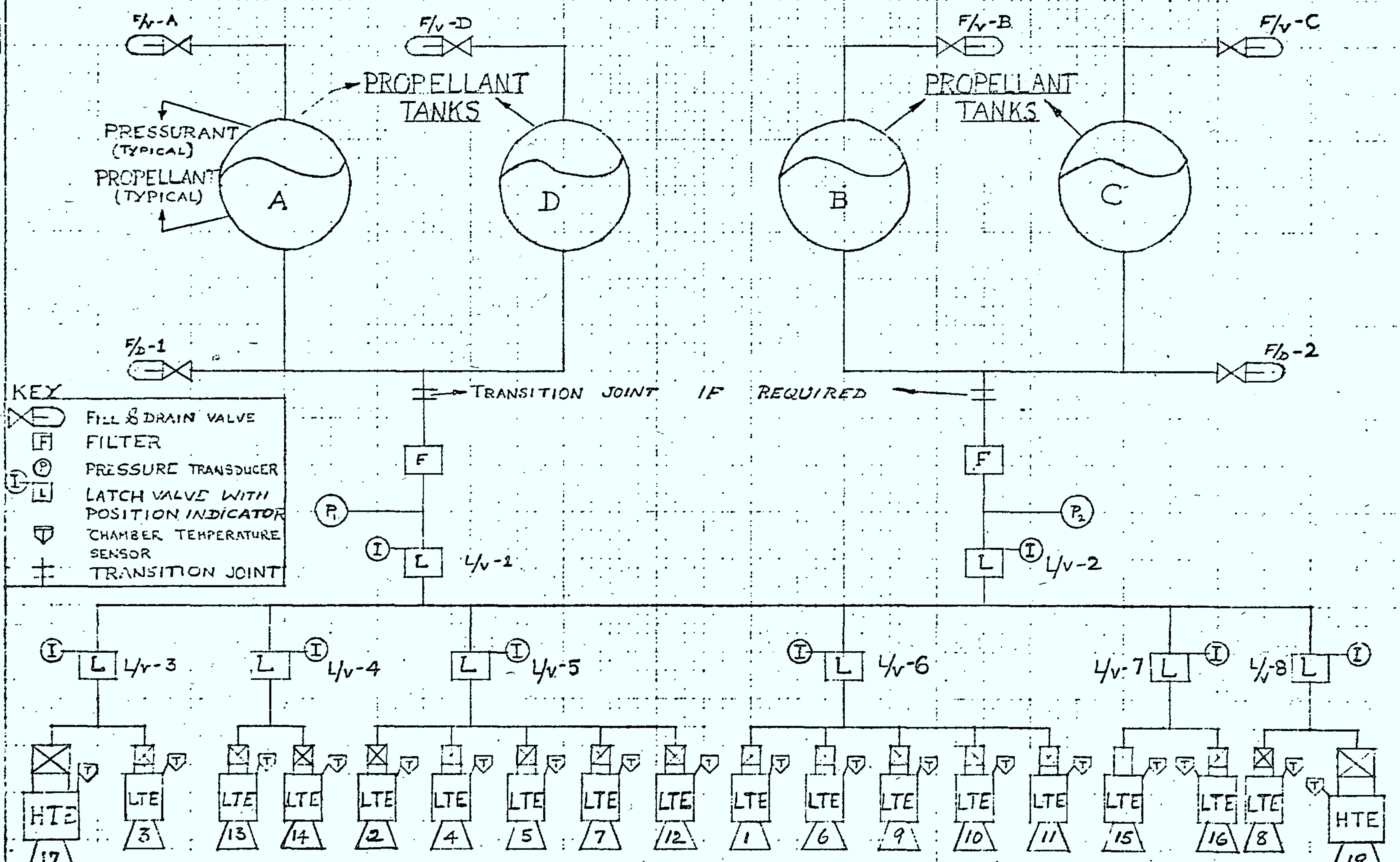
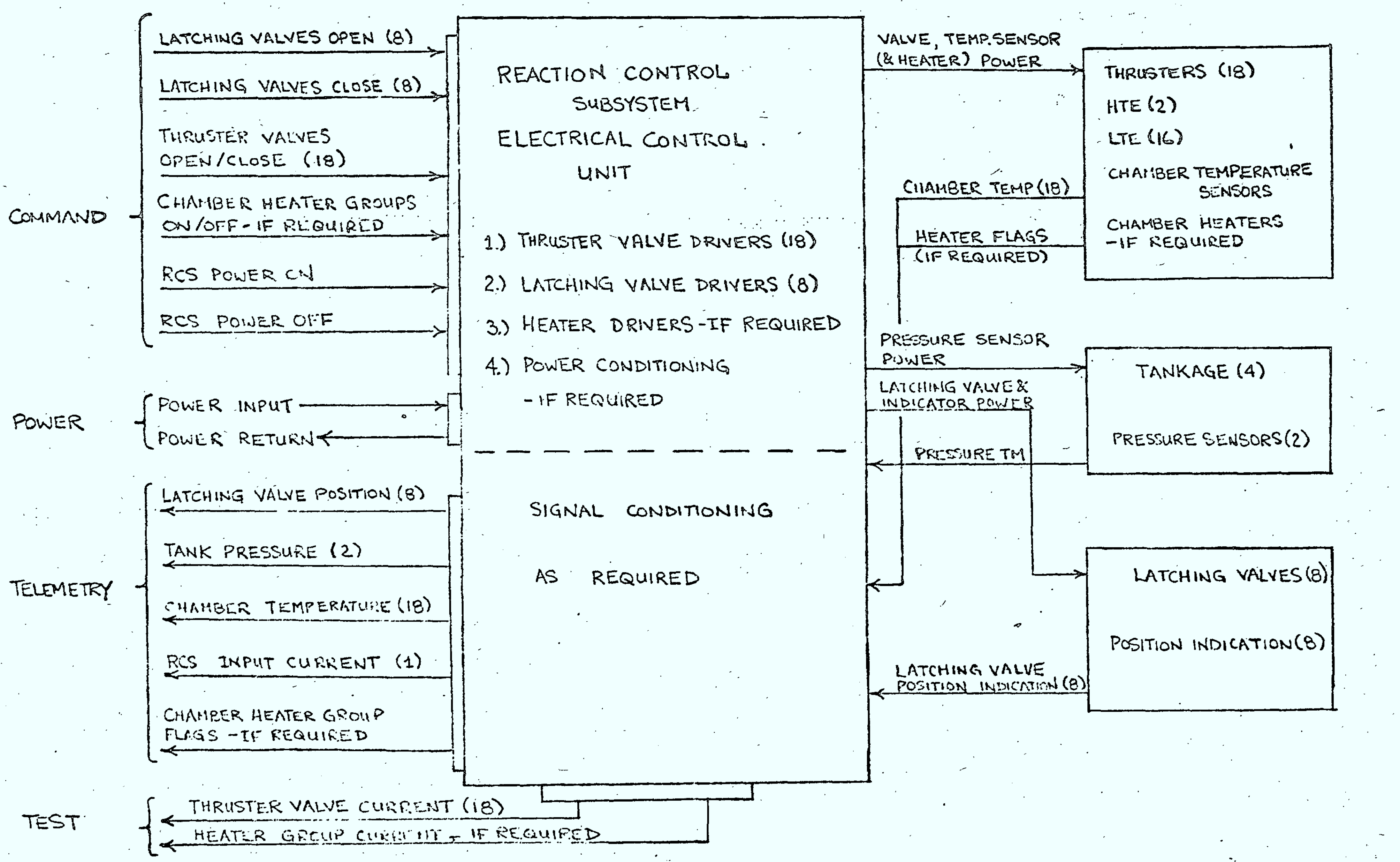


FIGURE 1 MECHANICAL SCHEMATIC DIAGRAM

FIGURE 2 RCS ELECTRICAL SCHEMATIC



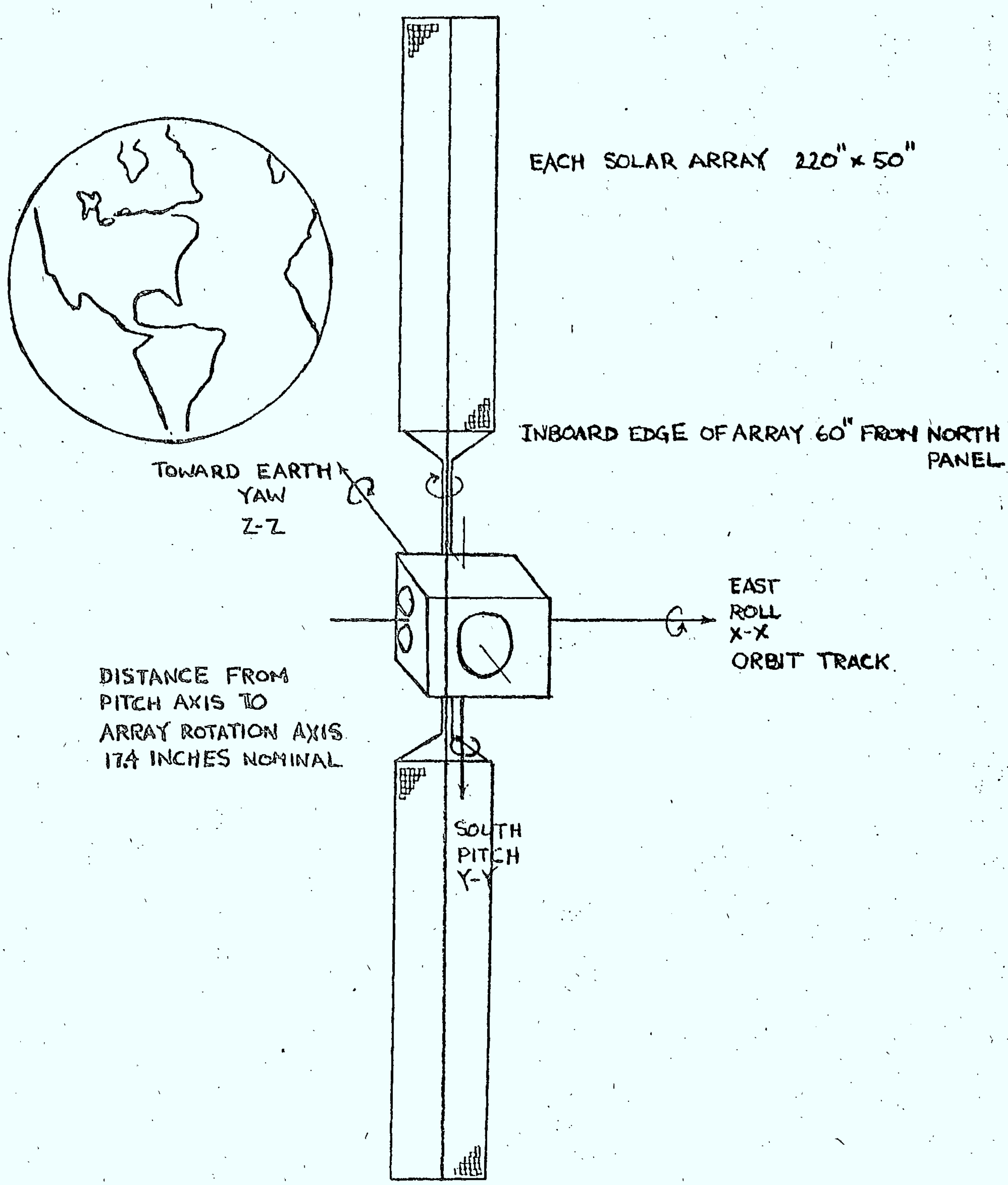
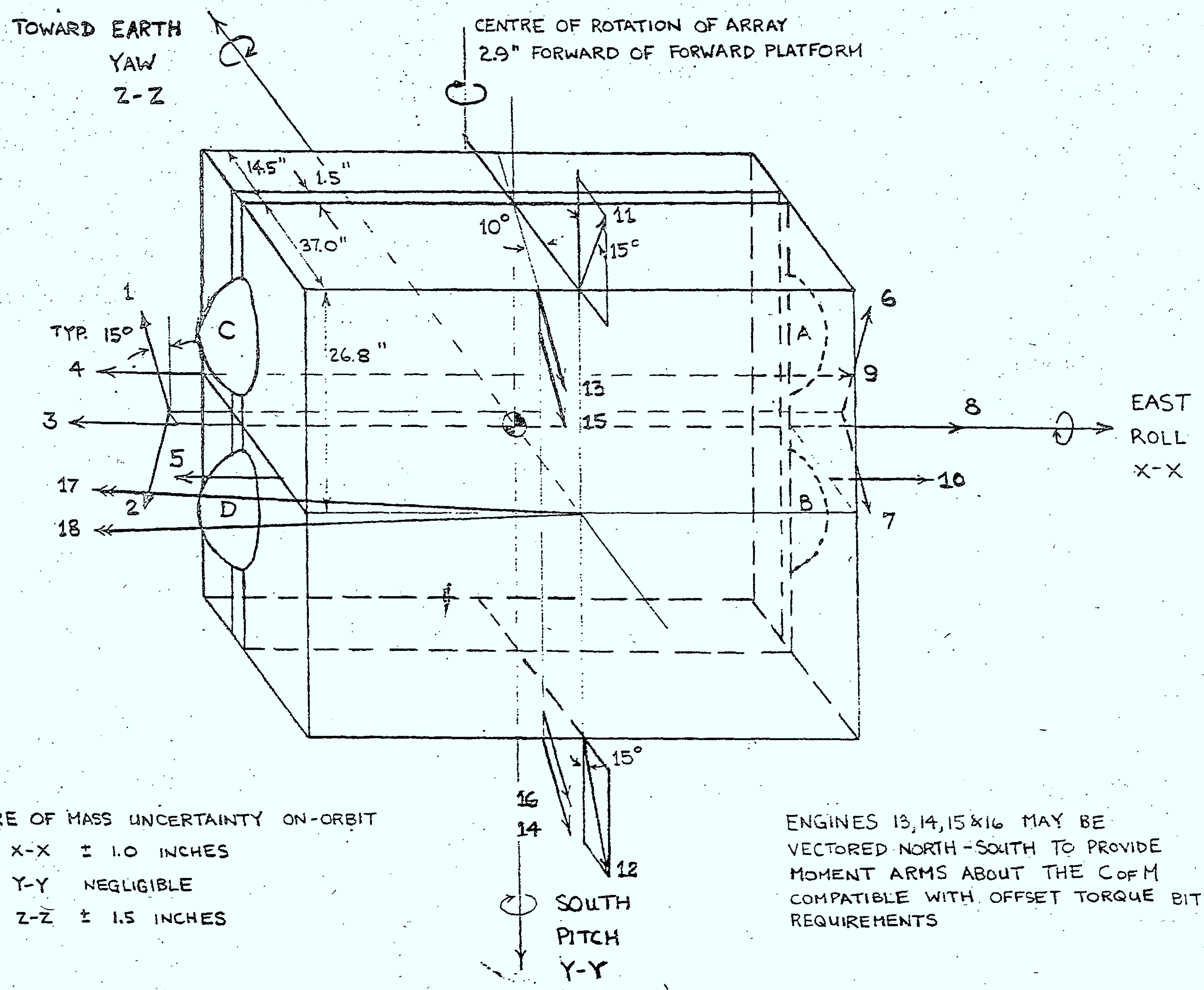


FIGURE 3 SPACECRAFT 3-AXIS STABILIZED CONFIGURATION



CENTRE OF MASS UNCERTAINTY ON-ORBIT

X-X	± 1.0	INCHES
Y-Y	NEGLIGIBLE	
Z-Z	± 1.5	INCHES

ENGINES 13, 14, 15 & 16 MAY BE VECTORED NORTH-SOUTH TO PROVIDE MOMENT ARMS ABOUT THE C of M COMPATIBLE WITH OFFSET TORQUE BIT REQUIREMENTS

FIGURE 4. RCS THRUSTER LOCATIONS & DIRECTIONS

PRELIMINARY

1.4 Deployable Solar Array Subsystem Performance Specification

CD/114

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SPAR

SPAR-SG.363

PERFORMANCE SPECIFICATION

DEPLOYABLE SOLAR ARRAY SUBSYSTEM

GENERAL-PURPOSE SATELLITE BUS

(FOR BUDGETARY PRICING PURPOSES)

Prepared by:........................
S.S. Sachdev
System Engineer,

Approved by:........................
P.A. McIntyre
Program Manager

May, 1975

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1.0 SCOPE

This is a performance specification for the Deployable Solar Array (DSA) Subsystem for use on the Canadian General Purpose Satellite Bus (GPB). The GPB, consisting of the DSA, Power, Telemetry and Command, Attitude Control, Reaction Control, Structure and Thermal Control Subsystems of a three-axis stabilized geosynchronous orbit spacecraft, is designed to cater for up to five different communications and experimental payloads.

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

2.0 APPLICABLE DOCUMENTS

TBD.

3.0 ASSEMBLY DEFINITION

3.1 General

For purposes of this specification, the DSA subsystem is divided into three basic assemblies. These are:

i) The Solar Cell Array
ii) The Stowage and Deployment System
iii) The Orientation and Power Transfer System

The DSA shall be stowed on and deploy from the spacecraft north and south surfaces and shall be rotated for sun orientation about a centreline forward of the spacecraft earth facing platform.

A functional block diagram of the subsystem is shown in Figure 1.

3.2 Solar Cell Array

The solar cell array shall consist of an assembly of n-on-p silicon solar cells with cover glasses, suitable interconnected in series and parallel and mounted on an insulating substrate. Flat conductor cable wiring shall be provided for transferring power from the solar cell strings along the array to the diode board interface on the stowage system. Design of the solar cell array shall be compatible with the stowage and deployment system, provision being made for attachment or bonding to the stowage system substrate and for stowage and deployment. The solar cell array shall be sized to meet the performance specified elsewhere in this document.

3.3 Stowage and Deployment System

The stowage and deployment system shall consist of the following:

i) A substrate for attachment/support of the solar cell array in the stowed, deploying and deployed modes.

ii) A tie-down/release system for restraining the stowed system during launch vibration, transfer and drift orbit spin and apogee motor firing, and for releasing the system ready for deployment after the spacecraft has been despun in synchronous orbit.

iii) A deployment system consisting of suitable interpanel hinges, torsion springs, deployment control mechanisms, electric motor and locking devices to achieve a smooth controlled deployment from the stowed to the fully deployed state.

iv) An elevation structure to place the in-board end of the deployed array at a sufficient distance from the spacecraft's north and south platforms to minimize UHF antenna shadowing and provide a sufficient thermal view factor as specified elsewhere in this document.

v) Diode boards, suitable terminal boards for interfacing with the solar cell blanket, flat conductor cable wiring for transferring power from the diode boards to the orientation and power system interface, and provision for installing a sun sensor to be used by the Bus Attitude Control subsystem for orienting the array.

The stowage and deployment system shall interface with the orientation and power transfer system and with the spacecraft power system as shown in the Functional Block Diagram, Figure 1.

3.4 Orientation and Power Transfer System

The array orientation and power transfer system shall be installed on the spacecraft forward or earth facing platform and interface with the array stowage and deployment system described above. It shall consist of the following:

i) Redundant drive mechanisms consisting of motors, bearings, position readout and support structure.

ii) Power transfer slip ring assemblies, and

iii) A torque tube or through shaft with appropriate support bearings and bracketry connecting the north and south arrays.

Interface connectors shall be provided at the non-rotating end of the power transfer assemblies for providing power to the deployment mechanism and analog sun sensor, array power to the spacecraft power subsystem, and deployment flag and analog sun sensor signals to the attitude control subsystem. Interface connectors shall also be provided at the drive mechanisms for controlling array rotation by the attitude control system electronics (see Figure 1, Functional Block Diagram, for clarification).

4.0 PERFORMANCE REQUIREMENTS

4.1 Life

The DSA shall survive all mission phases and ground testing (up to ten deployments before launch) and be capable of meeting the performance requirements specified elsewhere in this document after a maximum of ten transfer orbits and six years in synchronous orbital operation after having been stowed in a controlled environment for up to three years.

4.2 Power

4.2.1 Synchronous Orbit

The deployed DSA shall provide a minimum of 800 watts of DC power at 35 to 40 volts at the subsystem/ spacecraft interface (power transfer assembly connector) at the end of six years in synchronous orbit.

4.2.2 Transfer Orbit

The DSA shall have sufficient exposed solar cells in the stowed configuration to provide an average of 80 watts at 40 volts during the transfer and drift orbit phases of the mission.

4.2.3 Adaptability to Alternative Payloads

The design of the DSA shall allow the removal of at least one panel per side (two per spacecraft) to provide a minimum of 600 watts of power for an alternative payload. The transfer orbit capability shall not be compromised by this removal. Further, wiring of the in-board panels shall allow the use of only a sufficient number of solar cells to provide a minimum of 700 watts of power for another alternative payload.

4.3 Reliability

The DSA shall be designed to achieve an inherent mission reliability for the assembly described in Paragraph 3.0 of this specification of not less than TBD. Single point failure modes shall be minimized.

4.4 Weight

The weight of the DSA subsystem as defined in Paragraph 3.0 of this specification shall not exceed 125 lb for the 800 watt prime design and 104 lb for the 600 watt version.

4.5 Configuration and Size

4.5.1 Stowed

The DSA subsystem shall be stowed on the spacecraft north and south platforms. The north and south parts of the subsystems shall be symmetrical and interchangeable. Typically, four panels on each side, 50 inches wide will be folded concertina fashion and held down to the spacecraft structure at a minimum of four points. The outer surface of the top panel shall be used for providing power during the spinning phase. When stowed, the subsystem shall be within the 86 inches diameter dynamic envelope of the Thor Delta 3914 launch vehicle. The distance between the outer surfaces of the spacecraft north and south platforms is 53.5 inches.

The orientation and power transfer system shall be mounted on the outside surface of the spacecraft forward (earth facing) platform.

A typical configuration is shown in Drawing Number 31138J2.

Restriction in length of the stowed array (in direction of launch vehicle thrust axis) is placed by the spacecraft antenna on the forward platform - distance TBD from forward platform - and the limitations of the launch vehicle adapter as defined in the Delta Restraints Manual.

4.5.2 Deployed

When fully deployed, the rotational axis of the DSA shall be parallel to the spacecraft pitch axis (north/south). The in-board end of the DSA shall be elevated a distance of approximately 60 inches from the spacecraft platform to minimize shadowing of solar cells by the spacecraft UHF antenna. The elevation structure shall present a maximum thermal view factor between the spacecraft north and south heat dissipating platforms and array of less than 0.10.

4.6 Stowage and Release Requirements

4.6.1 Support and Attachments

Tie-down points for the stowed DSA shall be established in conjunction with the Design Authority and structure subsystem contractor. A minimum of four points on each platform shall be used.

4.6.2 Protection of Solar Cells

The stowage system shall be designed to provide a sufficiently benign environment to ensure no solar cell or coverglass breakage or other damage to the solar cell blanket occurs when the subsystem is subjected to the environment specified in Paragraph 4.11 of this specification.

4.6.3 Release System

The release system which will be pyrotechnically activated shall be designed to have no single point failure modes. The pyrotechnic firing circuitry will be provided by the spacecraft power subsystem (see Block Diagram, Figure 1).

A microswitch, or similar device, shall provide a flag signal to the spacecraft ACS to indicate successful release.

4.7 Deployment Requirements

The DSA shall deploy on command in a controlled, repeatable and predictable manner to its full extension in both synchronous orbit and in a suitably designed ground test fixture. The deployed array shall nominally be in a plane perpendicular to the spacecraft north and south platforms.

The rate of deployment, acceleration and deceleration at the end of deployment shall be so as not to damage the DSA subsystem.

A microswitch or similar device with associated electronics shall be provided to automatically shut-off the DC motor once deployment is completed. Simultaneously, a flag signal shall be provided to ACS to indicate successful deployment. During the period of deployment, no ACS thrusters will be fired, and the spacecraft will have been despun to essentially zero rates about all axes.

4.8 Deployed State Requirements

4.8.1 Stiffness and Free Play

When fully deployed, there shall be no free play in the joints of the DSA. The natural frequency of one half of the subsystem, i.e., one wing, shall be greater than 0.15 Hz assuming a constrained root.

4.8.2 Deflected Shape

The tip of the deployed array shall be within $\pm$ 2" of the nominal centreline of the root, allowing for dimensional tolerances and thermal bending. This may be achieved by adjustments on the space-craft. As a design goal, a deflection of $\pm$ 1" shall be attempted.

The twist of the tip of the array relative to the base shall be limited to $\pm$ 1°.

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4.8.3 Magnetic Moment

The solar array circuitry shall be arranged to such that the magnetic moment about the spacecraft centre of mass shall not exceed 0.5 X 10^{-6} ft/lb.

4.9 Orientation System Requirements

4.9.1 Rotation Rates

The orientation system shall be capable of rotating the deployed solar arrays at two rates; 1 revolution per day and 15 degrees per minute.

These rates will be commanded and controlled by the spacecraft Attitude Control System.

4.9.2 Configuration and Redundancy

There shall be no single point failure modes in the orientation system. To meet the reliability requirements, redundant mechanisms or at least redundant motors will be required. Only one mechanism shall operate both arrays by means of a through shaft or torque tube.

4.10 Power Transfer System Requirements

The power transfer assembly or assemblies shall be capable of transferring up to 1200 watt of DC power (Beginning of Life) at 40 volt from the rotating solar arrays to the stationary spacecraft. In addition, the assembly shall have a sufficient number of rings to transfer analog sun sensor and deployment mechanism power and signal voltages as shown in the block diagram. These are:

i) $\pm$ 15 V power to the sun sensor

ii) 0 - 5 V analog signal from the sun sensor

iii) +28 VDC power to the deployment mechanism

iv) 5 V flag signal from the deployment system

The above list is for each array.

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Slip ring noise shall not exceed 100 mV peak-to-peak for the power rings, 500 mV peak-to-peak for the +28 VDC power ring and 10 mV peak-to-peak for the signal rings, at test voltages and currents 1.5 times the required design voltages and currents.

4.11 Power Consumption

The average power required by the DSA during synchronous orbit shall not exceed 3.5 watt. Peak requirements up to 20 watt during the fast slew mode are allowed. This does not include sun sensor or orientation electronics requirements which are part of the ACS.

Power required during deployment shall not exceed 22 watt average.

Power required by the release system shall be less than 560 watt for 10 millisecond (four pyros, 5A each at 28 V).

4.12 Environment

The vibration, shock, acceleration, acoustic, humidity, thermal and radiation environments which the DSA must survive are given in the "Bus Specification", SPAR-SG.359.

4.13 Lubrication

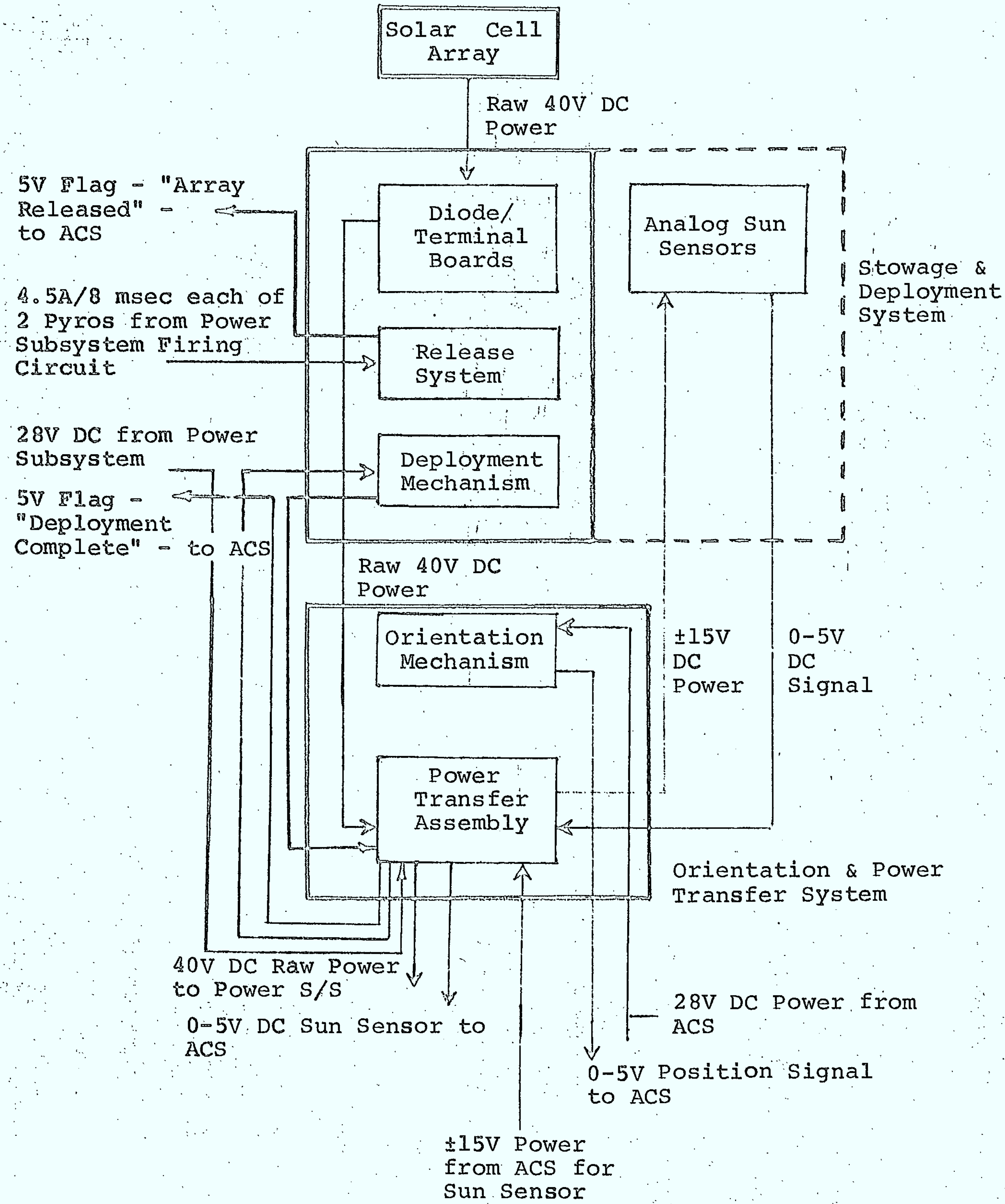
All moving joints and parts shall be lubricated with an acceptable space-proven lubricant system.

4.14 Electromagnetic Compatibility

The EMC requirement is specified in the Bus Specification, SPAR-SG.359.

FIGURE 1 - FUNCTIONAL BLOCK DIAGRAM

DEPLOYABLE SOLAR ARRAY SUBSYSTEM GPB



1.5 Structure Subsystem Performance Specification

MULTIPURPOSE BUS STUDY

SPECIFICATION REQUIREMENT FOR

STRUCTURE SUBSYSTEM (FOR

BUDGETARY ESTIMATES)

Prepared by: E. Quittner ....
for H. Borduas
Structural Engineer

Approved by: E. Quittner ....
E. Quittner
Group Leader
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Approved by: McIntyre ....
P. McIntyre
Program Manager

1.0 SCOPE

This specification summarizes the requirements to design, analyze, manufacture and test a structure for a communications satellite whose weight at lift-off on a Thor Delta 3914 launch vehicle is 2000 lbs +10% and whose specified life time in geosynchronous orbit will be eight years.

2.0 OTHER DOCUMENTS

Listed are documents which form part of this summary specificaton:

DAC61687	-	Delta Spacecraft Design Requirements
AFE TRM 127-1	-	Range Safety Manual
GSFC Spec S320-G-1	-	General Environmental Test Specification for Spacecraft Components
Fed Std 209	-	Clean Room and Work Station Requirements
NASA Spec NHB 5300.4 (1B)	-	Quality Assurance Requirements for Space Systems
MIL-HDBK-3A	-	Structural Sandwich Composites
MIL-HDBK-5A	-	Metallic Materials for Aerospace Vehicle Structures
MIL-D-1000	-	Military Drawing Standards
TBD	-	Spacecraft Payload and Mass Properties Configuration

3.0 DESIGN REQUIREMENTS

3.1 Configuration

- The structure shall provide the suitable interface for attaching the spacecraft to a Delta 3914 launch vehicle via a Delta 3731A adapter.

- The overall dimensions of the structure shall be such that the spacecraft on the launch vehicle shall lie within the dynamic envelope of the fairing.

- The structure design shall allow easy access to payload: equipment panels shall be easily removable from the primary thrust structure.

- In the launch configuration, the installation and removal of the apogee motor shall be facilitated, provision shall be made to fill, drain and pressurize the RCS tanks.

- The structure weight shall be minimized, 115 lbs maximum shall be used as a design goal.

- During transfer orbit the spacecraft will be spin stabilized and must have a favourable inertia ratio around the thrust axes, i.e. the roll to pitch inertia must be greater than 1.1 (assuming the roll axis to be thrust axis).

3.2 Design Loads

3.2.1 Definitions

Limit Loads, or 3 flight loads, are derived from the environment encountered by similar weight spacecrafts on the same launch vehicle.

Acceptance test loads are the loads imposed on the subsystem during acceptance testing. They are equal to limit loads.

Qualification test loads are applied to the subsystems for qualification.

Quasi-static loads:

Qual. loads = 1.25 x limit loads

Sinusoidal vibration loads:

Qual. loads = 1.50 x limit loads

Random vibration loads:

Qual power spectral density (PSD) = 2.25 x limit PSD

Qual. g RMS = 1.50 x limit g RMS

3.2.2 Safety Factors and Margins of Safety

Safety factors shall be applied to the stresses calculated from qualification loads. These shall meet the following criteria.

Yield Safety Factor	- 1.1	- no yielding
Ultimate Safety Factor	- 1.25	- no failure

Margins of safety shall be calculated using the most adverse combination of stresses and other environmental factors.

The Margin of Safety is defined as $M.S. = \frac{1}{R} - 1.0$

where R is the stress ratio that results when the designed stress is compared to the material or ultimate stress.

For all cases the M.S. shall be greater than zero.

3.2.3 Quasi-Static Levels

Mission Condition		3 Limit Load (g's)	Qualification Test Load (g's)
Liftoff	Axial	3.1, -1.2	3.9, -1.5
	Lateral	2.2	2.8
Meco	Axial	12.8	16.0
	Lateral	0.8	1.0
2nd Stage	Axial	TBD	
	Lateral		
3rd Stage	Axial	10.1	12.6
	Lateral	0	0
Apogee Motor	* Axial	+8	+10.0
	Lateral	.33	.4
Spin		66 RPM	120 RPM

3.2.4 Vibration Levels

3.2.4.1 Sinusoidal Vibration

The following qualification level inputs are to be applied at the base of the spacecraft/launch vehicle adapter.

i) Thrust Axis Input (Z direction)

Frequency Hz	g (0-Pk)
5-10	0.4" D.A.
10-15	2.3
15-21	6.8
21-250	2.3 (Notching may be considered)*
250-400	4.5
400-2000	7.5

Sweep rate: 2 octaves/minute

ii) Lateral Axis Input (X and Y directions)

Frequency Hz	g (0-Pk)
5-10	0.4" D.A.
10-12	2.0
12-250	1.5 (Notching may be considered)**
250-400	4.5
400-2000	7.5

Sweep rate: 2 octaves/minute

* A 16g qual. overall thrustwise response of the major spacecraft masses (North and South panels and payload, apogee motor) will not be exceeded.

** A 3g qual. lateral response of the spacecraft at its centre of mass will not be exceeded.

NOTE Flight acceptance input g's will be 2/3 of above values, and sweep rates will be 4 octaves/minute

3.2.4.2 Random Vibration

The qualification level inputs to be applied at the base of the spacecraft/launch vehicle adapters, along the X, Y and Z directions (separate tests) are:

20-350 Hz	:	+4 db/octave
350-700 Hz	:	.09 g^2/Hz
700-2000 Hz	:	-3 db/octave
Overall level	:	10.65 g_{RMS}

Duration: 2 minutes

Note Flight acceptance inputs (g^2/Hz) will be 4/9 of above values, overall g_{RMS} level 2/3 of above, and test duration will be 1 minute.

3.3 Structural Stiffness

For the spacecraft in the launch configuration, hard mounted at the separation plane (without the third stage adapter) the following natural frequency criteria will be met:

i) along the axial (thrust) axis the natural frequency of the structure shall be above 35 Hz.

ii) along the lateral axis the natural frequency shall be above 15 Hz.

iii) the lowest torsional natural frequency shall be above 35 Hz.

Prediction analysis will also be required for the axial and lateral natural frequencies with the spacecraft mounted on the third stage adapter, using stiffness values for the adapter which will be supplied by the L.V. agency.

3.4 Deformation

- The structure shall provide a stable environment for all equipment payloads. For dynamically inactive units hard-mounted to a structure panel, the qualifications level environment shall not exceed TBD g under sinusoidal or random vibration qualification inputs applied at the base of the spacecraft/launch vehicle adapter.

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- No permanent deformation shall ensue from qualification level vibration or thermal environments. Also, deformations will not be sufficient to produce interference between parts.

- Thermal distortion due to temperature gradients across the structure will be minimized by a judicious choice of materials and design features.

- The alignment of units which require a high degree of pointing accuracy, will be kept within a tolerance of ±TBD °.

The primary structure (thrust tube) will be designed in conjunction with the apogee motor to provide correct alignment of the spacecraft to facilitate correct apogee injection into synchronous orbit and also to provide correct alignment of the spacecraft to the launch vehicle third stage adapter.

4.0 ANALYSIS REQUIREMENTS

A computer model of the structure and payload shall be used to establish major natural frequencies of the spacecraft and to estimate design loads for the structure and for the payload when the spacecraft is subjected to sinusoidal vibration applied at the base of the spacecraft/launch vehicle adapter.

For local platform resonances, a value of c/c_{crit} of 0.025 (Q = 20) will be assumed for design purposes.

All structural components will be stress analyzed and tables of Margins of Safety, for yield and ultimate loading levels, will be prepared.

A thermal distortion analysis of the complete spacecraft structure will be carried out for temperature distribution cases considered critical.

5.0 TEST REQUIREMENTS

The spacecraft will be qualification tested in vibration to the levels given in Section 3.2.4. Personnel will be made available to carry out the following tasks.

i) Produce test prediction analysis for the test cases as defined.

ii) Support the test.

iii) Analyze results and prepare a comprehensive test report.

The contractor will also carry out a static test on the primary structure to quasi-static loads as defined earlier; this test will be carried out as early as possible in the program. In addition, development tests required to substantiate novel design features, mechanisms, attachments, etc. will be carried out and reported on.

6.0 PRODUCT EFFECTIVENESS

The contractor is to maintain a reliability and quality assurance program in accordance with the military specifications as supplied and as listed below.

6.1 Quality Assurance

A contractor must conform to NASA Specification N.H.B. 5300.4 (1B) or its equivalent plus have government source inspectorate. All materials, properties and failures (both in-house and at vendors) must be monitored and reported on.

6.2 Reliability

The structure subsystem must be capable of surviving the launch environment indicated and have a life span of 6 years in synchronous orbit and a reliability figure of .99. A failure mode and effects analysis will be carried out on items that are considered mission critical.

6.3 Materials and Processes

The contractor must prepare materials and process specifications for any special finishes and applications which are not covered by standard military specifications. In addition any adhesives or potting compounds used must have low outgassing properties in the environment associated with synchronous orbit.

1.6 Thermal Control Subsystem Performance Specification

SPAR-SG.364

MULTI-PURPOSE BUS STUDY - GENERAL SPECIFICATION

THERMAL SUBSYSTEM

FOR BUDGETARY PRICING PURPOSES

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Thermal Group

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Program Manager

May, 1975

GENERAL PURPOSE BUS-THERMAL SUBSYSTEM

GENERAL SPECIFICATION

1.0 SCOPE

This document summarizes the performance, design, analysis, manufacture and test requirements for a thermal control subsystem for a three-axis stabilized communications spacecraft operating at geo-synchronous altitude for eight years, having passed through a spin (transfer/drift) orbit phase for a period of 20 days.

2.0 APPLICABLE DOCUMENTS

Documents which also form part of this specification include:

DAC61687:	Delta Spacecraft Design Restraints.
Fed.Std. 209:	Clean Room and Work Station Requirements.
MIL-D-1000:	Military Specification for Engineering Drawings.
AFE TRM 127-1:	Range Safety Manual.
GSFC Spec. S320-G-1:	General Environmental Test Specification for Spacecraft Components.
NASA Spec. NHB.5300.4(1B):	Quality Assurance Requirements for Space Systems.
MIL-HDBK-5A:	Metallic Materials for Aerospace Vehicle Structures.
TBD:	Spacecraft Payload and Mass Properties Configuration.

D4/CAK/1

TBD: Requirements Spacecraft, Subsystem and Component Environmental Design and Test.

3.0 PERFORMANCE REQUIREMENTS

3.1 Thermal Performance Requirements

The thermal control subsystem for the general purpose bus shall maintain spacecraft component temperatures for all identified payloads within the specified design absolute temperature limits, and maintain inter-component temperatures within the requirements for minimization of structural distortion/pointing inaccuracy.

3.2 Weight Requirements

The general purpose bus thermal subsystem shall be designed for minimum weight. The total in-orbit weight of the thermal subsystem components (including sensors, heaters and thermostats but excluding any other power and electrical control components) shall not be greater than 25 lbs.

3.3 Power Requirements

The general purpose bus thermal subsystem shall be designed from minimum power. The total in-orbit power requirement of the thermal subsystem components (including sensors, heaters and thermostats, but excluding any other power and electrical control components) shall not be greater than 25 watts. Eclipse power requirements for the subsystem shall not be greater than 10 watts.

3.4 Operability Requirements

3.4.1 Reliability

The general purpose bus thermal subsystem shall be designed for maximum reliability, consistent with meeting specified design, performance and interface requirements. The reliability of the subsystem shall be no less than 99.

D4/CAK/2

3.4.2 Useful Life

The general purpose bus thermal subsystem shall be capable of meeting specified design/performance/ interface requirements within the specified mission for a period of not less than seven years after being stored for a period of up to three years and subjected to representative ground environments.

4.0 DESIGN REQUIREMENTS

4.1 The thermal subsystem shall employ passive thermal control materials, if possible, without the use of heat pipes or louvres. However, an optimization study shall be performed to determine the most cost/weight/reliability effective approach.

The thermal subsystem components shall continue to function without degradation below their specified performance when subjected to the specified ground, launch pad and flight environments.

The design shall accommodate the identified spacecraft payload characteristics and locations given below:

The main synchronous orbit dissipating components, i.e. the transponder components will be located on the north/south platforms of the spacecraft. In addition, as many of the high-powered housekeeping components as possible will also be located on these minimum solar load platforms. (Batteries will be located on these panels but will require separation from other spacecraft components in order to meet their maximum design temperature limit of 10°C).

Remaining housekeeping components will be located on convenient platforms/structural locations.

The antenna farm together with most of the attitude control subsystem sensors will be located on the forward (earth facing) platform.

Component dissipations and modes of operation will be defined at program commencement. Typical

D4/CAK/3

payload options and component dissipations and design/temperature limits are given in Tables 1 and 2 respectively. The following general statements shall also apply:

a) The minimum power dissipated in the spacecraft interior during the spin (transfer/drift orbit) phase will be 80 Watts, power being supplied by body solar arrays during sunlight, and batteries during eclipse.

b) The ratio between maximum and minimum power dissipated interior to the spacecraft during synchronous orbit will be in the order of 4:3; power being supplied by deployed solar arrays during sunlit periods and batteries during eclipse.

4.2 Mechanical Design Requirement

4.2.1 Strength

The thermal subsystem components shall be designed with sufficient structural strength and rigidity or with sufficient bond and tear strength to survive without degradation below specified performance, the loads induced by ground handling, launch vehicle vibration, acceleration and decompression environments.

4.2.2 Venting

The thermal subsystem shall be designed to permit venting of the entire spacecraft during the launch phase and to permit sufficient outgassing to meet component operating pressure requirements.

4.2.3 Outgassing

The thermal subsystem materials shall meet space standard outgassing criteria from material acceptibility. These are:

i) less than 1% total weight loss during exposure to a temperature of 125°C for 24 hours at a pressure of less than 10^{-6} Torr.

D4/CAK/4

ii) During the above exposure, less than 0.1% volatile condensed materials (VCM) collected on a plate cooled to 25°C.

4.2.4 Accessibility

The thermal subsystem shall be designed so as upon removal of multilayer insulation blankets, not to restrict access to the spacecraft components, and shall be designed for compatibility with spacecraft mechanical design interface requirements.

4.3 Electrical Design Requirements

4.3.1 Grounding

All thermal subsystem metallized blanket surfaces shall be electrically grounded to the spacecraft structure and shall have a maximum resistance from any point on the metallized surface of any blanket layer to the spacecraft structure of not more than 10 ohms.

All other thermal hardware materials shall meet standard grounding requirements.

4.3.2 Electro Magnetic Compatibility

The thermal subsystem shall comply with the electromagnetic compatibility requirements.

5.0 DESIGN AND PERFORMANCE VERIFICATION

The adequacy of the general purpose bus thermal design, to meet the specified requirements shall be substantiated by analysis and test.

5.1 Analysis

The contractor shall as a minimum perform the following analysis to substantiate the design.

5.1.1 Identification of all critical mission phases for all spacecraft components.

D4/CAK/5

5.1.2 Generate maximum and minimum temperature predictions for all spacecraft components for the above identified critical mission phases. As a minimum these shall include:

- launch phase fairing and aero heating,
- parking orbit pre-spinup conditions,
- transfer obrit maximum and minimum temperature cases,
- drift orbit maximum and minimum temperature cases,
- acquisition sequence,
- synchronous orbit BOL and EOL, maximum and minimum temperature cases.

Predictions will:

a) Assume isothermal components unless otherwise specified.

b) Assume design temperature limits to apply at component mounting surfaces unless otherwise specified.

c) Consider reasonable variation (e.g. measurement, uncertainty, degradation, sample variation etc.) in all thermal parameters including:
   - solar absorbtance,
   - IR emittance,
   - solar constant,
   - contact conductance,
   - honeycomb conductance.

d) During maximum temperature cases the designer shall assume the values of all parameters contributing to maximum spacecraft temperatures and vice versa in the minimum temperature cases.

5.2 Test

As a minimum the following test shall be performed:

5.2.1 Spacecraft thermal balance test performed in solar thermal vacuum with controlled boundary temperatures

D4/CAK/6

for non-representative/missing hardware, demonstrating under simulated environmental conditions that the thermal design of the spacecraft is adequate as demonstrated by comparison of test predictions and test data and by existence of positive design margin when the latter is corrected to the flight environment.

1.7 Apogee Motor Performance Specification

ISSUE

MULTIPURPOSE BUS STUDY

SPECIFICATION REQUIREMENTS

APOGEE KICK MOTOR

PRELIMINARY, FOR BUDGETARY ESTIMATES

PREPARED BY Staff

APPROVED BY D.A. Staley
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Program Manager

APPROVED BY

REASON FOR REISSUE

DATE

AMENDMENTS			
NUMBER	REASON FOR AMENDMENT	SIGNATURE	DATE

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MULTIPURPOSE BUS STUDY SPECIFICATION

REQUIREMENTS, APOGEE KICK MOTOR

PRELIMINARY, FOR BUDGETARY ESTIMATES

Prepared by:....................
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Approved by: D. A. Staley
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7/CS/01

May 6, 1975

TABLE OF CONTENTS

1.0 SCOPE

This specification establishes the design and test requirements for a solid propellant rocket motor to be used to inject a Satellite into a circular synchronous earth orbit at the apogee of an elliptical transfer orbit.

2.0 APPLICABLE DOCUMENTS

The following documents, of the exact issue shown or of the most recent issue when the date of issue is not shown, form a part of this specification.

MIL-STD-453	Inspection, Radiographic

U.S. Air Force

AFETRM 127-1	Range Safety Manual
AFM 127-100B	Explosive Safety Manual

3.0 REQUIREMENTS

3.1 General

The motor described by this specification shall be a product which meets the requirements of Section 3 and has passed all of the examinations and tests specified in Section 4.

3.2 Components

The motor shall consist of the following major components:

a) Case and insulation (chamber).
b) Expansion nozzle, including throat insert.
c) Ignition system consisting of igniter, pyrotechnic train, and safe and arm device.
d) Solid propellant.
e) Motor attachment flange.

3.3 Performance Requirements

3.3.1 General

The motor shall have the following performance characteristics based upon ignition and operation at vacuum conditions during and after exposure to the applicable environments.

3.3.2 Total Impulse

The total impulse of the motor in vacuum at +20°C (68°F) shall provide a nominal velocity increment of 6024 feet per second to a spacecraft that has an initial pad lift-off weight of 1925 pounds, including the apogee motor.

3.3.3 Total Impulse Reproducibility

The total impulse for all motors fired for this program shall be such that the resultant velocity increment is within ±1%, of the nominal over the operating temperature range (3.4.2).

3.3.4 Maximum Thrust Level

The maximum instantaneous thrust of the motor when fired over the operating temperature range shall be such that a spacecraft axial acceleration of 10.0 g's is not exceeded.

3.3.5 Thrust Rise Rate

Ignition shock and thrust shock on the spacecraft shall be minimized.

3.3.6 Ignition Time

The ignition time over the operating temperature range shall not exceed 0.300 second. The variation in ignition time over the operating temperature range shall not exceed ±0.075 second.

3.3.7 Motor Vibration

There shall be no shift in pressure or thrust level due to oscillatory burning. The following vibratory g levels shall not be exceeded at the motor attachment flange when the motor is fired:

Frequency	g level zero to peak
10-250	TBD
250-2000	TBD

There shall be no inflection in pressure indicating a momentary increase in motor static pressure during motor tail-off.

The above shall be verified by analysis and test.

3.3.8 Maximum Case External Temperature

A goal of the motor design shall be to assure that spacecraft heating is minimized. Emphasis shall be placed on minimizing heat transfer through the motor flange to the spacecraft. When fired in a thermal environment which reproduces the mission environment, the motor flange to spacecraft interfaces shall not exceed 150°F (66°C). The case external temperature shall not exceed a temperature which would cause insulation debonding. A maximum case external temperature of 260°C (500°F) after burnout is desirable. However, if analysis shows, to the Design Authority's satisfaction, that 370°C (700°F) maximum temperature is more effective from an overall spacecraft weight, cost and reliability point of view, then the higher temperature may be used. The exterior of the motor and exit cone shall be assumed to be perfectly insulated for thermal calculations on maximum case external temperature.

3.3.9 Motor Failure

There shall be no functional failure of any component with the motor conditioned within the operating temperature range during motor firing. Failures under this category include, but are not limited to the following:

a) Failure of the igniter to ignite the propellant.
b) Dislodgement of the nozzle throat insert or the exit cone, or any other part.
c) Burn-through or rupture of the motor case or closures.
d) Burn-through at the nozzle or igniter seals.

3.4 Environmental Requirements

3.4.1 General

The motor shall be capable of meeting the requirements of this specification after being subjected to any reasonable combination of the environmental conditions encountered during all phases in the motor's life as defined in Spacecraft Environmental Design and Test Specification requirements mentioned herein.

3.4.2 Operating Temperature

The motor shall meet the requirements of this specification over the operating temperature range of -7°C (20°F) to +38°C (100°F). The motor shall operate satisfactorily with a 16.5°C (30°F) temperature gradient across the propellant grain web. The motor shall also perform satisfactorily with the aft six (6) inches of the exit cone, including prediction uncertainty, ranging from -127°C (-197°F) to +82°C (180°F).

3.4.3 Flight Conditions

The motor shall meet the requirements of this specification in vacuum while oriented in any attitude.

3.4.4 Spin Condition

The motor shall meet the requirements of this specification while spinning at any rate between 50 and 110 rpm.

3.4.5 Time in Space

The motor shall meet the requirements of this specification after exposure to the transfer orbit space environment for a period of ten (10) days prior to motor firing. The transfer orbit perigee altitude is 100 $\pm$9 n.m. The transfer orbit apogee altitude is 19323 $\pm$590 n.m.

3.4.6 Acceleration

3.4.6.1 General

Limit loads imposed by the launch vehicle shall consist of the simultaneous application of quasi static thrust and lateral loads. Two flight conditions are structurally significant, and the apogee motor design must be capable of meeting both of them.

Flight Condition	Direction of Loads	Flight Level	Qualification Level	Design Yield Load	Design Ultimate Load
Liftoff	Axial	+2.9 -1.0	+4.35 -1.50	+5.23 -1.80	+6.53 -2.25
	Lateral	2.0	3.0	3.60	4.50
POGO	Axial	+13.0	+16		
	Lateral	0.80	1.0	1.20	1.50

Note that plus (+) indicates compression and minus (-) indicates tension for thrust axis accelerations. Lateral loads may occur in any direction in the lateral plane. All loads are to be applied at the C.G. of the apogee motor.

3.4.6.2 Motor Firing Loads

Motor firing loads shall consist of the combination of acceleration during burning as imposed by the motor thrust and 110 revolutions per minute maximum spin rate.

3.4.7 Vibration

3.4.7.1 Sinusoidal Vibration Levels

The qualification level input spectra of the spacecraft at the launch vehicle/adapter interface are as follows:

S/C Thrust Axis		S/C Lateral Axes	
Frequency (Hz)	Amplitude g's (0-PK)	Frequency (Hz)	Amplitude g's (0-PK)
5-10	0.4" D.A.	5-10	0.4" D.A.
10-15	2.3	10-14	2.0
15-21	6.8	14-250	1.5
21-250	2.3	250-400	4.5
250-400	4.5	400-2000	7.5
400-2000	7.5		

Sweep is upwards only at 2 octaves/minute in each of three orthogonal axes.

The spacecraft has been dynamically modelled, and preliminary analytical results indicate that the following qualification test spectra are appropriate inputs to the apogee motor at the apogee motor attachment flange:

a) Input at Apogee Motor Flange

S/C Thrust Axis		S/C Lateral Axes	
Frequency (Hz)	Amplitude g's (0-PK)	Frequency (Hz)	Amplitude g's (0-PK)
5-28	0.4" D.A.	0-15	0.4" D.A.
28-60	15.0	15-400	4.5
60-250	2.3	400-2000	7.5
250-400	4.5		
400-2000	7.5		

An acceptable procedure when inputting the above spectra would be to monitor the maximum vibration input by controlling the response of the C.G. of the apogee motor. It this approach is taken, the above spectra should be preprogrammed for input to the apogee motor, however an accelerometer located at the C.G. of the motor and recording response in the same direction as input, could limit the preprogrammed input. Consequently, within the frequency regions prescribed below the response of C.G. of the apogee motor need not exceed:

b) Maximum Response Required at Apogee Motor C.G.

S/C Thrust Axis		S/C Lateral Axes	
Frequency (Hz)	Amplitude g's (0-PK)	Frequency (Hz)	Amplitude g's (0-PK)
32-70	20.0	14-30	3.0
70-100	7.5	30-150	2.0
100-250	2.3	150-250	1.5

3.4.7.2 Random Levels

The random vibration levels at the base of the adapter are as follows:

Frequency (Hz)	Level (g^2/Hz)	Overall
20-300	.0029 to 0.045 increasing from 20 Hz at rate of +3dB/ octave	9.2 g rms
300-2000	0.045	

All axes - 2 minutes ±10° per axis.

The random vibration input shall be equalized such that the power spectral density is within ±3 dB of the specified levels everywhere in the frequency band and the overall gRMS level is within ±10% of that specified.

3.4.8 Humidity

The motor shall be designed to operate in an environment of up to 65 percent relative humidity at -7 to +38°C (20 to 100°F).

3.4.9 Storage Life

The motor shall meet the requirements of this specification after a minimum earth storage life of three years during which the ambient temperature may vary in the range of +16 to +38°C (60 to +100°F) with a relative humidity of 65 percent or less. Unis will be stored in shipping containers, and shall not suffer any detrimental effects from successive exposure to the temperature extremes.

3.4.10 Transportation

The motor shall be capable of being transported by truck or air, while packaged in its shipping container, and shall meet the requirements of this specification at point of delivery. Vibrations in shipment by common carrier should be designed for.

3.5 Physical Requirements

3.5.1 Mass Properties

3.5.1.1 Weight

The weight of the motor case shall be a practical minimum, and the system shall not exceed 960 lbs. when loaded for the nominal total impulse requirement. The motor design shall permit a 10% increase in propellant weight.

3.5.1.2 Centre of Gravity

Longitudinal locations of the prefire and postfire centres of gravity shall be determined and verified by measurement. Measurement shall be within ±.100 in. of predicted for loaded motors and ±.250 in. for post fire motors.

3.5.1.3 Static and Dynamic Balance

All motors shall possess a maximum static and dynamic imablance about the motor axis, including measurement inaccuracy, as indicated below.

	Before Firing Unloaded	Before Firing Loaded	After Firing
Static (lb-in.)	1.0	6.0	2.0
Dynamic (lb-in^2)	5.0	60.0	30.0

Ballast added to balance the empty case will be nonconsumable.

3.5.2 Interface

The motor shall conform to the envelope and interface requirements of Spar SCD Drawing TBD 'Apogee Motor 4 Interface Data'. The attachment flange shall enable mounting of the motor within the spacecraft, transmission of booster and motor thrust loads, alignment of the motor axis with the spacecraft spin axis, and lifting of the motor during ground handling operations.

The surface finish and flatness for the motor flange shall equal or exceed those of the spacecraft mating flange.

The Contractor shall prepare a drawing bearing his Drawing No. titled: 'Apogee Motor Electrical Interface Requirements' in accordance with Military Specification MIL-D-1000, Category B, Form 3. The original shall be submitted for approval and then maintained.

3.5.3 Thrust Alignment

Nozzle centreline and thrust axis shall be considered coincident and are defined as the axis connecting the centroid of the nozzle throat and the centroid of the nozzle exit plane. Thrust axis displacement from the motor axis, in the plane of the throat shall not exceed .020 inch. Prior to firing, the
thrust axis shall be perpendicular to the attachment plane within .002 inch per inch.

3.5.4 Moments of Inertia

The moments of inertia of the motor shall be determined by analysis (before and after firing) about three orthogonal axes through the centre of gravity, one perpendicular to the attachment plane, with an accuracy of $\pm 3\%$.

3.5.5 Factors of Safety

For attachment and associated structures the following factors of safety shall be applied for the most sever loading conditions:

1.20 based on 0.2% yield stress: no excessive elastic deformation

1.50 based on ultimate stress: no failure or excessive inelastic deformation.

All pressurized components shall have a proof equal to the maximum expected chamber pressure (3 sigma) at the maximum test firing temperature and a minimum burst pressure 1.3 times the proof.

3.5.6 Natural Frequency

The natural frequency of the loaded motor when assembled to the spacecraft mounting flange (details provided by the Design Authority) shall not be less than 80 Hz in the thrust axis direction and 60 Hz in the lateral direction.

3.6 Component Design and Construction

3.6.1 General

The motor shall be designed to provide the highest practical specific impulse and propellant mass ratio consistent with the performance, operation and reliability requirements.

3.6.2 Structural Integrity

The apogee motor shall be designed to assure structural integrity of the motor case, propellant grain and nozzle during exposure to all environmental conditions, such that the motor will reliably perform its function and the spacecraft is not adversely affected physically or operationally as a consequence of motor firing. Internal or external separation of parts or pieces from the motor and excessive outgassing from the motor case or nozzle will constitute non-compliance with post-fire structural integrity criteria. Additionally, the post-fire minus three sigma value of internal insulation thickness shall not expose the case or be less than the insulation required to meet the thermal design criteria for the case or case-insulator bond. The nozzle throat and exit cone shall remain in place after firing. The motor subcontractor shall submit the criteria for approval prior to the initial design review.

3.6.3 Safety

The motor shall satisfy the following safety requirements:

a) The motor shall be designed to satisfy all applicable range safety requirements as defined by AFETRM 127-1 and shall meet the design and testing requirements for hazardous explosives as defined by AFM 127-100.

b) Propellants containing beryllium are unacceptable.

c) Electrical safety between the power supply and the initiators shall be provided by an electromechanical safe and arm device. The safe and arm device shall be capable of mechanically disarming the igniter.

3.6.4 Nozzle

The exterior of the nozzle exit cone shall have an emissivity of less than 0.1.

3.6.5 Propellant

The propellant shall not ignite while conditioned to +120°C (250°F) for eight hours. Proof of space environment compatibility will be required for propellants that have not been used in other apogee motors. (Vacuum storage and testing will then be required, along with an analysis and possible testing for the effects of space radiation).

3.6.6 Ignition System

3.6.6.1 General

The ignition system for the motor shall be an electromechanical safe and arm device containing redundant electrically initiated squibs.

The ignition system will not ignite when held for eight hours at any temperature up to ±120°C (250°F).

3.6.6.2 Initiator

Two squibs shall be used for redundancy, each with a design capability of achieving a 0.995 reliability at a confidence level of 0.90. One squib shall be capable of meeting the ignition requirements of this specification.

3.6.6.3 Electrically Initiated Squib

3.6.6.3.1 Operation

The squib shall demonstrate compliance in electromagnetic fields as specified in AFETRM 127-1. The initiator shall operate from 28 VDC ±2% source and shall require not more than 4.5 amperes for 0.010 second per squib.

3.6.6.3.2 Squib No-Fire Power

The initiator shall not fire when either bridgewire is subjected to a current of 1.0 amp dc for 5 minutes.

3.6.6.3.3 Squib All-Fire Current

The squibs shall fire satisfactorily when any electrical current between 4.5 amperes and 22 amperes is applied for a maximum duration of 10 milliseconds.

3.6.6.3.4 Safety

The initiator shall be shorted, or otherwise protected during shipment and handling to prevent firing due to current accidentally applied or induced by electromagnetic environment. Any cabling associated with the initiators shall have a ground shield. Polar bosses and the mounting flange shall have electrical continuity between them.

3.6.7 Safe and Arm Device

3.6.7.1 Design

The apogee motor shall utilize a safe and arm (S&A) device with the following design features:

a) S&A device must operate by a mechanical alignment of an explosive train and electrical continuity from the firing circuit to the initiator within the S&A.

b) Monitoring and controlling circuits must not be routed through the same connectors as the firing circuit.

c) The device must provide a means of mechanical saving in the event of a malfunction or abort.

d) A positive mechanical lock (safety pin) will be used in the S&A device to prevent movement from the safe to the armed position. Removal of the safety pin from the S&A shall be from an angle perpendicular to the motor centreline and shall be impossible if the sensing circuit is energized. Removal of the pin must not cause the unit to go to the arm position.

e) The design must indicate its arm or safe status by simple visual inspection.

f) In the safe position, the initiator must be internally shorted, and the explosive train mechanically misaligned.

g) The monitor circuit must provide both arm and safe indications.

3.6.7.2 Insulation Resistance

The resistance of all electrical points to ground and between mutually insulated points shall be equal or greater than 2 megohms when measured at 500 VDC.

3.6.8 Environmental Seals

The motor shall contain seals to prevent damage to the propellant during storage and handling. The seals shall not be used as ignition aids.

3.7 Reliability

3.7.1 Motor Reliability

The motor, including the safe and arm device, shall have a probability of at least 0.995, with a 0.50 confidence level of surviving launch and meeting all performance requirements while being subjected to the environmental conditions, tests and measurements specified herein.

3.7.2 Calculated Reliability

Using failure rates established by test results and/or correlatible programs of similar components and scope conducted by the subcontractor, the reliability analysis of the motor shall demonstrate analytically the reliability requirements given in para. 3.7.1 above.

3.8 Interchangeability

Each safe and arm device and each motor shall be directly inter changeable with regard to form, fit, and function with other subassemblies of the same part number. Cases, nozzles, and safe and arm devices may be considered a matched set after empty balancing.

3.9 Electromagnetic Compatibility

Electrical and electronic components of the motor shall comply with Requirements, Spacecraft Electromagnetic Compatibility (T.B.D). Compliance with these requirements shall be verified by test or accomplished by proof of similarity.

3.10 Maintainability

The motor shall be designed such that field maintenance is not required except for removal and replacement of the factory assembled motor from the spacecraft and removal and replacement of the safe and arm device from the motor. The design shall permit leak checks, radiographic inspection, and checkout of the safe and arm device prior to installation of the motor in the spacecraft.

Provision shall be made for a one position installation of the motor in the spacecraft by the use of holes and pins on the motor attachment flange.

3.11 Identification of Product

3.11.1 Motor Identification

The motor shall be marked for identification in accordance with the manufacturer's standards.

The identification shall include, but not be limited to, the following:

a) Customer Specification
b) Customer Part Number
c) Contract Number
d) Manufacturer's Name or Trademark
e) Manufacturer's Part Number
f) Serial Number
g) Date of Casting
h) Actual weight, expressed in pounds and decimal form to the nearest 0.1 pound
i) Storage temperature range and expiry date
j) Propellant Batch and/or Lot Number

3.11.2 Component Identification

Components (such as the safe and arm assembly) which may be packaged separately shall be marked for identification. The identification shall include, but not be limited to, the following:

a) Contract Number
b) Manufacturer's Name or Trademark
c) Manufacturer's Part Number
d) Serial Number
e) Date of Manufacture
f) Lot Number

3.12 Workmanship

The motor, including all parts and assemblies, shall be constructed, finished and assembled in accordance with highest standards. Particular attention shall be paid to neatness and thoroughness of soldering, wiring, marking or parts and assemblies, plating, painting, machine screw assemblage, and freedom from burrs and sharp edges. The motor shall not contain cracks, chips, or voids that in the judgement of the procuring activity will render the motor unsuitable for the purpose intended.

4.0 PRODUCT ASSURANCE REQUIREMENTS

The contractor shall establish and conduct a Product Assurance Program and submit a detailed program plan to the Design Authority to comply with Spar Requirements, Product Assurance Program, and its supporting applicable documents.

4.1 Responsibility for Inspection and Test

Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may utilize his own facilities or any laboratory acceptable to the customer, who reserves the right to witness, perform or review the documentation of any of the inspections set forth in the specification where such inspections are deemed necessary by the Design Authority to assure that supplies and services conform to prescribed requirements.

4.2 Test Conditions

4.2.1 Temperature Conditions

4.2.1.1 Low Temperature

Low Temperature shall be conducted with the temperature of the unit at -7 (+0, -5)°C (+20(+0, -10)°F).

4.2.1.2 Ambient Temperature

Unless otherwise specified, tests shall be conduced at local ambient temperatures within the range of 21 ±3°C (70 ±5°F) and a relative humidity of not more than 95%.

4.2.1.3 High Temperature

High temperature tests shall be conducted with the temperature of the unit at +38 (+5, -0)°C (+100 (+10, -0)°F).

4.2.1.4 Temperature Cycling

Motors subjected to temperature cycling shall be conditioned to the temperature extremes for six cycles. Temperature cycling may begin at either the high or the low temperature. One cycle, beginning at the low temperature, is defined as follows:

Low Temperature
High Temperature
Low Temperature

4.2.1.5 Temperature Conditioning Time

Temperature conditioning for the motor shall be such that all components have reached a uniform temperature within the specified temperature tolerance when ignited. During conditioning time, the temperature of the conditioning chamber shall not vary outside the specified temperature tolerance.

4.2.1.6 Thermal Insulation

One motor fired at altitude shall be tested with a customer supplied motor thermal blanket installed and with a representative thermal mass at the attachment ring for typical thermal soak back conditions. The customer will provide the design parameters and the contractor shall provide the fixture.

4.2.2 Pressure Conditions

4.2.2.1 Sea-Level Pressure

Sea level tests shall be conducted with the motor exposed to the altitude of the test facility. Unless otherwise specified, all inspections and tests shall be conducted at sea level pressure.

4.2.2.2 Altitude Tests

Altitude tests shall be conducted with the motor exposed to a pressure simulating altitude conditions of at least 100,000 feet for five minutes prior to firing. The pressure shall be maintained at a minimum of 50,000 feet throughout tailoff, or until blowback occurs. All tests shall be conducted with the weather seal punctured.

4.2.2.3 Burst Test

Pressurized components shall be subjected to burst tests to demonstrate conformance to paragraph 3.5.5. Chambers shall have simulated thrust loads applied during burst.

4.3 Test Measurements and Equipment

4.3.1 Thrust Measurement

The motor thrust shall be measured at the plane of the attachment fitting. A minimum of two (2) channels of thrust data shall be measured on both

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digital and analog recording systems. The axial load cell shall be a multiple output, strain gauge type that is accurate to within ±100 pounds for all tests.

4.3.2 Chamber Pressure Measurement

Chamber pressure measurements shall be made. A minimum of one high range (0-1000 psi) and one low range (0-100 psi) pressure transducer shall be used and the output recorded on digital and analog systems. Transducer accuracy shall be within ±0.5 percent for all tests with a frequency response for the transducer and any connecting gas line as an assembly of 400 cps for all tests.

4.3.3 Ambient Pressure Measurement

Ambient pressure shall be measured within six inches of the nozzle exit and at the forward end of the motor during all test firings. Transducer accuracy shall be within ±0.05 psi for all tests.

4.3.4 Motor Temperature Measurements

During all engine static firings, motor temperature shall be measured at a minimum of four locations on the outside of the case and three locations on the outside of the external portion of the nozzle. The accuracy shall be within ±5°C (±10°F).

4.3.5 Recording Equipment

Suitable recording equipment shall be used which shall have an accuracy within ±0.3 percent of reading with ±2 count resolution error.

4.4 Quality Assurance Inspection and Tests

4.4.1 General

Each motor selected for qualification testing, development testing or delivery shall meet the following requirements.

4.4.2 Examination of Product

Each motor shall be visually examined for conformance to applicable drawings and specifications, workmanship, and finish. Where applicable, piece parts inspection shall be accomplished prior to completion of assembly.

4.4.3 Proof Pressure Test

Each motor case shall be subjected to static proof pressure tests as specified in paragraph 3.5.5.

4.4.4 Motor X-ray

Each motor shall be subjected to radiographic inspection after curing of the propellant. Test motors shall be x-rayed after being subjected to vibration and temperature cycling. The x-ray inspection shall be accomplished with a 10 MEV LINAC or equivalent x-ray machine. Film exposure shall be in accordance with MIL-STD-453 to obtain a density of 1.8 to 3.0 H and D and a quality level of 1-1T. Voids or combination of voids observed in the propellant that would increase the burning surface, reduce the web thickness, or reduce the structural integrity may be cause for rejection. Criteria for acceptance shall be developed by the manufacturer and approved by the Design Authority. Cracks or separations in the propellant, between the propellant and liner between the liner and insulation, or between the case and insulation shall be cause for rejection.

4.4.5 Electrical Inspection and Safety

The safe and arm device of each motor shall be tested to verify conformance with electrical inspection and safety requirements.

4.4.6 Weight Determination

Each motor shall be weighed and the motor manufacturing record shall be examined to determine propellant weight and propellant mass ratio. The required weights shall be determined using test instruments and procedures which provide an accuracy of ±.25%. Motor weight after firing shall also be recorded.

4.4.7 Centre of Gravity

The longitudinal and lateral centre of gravity of each loaded and unloaded motor, and fired motors which have been subject to spin tests during qualification shall be determined using procedures and equipment which provides a measurement accuracy within ±0.010 in. loaded and ±0.010 in. burnout. Measurements shall be made relative to the plane of the motor attachment fitting and spin axis.

4.4.8 Static and Dynamic Imbalance Tests

Each motor shall be subjected to a static imbalance test and a dynamic imbalance test to demonstrate compliance with the requirements of 3.5.1.3. Motors which have been subjected to spin firings shall have their static and dynamic imbalance determined after firing. Dynamic imbalance tests shall be conducted at a minimum spin speed of 150 revolutions per minute. Balancing of loaded motors shall be accomplished by the adjustment of propellant distribution; i.e. no inert material may be added or removed.

4.4.9 Nozzle Alignment

Each motor shall be examined to determine the mechanical alignment of the nozzle axis relative to the motor axis, and to demonstrate compliance with the requirements of 3.5.3. Compliance with the alignment requirement during firing may be demonstrated statistically by measurements of nozzle deflection due to case pressurization during the case proof pressure tests of 4.4.3, or may be measured during static firing.

4.4.10 Moments of Inertia

Longitudinal and lateral moments of inertia shall be determined by analysis about the centre of gravity before and after firing, with an accuracy of ±3%.

4.4.11 Leak Test

Each motor shall be subjected to a 50 psi leak test using a mixture of 90 percent nitrogen and 10 percent Freon. No leakage greater than a rate of one ounce per year (Freon) shall be permitted when using a halogen leak detector (G.E. Model H2 or equivalent), having a working sensitivity for detecting Freon leakage at a rate of one ounce per year and having a 100:1 range of sensitivity adjustment.

4.5 Testing

4.5.1 General

Motors subjected to Development and Qualification testing shall meet requirements of paragraph 4.4 and the following.

4.5.2 Temperature Cycling

The motor shall be subjected to temperature cycling tests as outlined in paragraph 4.2.1.4.

4.5.3 Spin

The motor shall be mounted on a fixture which will enable rotation about its spin axis. It shall be rotated at a minimum speed of 150 revolutions per minute for at least five minutes prior to ignition, while at the specified prefiring environmental temperature. The spin rate shall then be reduced to 110 rpm ±10 rpm and the motor shall be fired in accordance with 4.5.5.

4.5.4 Vibration Testing

4.5.4.1 General

Motors shall be subjected sinusoidal and random vibration levels defined in paragraph 3.4.7. Dynamic parameters of the spacecraft and launch vehicle adapter will be provided by the customer to enable the motor manufacturer to design and manufacture a test fixture representing the spacecraft and adapter if desired. Alternatively, the motor manufacturer may conduct tests with the motor hard mounted, and apply the calculated flange response levels in paragraph 3.4.7.1 a) and limit motor centre of mass response to levels in paragraph 3.4.7.1 b).

4.5.4.2 Sinusoidal Vibration

During the sinusoidal sweep tests the frequency end points shall be within ±2% of the specified end points in the frequency range of 20 to 2000 Hz, and shall be within ±1/2 Hz for frequency end points below 20 Hz.

All sine test durations will be within ±10% of the specified test durations.

4.5.4.3 Random Vibration

The random vibration test shall consist of input amplitudes and durations as specified in paragraph 3.4.7.2.

4.5.5 Firing Test

4.5.5.1 Procedure

Motors shall be subjected to firing tests under the following conditions:

a) Motor and test temperature conditions shall be stabilized per paragraph 4.2.1.

b) Motors shall be fired at pressure conditions specified in paragraph 4.2.2.

c) Motors shall be fired while spinning at 110 rpm ± 10 rpm.

4.5.5.2 Firing Performance Data

During firing, the performance of the motor will be monitored for compliance with the requirements of paragraph 3.3. Data measurement shall at least include:

a) Thrust versus time
b) Chamber pressure versus time
c) Spin rate
d) Cell pressure near nozzle exit plane.

4.5.5.3 Firing Temperature Measurements

As specified in paragraph 4.3.4.

4.5.5.4 Cold Nozzle Tests

Motors fired to verify the integrity of the exit cone at cold conditions shall be tested in accordance with the conditions specified in paragraph 3.4.2. This requirement may be waived in the event that the same or similar configuration with the same materials has previously been qualified to the same or more severe environment.

4.5.5.5 Post Fire Measurements

After firing, the following measurements shall be recorded:

a) Weight per paragraph 4.4.6.
b) Centre of gravity per paragraph 4.4.7.
c) Balance per paragraph 4.4.8.
d) Nozzle Alignment per paragraph 4.4.9.
e) Moments of Inertia per paragraph 4.4.10.

4.6 Acceptance Tests

4.6.1 General

All motors will be subject to acceptance by the Design Authority prior to static test or delivery. These units shall meet the requirements of paragraph 4.4 and the following.

4.6.2 Firing Tests (Sub-Scale)

Firing tests shall confirm that the performance of the propellant batch evaluated meets the requirements of this specification.

4.7 Rejection and Retest

4.7.1 Rejection

The motors which fail to comply with the requirements specified herein may be rejected by the Design Authority.

4.7.2 Retest

4.7.2.1 General

If a motor from a lot fails to comply with the tests specified herein, acceptance of the lot shall be withheld until the extent and cause of failure has been determined. Full disclosure of corrective measures proposed shall be made to the Design Authority for approval. After corrective action is taken, the test in progress at the time of failure shall be repeated at the option of the Design Authority. If in the opinion of the Design Authority corrective action could significantly change results of previous tests, all qualification tests shall be repeated.

4.7.2.2 Loss of Data

Loss of data because of instrument malfunction or other unforseen test problems not related to the motor design may not be considered a failure, if in the opinion of the Design Authority the motor completes the firing so that the reliability of the motor under test may be adequately assessed.

5.0 PREPARATION FOR DELIVERY

5.1 Handling and Transportation Safety

Interstate commerce commission approval shall be obtained for the method of delivery of the motor and initiator. Evidence of such approval shall be submitted 60 days prior to shipment of any motor.

5.2 Manufacture Packaging for Shipment

5.2.1 General

Preservation, packaging and marking shall be conducted to assure compliance with the requirements of Requirements, Product Assurance Program, SR.01-01. Shipping containers shall include a shock recorder.

5.2.2 Marking for Shipment

The packaged unit shall be marked for shipment in conformance with manufacturer's standard and U.S. Department of Transportation regulations.

5.2.3 U.S. DOT Approval

The subcontractor shall obtain U.S. Department of Transportation explosive classification for delivery of the unit and shall submit this to the Design Authority.

5.3 Motor Log Book

A motor log book shall be prepared for each motor.

6.0 NOTES

6.1 Intended Use

The motor described in this specification is intended for use in a General Purpose BUS System, for future satellites.

6.2 Procurement Data

Documentation required to procure apogee motors shall specify the title, number and date of this specification.

6.3 Definitions

For the purpose of this specification, the following terms are defined:

a) Total Impulse

Total impulse is the area under the thrust-time curve.

b) Action Time

Action time is the time duration between the first indication of thrust on the rising portion of the thrust-time curve and the delivery of 99.5% of total impulse.

c) Average Thrust

Average thrust is the total impulse divided by action time.

d) Thrust Time

Thrust time is the instantaneous thrust level at any point in time during operation of the motor.

e) Load Factor

Load factor is the instantaneous spacecraft and motor mass divided by thrust.

f) Thrust Rise Rate

The thrust rise rate is the calculated value of the average rate of change of thrust (lb) per unit time (sec) derived from the slope of the straight line drawn tangent to the pressure-time curve at any given instant over a 5 millisecond interval.

g) Ignition Time

Ignition time is the time duration between the application of the firing current at the igniter plug and the attainment of 200 psia.

h) Thrust Axis

The thrust axis is a straight line connecting the centroids of the nozzle throat and exit plane diameters.

i) Motor Axis

The motor axis is a straight line perpendicular to, and passing through the centroid of, the motor/spacecraft attachment ring.

j) Failure

An item is considered to have failed when:

i) It no longer can perform its intended function.
ii) Any part of it has cracked, ruptured, or collapsed.
iii) Permanent misalignment occurs which impairs the CTS mission accuracy criteria.

k) Excessive Deformations

i) Elastic Deformations

Elastic deformations under the application of qualification level accelerations or loads shall be considered excessive when they produce contact or interference between parts of the structure and/or items of equipment.

ii) Inelastic Deformations

Inelastic deformations of the structure are considered excessive when any part can no longer perform its intended function.

l) Limit Load

The limit load is the most severe structural loading to the spacecraft and/or its subsystems and components during each critical mission phase. (Loads imposed by testing are not considered as limit loads and are defined separately.) The limit load is typically a 3-sigma flight load, predicted by rational analysis. The analysis is supported with similar-mission flight data when available.

m) Qualification Test Load

The qualification test load is the most severe loading imposed on the structure and/or its subsystems during qualification testing. It is by intent, a more severe loading condition than that imposed by limit loads. To derive a qualification test load from a limit load,the following factors apply:

Type of Limit Load	Multiplied By = Qualfication Load
Quasi Static Acceleration	1.25
Amplitude (0-PK) of sinusoidal vibration	1.50
Amplitude of random vibration PSD spectra	2.25

n) Acceptance Test Load

The acceptance test load is the load imposed on the spacecraft and/or its subsystems and components during acceptance testing. Acceptance test levels are equal to limit load levels.

o) Factor of Safety

A factor of safety is an arbitrary multiplier dependent on the type and use of the structure to account for variaties in material, manufacturing and load distribution as well as inaccuracies inherent in analysis. Its purpose is to establish design loads.

p) <u>Design Loads</u>

Design loads are those to which a spacecraft and its subsystems are designed. A properly designed system will never experience design loads during destructive testing, since they are in excess of qualification test loads. Design loads are categorized as ultimate loads, and yield loads. For the apogee motor:

Ultimate Load = 1.50 x Qualification test load

Yield Load = 1.20 x Qualification test load

q) <u>Margin of Safety</u>

The margin of safety (MS) is defined as:

$$M.S. = \frac{1}{R} - 1.0$$

where R is the stress ratio that results from the application of the design load to the allowable stress.

Effects of combined (interacting) loads or stresses are included in the computation of "R". The M.S. must be equal to or greater than zero for both ultimate loads and yield loads.

Ultimate load stresses are to be compared to material ultimate or component buckling stresses; while yield load stresses are to be compared to 0.2% permanent deformation material yield stresses.

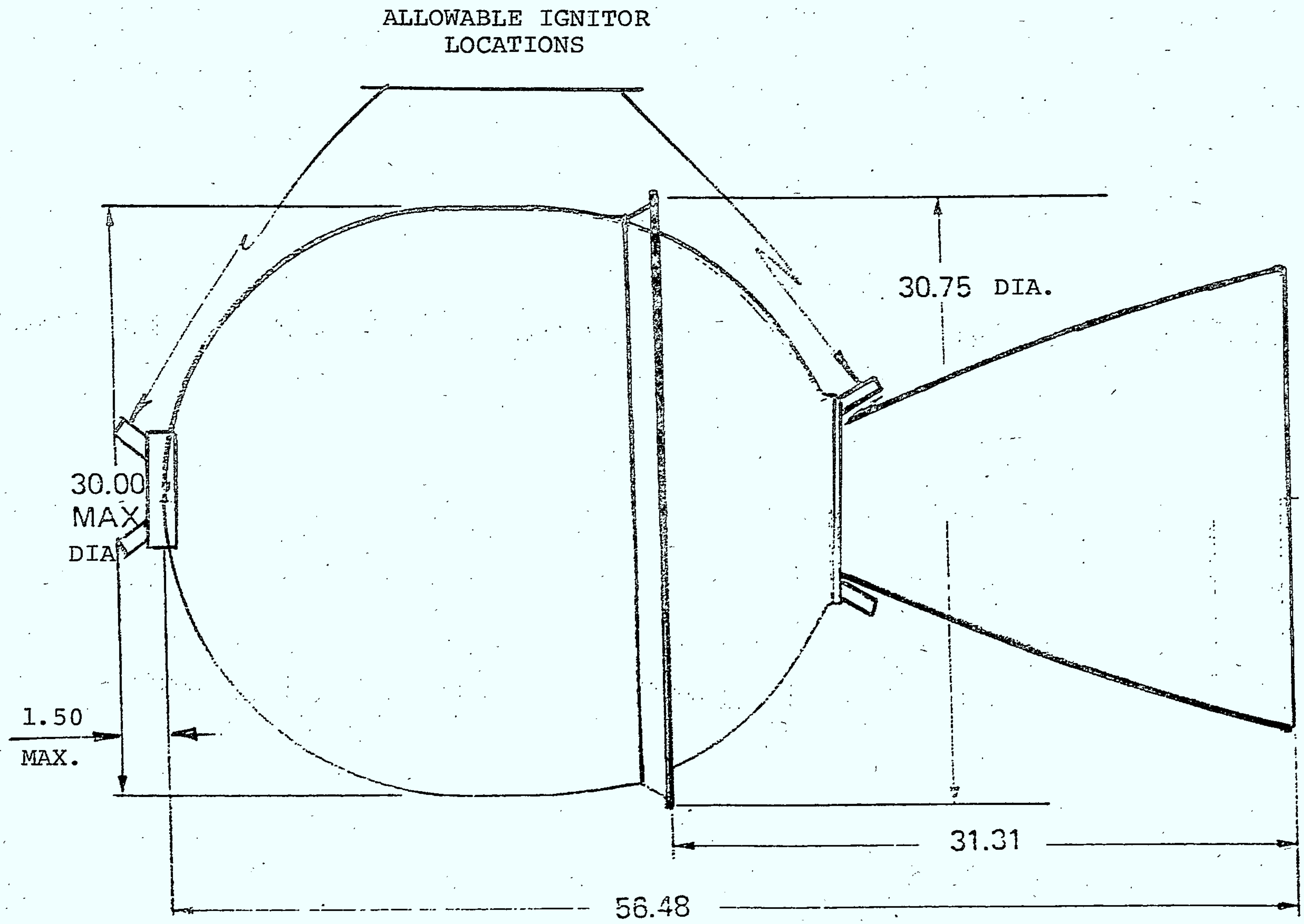

FIG. 1 - APOGEE MOTOR ALLOWABLE DIMENSIONS

2.0 POTENTIAL VENDORS, ASSESSMENT AND RESPONSES

This section lists potential vendors of components and assemblies of the RCS, DSA and Apogee Motor subsystems.

An assessment of a general nature has been made where possible of the vendors' experience and capabilities.

2.1 Reaction Control Subsystem Vendor Assessment And Baseline Design

2.1.0 General

This section of the report documents the release of the RCS RFP to potential vendors for budgetary estimates, their responses, a tradeoff of designs and the recommended baseline configuration for the General Purpose Bus (GPB) satellite.

2.1.1 RCS RFP Package For Budgetary Estimate, And Potential Vendors

SPAR-SG.350, Multi-Purpose Bus Study, Specification Requirements, Reaction Control Subsystem, Preliminary, for Budgetary Estimates, April 15, 1975 and SPAR-SOW.071, Multi-Purpose Bus Study, Statement of Work, Reaction Control Subsystem, Preliminary, for Budgetary Estimates, April 15, 1975 presented in Section 1.3, along with Spar Drawing 31138J1, Sheets 1 and 2, Structure and RCS, General Purpose Satellite Bus and a covering letter, were released as Spar RFP No. 5427 on April 18, 1975 to six potential RCS subsystem suppliers for budgetary responses. The companies, as listed below, had all been previously briefed in general on the requirements for the RCS and the nature of this study phase and had indicated an interest in responding.

a) AVCO Corporation
201 Lowell Street
Wilmington, Mass. 01887

b) Bell Aerospace Company
P.O. Box #1
Buffalo, New York 14240

c) Hamilton Standard Division
United Technologies Corporation
Windsor Locks, Conn., 06096

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d) The Marquardt Company
16555 Saticoy Street
Van Nuys, California 91409

e) Rocket Research Corporation
York Center
Willows Road, N.W. 116th Street
Redmond, Washington 98052

f) TRW Systems Group of TRW Inc.
1 Space Park
Redondo Beach, California 90278

Because of the short time frame allotted for this study, it was necessary to request that the responses be mailed by May 5, 1975 which gave very little time to prepare the proposals. Responses were received from all companies except Marquardt who were unable to bid at this time due to previous commitments, but who contacted Spar by telephone to indicate their continuing interest in the program and to comment on the possible use of a bipropellant subsystem. The ambiguity which existed in the documents at RFP release regarding the number of flight models and spares was subsequently cleared up by Telex. This telex is included in Section 1.3.

2.1.2 RCS Trade-Off and Baseline Design

2.1.2.0 General

Because of the state-of-the-art at the time of contract award for CTS and because of the minimal, two year, no north/south stationkeeping mission of that satellite, there was no major incentive to be the first in flight to advance the state-of-the-art past catalytic hydrazine. Indeed, to produce a catalytic LTE capable of 350,000 repeatable pulses delivered at very low I_{BIT} and duty cycle was advnacement enough at that time. However, the state-of-the-art has changed and so have spacecraft requirements. GPB, high power, six year life with north/south stationkeeping to $\pm$ 0.1° and negative weight margin using an all catalytic RCS, is a

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prime candidate for application of rapidly developing high performance RCS technology. The potential for weight saving in the RCS is much greater than in any other subsystem on the spacecraft. Because of these conditions, it is imperative to carefully investigate the use of:

a) liquid electrothermal hydrazine thrusters

b) super-heated liquid electrothermal hydrazine thrusters

c) electric propulsion - including ion, colloid and pulsed plasma thrusters

d) bipropellant thrusters and

e) gas electrothermal thrusters fed from a catalytic or electrothermal hydrazine plenum

At the time of release of the RFP, each appeared to have the potential of saving weight compared to an all catalytic design to some degree as well as having other possible performance enhancements. These thruster types will be discussed in Section 2.1.2.4 to 2.1.2.7.

The intent of this study is to establish RCS design feasibility and assess current state-of-the-art to meet the GPB requirements. The intent is not to select an RCS vendor at this time. Requirements are not well enough defined and candidate suppliers have not been given sufficient time in which to respond formally and fully to accomplish this latter task. The requirements are, however, well enough defined to ensure that responses to Spar RFP No. 5427 do provide accurate state-of-the-art design information pertinent to the GPB requirements. This report does not contain all of the detailed information presented in the vendor budgetary estimates, but does concentrate on weight, power, engine type and qualification status.

From the budgetary estimates received and from previous contact with the five companies who responded, it can be stated that each of these contractors is capable of producing an RCS for GPB. They have all, with the exception of Bell Aerospace Company, built and successfully flown satellite reaction control subsystems. Bell Aerospace Company, with the on-going Minuteman RCS contract, has shown that it has the facilities and systems capability to be a successful subsystem supplier. Each of these companies manufactures and tests RCS engines in the GPB required thrust regime.

2.1.2.1 Vendor Background and Responses to the RFP

a) AVCO Corp. has built and flown ammonia resistojet RCSs and has been working since 1969 on electrothermal hydrazine thrusters (EHT) in the millilbf thrust range. These EHT were proposed for CTS. Currently, AVCO is under contract to the AFRPL to demonstrate performance and life on this type of engine in the lbf thrust range for high total throughput missions. They are presently working on a modification to the EHT, a super-heat electrothermal (SHE) engine in the millilbf thrust range, which, through the addition of heater power, could significantly reduce RCS subsystem weight. AVCO, therefore, proposes to use catalytic HTEs for the precession (which they do not manufacture), SHE engines for north/south and east/west station acquisition and keeping (large throughput manoeuvres), and EHT engines for all other LTE (low I_{BIT} manoeuvre) functions. They presented a propellant weight comparison between this option and the all catalytic subsystem design. However, they did not submit a subsystem design at this time.

b) Bell Aerospace Company builds the bipropellant RCS for the Minuteman missile and have recently built a 0.2 lbf thrust catalytic monopropellant engine to be qualified this summer by GSFC as back-up to the Rocket Research "standardized" 0.2 lbf catalytic

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engine. They have been involved in parametric studies of catalytic monopropellant engine designs for the AFRPL. They propose the use of an all catalytic RCS for GPB, using their own engine for the LTEs and purchasing the HTEs from one of the several qualified suppliers. They also presented their views on the use of a bipropellant RCS for GPB. They submitted a creditable subsystem proposal indicating they are very interested in the program.

c) Hamilton Standard Division has built and is building several all catalytic RCSs including RAE-B, CTS, BSE and IUE as well as delivering engine assemblies for numerous programs. They propose the use of an all catalytic RCS for GPB using their own engines for both HTE and LTE applications. They fully understand the GPB requirements and have submitted the most complete subsystem response to the RFP indicating keen interest in maintaining their position in the Canadian Space Program.

d) Rocket Research Corp. has built the ERTS and ATS-F all catalytic RCSs as well as delivering engine assemblies for numerous programs including the RCA SATCOM. They are presently under contract to JPL to qualify a "standardized" 0.2 lbf catalytic hydrazine thruster which will be used first on the JPL MJS77 mission and later on the Rockwell GPS. They propose the use of an all catalytic RCS for GPB using their own engines, including the thruster mentioned above, for both HTE and LTE applications. Due to previous commitments, Rocket Research were not able to provide a thorough subsystem design at this time, although their responses did provide the essential engine and subsystem information including preliminary schedule and budgetary cost estimate.

e) TRW Systems has built and is building many propulsion systems using several in-house developed propulsion devices. These include:

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- Mariner 1969, DSCS II, Intelsat III, Pioneer F&G, Model 777 and Atmospheric Explorer single thrust level catalytic hydrazine RCSs

- Fleet SATCOM dual thrust level catalytic hydrazine RCS

- DSP propulsion system, classified, using high level catalytic hydrazine and low level warm gas thrusters with gas supplied by pressure regulated catalytic hydrazine plenum gas-generator

- VASP warm N_2 gas electrothermal thrusters, and

- Spacecraft employing high thrust bipropellant thrusters, and ammonia resistojets (LES8).

TRW have held contracts with GSFC since 1971 amounting to over $1/2M for development of amounting to over $1/2M for development of a millilbf thrust class liquid electrothermal hydrazine thruster. These EHT were proposed for CTS. They are presently working on a modification to the EHT, a super-heated liquid electrothermal thruster (HiPEHT) in the millilbf thrust range, which, through the addition of heater power, could significantly reduce RCS subsystem weight.

In their response, TRW concentrated firstly on presenting a weight trade-off of seven propulsion system options, namely:

a) All catalytic hydrazine based on Flt SatCom technology

b) Catalytic hydrazine HTEs + EHT LTEs

c) Catalytic hydrazine for precession control and on-orbit control with thermally augmented electrothermal (HiPEHT) for north/south stationkeeping

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d) (b) above, but with HiPEHT for north/ south stationkeeping

e) (b) above, but with HiPEHT for all on-orbit V requirements

f) "Symphonie" propulsion system with 2 lbf bipropellant thrusters for precession control and orbital V functions and cold gas for on-orbit attitude control

g) An all bipropellant RCS

This was a preliminary study based upon propellant ISP and did not take into consideration hardware weight differences especially for the bipropellant system. It did, however, serve to emphasize the potential weight savings of the design they finally recommended, namely Option (e) (note that this is the same complement as was proposed by AVCO Corp.). Cost, development status and schedule information were then presented for the recommended subsystem design. TRW are, of course, anxious to sell propulsion subsystems in general, including an all catalytic design, if requested.

2.1.2.2 Baseline Design

At this point in time because of the considerable weight advantages to be gained, the GPB will be using as its baseline a system which uses super heated electro-thermal engines. These engines, although at this time considered advanced technology, should be developed and available for the GP Bus and a launch in 79/80 time-period. However, in order to understand the RCS and for the purpose of this report, the Hamilton Standard (HS) design is presented on behalf of all catalytic vendors, as the baseline. See HSPC75R15, Multi-Purpose Bus Satellite, Reaction Control Subsystem, Technical Program Proposal contained in Volume II, Appendix B. This is done for several reasons:

a) HS have presented a complete subsystem technical proposal responding to all of the requirements stated in the RFP covering letter.

b) The HS design is most probably the lightest weight design of the catalytic bidders. Their engines typically run slightly hotter and achieve slightly better ISP than other catalytic engines (although this is somewhat masked by data reduction methods and degrees of conservatism elsewhere within the industry). HS achieved a very light-weight subsystem design implementation on CTS and their proposal for GPB reflects the same emphasis on weight saving.

c) HS have endeavoured to use CTS technology hardware wherever possible even down to the level of several of the printed circuit boards). Given the short time period for this study, Spar were able to gain most confidence in the HS design because of their familiarity with CTS.

However, each of the all catalytic competitors has advantages and disadvantages to their engine and subsystems designs and should be given full opportunity to prepare a full proposal for the RCS for GPB should there indeed be a project.

The main features of the HS design include:

(a) A total subsystem wet weight for the worst case six year mission of 258.69 lbm which is approximately 21.5 lbm lighter than CTS technology hardware. This saving is attributable to two main factors. Firstly, it is due to the use of surface tension tankage as opposed to elastomeric diaphragm. Secondly, it is due to a modification to the design of the LTE for offset operation (reduction in injector ID, throat diameter and use of double the CTS chamber heater power) to reduce min I_{BIT} at beginning of life to below 2×10^{-3} lbf sec so that no reduction in engine moment arm is necessary to avoid the momentum wheel hardware weight penalty. Spar concurs with HS in their selection of surface tension tankage for the following reasons:

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- materials of construction are fully compatible with an eight year hydrazine environment; there remains some doubt as to the compatibility of elastomerics (EPT-10 or AFE-332) for this length of mission.

- the inherent difficulty with surface tension tankage is to qualify the design in a 1 g environment. Both RCA/Fansteel (for RCA SATCOM) and LMSC/PSI (for a classified project) have qualified such a design. The SATCOM design has been designed for essentially the same mission as GPB including the Thor Delta 3914 launch vehicle, with tank size optimum for GPB, and should be flight proven by early 1976 prior to possible GPB RCS RFP release.

- the potential weight saving compared to the elastomeric diaphragm tanks of approximately 14 lbm for GPB is very attractive considering the weight situation.

It should be pointed out that TRW, RR and Bell did not recommend surface tension tankage as a baseline, although they all pointed out that they might choose it as such prior to contract award. A complete and acceptable HS bid on GPB in 1976 would be contingent upon presentation, with their proposal, of satisfactory development test results on the modified CTS LTE showing the capability of meeting the eight year cycle life requirement of approximately 630,000 pulses, plus margin, at $I_{BITS} \leq 2 \times 10^{-3}$ lbf sec at very low duty cycles over the full operating temperature range of the engine. The CTS LTE was qualified to 350,000 pulses of less than 4×10^{-3} lbf sec and one hour of steady state operation. In addition, it would be necessary for HS to show that the CTS LTE could operate successfully for 31.5 hours steady state firing, plus margin. It is not presently certain as to

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whether HS intends to perform these tests in the near future. It should be noted that this total weight includes 11.95 lbm for structure of which 7.70 lbm is allocated for tank struts, strut end fittings and bulkhead tank mounting brackets.

b) A reliability of 0.95 for the eight year mission.

c) The use of flight proven hardware in most cases identical to CTS for all components except the propellant tank and the offset engine and its temperature sensor.

The main departure of the HS design from the RFP is the use of six latching valves instead of eight (the offset engines are latched with the other two LTE groups rather than by themselves). This was done:

- because a six LV module has already been qualified for BSE and would save money;

- the reliability analysis indicates negligible decrease in subsystem reliability is obtained by deleting these two latching valves; and

- 2.29 lbm weight savings can be realized by deleting them.

Spar is still evaluating the merit of retaining these latching valves. Qualitatively, since the offset engines are used continuously throughout the life whereas the other LTEs are only used periodically, it would appear beneficial to isolate the two sets. A second departure is the removal of RCS power on/off switch from the ECU. Spar concurs with the HS recommendation for the reasons they discuss. This switch was originally included because of the uncertainty in design of the power subsystem for GPB.

The HS design shows the latching valve module and electrical control unit in an unacceptable location

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because of encroachment on valuable north panel real estate. Relocation of these units would be expected to have no impact on the weight, cost or reliability of the design.

HS proposes to qualify 1 LTE for pulsed mode cyclelife and 1 LTE for steady state operation. Spar agrees that there is no purpose to be served in performing both qualifications on the same engine but recommends that 2 LTE are qualified for each requirement.

After receipt of the HS proposal, several technical questions were posed to HS by Spar. Response to these questions is attached in Appendix B.

2.1.2.3 Catalytic Hydrazine Engine State-of-the-Art

The engines are the heart, or rather muscle, of the RCS. Several companies, HS, RR, TRW, Hughes, etc., manufacture catalytic hydrazine thrusters qualified for the HTE mission for GPB.

The present qualification status of the 0.3 lbf to 0.1 lbf thrust class, catalytic hydrazine thrusters to meet the GPB requirements is shown in Table 2.1-1. All engines would require some modification to meet the GPB requirements, either in the hardware or in the planned test programs.

2.1.2.4 Bipropellant RCS

Dr. L. Kayser, Technologieforschung, Stuttgart, Germany, claims that he has developed a MMH/N_2O_4, 0.5 lbf bipropellant thruster which will produce a steady state ISP of 290 lbf sec/lbm at this thrust level and has been operated in this mode for 1M seconds continuous burn (278 hours). This same engine has operated without degradation for 10M pulses (the engine was pulsed in a vacuum every day, once per second and ten times per second for one year). This engine, TIROC, employs tiroidal injection to mix the fuels and cool the outer walls and Dr. Kayser states that these walls do not exceed 270°F during firing with the hottest point being at the

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Table 2.1-1

CATALYTIC LTE QUALIFICATION STATUS

Performance Characteristics Manufacturer and Engine	Steady State Thrust Range (lbf)	Steady State Operating Life (Hours)	Pulsing Cycle Life	Minimum Lin. Impulse Bit (lbf. sec.) At Maximum Thrust	Comments
Hamilton Standard - CTS Engine	0.25 to 0.096	1 Hour (Q), CTS, High P_C roughness of 30%	450,000 (D), 350,000 (Q), Min I_{BIT} Pulses, CTS	3.8×10^{-3}	This engine also used on BSE, IUE and Solrad X
Hamilton Standard - Proposed Offset Engine Redesign, Modified CTS		YET TO BE BUILT AND TESTED		2.0×10^{-3}	
Bell - Belltex DVT	0.24 to 0.06	40 Hours (D), Very low P_C roughness (2%)	506,000 (D), Most pulses not done at MIN I_{BIT}	3×10^{-3} High I_{BIT} Variability (25%)	Engine to be qualified during summer of 1975 by GSFC
Rocket Research - MR-74	0.15 nominal		NOT AVAILABLE		This engine flown on ATS-F and will be flown on RCA SATCOM
Rocket Research -0.2-lbf T/VA, Modified MR-74	0.20 to 0.04	60 Hours to be demonstrated by June, 1975	750,000 to be demonstrated by June, 1975 (10 to 40 msec on time)	3.0×10^{-3} at 10 msec on, 2.0×10^{-3} at 8 msec on, but no planned testing at this on time	380,000 pulses, 30 hour steady state accomplished pre-contract, this engine to be the "standardized" 0.2 lbf engine
TRW-MRE-.1	0.25 to 0.1	40 Hours (D)	200,000 (Q) 350,000 (D)	2.0×10^{-3}	This engine will fly on FLT SATCOM
GPB Requirements	0.3 to 0.01 allowed	40 Hours at 0.25 lbf BOL	700,000 at 2×10^{-3} lbf sec BOL	2.0×10^{-3} or 4.0×10^{-3} with weight penalty	

Note:

(D) Demonstrated
(Q) Qualified

throat (1000°F). There are two flexure guided valves on each engine, not linked, which open in approximately 0.5 msec and are synchronized to 0.05 msec. During pulsing operation of 15 msec on time, long off times, he claims that the ISP only drops to approximately 260 lbf sec/lbm. The engine requires no thermal shield and the propellants, he states, need no thermal conditioning down to approximately -20°C. A pressure regulation or repressurization system is however required to enable the engine to operate at its design point, see Reference 5*.

Using this engine for low thrust functions at the 0.5 lbf thrust level, and the "Symphonie" bipropellant thruster at 2 lbf thrust or the Aerojet Liquid Rocket bipropellant thruster at 5 lbf thrust, for precession manoeuvres (see Reference 6), it would be possible to provide a bipropellant RCS for GPB which would weigh approximately 248 lbm for the worst case six year mission (hardware for eight years) taking into consideration a 31.5 lbm hardware weight penalty associated with the pressure regulation equipment and dual fuel distribution system. This design would use equal volume propellant and oxidizer tankage (mixture ratio of 1:6). Figure 2-1 presents a barchart representation of the subsystem weights for various options. This would be a 10 lbm weight savings compared to the baseline design.

Presently, the AFRPL is testing two of Dr. Kayser's engines to determine whether the ISPs quoted can really be achieved. They have had some facility problems and the tests are scheduled to resume at NASA-LeRC at the end of May. There appears to be general scepticism within the North American community regarding TIROC performance. This is based upon R&D work on low thrust bipropellant engines. Their work would indicate that a maximum achieveable steady state ISP for a bipropellant engine in the 0.5 lbf thrust class if designed for the long life requirements of GPB would be approximately 260 lbf sec/lbm and that pulsed mode performance would drop off severely to less than 200 lbf sec/lbm. However, it is fair to say that U.S. are

* For List of References, See Volume I, Section 5.2.4.

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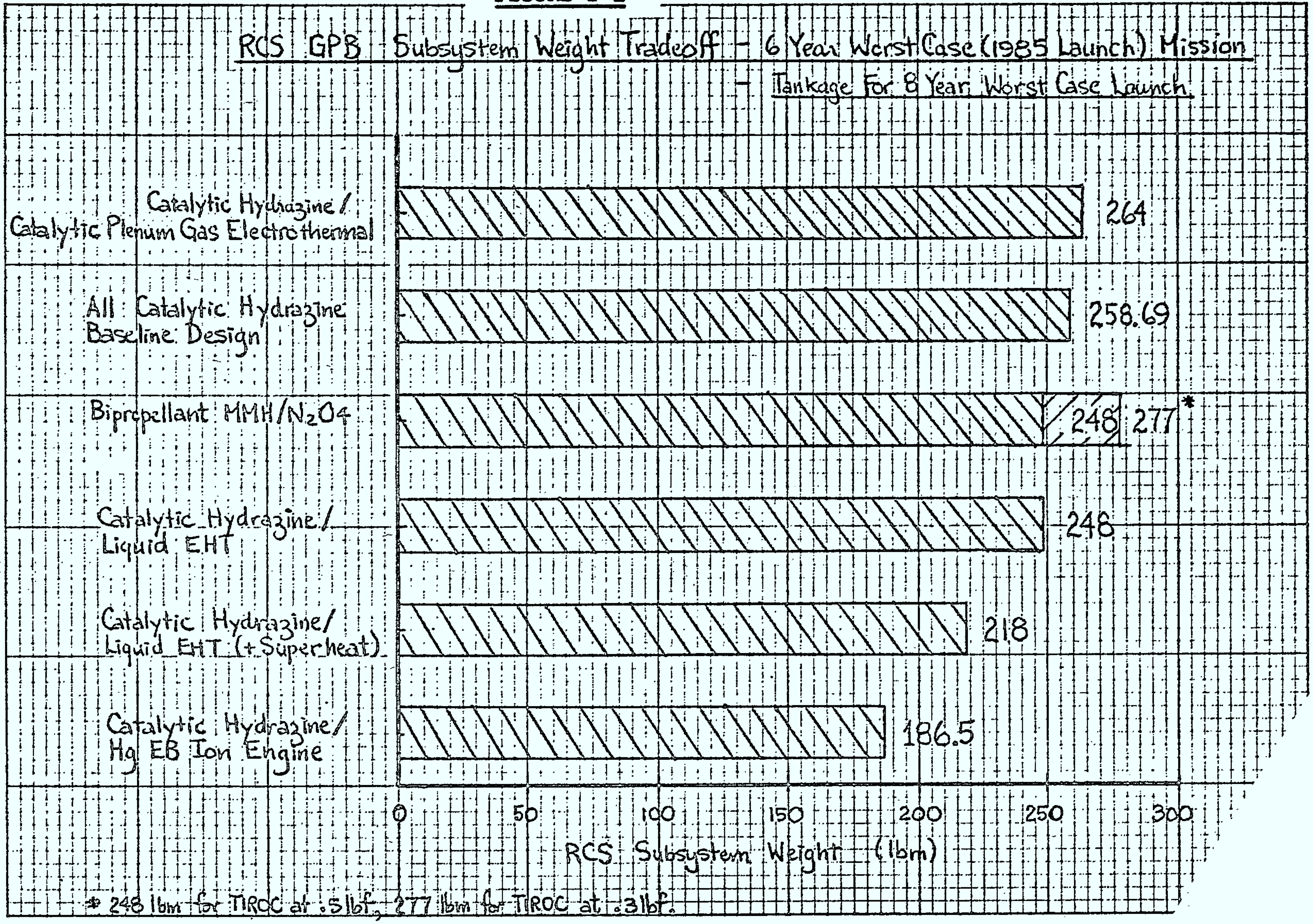
FIGURE 2-1


RCS GPB Subsystem Weight Tradeoff - 6 Year Worst Case (1985 Launch) Mission
- Tankage For 8 Year Worst Case Launch
Catalytic Hydrazine / Catalytic Plenum Gas Electrothermal
264
All Catalytic Hydrazine Baseline Design
258.69
Bipropellant MMH/N2O4
248
277*
Catalytic Hydrazine / Liquid EHT
246
Catalytic Hydrazine / Liquid EHT (+ Superheat)
218
Catalytic Hydrazine / Hg EB Ion Engine
186.5
0
50
100
150
200
250
300
RCS Subsystem Weight (lbm)
* 248 lbm for TIROC at .5lbf, 277 lbm for TIROC at .3lbf.

not completely familiar with the fluid dynamics of the TIROC engine. Acceptance test data obtained on the AFRPL engine prior to shipment from Germany two years ago indicates that steady state performance drops off rapidly with decreasing thrust such that, at the 0.3 lbf thrust level (the maximum thrust allowed by ACS for GPB) the steady state ISP is 248 lbf sec/lbm instead of 290 lbf sec/lbm. The TIROC engine has been improved since that time but unfortunately more recent data is not available to Spar at this time. Using the lower ISP values associated with the 0.3 lbf thrust level the bipropellant subsystem becomes heavier than the baseline by 18.6 lbm at 277.3 lbm.

It has been pointed out by Marquardt that, although the oxidizer, with N.O. added, may not freeze above -30°C, iron nitrate can precipitate out if the oxidizer is passed through a negative temperature gradient and will agglomerate causing flow restrictions in small diameter passages such as the TIROC engine injector tube, etc. It would appear that thermal conditioning might therefore be required for these engines. However, no thrust chamber heater or heat shield would be required.

Because of the hypergolic chemical reaction of the bipropellant (compared to the catalytic or thermal decomposition of the hydrazine monopropellant) unreacted oxidizer or propellant can be expelled from the thruster if the valves are not synchronized. This could cause contamination on solar cells, etc. This situation is therefore more likely to occur in pulsed mode operation. Because of this potential problem, U.S. manufacturers would prefer to see all low thrust bipropellant engines use linked valves to ensure synchronization. If the TIROC valves are synchronized to 0.050 msec throughout life, however, there should not be an unburned fuel contamination problem.

The numerical reliability of a bipropellant subsystem will be lower than for a monopropellant subsystem with the same complement of engines because of the increased hardware complexity of pressure regulation

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and dual latching and engine valves. The constant supply pressure would, however, provide constant I_{BIT} throughout life which could reduce cycle life and SS life requirements.

The long term, eight years, compatibility of the oxidizer with the hardware, especially valve components. The oxidizer would definitely require all metal tankage. For the purpose of the weight estimate, the HS surface tension tankage weight was assumed. Should a metallic bellows tank be required, it would be considerably heavier (competitive in weight to an elastomeric diaphragm tank).

In summary, the U.S. vendors conclude that there does not presently exist a bipropellant thruster which has been proven to be fully developed to be suitable as the LTE for GPB, and which could provide overall subsystem performance enhancement over an all catalytic design. Spar concurs with this recommendation but will watch, with interest, the upcoming TIROC tests at NASA-LeRC and the reactions of the U.S. bipropellant vendors to the results of these tests.

2.1.2.5 Catalytic Hydrazine/Catalytic Plenum Gas Electrothermal Hydrazine Subsystem

Although no potential RCS vendor discussed this type of subsystem directly in his proposal, it has some advantages which should be investigated. As stated in Section 2.1.2.1, TRW has built and flown the DSP propulsion system using blowdown mass expulsion main tankage and catalytic thrusters for high thrust manoeuvres and unheated gas thrusters in the millilbf thrust range for attitude control. The gas used for attitude control, stored in a plenum tank, is produced from catalytic decomposition of hydrazine supplied from the main tanks and the pressure in the plenum tank is regulated. In addition, they have produced and flown gas electrothermal thrusters employing a thrust chamber heater to boost the C* (and thus ISP) of the gas using nitrogen as propellant. These two technologies can be combined to develop thrusters which could

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produce the min. I_{BITS} required for GPB at very low thrust levels (0.01 lbf). The I_{BIT} would be constant throughout the mission, thereby minimizing the cycle and steady state life required and would be delivered at an ISP of 150 lbf sec/lbm using a 3 watt heater on each engine. This ISP is higher than that quoted by the catalytic engines for this mode of operation, see Section 2.1.2.2. AVCO Corp. has been developing this type of system, employing an electrothermal gas-generator, for future NRL applications. This subsystem design, if configured to use catalytic thrusters for all functions except offset operation and gas electrothermal thrusters for offset operation, would weigh (for the six year worst case mission, surface tension tankage for the eight year worst case mission) approximately 264 lbm (including a 7 lbm hardware weight penalty for the plenum assembly and associated plumbing), see Figure 2-1. This design is, therefore, approximately 5 lbm heavier than the baseline design. The advantages of this configuration are greatly overshadowed by the disadvantages of:

- increased weight,
- reduced reliability due to increased complexity of the pressure regulated plenum assembly
- possible long term gas leakage with offset thrusters, and
- possible problem with operating at high duty cycle during capture because of throughput rate limitations of the plenum assembly.

Spar, therefore, recommends that no further consideration be given to this design.

2.1.2.6 Catalytic Hydrazine/Electric Propulsion Subsystem

The term electric propulsion is used in this report to refer to colloid, pulsed plasma and ion engine propulsion. Dr. W.F. Payne, who was Supervisor for the ion engine experiment for CTS, under

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subcontract to Spar, attended the AIAA 11th Electric Propulsion Conference held in New Orleans, La. on March 19-21, 1975 and subsequently prepared and presented his report on the status of electric propulsion to meet the requirements of north/south stationkeeping for GPB. His report, WFP 75-01, is contained in Appendix A. As can be seen from the document:

a) Dr. Payne believes that the mercury electron bombardment ion engine, with three possible vendors, shows the most promise for use on GPB, with the Hughes/LeRC SIT-8 thruster perhaps the forerunner at this point in time. In addition, he believes that "this technology is currently sufficiently advanced...such that with extensive ground testing on engineering model equipment, a sufficiently reliable subsystem can be produced". The SIT-8 design was based on the CTS requirements and facilities exist at the UTIAS, University of Toronto, for life testing the engine to provide significant Canadian content.

b) The major advantage of this type of engine is weight saving. The SIT-8 thruster operates at 2900 lbf sec/lbm ISP. Dr. Payne presents a baseline design for GPB which will be discussed below.

c) On the other hand, Dr. Payne states that the engine; requires 150 watts of power at high voltage requiring power processing, does not provide commonality between hydrazine fuel and north/south stationkeeping fuel, has not been unequivocally demonstrated by space flight and presents a contamination hazard to the spacecraft requiring use of baffles to protect solar arrays.

The subsystem design considered here is catalytic hydrazine (using the baseline engine complement) with blowdown mass expulsion tankage for all manoeuvres except north/south stationkeeping which is performed using the SIT-8 mercury ion thruster

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at ISP = 2900 lbf sec/lbm. With the spacecraft design proposed for GPB, a four thruster configuration would likely be chosen with two completely cross straped power processors. The location, as shown on Page 30 of WFP 75-01, is dictated by shroud dimensions, contamination considerations and location of the stowed solar arrays. As a result, the thrusters would be nominally vectored 46 degrees from north/south. Each thruster would occupy an area on the north or south panel of approximately 8" X 8". Mechanical gimballing of $\pm$ 10 degrees would be provided to account for C of M uncertainties. The total subsystem weight for the six year worst case mission would be as shown in Table 2.1-2. This assumes HS catalytic weights and two CTS tanks which would be appropriate for the hydrazine fuel required. The ion engines would be operated one engine at a time but at both nodes of the orbit each day for approximately 2.85 hours per burn yielding:

Thrusting Time per Orbit		5.70 hours
Total Thrusting Time		
(six years)	≈	12,480 hrs ≈ 3120 hrs. per thruster
(eight years)	≈	16,645 hrs ≈ 4160 hrs. per thruster

It is assumed that all of the 150 watts must come from the arrays and extra solar array weight has been added at 30 w/kg. The total subsystem weight would be approximately 186.5 lbm thereby saving approximately 72 lbm compared to the baseline design. A further 11 lbm could be saved if extra array power is not needed, see Figure 2-1.

Contained in Appendix A is a report by William Kerslake, Chairman, AIAA Electric Propulsion Committee on the results of a poll sent to spacecraft builders on the use of electric propulsion. This was compiled in 1975 in preparation for the March, 1975 conference. In summary, the builders call for a successful flight test and full mission cyclic system life test on the ground prior to acceptance for an operational flight. At present,

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Table 2.1-2

CATALYTIC HYDRAZINE/ELECTRIC PROPULSION SUBSYSTEM WEIGHT

Item		Weight (lbm)
Catalytic Subsystem Hardware		42.6 lbm*
Hydrazine Propellant		51.5 lbm
Nitrogen Pressurant		.7 lbm
	Catalytic Subsystem Total	94.8 lbm
Ion Engine Subsystem Hardware		65.2 lbm**
Additional Array 150 W at 30 W/Kg		11.0 lbm
Mercury Propellant		15.5 lbm***
	Ion Engine Subsystem Total	91.7 lbm
	Total RCS	186.5 lbm

* 4 Surface Tension Tanks and Mounting Struts Replaced by 2 CTS Tanks

** 3 lbm Added to Dr. Payne's Estimate for Engine Mounting Brackets and Baffles

*** Operating at Both Nodes Each Day. Dr. Payne Assumed Only Operating at One Node

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such a test has not been committed to any flight spacecraft. The recent failure of the Cesium bombardment ion engine system on ATS-F has done a disservice to the electric propulsion community and spacecraft primes are very leary of working with the new technology. Spar concurs with the results of this poll and would not recommend, at this time, the use of electric propulsion on GPB for a launch in 1979.

2.1.2.7 Catalytic Hydrazine/Liquid Electrothermal Hydrazine Subsystem

2.1.2.7.1 Liquid Electrothermal Hydrazine Engine Development

As stated in Section 2.1.2.1, both AVCO Corp. and TRW have been developing electrothermal hydrazine thrusters (EHT) for over five years. These engines use electrical energy converted into thermal energy (without catalyst) to vaporize and initiate the decomposition of liquid hydrazine. Once the reaction is initiated, it is then self-sustaining without further electrical heating providing the duty cycle is high enough to maintain the thrust chamber at a minimum preheat temperature of approximately 800°F. Both companies had, until recently, been concentrating of developing a 10 to 70 millilbf range thruster. AVCO was funded until 1971 by GSFC and have been working on their own funds since that time. In 1971, TRW received the GSFC funding and is still funded today on the third of such contracts.

TRW and AVCO both proposed to use these thrusters for the CTS LTE application. At that time, TRW had demonstrated over 1M cycles on a prototype EHT at the 35 mlbf thrust level and with repeatable I_{BITS} as low as 2 X 10^{-4} lbf sec. Since that time, TRW has built an engineering model configuration EHT for GSFC (it was to fly as an experiment on ATS-G had the program not been cancelled), see references 7, 8 and 9. This engine was successfully operated for 315,000 pulses and 30 hours steady state, although some nitriding occurred in the injector tube during the test. GSFC agrees with

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the TRW conclusions to the first study, contained on Page 61 of reference 8, which are:

- The general design of the EHT is adequate for flight qualification in terms of thermal, mechanical, and electrical interfaces.

- The steady state specific impulse of the thruster exceeded 200 seconds throughout the entire life test. The overall performance compares very favourably with that of equivalent-sized and larger catalytic thrusters.

- Pulse-mode performance is considerably better than that typical of small catalytic hydrazine thrusters in terms of response times, pulse repeatability, and specific impulse. The minimum impulse bit capability is considerably better than state-of-the-art catalytic thrusters.

- The reliability of the heater design for the EHT has been verified. No failures of the basic 30 ohm unit have been experienced in nearly 2 years of development.

- Performance and life capability of the Parker valve design utilized for the program has been verified. No anomalies were experienced during the estimated 3 x 10^6 cycles accumulated on two units during development and the Engineering Model tests.

- Haynes 25 proved sufficiently nitride-resistant to meet the performance and life goals of the program. With the substitution of noble metals in critical areas, it is believed that EHT life-time would be extended significantly.

A TRW company-sponsored test resulted in the successful demonstration of 100 hours of steady state life for the EHT. For the worst case 8 year mission and at thrust levels discussed below the steady state life requirements per thruster, without margin, would be approximately 250 hours.

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This engine can maintain 800°F with a 3 watt chamber heater (note that the catalytic design for 500°F is 2.2 watts) for offset operation and achieve an ISP of 140 seconds when operated in this mode. Steady state ISP of 210 to 230 seconds has been demonstrated. It is important to note that the bed heater for this engine is external to the chamber and does not come in contact with the liquid hydrazine or its exhaust products. A screen pack inside the chamber transmits the heat to the propellant.

AVCO, see Reference 10, initially designed their EHT with an internal heater, employing radial injection of liquid propellant. One of the major reasons why GSFC chose the TRW engine in 1971 was because of the use of an external heater; they were leary of heater life when in contact with the NH_3 exhaust product at high temperature. Since that time, AVCO have redesigned the engine so that the heater is no longer in contact with either the liquid propellant or the gaseous propellant or exhaust products. This engine has completed engineering and preflight qualification testing and delivers a steady state ISP of 235 to 240 lbf sec/lbm and min I_{BIT}, low duty cycle ISP of 160 lbf sec/lbm for a three watt chamber heater (chamber holding temperature 900°F). In this engine, the heat transfer is through the working fluid which has been radially injected. This engine and the TRW engine were optimized for the 20 to 40 mlbf thrust range.

Spar concurs with the conclusions drawn in reference 8. However, as yet, these devices have not flown even though the NH_3 resistojets on which technology they are based are currently in flight operation.

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2.1.2.7.2 Super-Heated Liquid Electrothermal Hydrazine Engine Development

The information contained within this paragraph, and elsewhere in the report regarding these engines, is considered company proprietary to each of the companies mentioned. It is Spar's request that this data not be transmitted beyond the personnel within the DOC and Telesat who have a need to know for GPB purposes.

The EHT offers the advantages of lower perturbation torques on the spacecraft, possible longer engine life than catalytic engines (where catalyst attrition can be a life-limiting factor), slightly higher steady state ISP and smaller repeatable linear impulse bits for offset operation than catalytic engines. A design with catalytic HTEs and EHT LTEs would have a weight savings potential compared to the baseline of approximately 10 lbm, see Figure 2-1.

AVCO and TRW have been developing, over the last year, a super-heated EHT engine operating in the millilbf thrust range, which, if used for GPB, could offer very significant weight savings compared to the baseline design. The EHT steady state chamber temperature is approximately 1800°F, governed by the adiabatic flame temperature of the decomposing hydrazine. If extra heat is applied, the remaining NH_3 can be dissociated (endothermic reaction) thereby reducing the molecular weight of the exhaust products and the temperature of these products can be boosted. This concept is not new, being in flight at present in the form of NH_3 resistojets.

The TRW approach (HiPEHT engine) is to use the EHT as a gas generator and feed the gas through a second injector tube to a gas vortex thrust chamber containing a bare element tungsten-rhenium heater element. This super-heating vortex chamber requires significant quantities of power and heats the exhaust gases to 3000° to 4000°F. Development tests are presently underway and the results are shown in Figure 2-2. This development engine does not have flight configuration coupling between the two chambers, nor does it have insulation. Therefore, the results are very encouraging, pointing towards achieving the performance goal with the final flight configuration hardware. One drawback to this design is the fact that the engine cannot be pulsed in the super-heated mode.

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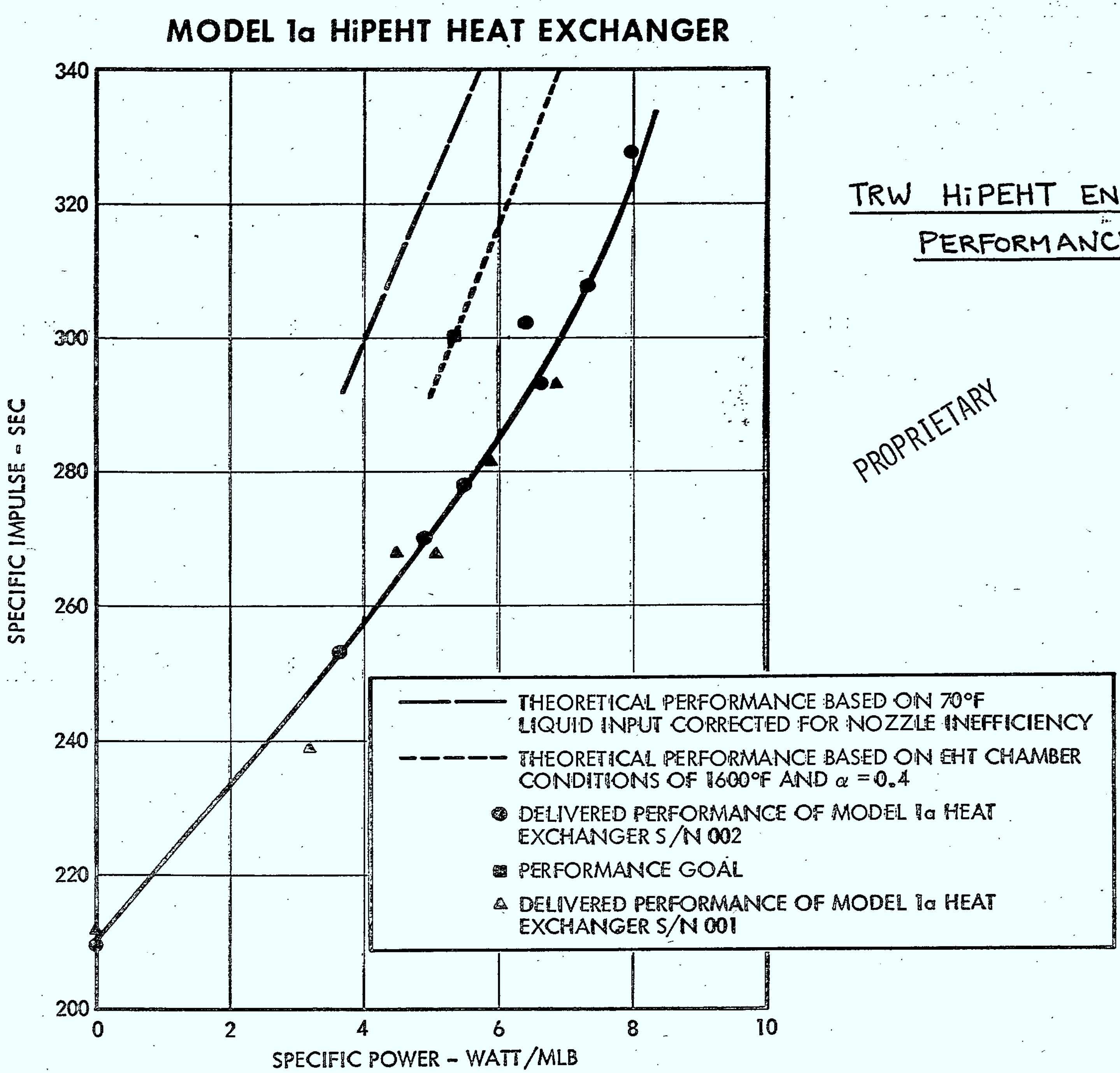

TRW HiPEHT ENGINE PERFORMANCE

PROPRIETARY

FIGURE 2-2

SPAR

The bare element, vortex chamber heater design requires gas flow to dissipate its heat by conduction. The power supply for this heater is interlocked with the engine valve to prevent it from coming on when the valve is closed. In the future, consideration will be given to developing a pulsing engine. However, if the super-heater is not energized, but the decomposition chamber heater is, the engine may be operated as a conventional EHT with a slight loss in performance (steady state ISP of 215 lbf sec/lbm). TRW are funding this engine development extensively in-house to support their upcoming prime bid for Intelsat-V. COMSAT have shown spontaneous interest in the design. A design verification program with engineering model hardware is planned during the third quarter of 1975 with completion of life testing at year end. In addition, heater and superscale fluiddynamic tests are presently in progress.

AVCO have also been developing a super-heated liquid electrothermal engine (SHE). Their approach is to increase the power at the decomposition chamber with heaters external to the fluid flow. Figure 2-3 shows the results of engineering model testing. This engine can be pulsed in the super-heated mode. As can be seen from a comparison of Figures 2-2 and 2-3, the TRW engine is, at present, more efficient than the AVCO engine, producing ISP of 300 lbf sec/lbm at 6.25 watts/mlbf compared to 7.4 lbf sec/lbm. This is, at least in part, a consequence of the internal versus external heater approach. AVCO shall be equipping the engine with three commandable, 75 watt, 35 VDC heaters. This arrangement will permit both step level power changes to correspond with blowdown thrust reduction and heater back-up flexibility. AVCO will invest prototype development funds during the next three months to fully establish the operating parameters and performance characteristics. They state that if sufficient interest is generated from potential users (ComSat, DOC, Telesat, etc.) preflight qualification would be performed through a flight demonstration.

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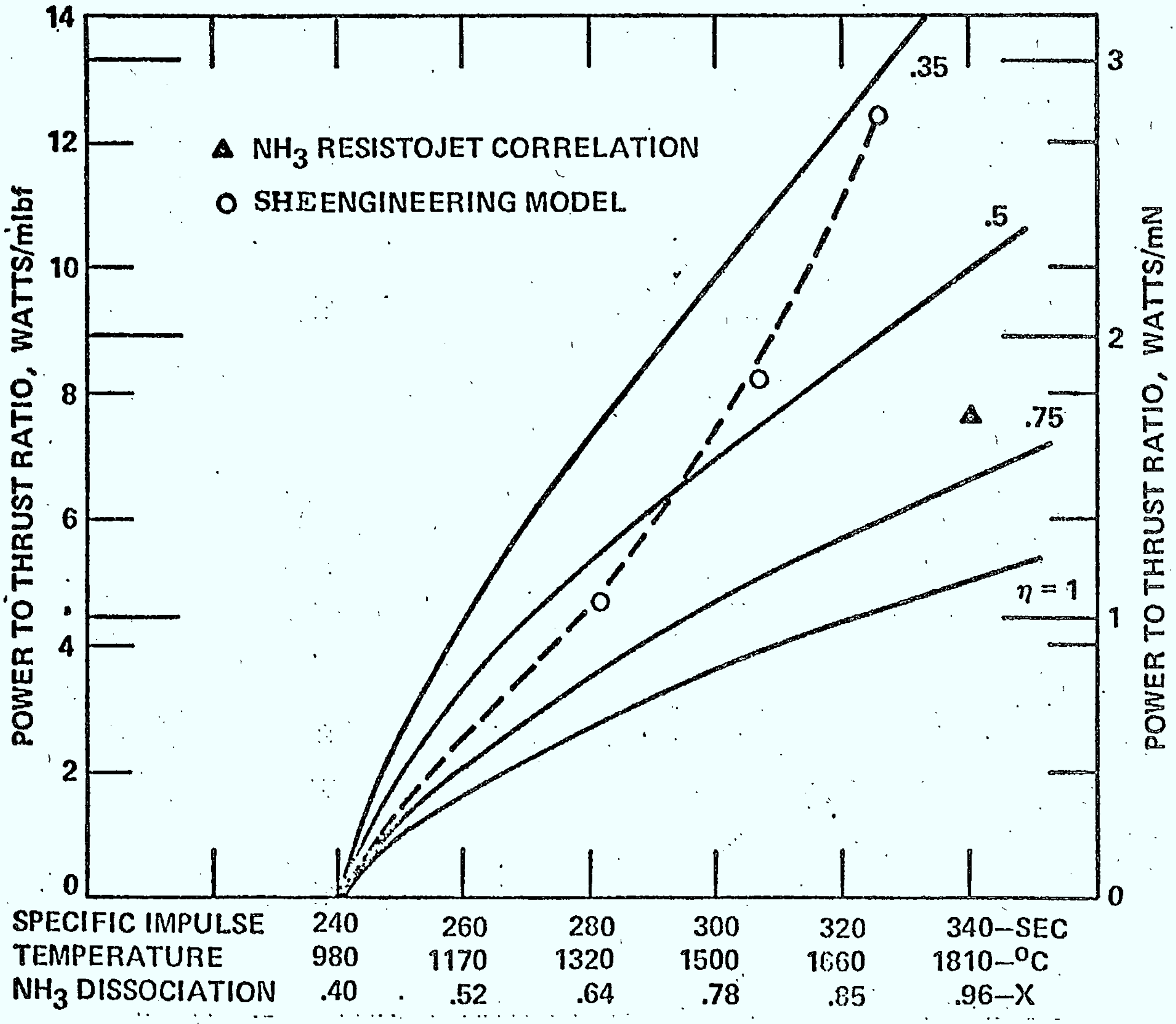

AVCO SHE ENGINE PERFORMANCE

FIGURE 2-3

2.1.2.7.3 RCS Design Using Super-Heated Liquid Electrothermal Hydrazine Thrusters

The super-heated liquid electrothermal hydrazine thrusters, with their high specific impulse, can very profitably be used for high impulse steady state manoeuvres. On GPB, the prime candidates are north/south and east/west station acquisition and keeping. Consider the GPB thruster complement with Engines #1, #2, #3, #6, #7 and #8 replaced by super-heated EHTS. Engines #11 and #12, required for on-pulse width modulation, would be conventional EHT devices as would all other LTEs.

A preliminary investigation of the availability of power to supply the super-heated EHTs follows. A more detailed analysis should be carried out once the power subsystem and thruster performance have been better defined. Trade-offs, which are dependent upon voltage and energy available from the power subsystem, should be performed between thrust level, power level, ISP, and number of days between manoeuvres. In this report time permits only the choosing of one set of values and a demonstration of the feasibility of the design.

Assume:

a) Beginning of life thrust, two engines = 30 mlbf, therefore end of life thrust, two engines ≃15 mlbf for GPB blowdown.

b) Multiple heater elements to maintain desired power to thrust ratio.

c) Use worst case value of 7 watts/mlbf BOL to achieve 300 lbf sec/lbm, perhaps 8 watts/mlbf EOL due to heater step size.

d) Worst case EOL conditions:

- only 35 watts of solar array power available

- equinox condition (72 minute eclipse)
- 800 watt eclipse power required at DOD = 50% on batteries at end of eclipse
- charge rate of all three batteries = C/20

e) All three batteries operational, total capacity 1920 watt hours

f) Both north and south stationkeeping are performed.

g) Frequency of north/south stationkeeping once every 21 days.

h) Worst case six year mission requiring ΔV_{n-s} = 999 ft/sec

i) At this thrust range, plume impingement, as discussed in Section 5.2.2.4, Volume I, will not be as significant allowing a reduction in the cant angle of engines #1, #2, #6, #7, #11 and #12 and reduction of the additional 2.5% impingement penalty. Assume that the cant angle will remain 15° but the additional penalty will be removed.

The choice of 30 mlbf BOL thrust is consistent with the heater power being provided in the development engines for 7 watts/mlbf. In the 15 to 30 mlbf thrust range with stationkeeping once every 21 days, a small increase in propellant of 0.5 to 1.5% can be expected due to not firing impulsively at the nodes; (since the batteries are watt hour and not watt limited, it would be beneficial to increase the thrust level as much as is practical to reduce this inefficiency).

At equinox, north/south stationkeeping will be performed centred around 6 a.m. and 6 p.m., see Section 5.2.1.1 (q), Volume I. The manoeuvre need never be performed in eclipse. At the end of life, one pair of North or South thrusters would be fired for approximately 2.5 hours centred at 6 a.m. and the other pair of South or North thrusters

CFC/26

for 2.5 hours centred at 6 p.m. There would be a 1.5% non-impulsive cosine loss. At 8 watts/mlbf the total energy required per burn would be 300 watt hours. The arrays could supply 87.5 watt hours leaving the battery to supply 212.5 watt hours. Table 2.1-3 shows the battery state of charge as a function of time of day. Note that the maximum depth of discharge occurs as a result of eclipse operation and not stationkeeping and that the batteries go into eclipse in a state of overcharge. It is Spar's understanding that these extra two discharge-charge cycles for the batteries every 21 days might promote longevity. In fact, more power will be available from the array for most, if not all, of the mission. Should one battery or one N/S stationkeeping thruster fail, it might be necessary to increase the frequency of N/S station-keeping. Alternatively, the manoeuvre could be arranged so as not to occur at maximum eclipse. N/S station acquisition and E/W station acquisition and keeping impose less severe drains on the power subsystem.

In summary, with either of the two super-heated EHT designs, it is feasible to perform N/S and E/W station acquisition and keeping at an ISP of at least 300 lbf sec/lbm without requiring additional solar array hardware.

Considering the AVCO engine which can pulse, the subsystem weight for the six year worst case mission, tankage for eight years (assuming; SHE I_{sp} of 300 lbf sec/lbm, EHT steady state I_{sp} = 235 lbf sec/lbm, min I_{BIT}-low duty cycle I_{sp} = 160 lbf sec/lbm, HS catalytic weights and surface tension tankage) would be approximately 218 lbm. This would save approximately 41 lbm compared to the baseline design, see Figure 2-1. It should be pointed out that the HS weights are being used as part of the subsystem weight for comparison purposes only. Weight savings for the eight year mission would be even more significant. The TRW engine presently unable to pulse while super-heated, might be less efficient because of the possible need, due to thrust mismatch and/or X-X axis C of

Table 2.1-3

BATTERIES STATE OF CHARGE DURING NORTH/SOUTH STATIONKEEPING

Event	Time of Day (Hour:Min)	Batteries Charge Level (Watt Hours)	Depth of Discharge	Percentage Overcharge
Beginning of Eclipse	23:24	1920	0%	0%
End of Eclipse Start Charge at C/20	00:36	960	50%	-
Stop Charging Start Stationkeep	4:45	1358	29%	-
Stop Stationkeep Start Charge at C/20	7:15	1146	40%	-
Stop Charging Start Stationkeep	16:45	2058 (1920)	-	7%
Stop Stationkeep Start Charge at C/20	19:15	1708	11%	-
Stop Charging Beginning of Eclipse	23:24	2106 (1920)	-	10%

M excursions, to pulse an inefficient engine. The Catalytic Hydrazine/Liquid Electrothermal Hydrazine (with Super-heat) Subsystem design has been inserted as the last contender in this report not because of lack of merit, but rather because of merit and for emphasis. Although the development status has not advanced to a point where Spar would recommend the design as the baseline, we are looking very closely at the testing and analysis work presently in progress. We understand that several other primes are also doing so. Spar believes that these designs are based on proven technology and that if development and prequalification prove successful prior to RFPs being released for GPB, the mechanical prime should consider such a subsystem as a prime candidate for the RCS.

2.2 Potential Vendors - DSA Subsystem

2.2.1 General

This section lists potential vendors for various parts of the General Purpose Bus Deployable Solar Array subsystem. A general assessment is made of each one's capabilities based on previous experience.

2.2.2 Solar Cell Array

Component/ Assembly	Potential Vendor	Assessment
Solar Cells	AEG-Telefunken, Heilbronn, West Germany	- Experienced solar cell supplier for space projects - European, Canadian and U.S. Are developing new High Efficiency cells. Usually more expensive than U.S. suppliers.
	Societe Anonyme de Telecommunications, Paris, France	- Have supplied solar cells for various European and national programs. Also some U.S. programs. Are developing new High Efficiency cells. Usually more expensive than U.S. suppliers.
	Heliotek/Spectrolab Sylmer, California, U.S.A.	- Most experienced solar cell suppliers for U.S. and Canadian space programs. Are now producing the Violet Cell under licence from Comsat. Most cost competitive.
	OCLI/Centralab, Santa Rosa, California, U.S.A.	- Have supplied solar cells for U.S. space programs. Provide U.S. competition to Spectrolab.
Solar Cell Arrays	AEG-Telefunken, Wedel, West Germany	- Supplier for the CTS Project, European and German programs. Experienced in both rigid and flexible arrays. Leader in

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	welded interconnect technology. May be more expensive than U.S. suppliers.
S.A.T., France	- Have developed a flexible, fold-out solar array for Comsat (via Aerospatiale). Some experience on European space programs.
Ferranti Ltd., U.K.	- Have developed a flexible, flat fold solar array with R.A.E. Farnborough using wrap-around interconnects, substrate with holes. A 64W model has flown on a low orbit U.K. experimental satellite.
Spectrolab, U.S.A.	- Experienced house for lay-down of solar cells on rigid substrates for many U.S. and Canadian programs.
Hughes Aircraft, El Segundo, California, U.S.A.	- Leader in developing flexible roll-up arrays (FRUSA). Were very interested in initial CTS RFPs as flexible array suppliers.
Lockheed Missiles & Space Co., Sunnyvale, California, U.S.A.	- Performed considerable development work for large flexible solar arrays for Intelsat V and the Space Station. Have developed rigid array designs for Intelsat V.
TRW Systems, Redondo Beach, California, U.S.A.	- Experienced in rigid arrays for U.S. Space programs. Working on a new rigid frame/semi-flexible substrate lightweight design.
Boeing Aircraft, Seattle, Washington U.S.A.	- Were interested in CTS RFPs for flexible arrays. Not very well known for solar arrays, but have experience in U.S. programs.

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	Messerschmitt-Bolkow, Blohm, Germany	- Have done development work under European and German contracts. Have European space program experience. (AEROS, Dial)

2.2.3 Stowage and Development System

Honeycomb Substrates	Fleet Aircraft, Fort Erie, Canada	- Suppliers of honeycomb substrates and panels for Canadian space programs and other aircraft programs. Major supplier for CTS. May be more expensive than U.S. suppliers for lightweight core, skins and adhesives.
	Boeing Aircraft, Winnipeg, Canada	- Have honeycomb panel manufacturing capability. Have not supplied for space programs but have good aircraft panel background.
	Boeing Aircraft U.S.A.	Experienced honeycomb panel suppliers for space and aircraft programs.
	Hughes Aircraft U.S.A.	Experienced honeycomb panel suppliers for space and aircraft programs.
	Heath-Tecna, Seattle, Washington U.S.A.	Have built honeycomb substrates for the G.E., B.S.E. spacecraft. Build to print.
	Parsons, Stockton California, U.S.A.	Have built structure and array substrates for R.C.A. SatCom to print.
Other Rigid Substrates	McDonnell Douglas, U.S.A.	Have made an experimental isogrid panel substrate. Not flown.

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Pyrotechnic Actuators and Cartrides	Holex Corp. U.S.A.	Experienced pyrotechnic actuator vendor for aerospace missions. Supplied CTS DSA pyros.
	Teledyne McCormick Selph, U.S.A.	Proposed on CTS DSA pyros. More expensive than Holex. Experienced supplier.
	Hi-Shear, U.S.A.	Experienced supplier. Proposed on CTS DSA pyros. Cost competitive with Holex.
	Conax Corporation, U.S.A.	Potential supplier. Did not respond to CTS DSA RFP due to shortage of time.
Deployment Motors-D.C.	Singer-Kearfott U.S.A.	Experienced vendor for space programs. Supplied CTS DSA D.C. Motor. Developed brush problems (incorrect selection).
	Clifton Motors, U.S.A.	Experienced vendor. Used on many Spar STEM antennae programs.
	Globe Motors, U.S.A.	Experienced vendor. Used on Spar STEM antennae programs.
	Nash Controls, U.S.A.	Experienced vendor. No bid on CTS DSA RFP.
Stowage and Deployment System	Engins MATRA France	Have developed and tested a lightweight rigid aluminum honeycomb, fibreglass skin design. Do not seem to be active in this field at present.
	MBB, Germany	Supplier of OTS and MAROTS rigid arrays using carbon fibre composite skins, aluminum honeycomb substrates, cable deployment system. Developing ultra-lightweight design for higher power arrays. Have also developed self-rigidizing arrays.

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	Lockheed Missiles & Space, U.S.A.	Have developed a rigid array design for Intelsat V. Pantograph deployed.
	TRW Systems U.S.A.	Have flown rigid deployable arrays on U.S. space programs - Fleet SATCOM, Skylab
	Spar Aerospace Products Ltd. Canada	Suppliers of CTS Solar Array Mechanical Assembly. Have designed a lightweight rigid frame/flexible substrate array - awaiting tests. Have a preliminary design for a rigid substrate array.

2.2.4 Orientation and Power Transfer System

Drive Motors	Singer-Kearfott U.S.A.	Suppliers of CTS DSA stepper motors
	Superior Electric U.S.A.	Suppliers of stepper motors. No bid CTS DSA RFP.
	Novatronics, U.S.A.	Suppliers of stepper motors.
	Sigma, U.S.A.	Suppliers of stepper motors.
	IMC, U.S.A.	Suppliers of stepper motors.
	Inland Motors U.S.A.	Suppliers of DC Torque Motors.
	Aeroflex, U.S.A.	Suppliers of DC Torque motors.
Encoder	Conrac-Duarte, U.S.A.	Supplied CTS DSA optical encoder.
	Singer-Librascope U.S.A.	Proposed on CTS DSA encoder - magnetic encoder.
	Litton, U.S.A.	Proposed on CTS DSA - contact encoder.
	United Aircraft, U.S.A.	Proposed on CTS DSA - contact encoder

Slip Ring Assembly	Polyscientific, U.S.A.	Supplied CTS DSA slip ring assembly via Ball Brothers. Very experienced suppliers for space programs.
	Spar Aerospace Products Ltd. Canada	Have developed several designs. Life tested a breadboard for Intelsat V program with Lockheed.
Orientation & Power Transfer	Ball Brothers Research Corp. U.S.A.	Experienced U.S. supplier of drive mechanisms for U.S. space programs. Supplying a direct drive brush torque motor drive system to Rockwell for the Global Positioning Satellite.
	Scheaffer Magnetics Inc., U.S.A.	Experienced U.S. supplier. Recently supplied a mechanism for Lockheed.
	General Electric, U.S.A.	Have developed a stepper motor driven, harmonic gear reduction device. Being used on the Japanese BSE spacecraft.
	TRW Systems, U.S.A.	Have flown mechanisms on Nimbus, ERTS and Fleet SATCOM. Developing a mechanism for Comsat.
	Marconi Space & Defence Systems U.K.	Have developed mechanisms under contract to ESTEC. Use lead lubrication and a brush torquer.
	Hawker Siddeley Dynamics, U.K.	Supplying the system for OTS and MAROTS. Use lead lubrication and brushless torquer/resolver system.
	Lockheed Missiles & Space, U.S.A.	Not very experienced in this field. Use Scheaffer Magnetics as a vendor,

Bendix Corp. U.S.A.	Have developed a torquer drive unit with complex electronics for Grumman. Not flown.
R.C.A., U.S.A.	Have developed a brushless torquer resolver system for the SATCOM Spacecraft. Wet lubricated.
Spar Aerospace Products Ltd. Canada	Have developed systems for Lockheed (Intelsat V) and CTS. The former has been life tested. The latter is undergoing life test. Uses a stepper motor with a spur gear reduction.

2.3 Vendor Review and Assessment - Apogee Motor

For this study specification Requirement SPAR-SG.356 was prepared and four vendors were solicited for response to the requirements. They were:

Bristol Canada, Winnipeg,
United Technology/United Aircraft Canada,
Aerojet General,
Thiokol, Elkton Division.

Replies have been received by all companies solicited and their responses are available on request. To summarize it can be said that two companies Bristol and United Technology indicated that because of insufficient time they were unable to respond with a formal proposal, however, UTC have supplied a budgetary cost. With the other two vendors it can be said that their responses are good; both Aerojet General and Thiokol have motors developed and tested, Aerojet being the supplier for RCA's Domsat program. There is little difference between the two proposals received, and it is felt that at this time a selection is difficult to make since both proposals covering price and motor performance indicate a most economical and cost effective program.

It is also unfair at this time to expose competitors prices and proposal details, nevertheless in order to provide the readers of this volume with sufficient information on Apogee motors Thiokol's technical proposal is provided as an appendix and forms part of this study report.

3.0 APPENDICES

This section includes:

Appendix A	-	Ion Engine And Poll on Electric Propulsion
Appendix B	-	RCS, Hamilton Standard Proposal
Appendix C	-	Apogee Motor, Thiokol Proposal

APPENDIX A

ION ENGINE REPORT

AND POLL ON ELECTRIC PROPULSION

WFP 75-01

March 1975

AN ION ENGINE SUBSYSTEM

for

THE UHF BUS

Report prepared for SPAR Aerospace Products Ltd.

under Sub-contract 37234-X

Dr. W.F. Payne, P. Eng.

W. F. PAYNE

PROVINCE OF ONTARIO

I N D E X

A. Introduction

Purpose

The purpose of this report is to assess the state of readiness of electric propulsion for station-keeping tasks on a geosynchronous communications satellite.

Summary Conclusion

The current technology of mercury ion thrusters is sufficiently advanced that this form of propulsion must be seriously considered as an alternative to hydrazine for long-term station-keeping tasks on synchronous satellites. A typical design is presented as a baseline for trade off studies.

B. Advantages and Disadvantages of Ion Propulsion

1. Advantages

1.1 Propellant Savings

The most significant feature of an electric propulsion system is the mass saving that can be realized when compared to a chemical propellant system that is sized to provide the same total impulse. It must be borne in mind, however, that the absolute thrust level is very much lower for typical electric propulsion device, and that it must, therefore, thrust over appreciable mission periods. For a 450 Kg geosynchronous spacecraft with a north-south station-keeping requirement, it is expected that the useful payload may be increased by 45 kg through the use of electric propulsion instead of hydrazine. This extra payload may be then used for additional communications capability - thereby generating increased revenue.

1.2 Spacecraft Charging Control

It has been demonstrated that the charging of geosynchronous spacecraft during eclipse periods can be controlled by the neutralizer of an ion engine. It is expected that the differential charging of adjacent spacecraft surfaces could be reduced, thus lowering the probability of discharges across the spacecraft skin.

1.3 Time Sharing of Spacecraft Battery

The total power requirement of the ion engine need not be derived from the solar array. As communications satellites will carry relatively large batteries (e.g. 400 W-hrs.), this capacity could be tapped intermittently by the ion engine during non-eclipse periods when the power is not required for the communications equipment. Additional solar array capacity may thus not have to be charged to the ion engine system.

1.4 East-West Station-keeping

Although the velocity change for this task is much smaller than that needed for north-south station-keeping, most configurations for ion engine mounting realize thrust in the east-west direction as well. By judicious management of the thruster on-times, east-west station-keeping can be accomplished at the same time as north-south.

Limited spacecraft repositioning is also a possibility.

1.5 Momentum Dumping

If the satellite is three-axis stabilized and a momentum wheel is used in the attitude control system, then the wheel speed can be adjusted during thruster operation by vectoring the ion engine. All the systems listed in Appendix I have the capability of thrust vectoring.

2. Disadvantages

2.1 Plume Impingement

The ion engine must be oriented so that the accelerator grid of the ion optics does not have a direct line of sight to sensitive spacecraft surfaces. On the positive side, however, the torque produced by any propellant which impinges on an adjacent surface

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should be lower than that produced by the plume of a conventional thruster.

2.2 Commonality of Propellant

The electric propulsion system cannot normally share the propellant used by the attitude control system.

2.3 Requirement of Electrical Power

Additional electrical power is required for these devices. In addition, power processing is also necessary.

2.4 Reliability

The use of electric propulsion has not been unequivocally demonstrated by space flight.

C. Types of Thrusters Available

Electric propulsion encompasses a wide range of devices, all of which produce thrust by the expenditure of electrical energy. This report will be limited to consideration of a smaller group of electric propulsion devices which are suitable for north-south station-keeping tasks on a 454 Kg (1000 lb) geosynchronous spacecraft. It will be shown that a typical system of this type will have significantly lower mass than a hydrazine system designed to accomplish the same station-keeping task.

At the present time there are five types of electric propulsion thrusters which have been developed for station-keeping.

They are:

- pulsed plasma, or Hall accelerator devices
- resisto jets
- colloid thrusters
- contact ionization engines
- ion engines (electron bombardment engines)

1. Pulsed Plasma Devices

These devices work by producing an intermittent capacitive discharge which vapourizes and ionizes a solid propellant. Then, by a combination of gas dynamic and magnetohydrodynamic forces, accelerates this gas to produce thrust. This type of thruster has been used successfully on several spin-stabilized satellites for east-west station-keeping. Thrust levels are generally too low for north-south station-keeping tasks, and thus will not be considered further here.

2. Resistojets

Resistojets are small thrusters that obtain higher specific impulse by means of resistive heating elements which raise the enthalpy of the

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of the propellant prior to gas dynamic expansion. The specific impulse of this device is intermediate between a pure chemical propellant and the remainder of the thrusters under consideration.

Derivative devices of this nature are more closely associated with conventional thruster systems, and will enter a propulsion trade-off analysis from that point of view.

3. Colloid Thrusters

Colloidal thrusters achieve useful thrust by accelerating charged droplets of a substance such as glycerine through electrostatic potentials greater than 10 KV. These devices have not been under development for as long a time as the other thrusters. Thus, significant life test data is not yet available. The thruster has the advantage of extreme simplicity. Future development of this thruster should be monitored, but it is not considered ready for flight at this time. Present details of this system are presented in the first Appendix.

4. Contact Ionization Thrusters

These ion engines use cesium as a propellant which is ionized by means of a heated porous tungsten plug. The ions are subsequently accelerated and the ion beam neutralized to a plasma by an electron-emitting filament. Development on this device was stopped in 1970 in North America; work is proceeding only in France. Procurement of flight quality hardware for the type of device would be difficult.

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5. Electron Bombardment Thrusters

At this time, the major development effort is centered on electron bombardment thrusters. These devices produce a source of ions by means of an electrical discharge sustained (in most cases) by an auxiliary electron source. Ions are accelerated from the plasma produced by the discharge and the ion beam is the neutralized at some point downstream by a device called a plasma bridge neutralizer. Either cesium or mercury are used as propellants, although heavy noble gases such as Argon or Xenon have been tried in the laboratory.

At this time, there are four distinct centers that have developed bombardment thrusters for auxiliary propulsion. These centers and their thrusters are:

- Electro-optical Systems EOS - cesium bombardment
- University of Giessan W. Germany - radio frequency thruster
- RAE Farnbourough, Culham Labs - mercury bombardment
- Lewis Research Center, LeRC - mercury bombardment

Some of the characteristics of these devices are given in more detail in the Appendix I.

D. Current Progress in Ion Engine Technology

Two parallel approaches have been followed in the recent development of ion thrusters for auxiliary propulsion.

Firstly, life-time testing of flight representative hardware has been undertaken. This testing is designed to simulate the cyclic requirements of station-keeping tasks, as well as to establish representative wear-out modes and typical life-times. A short list of these tests is given in the section on Reliability.

Secondly, a number of recent design improvements have been incorporated in the thruster design. The following features are particularly to be noted:

1. <u>Compensated Dished Grids</u>

   This design of accelerator-screen-grid-pairs is of greater importance for the larger prime propulsion thrusters. It has been shown, however, that by designing the smaller grid systems with a shallow convex profile, the grid spacing is more constant during thermal cycling. The propellant "ion" trajectories are more stable with time. Beam divergence is reduced by compensating the hole alignment near the edge of the grid system. Smaller divergence leads to higher thrust per ion (of the order of 2%), and more importantly reduces the number of beam ions that may impinge on spacecraft surfaces.

2. <u>Small Hole Accelerator Grid</u>

   An optimization in ion optics design has led to a grid design with a smaller percentage open area. This design allows fewer neutral propellant molecules to exit from the discharge chamber. The resulting higher propellant utilization thus raises the electrical efficiency, and in addition reduces the number of charge exchange ions that can impinge on spacecraft and thruster surfaces. Less impingement implies longer component life-times.

This small hole geometry also allows the accel grid to be run near the spacecraft surface potential because the neutralizing electrons are more effectively screened from the high positive discharge chamber potentials. This further reduces charge-exchange erosion and may even allow the deletion of one power supply for the thruster.

3. Optimized Neutralizer Location

Continuing studies of the neutralizer location have resulted in an optimized location for two of the mercury bombardment designs. This location again minimizes the beam erosion and charge-exchange sputtering that was observed on the SERT II thrusters. The neutralizer-beam coupling voltage has been shown to remain acceptably low (about 10 volts).

4. Cathode Impregnation and Ignition

The usefulness of thrusters as station-keeping devices requires that the cathode emission does not drop with time, and that reliable restarts can be readily accomplished. To this end, two types of emissive inserts were evaluated, a high voltage pulse ignition system was tested and a series of contamination tests of cathodes were performed. A life test of a complete Cathode-Isolator-Vapourizer (CIV) assembly was run for 20,000 hrs.

5. Reduction in Discharge Chamber Sputtering

Success has been achieved in reducing the effect of sputtered material within the discharge chamber. The use of tantalum and graphite components, as well as surface treatment of stainless wire mesh anodes, has resulted in a reduction in the formation of loose material within the chamber. Material which does form in either fixed to the chamber walls, or is of sufficiently small size that it cannot interfere with the thruster operation.

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6. Vectoring Capability

A change in the philosophy of thrust rectoring is apparent in the most recent designs. It would appear that mechanical gimbals now represent the most reliable approach to vectoring. Gimbal designs have been built and vibrated in the past year. These advances represent a part of several on-going programs of thruster development.

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E. Reliability of Ion Propulsion

The most conservative spacecraft design is always considered to be the most reliable. Thus the spacecraft designer is expected to choose flight-proven hardware. Against this bias, he must weigh the anticipated benefits of newer technology devices. Electric propulsion presents a classic case of this trade-off process.

To the spacecraft designer, ion engines represent a new and relatively unfamiliar solution to auxiliary propulsion requirements. It is this unfamiliarity that mitigates against its adoption as a baseline subsystem. In addition, the operating life-time requirements for an electric propulsion system are orders of magnitude longer than for higher thrust chemical systems, thus making reliability goals more stringent for electric propulsion. These two hurdles, unfamiliarity and life-time are addressed in turn.

1. The Ion Engine Enigma

A number of objections to the use of ion propulsion have been raised by spacecraft designers unfamiliar with these devices. It is felt that the existing flight data should be sufficient to give confidence in the capability of ion propulsion to be successfully integrated on spacecraft. Table I lists the known flights of operating ion thrusters. Some of the better known criticisms of ion thrusters are discussed below.

1.1 Beam Neutralization

In order for a thruster to be able to produce thrust, it is necessary that the ion beam be neutralized by some means. The

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TABLE I

NO.	DATE	LAUNCH VEHICLE	ION ENGINE	MANUFACTURER	AGENCY	RESULT
1.	18 Dec 62	Ballistic (Scout)	Cs - Contact	EOS	Air Force	- failed - battery vapour caused P.C. arcs
2.	20 Jul 64	Ballistic (Scout) SERT I	a) Cs- Contact b) Hg - Bombardment	HRL LeRC	NASA NASA	- a) failed - arc in high voltage connector - b) successful
3.	29 Oct 64	Ballistic (Scout)	Cs - Contact	EOS	A.F.	- successful
4.	21 Dec 64	Ballistic (Scout)	Cs - Contact	EOS	A.F.	- successful
5.	3 Apr 64	Orbital Snap 10-A	Cs - Contact	EOS	A.F.	- telemetry failed, ion engine test prematurely terminated
6.	10 Aug 68	Synch Orbit ATS-4 (Atlas-Centaur)	a) CS - Contact b) CS - Contact	EOS EOS	NASA	- successful for two month S/C life
7.	12 Aug 69	Synch Orbit ATS-5 (Atlas-Centaur)	a) Cs - Contact b) Cs - Contact	EOS EOS	NASA	- S/C in flat spin, meutralizer only operated
8.	3 Feb 70	Orbital SERT II (Thorad/Agena)	a) Hg - Bombardment b) Hg - Bombardment	LeRC LeRC	NASA	- a) 3781 hrs then shorted b) 2011 hrs then shorted* *restarted 16 Aug 74
9.	30 May 74	Synch Orbit ATS-6 (Titan 3E-Centaur)	a) Cs - Bombardment b) CS - Bombardment	EOS EOS	NASA	- 1 hr on one thruster 92 hr on second

SERT I test has proved that the beam can be neutralized and that thrust can be produced. All subsequent tests have shown this as well.

1.2 Plume Impingement

The interaction of the exhaust plume of the thruster and various spacecraft surfaces has been investigated by theoretical studies and measured explicitly on the SERT II and the ATS-6 flights. The SERT II data show that it is important to consider the effects of sputtered eflux from the thruster, but that this constraint can be accommodated. ATS-6 has shown that the thruster can be integrated on a satellite not intended specifically as an ion engine test bed. Polaris star tracker and radiation detector interfaces were successfully dealt with.

1.3 Interference with Communications

In all flights but one, interference with the command and telemetry systems was eliminated during EMC testing. Careful attention to electromagnetic compatibility requirements and to ground testing will ensure that this is not a problem. No interference between the ion engine and the communications experiments was measured during thruster operation on ATS-6.

1.4 Tankage

ATS-6 has demonstrated difficulty with a capilliary feed system using cesium as a propellant. It is significant that the tankage on SERT II is continuing to provide mercury propellant to all cathode and neutralizers after five years in space. Ongoing

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ground test data also confirm that mercury propellant storage is not a problem.

1.5 Restart Capability

SERT II has been able to demonstrate restart capability after five years in space. Cyclic restart capability has been testing in space and extensively on the ground.

2. The Life-time Requirement

In addition to the specific criticisms raised against ion engines, the requirement for a demonstrated life-time exists. It is felt that the recently reported endurance testing results are significant in this regard. This is particularly true because the required thrusting times have been somewhat reduced by raising the thrust level of the devices. Thus the required thrusting time per thruster has been lowered to about five thousand hours, while at the same time the demonstrated life-times have been raised to at least 10 thousand hours. The most significant tests in this light are the currently continuing tests listed in Table II.

TABLE II

GROUND LIFE TESTS

- SIT 8 - Hg - Bombardment	- 10,700 hrs. 317 cycles
- 30 cm - Hg - Bombardment	- 8,900 hrs.
- SIT 5 - Cathode, isolator vapourizer	- 20,000 hrs.
- T-4 - Hg - Bombardment	- 1,050 hrs. no failures
- EOS 8 cm - Cs - Bombardment	- 4,348 hrs (including 2,614 hrs full thrust 471 cycles)
- RIT 10 Hg - Bombardment duration test	- 1,000 hrs.

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It can be seen that the current development effort is strongly oriented towards proof of adequate useful life-times (including cyclic operation). In addition to these tests, the thruster wear out modes themselves are being investigated and some novel solutions have been tested. The most notable recent development involves the internal finish of the Hg bombardment discharge chamber. It has been shown that grit-blasted wire mesh does not give rise to "flakes" of sputtered grid material. As this was the failure mechanism demonstrated for the SIT 5, this development should mean a significant advance in projected thruster life-time. Wear out times of 20,000 to 30,000 hrs. are now being projected. More importantly though, demonstrated ground test life-times for thrusters have now reached almost 10,000 hrs. in two separate tests.

The power processing of the ion engine subsystem must also be assessed from a reliability stand point. The power processing requirements of ion engines should not be dismissed lightly. However, it is considered that the electronics is well enough developed that reasonable reliability numbers can be assigned. An assessment of the state of progress is given for each of the thrusters considered in Appendix I. Specifically a reliability analysis has been performed for at least three types of power processors. Standard electrical engineering design practices can be used to increase the reliability of the electronics, if this is deemed necessary.

A very good assessment of the reliability of ion engines is found in "Ion Propulsion Flight Experience, Life Tests and Reliability Estimates", J.H. Molitor AIAA Paper 73-1256. Nov. 1973. This paper is attached to

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the present report because it is felt that ion engine reliability is such an important aspect when considering the adoption of ion engines as a baseline. The present report lists the current status of some of the tests discussed in this paper. (Please see Tables I and II.)

3. <u>More Conservative Mission Approach</u>

One final point with regard to reliability can be made. Another trade-off study which has been performed recently presented a more conservative mission planning approach. As the ion propulsion system, even with extensive ground test data, will not have been experimentally flown, it was proposed to carry sufficient chemical propellant for a two-year mission. The ion engines were then sized for an additional 6 years of station keeping. The mission was thus planned for 8 years with a "guarantee" of 2 years minimum life.

F. The CTS Legacy

Although the ion engine experiment was removed from the Communications Technology Satellite (CTS), the effort that was expended prior to termination has had some effect in the direction of development of electric propulsion in the last three years. Indeed, the CTS ion engine specification was used as the basis for a subsequent power processor development contract at Hughes Research Labs (HRL).

Specifically the command, telemetry and power interfaces were derived directly from CTS work. This has some interest for Canadians, as it is not unreasonable to expect other subsystems such as the command and telemetry subsystems would be similar to CTS on a future Canadian satellite.

The trade-off studies produced for the CTS SIT-5 engine are germaine to the present trade-off for location of the thrusters on a future communication satellite. As well, assessment of the various thruster technologies was made simpler by past work in this area.

A study produced to investigate the ecological effects of mercury in the upper atmosphere is still pertinent to any future application.

On another study a design was prepared in Canada for a unique ground support device that would allow great flexibility for Electromagnetic Compatibility testing (EMC) of all spacecraft systems with the ion thruster system. This design is expected to be directly transferable to a future project. Further, the capability exists for the manufacture of this system in Canada.

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Interface requirements for ion engines on the CTS program required planning for cyclic operation, thruster eflux studies, a long term ground life test, and planning for software controlled remote turn-on. All these tasks are expected to be similar on another program.

In short, the CTS ion engine experiment, even when not carried to completion, has been able to serve as a logical starting point for the integration of a ion engine subsystem on a Canadian satellite.

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G. Canadian Test Facilities

The Institute for Aerospace Studies (UTIAS) is equipped with an unique facility capable of testing ion engines. This Multi Purpose Space Simulation Facility is equipped in part with a nine foot diameter by nine foot long ultra high vacuum chamber, associated pumps and instruments sufficient to monitor the operating environment of a thruster.

UTIAS also has a partially built laboratory-type breadboard power supply. This supply was being made to a design provided by NASA and intended for ground testing. The completed supply, when combined with the simulation chamber, would comprise a complete thruster assessment facility.

The chamber with proper fixturing would be ideal for acceptance testing delivered ion engines.

UTIAS intended to measure the thrust vector of the ion thruster with cylindrical Langmuir probes, double ion concentration and beam profiles could also be mapped.

It is not recommended to perform life testing in Canada, however. This is because any procurement effort would be biased toward a design with a proven ground life test history.

H. Conclusion

Although ion propulsion does not have extensive flight test experience, it is felt that the technology is sufficiently advanced that it must be considered as an alternative to more conventional auxiliary chemical propulsion systems.

Perhaps one of the most significant bell-weathers of the fact that electric propulsion is technically ready is the fact that several cost-benefit-analysis have been performed in which various auxiliary propulsion systems have been compared. The results of these studies always favour ion propulsion on the basis that the useful (revenue producing) payload is the largest with typical ion thruster subsystems.

This conclusion was reached in one such study even though the projected cost of the electric propulsion system was four times the cost of a hydrazine system.

Thus arguments of cost and weight now would seem to favour electric propulsion. If extensive ground test experience on engineering model equipment is undertaken, a sufficiently reliable subsystem can be procured.

Appendix I

Typical Thruster Systems

Table III lists the thruster systems that may be expected to be available for auxiliary propulsion on spacecraft with projected launches before 1980.

A short status account is presented on the five ion propulsion systems. Any cost data must be considered as a very rough estimate at this time.

TABLE III

AVAILABLE THRUSTER SYSTEM

Developer/Manufacturer	Propellant	Description	Thrust mN	Thrust mlb	Isp sec	Grid Diameter cm	Thruster Power (Watts)	PPS Eff. %	Total Power	MASS Kg Less Propellant
LeRC/Hughes	Hg-B	SIT 8	4.45	1	2900	8	120	87	141	9.05
Giessen/MBB	Hg-RF	RIT-10	5	1.12	3000	10	145	72	200	8.5
RAE,Cullam/ Mullard, Marconi	Hg-B	T-4	7	1.57	2620	10	175	87	200	8.0 (Est.)
TRW/TRW	Colloid	ADP	4.4	1	1400	20 (square)	50.6	72	70	11.5
EOS/EOS	Cs-B	ATS-6	4.45	1	2600	8	123	85	150	12.33

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a. Structurally Integrated Thruster SIT 8

This thruster represents the latest version of the Kaufman thruster which was developed at the Lewis Research Center (LeRC). The technology has been transferred to Hughes Research Labs where a strong on-going program is maintained.

No projected flights are forseen. The design has been virtually flight ready for an experimental flight since it was selected for CTS in 1971. Four years of ground test data have since been accumulated. The ground life test data is this quite impressive (5 cm variant had run 9800 hrs.).

Gimbal vectoring has been built and tested, command, telemetry and power interfaces have been defined.

Costs can be expected to be $70 K per thruster. Power processors may run about $400 K per copy, but the costs are very sensitive to semi-conductor prices.

It is possible that the fabrication of the power processor could be done in Canada. HRL has suggested this approach before. A very strong support effort would exist within NASA for this thruster.

Thermal analysis has been performed and some thermal testing has been done.

This system has been used as the baseline subsystem for comparison.

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b. The Radio Frequency Ion Thruster : RIT 10

The RIT 10 thruster is an interesting variant of the Kaufman electron bombardment thruster. It has been developed in Western Germany at the University of Giessen. Ions are produced by means of a radio frequency discharge instead of a direct current arc. This process has the advantage of producing a greater proportion of singly to doubly ionized ions. In addition, the energy is fed into the discharge chamber in such a manner that the ion density is more uniform across the screen grids. Against these advantages are the unknown EMC problems and a poorly defined discharge chamber life-time.

The technology is being transfered to MBB and it is from this source that the system would be procured. Funding for the program is from the German space agency. This funding has doubled for 1975. At this time power processors are being developed and plans include vibration, EMC, and extensive life tests.

The Giessen group hopes to have a test flight on the European launcher in the 1979 time frame. For this reason, it is expected that the thermal design and an adequate vectoring system will be concluded this year. The technology should be ready and competitive with alternate suppliers.

Canadian content in the thrusters would consist of acceptance testing only.

Cost data was not available from the MBB representative contacted.

c. British Thruster T-4

This thruster is very similar in basic design to the SIT 8 type of Kaufman thruster. It has been under development at the Royal Aircraft Establishment RAE Farnbourough and at Culham Laboratories. The technology is being transferred to Marconi and Mullard in England.

It is expected that this thruster may be given an opportunity to "fly off" against the RIT on a European launcher qualification flight. These launches are expected in the 1979-80 time frame.

Life-test data is not as extensive as for the SIT 8, but on-going programs should rectify this.

Vectoring will be done by means of a gimbal system; this has not yet been designed.

The development of a breadboard power processor is under way and test data should be available within a year.

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d. Colloid Thruster

This thruster design is being built at TRW under Air Force funding. The design differs from the remainder of the ion thrusters in that the propellant is accelerated in the form of liquid droplets. The basic thruster design, including the power supply is simpler than an ion thruster. However, this simpler, lighter design must be traded off with a lower Isp, and thus a higher propellant mass for a given total thrust.

Gimballing will be effected by mechanical means.

It is possible that this system may be given a flight test as early as 1978. Considering the shorter development time that this thruster has had in comparison to the ion engine, it is surprising that the system will be ready for a flight at that time.

e. Cesium Bombardment Thruster ATS-6

This thruster is basically similar to the mercury bombardment type except that cesium is used for the propellant. Handling and control of this propellant is more difficult than with mercury. The thruster was developed at Electro-Optical Systems (EOS) initially under Air Force and then under NASA funding.

The system was flown on ATS-6 with a limited success. All the experimental objectives except restart and life-time were achieved. The life test for this thruster will be performed on the ground in parallel with a flight because of the conflicts in experimenter's time requirements. Space restart has not been possible with this design because of a propellant feed line problem. The design has been modified.

Vectoring is accomplished by means of a grid translation system and momentum dumping has been demonstrated.

The cost of the subsystem was about $2.7 M which included a lot of non-recurring enginering.

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Appendix II

Baseline Design

The SIT 8 thruster is used as an example for a baseline design.

Assumptions

Spacecraft mass	448 Kg (985 lb)
Thrust of ion engine	4.45 mN (1m lb)
Specific impulse	2900 sec.

Configuration Is as shown in the attached figure. The two power processors are completely crosstrapped. Thrusters are gimballed through ± 10^{0} in two orthogonal directions.

Mass Properties

			Kg		Kg
Thrusters	4	@	1.69	=	6.76
Gimbals	4	@	0.68	=	2.72
Power Processors	2	@	6.49	=	12.98
Tanks (8 yr size)	2	@	0.92	=	1.84
Switches	4	@	.8	=	3.20
Harnesses	2	@	.4	+	.80
					28.30
6 yrs propellant					8.78
					37.08
150 watts array allocation at 30 Kg/KW (if no battery allocation)					4.5
					41.58 Kg = (91.47 lb)

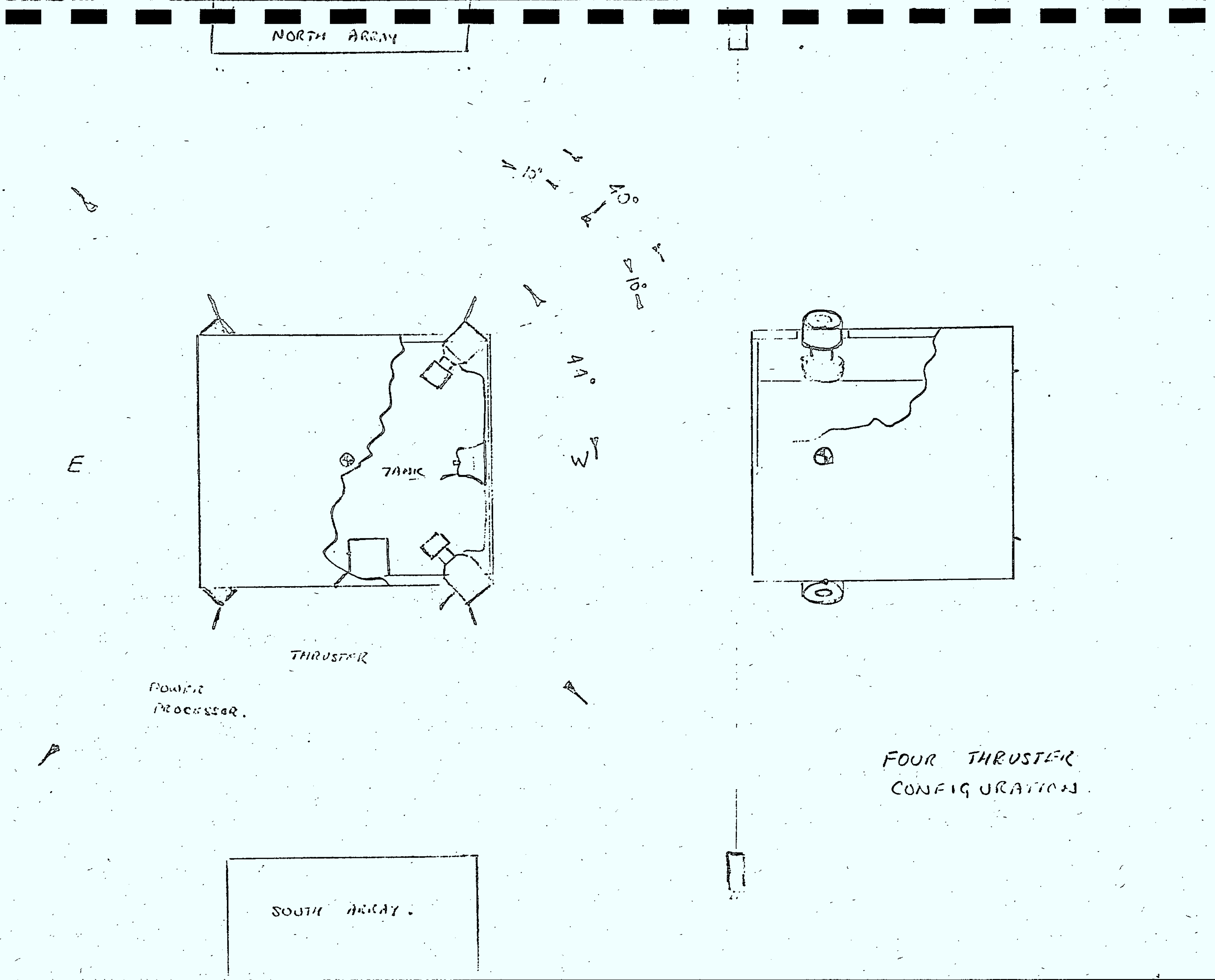
Operation

Thrusting time per orbit	-	6.09 hrs
Total thrusting time*	-	13,362 hrs
Total preheat time (6 yrs)	-	547 hrs
Propellant required (6 yrs)	-	8.78 Kg
Propellant required (8 yrs)	-	11.70 Kg

*Note that the firing time is divided approximately equally between a thruster on each of the east and west faces. As each unit is redundant, the average operating time required is 13,362/4 = 3340 hrs.

Power Requirement

150 watts (assuming vectoring gimbals are operated during thrust periods).



FOUR THRUSTER CONFIGURATION.

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ION PROPULSION FLIGHT EXPERIENCE LIFE TESTS AND RELIABILITY ESTIMATES

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ION PROPULSION FLIGHT EXPERIENCE LIFE TESTS AND RELIABILITY ESTIMATES

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Abstract

The application of low-thrust ion propulsion systems to space missions requires long duration (~5,000 to 20,000 hours) component operation. Thus, components must be developed with wear-out mean times to failure in excess of these required mission times. Furthermore, chance failures which occur during the useful life of components must be minimized. This paper makes an assessment of both early and wearout failure modes of ion propulsion systems by examining the results of existing developmental and long duration testing. Estimates of chance failure rates of system components are also presented along with design concepts which maximize total propulsion system reliability.

I. Introduction and Summary

Because of the long time operation (5,000 to 20,000 hours) required for most space applications, ion propulsion system reliability is a major concern. For this reason, the long duration and space testing of critical components, subsystems and systems associated with primary and auxiliary ion propulsion systems are of great importance. It should be recognized, however, that although a large number of long duration tests have been conducted to date on ion propulsion systems, subsystems, and components, these tests have been part of a technology development phase and are not to be considered statistical reliability testing of fully developed systems. Thus, even though space-qualified ion propulsion systems are becoming available, system reliability can only be inferred from the results of past and present developmental testing and estimated by analytical techniques.

In an effort to put ion propulsion system reliability estimates in perspective and to provide confidence in the ability of these systems to function for the long periods required, this paper attempts to:

1. Present the results of ion propulsion flight tests.

2. Compile the impressive list of long duration developmental and life testing of ion propulsion systems, subsystems, and components.

3. Predict from the available data the early and wearout failure modes of ion propulsion systems and eliminate them from further concern by establishing proper preflight test procedures and by specifying limited but adequate operating times.

4. Estimate, by analytical techniques, the chance (or random) failure rates of the major subsystems, namely, ion thruster, propellant reservoir, and power processor unit.

5. Present ion propulsion system design techniques which, through the use of redundancy, results in total system reliabilities acceptable to mission planners.

While many detailed reliability questions may be left without final and definitive answers, the information accumulated in this paper supports the contention that long life, reliable ion propulsion systems, based on present technology, can be made available for the numerous primary and auxiliary propulsion applications under consideration at the present time.

II. Flight Experience

Beginning in 1962 there have been eight space tests of ion engine systems culminating in the flight of the Space Electric Rocket Test (SERT) II launched on February 3, 1970 into a near polar orbit of 1000 km. A chronological listing of these ion engine space tests is shown in Table 1. As noted, the two types of thrusters tested to date are the cesium-contact and the mercury-bombardment ion engines. (The Table also indicates the planned test in the near future of a cesium-bombardment ion engine.)

Ion engine flight testing started with a series of three ballistic flights launched with the Scout rocket. These three flights took place on December 18, 1962, October 29, 1964, and December 21, 1964. The engines tested were Electro Optical System's 2 mlbf contact ion engines which were provided approximately 40 min of operation above 100 nm. The first failed through high voltage arcing because of venting of the battery compartment into the power conditioning unit. The second functioned according to plan as verified through electrical measurements. The third flight gave only 10 min of operation because of a launch failure, but all measurements indicated normal operation up to that time.

The SERT I was launched from Wallops Island on July 20, 1964 on a four-stage Scout rocket. It followed a ballistic trajectory for 47 min and had the distinction of being the first successful flight of an ion engine. The Hughes cesium-contact ion engine was unable to be operated because of a high-voltage short circuit. However, the NASA Lewis Research Center mercury electron bombardment thruster operated for 31 min at a thrust level of 4.5 mlbf providing data for ion beam neutralization, thrust level, radio communication interference, and differences in performance between ground and space testing. Thrust was measured by determining changes in spin rate. From these measurements, it was determined that complete beam neutralization was achieved in agreement with vacuum chamber tests. In addition, there was no interference with radio communication.

The next flight test was the SNAP-10A orbital flight test of a 2 mlbf contact ion engine system. Electrical incompatibility between the ion engine and spacecraft resulted in an interruption of telemetry during its initial operation and for this reason, it was subsequently not activated.

Table 1. Ion Propulsion Space Tests

Date	Flight
18 Dec 1962	Air Force ballistic flight to test an Electro-Optical Systems cesium-contact ion thruster system (No. 1)
20 Jul 1964	NASA SERT-I ballistic flight to test a Hughes Research Laboratories cesium-contact ion thruster system and NASA Lewis Research Center mercury electron bombardment thruster system
29 Oct 1964	Air Force ballistic flights to test an Electro-Optical Systems cesium-contact ion thruster system (No. 2)
21 Dec 1964	Air Force ballistic flights to test an Electro-Optical Systems cesium-contact ion thruster system (No. 3)
3 Apr 1965	Air Force SNAP-10A orbital flight with Electro-Optical Systems cesium-contact ion thruster as an auxiliary experiment
10 Aug 1968	NASA ATS-4 gravity gradient stabilized synchronous satellite containing two Electro-Optical Systems cesium-contact ion engines for stationkeeping.
12 Aug 1969	NASA ATS-5 gravity gradient stabilized synchronous satellite containing two Electro-Optical Systems cesium-contact ion engine for stationkeeping
3 Feb 1970	NASA SERT-II orbital flight to test two NASA Lewis Research Center 15 cm mercury electron bombardment ion engines during a six-month polar orbit
FUTURE Apr 1974	NASA ATS-F flight to erect 30 ft dish antenna and test UHF system. North-south stationkeeping will be provided by two Electro-Optical Systems cesium electron bombardment ion engines of 1 mlbf thrust

The two 5 to 20 μlbf cesium contact ion engines launched on ATS-4 in 1968 functioned normally in all respects for the two-month period prior to its decay from orbit. This engine is identical to the one used in ATS-5. The engines on ATS-5 were never turned on because of a malfunction of the nutation damper which caused the spacecraft to go into a flat spin.

The most recent space test was carried out with the launching of two 6.3 mlbf mercury electron bombardment ion engines on 3 February 1970 aboard SERT II. The ion engines were placed into a near circular, near polar orbit of 1000 km by a Thorad/Agena launch vehicle. The thrusters, along with related test equipment, were mounted on the spacecraft support unit which is in turn joined to the forward end of the Agena which serves as a spacecraft support platform. The Agena provided 1500 W of power from its solar arrays to power the thrusters and ancillary equipment along with the use of its horizon scanners for attitude determination. The primary purpose of SERT II was to demonstrate the long life space operation of either one of the two ion engines on board. In addition, data were obtained on the flight characteristics of the thrusters, contamination, and interference with radio communication. The first thruster operated continuously for 3781 hours before an electrical short caused an early shutdown with less than a month remaining to complete the goal of six month's operation. The second engine also failed because of an electrical short after 2011 hours.

Since cesium-contact thrusters are no longer under development in this country, the results of the SERT II are the most significant of all ion engine flight tests to date. It is appropriate, therefore, to emphasize the accomplishments of SERT II and to discuss further the failure modes of the thrusters as well as the subsequent development efforts which have eliminated these failure modes from present-day mercury bombardment ion engines.

First, SERT II provided long-term operational experience with solar-powered, space-qualified ion thrusters. This experience has established the viability of long duration operation of ion thrusters in space. Second, both thrusters, in general, performed the same in space as in ground tests. This correspondence between laboratory and space performance provides confidence in the specification of thruster operating parameters prior to launch for future missions.

The failure mode (as established in ground tests) of the SERT II thrusters was a high voltage short between the screen and accelerator electrodes. This short was caused by the erosion of the accelerator grid (in a localized area under the neutralizer) resulting in the bridging of the electrode gap by a piece of the grid webbing. This particular failure mode has since been eliminated by the repositioning of the neutralizer (e.g., as proved by subsequent long duration testing of various sized mercury bombardment thrusters). With the elimination of this failure mode, projected SERT II thruster lifetime is 15,000 to 20,000 hours.

A final significant result of the SERT II flight was the fact that during the past 2-1/2 years, all cathodes and vaporizers have remained functional and have been periodically ignited and tested, giving over 180 restarts of the cathode subassemblies.

III. Life Tests

During the past eight years, a substantial number of long duration tests (i. e., > 500 hours) have been carried out on both mercury- and cesium-bombardment thrusters. The majority of these tests have been carried out as part of the development phase of ion thrusters and, therefore, should not be considered in the realm of statistical testing of fully developed, quality controlled, flight qualified devices. However, these tests did serve to uncover wearout failure modes which through subsequent development were eliminated, to add to the confidence in those component designs which repeatedly survived, and, finally, to provide data on the fundamental and limiting failure modes of ion thrusters. Since in many cases the test periods approached the operating times required by typical mission applications, these latter data allow projections of ultimate thruster life to be made with reasonable accuracy.

During the course of the development of the 15 cm mercury bombardment thruster, which was ultimately to be flight tested on SERT II, numerous long duration tests of thrusters, subassemblies, and components were carried out. A comprehensive listing of these tests is given in Table 2. The testing associated with the two Experimental Thrusters was performed during the early development phases of the thruster. Thus, although the thruster bodies were operated for an accumulated time of 5000 hours and 5480 hours, respectively, the cathodes and grids were replaced with advanced designs when, and if, failures occurred. Based on the results of these tests, a flight thruster design was generated. Evaluation of this design was carried out primarily during the "hands off" testing of Prototype Thrusters P-10 and P-20. These two tests, both of which were completed without failures, lasted for 5412 hours and 6787 hours, respectively. (Actually, most of the thruster subassemblies including the neutralizer, grids and body were pretested giving them, in one case, an accumulated test time of approximately 8000 hours.)

Several observations, based on the SERT II thruster development and life testing, can be made concerning potential mercury bombardment ion thruster wearout phenomena. Initially, the most critical components in terms of inherent wearout mechanisms were expected to be the hollow cathodes and the screen and accelerator electrodes. The life tests of the SERT II thrusters confirmed these expectations. A list of the potential wearout failure mechanisms associated with these components is given in Table 3 along with an evaluation of their significance as basic life-limiting phenomena. As presented in Table 3, the life-limiting wearout mechanism of the SERT II thrusters was shown to be the erosion of the accelerator grid in a localized area beneath the neutralizer. (Actual failure caused by this erosion did not occur in ground tests because gravity prevented eroded accelerator grid webs from shorting out the electrode system.) Investigation of other potential wearout phenomena indicated mercury-bombardment thruster lifetimes of 15,000 to 20,000 hrs.

In addition to the 15-cm SERT II thruster development and life tests, a large number of test hours have been accumulated on mercury-bombardment ion engines, for example, the 5-cm, 15-cm, 20-cm, and 30-cm devices under development for auxiliary and primary propulsion applications. Tables 4 and 5 list the long duration testing performed on these thruster sizes and their critical components. Most notable (from a thruster life standpoint) of the tests listed in these two Tables are the 9715 hour test of a 5-cm thruster and the 1100 hour test of a 30-cm thruster. The significance of these two tests is that they established that the enhanced localized erosion of the accelerator grid beneath the neutralizer has been completely eliminated and that they confirmed the projected lifetime (based on wearout mechanisms) of 20,000 hours for mercury bombardment thrusters was attainable. This conclusion has been further supported, of course, by the numerous additional long duration tests of critical components and subassemblies such as mercury propellant reservoir, electrode systems, translating vector grids, cathode-isolator-vaporizer assemblies, neutralizers, and cathodes. In many cases, these tests have exceeded 10,000 hours, thereby, giving additional credibility to the estimates of ultimate wearout lifetimes of greater than 20,000 hours for mercury bombardment thrusters.

Two life tests of 30 cm thrusters are planned during the 1974-1975 time period. The first test is a long duration run of a development model thruster. The second (and most significant) is a 10,000 hour test of a fully developed Engineering Test Model (ETM). Successful completion of this latter test will establish the availability of long life, flight qualified mercury-bombardment thrusters.

One final test to be noted is the modularized thrust subsystem listed in Table 5, which employed 20-cm mercury-bombardment thrusters. This system consisted of a three thruster array (two operating and one standby), including array translator and thruster gimbals, two flight-type transistorized power processor units, and a liquid mercury reservoir. The system test also included simulation of spacecraft command and control system functions and of spacecraft dynamics with control loops closed to the thrust vector control system (i. e., translator and gimbals). The test was run for 1500 hours with two thrusters (two of which accumulated 1200 hours and one 600 hours), as well as the two PPU's and single reservoir operating simultaneously. During the 1500 hour period two failures occurred; one was the loss of a thruster cathode, the other was the loss of a discharge supply. The latter was a system (i. e., not a PPU) failure mode caused by a transient interaction between the computer and PPU and was subsequently remedied by a software modification.

A summary of the life test history of cesium bombardment thrusters and components is given in Table 6. As shown in the table the longest tests were conducted on three 12-cm thrusters and reservoir combinations. These tests, which

Table 2. SERT II 15-cm Mercury Bombardment Thruster and Component Life Tests (1967-1970)

Component	Hours	Reason for Termination	Comments
Experimental Thruster (No. 1)	5000	End of experimental thruster program	Test used same vaporizers, magnets, and thruster body
Neut. cathode heater	762[1]	Heater lead short at end terminal	End terminal design improved
Main cathode heater	2067[1]	Heater lead short at end terminal	End terminal design improved
Main cathode heater	763[1]	Corrosion at junction of Al_2O_3 and Ta tube	Added tungsten coating to Ta tube
Accelerator grid	3087	Replace grid surface to study "groove"	Basic accel grid life extrapolates to 20,000 hr
Neut. cathode heater	1821[1]	Heater lead short at end terminal	Still using old-design end terminal
Main cathode heater	1411[1]	End of program (no failures)	Used new tungsten coating and end terminal design
Neut. cathode heater	1623[1]	End of program (no failures)	Used new tungsten coating and end terminal design
Screen/accel grids	1800	End of program (no failures)	13 tests made on various shields to prevent "groove" wear
Experimental Thruster (No. 2)	5480	End of experimental thruster program	Test used same vaporizers, magnets, and thruster body
Neut. cathode tip	461[1]	Cathode orifice plugged	Reduced coupling voltage to lower tip erosion rate
Neut. cathode heater	899[1]	Heater lead short at terminal	End terminal design improved
Main cathode assembly	1514[1]	Cathode assembly "lost" during thruster inspection	Cathode functioning normally when lost
Accelerator grid	2104	Replace grid surface to study "groove"	Screen-accel always replaced as matched set
Main cathode heater	1451[1]	Corrosion at junction of Al_2O_3 and Ta tube	Added tungsten coating to Ta tube
Neut. cathode heater	3438[1]	Heater lead broke in handling during inspection	Added extra strap to heater lead for support
Screen/accel grids	3300	End of thruster program (no failures)	12 test segments or inspection points
Main cathode assembly	2000	End of thruster program (no failures)	Used new tungsten coating and heater improvements
Neut. cathode assembly	600	End of thruster program (no failures)	Used new tungsten coating and heater improvements
Prototype Thruster (P-3)	1000	End of 1000-hr demonstration test	All components functioning normally
Prototype Thruster (P-5)	208	Neutralizer vaporizer passing liquid Hg	Neutralizer lines froze due to LN_2 overnite exposure
Prototype Thruster (P-5)	3200	Increase in main discharge current	Lowered cathode heater current and discharge level for neutralizer test
Prototype Thruster (P-5)	1240	Replace main cathode with final flight design	Thermally redesigned main vaporizer and cathode insert (5736 hr on T/S body; 4729 hr on grids)
Prototype Thruster (P-10)	5412[2]	End of program (no failures)	T/S and PPU identical with flight systems
Neut. system	6331[2]	End of program (no failures)	919 hr prior testing on neutralizer system
Grids and body	7346[2]	End of program (no failures)	1534 hr prior testing on grids and thruster body
Prototype Thruster (P-20)	779[2]	Internal arc damage to PPU; no T/S failure	Prototype PPU used on test
Prototype Thruster (P-20)	6787[2]	Main propellant tank emptied	T/S and PPU identical with flight systems
Neut, grids, body	7994[2]	Main propellant tank emptied	1207 hr prior testing on T/S except main cathode
Various Prototype Thrusters (on prototype SERT II spacecraft)	2400 hr[3] in ~50 tests	One T/S failed due to facility failure	Cathode inserts chemically damaged by H_2O—entire cathode assembly replaced to restore normal performance
Neutralizer Cathode in Bell Jar	12,979[4]	Facility failure exposed hot cathode to air, heater failed on restart attempts	Projected lifetime >40,000 hr based on tip erosion rate

Table 3. Potential Wearout Mechanisms (Based on SERT II Thruster Testing)

Component/Wearout Mechanism	Comments	Projected Life
Hollow Cathode		
Tip wear	Changes orifice geometry	20,000 hrs.
Insert Material Depletion	Loss prevents restart	15,000 hrs.
Heater Corrosion	None seen	Indefinite
Electrode System		
Accelerator Grid Erosion		
1. Direct beam ions	None seen	Indefinite
2. Charge exchange ions	Uniform erosion	20,000 hrs.
3. Neutralizer ions	Localized erosion	2,000 hrs. *
Screen Grid Erosion from Discharge Chamber Ions	Minimal effects	50,000 hrs.
Metal Flakes from Discharge Chamber Shorting Grids	None seen	Indefinite
Grid Insulator Shorts	None seen	Indefinite

*Could occur anytime after ~2,000 hrs. when, and if, an eroded web becomes trapped between electrodes.

Table 4. 5 cm Mercury Bombardment Thruster and Component Life Tests (1972 - 1973)

Component	Hours -- Mode		Reasons for Termination	Comments
Thruster (SIT-5II S/N 101)	9715(5)	1 Shutdown	Elec. vector grid element severed	E. V. grid installed at 2027 hrs
Translating Vector Grid	5027(5,6)	Accumulated	Tests completed	2027 hrs in SIT-5 life test; 3000 hrs on LeRC thruster; Projected life >20,000 hr.
Electrostatic Vector Grid	7688(5)	Continuous	Vector grid element severed	Beam diversion by metal flakes
Propellant reservoir	13500(7)	Accumulated		Companion test for CIV in bell jar; Projected life - indefinite.
Thruster (SIT-5II S/N 204)	927(14)	Continuous	End of test	Vaporizer heater subsequently redesigned to operate at lower temperature
Main Cathode		594 cycles	Vaporizer heater failure	
Neutralizer		1009 cycles		
Cathode Isolator-Vaporizer Assemblies				
S/N 101	5400(7)	Continuous	Switch to 2nd generation isolator	1300 V isolation - no leakage current
S/N 109	13700(5)	1 Exposure Many restarts	Removed from SIT-5 life test	Durability and restart tests continuing in bell jar. Projected life >20,000 hrs.
S/N 113	1750(8)	190 Starts	Convert to impregnated insert	Cathode broke during disassembly
S/N 205	1664(9)	Accumulated	Set up for next test series	Incremental sputter erosion tests
S/N 207	8100(7)	Continuous		1600 V isolator test with SIT-5 reservoir
Neutralizer-Vaporizer Assembly				
S/N 111	9715(5)	1 Exposure Many restarts	Removed from SIT-5 life test	Tip disk eroded - sectioned for examination
Misc. Components				
First gen. E. V. Grid	1000(10)	Continuous	End of test - inspection	Problem areas delineated
1-axis E. V. Grid	1367(11)	Accumulated	End of test - inspection	Comparison of grid end designs
LeRC cathode	10000(12)	184 cycles		Bell jar test, impregnated cylindrical insert
LeRC cathode	2150(12)	943 cycles		Bell jar test, impregnated cylindrical insert
Pulse ignition cathode		10,000 cycles(13)	End of test - inspection	Bell jar test, 9.5 W tip heat, 11 mA Hg flow
LeRC cathode	2650(12)	59 cycles	More study required	Impregnated tip and insert
HRL cathode	1104(15)	Continuous	End of test	Life projection >20,000 hrs at operating temperature (850°C)

were conducted in a continuous mode, were completed with only one failure (a cathode heater open). Examination of these thrusters showed that, based on extrapolation of grid erosion due to charge exchange erosion, a thruster lifetime of 20,000 hours can be expected.

The last three tests listed in Table 6 were conducted on essentially the same design thrusters and were intended to demonstrate absence of any short-term problems rather than provide information on ultimate life-limiting phenomena. As in earlier tests, charge exchange erosion was observed, but at such a rate that extrapolated lifetime of 10,000 to 20,000 could be expected. No problems associated with cyclic operation were observed for up to the 90 cycles performed on these tests.

Although difficulties have been experienced with cathode heater cyclic lifetime (e.g., one design consistently failed after a few hundred cycles), a redesigned version has been operated for 2534 cycles.

More useful information will be gained during the next two years in the course of a two-year ground test of the ATS-F Ion Engine during which it is planned that approximately 1400 cyclic operations and 12,000 hours at full thrust will be accumulated.

IV. Subsystem Failure Rates (Reliabilities)

The discussion of ion propulsion system reliability will be based on classical concepts currently used in reliability engineering and will be limited to mercury-bombardment ion engine systems. For the sake of completeness, a brief summary of some of these basic ideas is presented.

Classical reliability theory divides the possible failures of a component into three modes, each corresponding to a specific cause of failure. The three types of failures are given as follows:

1. Early failures occur early in the life of a component and in most cases result from poor quality control in the manufacturing process. An early failure is eliminated by prerunning components for some time before they are actually used and keeping only those that survive.
2. Wearout failures are a result of the natural fatigue of a component toward the end of its life and are dealt with by life testing. That is, many components are run under simulated conditions until they fail (their time of failure is recorded). It will turn out that the failure times are normally distributed about some value called the mean time of failure. Failure due to

Table 5. 15 cm, 20 cm, and 30 cm Mercury Bombardment Thruster and Component Life Tests

Component	Hours - Mode	Reasons for Termination	Comments
15 cm Thruster, Reservoir, and PPU (1966)	500[16] Continuous	End of Planned Test	Flight-type transistorized power processor employed; oxide cathode degraded
15 cm Thruster (1967)	1000[17] Continuous	End of Planned Test	Test of oxide cathodes
20 cm Thruster (1964)	4000[18] Continuous	End of Planned Test	Liquid mercury cathode; no failures; projected life >10,000 hours
20 cm Thruster System (1971) (including 3 thrusters, 2 transistorized PPU's, 1 reservoir, translator, and gimbals)	1500[19] Continuous	End of Planned Test	2 thrusters operating simultaneously (2 thrusters - 1200 hrs, 1 thruster - 600 hrs); 1 cathode failure
30 cm Thruster (1969) Glass Grid	500[20] 3 Restarts	Glass Coated Grid failures	No cathode erosion
30 cm Thruster (1970) Dual Grid (with center support)	450[21] Continuous	End of Planned Test	Heater failure; no erosion under neutralizer; erosion at center support
30 cm Thruster (1973) a) Dual Grid (with insulated center support)	500[22] Continuous	End of Planned Test	Erosion at Center support
b) Dished Grids (no center support); sheathed Cathode heater	500[22] Continuous	End of Planned Test	Sheathed cathode heater reacted with braze
c) Dished Grids; plasma sprayed cathode heater	1100[22] Continuous	Vacuum Facility Failure	Stable operation with flight-type transistorized PPU; minimal electrode erosion; no failures; projected life >10,000 hrs.
30 cm Thruster (1973)	800 Continuous	End of Planned Test	Minimal Grid Erosion; Stable operation; no failures
30 cm Thruster Components			
Cathode (A)	1600[23] Continuous	Test Completed	No cathode insert
Cathode (B)	5 Restarts 1980[23] Continuous	Test Completed	No cathode insert
Cathode (C)	6 Restarts 3880[23] Continuous	Test Completed	No performance variation; no failure
Cathode (D)	19 Restarts 3950[23] Continuous	Continuing	No performance variation; no failure
Neutralizer	1400[23] Continuous	Test Completed	No performance variation; no failure
Isolator	17 Restarts 1800[24] Continuous	Continuing	No Leakage Current; no failure
Dished Grid (A)	1552[25] Accumulated	Sporadic tests	Minimal Erosion; no failure
Dished Grid (B)	1580[25] Accumulated	Sporadic tests	Minimal Erosion; no failure

wearout is made arbitrarily small by using components with mean times of failure far in excess of their required life.

3. Chance failures occur during the useful life of the component and are caused by random stresses.

A typical curve of failure rates versus operating time is shown in Fig. 1. Since it is unlikely that the amount of testing required to generate a failure rate curve (such as shown in Fig. 1) for ion propulsion systems will take place in the near future, a different approach must be taken to satisfy potential users that these systems will survive to perform their required functions. For example, if it is assumed that the early and wearout failures are eliminated and that the chance failure rate λ_c in the interval (T_E, T_W) is constant, the reliability of a device is given by the well known exponential law

$$r(t) = e^{-\lambda_c t} \qquad (1)$$

This exponential law is derived on the assumption that the failures are random (i.e., chance failures) and that their statistics are given by the Poisson process. The "elimination" of early and wearout failures can be accomplished by proper

Table 6. 8 cm and 12 cm Cesium Bombardment Thruster and Component Life Tests

Component	Hours - Mode	Reasons for Termination	Comments
12 cm Thruster (DF-1) (1965)	2610 - Continuous	Propellant tank emptied	One cathode heater failure; projected grid life (based on charge exchange erosion) >20,000 hrs
12 cm Thruster (DG-1) (1966)	3700 - Continuous	Propellant tank emptied	
12 cm Thruster (DG-2) (1966)	8200 - Continuous	Propellant tank emptied	
8 Cathode Heaters (1967)	<1000-7000 cycles each	Availability of Facility	Established cycling capability of Cathode heaters, one failure; heavy duty heater design
8 cm ATS-F Thruster (1970)	465 - Continuous 18 cycles	End of Planned test	No failures; extrapolated lifetimes 10,000 hrs to 20,000 hrs
8 cm ATS-F Thruster (1972)	750 - Continuous 90 cycles	End of Planned Test	
8 cm ATS-F Thruster (1973)	1100 - Continuous 90 cycles	End of Planned test	

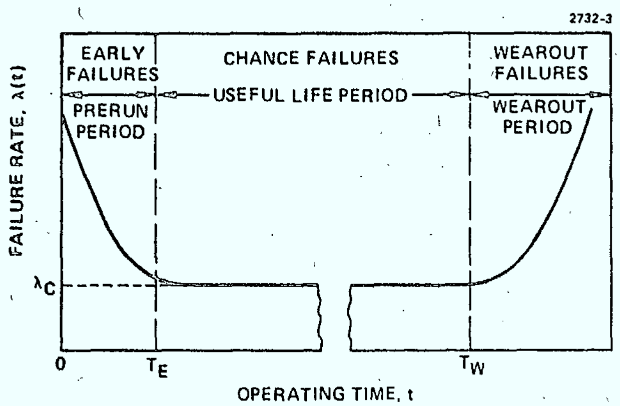


Fig. 1. Typical component failure rate versus operating time curve.

preflight checkout and by limiting required operating times to an interval less than the established wearout life of the major subsystems (i.e., by lifetest). The estimating of failure rate values can then be limited to random or chance failures that might occur during the useful life of the system. Thus, the approach to be taken here is the following:

1. Early Failures. Indicate potential early failure modes, establish proper checkout procedures, and specify required preflight test periods (i.e., determine time T_E).

2. Wearout Failures. Indicate life-limiting wearout failure modes and, based on existing life-test data, project minimum wearout life expectancy (i.e., determine time T_W).

3. Chance Failures. By analysis of the various component parts that make up an ion thruster, a propellant reservoir, and a power processing unit, estimate the chance failure rates of these three major ion propulsion subsystems (i.e., estimate subsystem failure rates λ_c).

Early Failures

Since the potential early failures associated with the power processing unit are typical of space-qualified electronic systems, only the thruster and reservoir subsystems will be analyzed here. First, any ion thruster (or reservoir) considered for a flight application will have been based on designs that have passed normal flight qualification tests. Furthermore, quality control procedures governing fabrication and assembly of thruster systems will have been established during development programs to minimize the probability of both early and random failure modes. However, since complete elimination of these failure modes would require an excessive expenditure of time and money on quality control testing for each unit produced, some preflight testing is required.

Most early failure modes have been experienced in developmental testing and a substantial background of data is available for detection of potential early failure by proper prelaunch testing of the thruster system or subsystems. Table 7 lists general categories of components for a mercury electron-bombardment ion thruster and propellant reservoir and the probable failures and symptoms that could be detected by appropriate diagnostics in prelaunch testing. The details of this testing procedure should be tailored to fit the requirements of the mission application. However, a format similar to that used in the SERT II program is recommended. The schedule of prelaunch testing under that program proceeded as follows:

1. Preliminary operation of assembled thruster system for calibration of operating parameters over full operating range with laboratory power processor.

2. Preliminary operation of assembled thruster with flight power processor to calibrate thruster and telemetry system.

3. Vibrational and thermal-vacuum test of thruster system mounted on spacecraft.

4. Operation of all thruster and system components while installed on spacecraft to accumulate about 50 hours operating on each subsystem.

The amount of operating time logged on each thruster or thruster system under the preliminary testing phase will depend on the test facility available. Operation should be kept to a minimum if testing must be done in a small vacuum facility where the ion beam impacts and sputters a metallic collector during test. It may not always be practical to install the assembled spacecraft in a vacuum facility for operation of all thruster systems in situ following the vibrational and thermal-vacuum tests. If the final test with the entire thruster system in situ is not possible, then in order to minimize early failures, the thruster system must be demountable in such a manner that minimal disconnection of wiring and propellant lines is necessary. Using these guidelines and a catalog of "normal" thruster parameters, it should be possible to virtually eliminate early failures.

Wearout Failures

Based on the thousands of hours of accumulated developmental and life testing, it has been concluded that at this time the fundamental or limiting wearout mechanisms of an ion propulsion system (i. e., thruster, reservoir, and power processor unit) are associated with three critical components of the thrust device. These components and their wearout failure modes are listed in Table 8. Estimates of the wearout life (i. e., the value of T_W) of present-day thrusters range from 15,000 to 20,000 hours. These estimates are based on straightforward physical extrapolation of the erosion and depletion phenomena indicated in Table 8.

It should also be noted that research and development efforts devoted to increasing thruster life beyond 20,000 hours are being pursued vigorously. It is expected that by increasing propellant mass utilization efficiency, reducing cathode starting and operating temperatures, reducing discharge voltages, and using low sputtering yield materials, thruster life can be extended to values much greater than 20,000 hours in the near future. It is, of course, important that life tests of fully developed thrusters be conducted for periods of 20,000 hours or more so that thruster life can be established without reliance on extrapolation. Furthermore, because of the potential deleterious effects of life testing in ground-based test facilities (e. g., backsputtered vacuum tank material), laboratory tests should continue to be augmented by space testing.

Chance Failures

In order to obtain an estimate of the chance (or random) failure rates for ion thrusters, propellant reservoirs, and power processor units without the benefit of statistical testing, an analysis of each of these subsystems was made on a component level. In the case of the thrusters and reservoirs the resulting total failure rates were essentially the sum of the failure rates of the components and their assemblies. The values obtained in this manner for the thrusters should be considered a probable minimum since the thermal and electrical stresses associated with operation were not considered. The reservoir failures, however, can be considered representative. Furthermore, in the case of the power processor units, where individual component failure rates are directly available and where circuit designs including redundancy are known and amenable to reliability analysis, the failure rates (and reliabilities) should be reasonably accurate.

1. Thruster Failure Rates

Two ion engines, both of which have been developed to a flight qualification state, were analyzed. These are the 5 cm and 30 cm mercury-bombardment thrusters. The failure rates of these thrusters were calculated by determining the failure rates of their component parts and certain construction features (such as welds). The failure rates for component parts were obtained from data in the following references:

1. "A Study of Liquid Mercury Isolator Development," Final Report, Contract NAS 7-539, September 1967.

2. W. Yurkovisky, "Data Collection for Nonelectronic Reliability Handbook (NEDCO I & II)," Technical Report No. RADC-TR-68-114, Hughes Aircraft Company, June 1968.

3. MIL-HDBK-217A, "Reliability Stress and Failure Rate Data for Electronic Equipment," 1 December 1965.

4. "Reliability Data Book," Martin Company, June 1962, Baltimore, Maryland.

5. "Reliability Application and Analysis Guide," Martin Company, July 1961, Denver, Colorado.

These reports are extensive compilations of failure rates for various components in airborne, ship, and ground applications. They also contain a limited amount of space and missile application data.

Since reliability data for many of the components used in an ion engine do not exist, the component failure rates of similar equipment used in "airborne" applications were selected. Because components are subjected to far greater stress in an airborne environment than in the space environment (other than the relatively short boost phase), the failure rates were adjusted by a K factor of 1/200. This factor, which is suggested in Martin Company's "Reliability Data Handbook" to convert data from airborne to space environment, brings the NEDCO I & II failure rates into good agreement with the "generic" failure rate range of the Martin data. Furthermore, experience by the HAC Space and Communication Group has shown that observed failure rates are in good agreement with the Martin data, scattered randomly between the specified upper and lower limits. In final analysis, the NEDCO I & II data were used because they are more extensive than any other compilation of mechanical reliability data available at present.

The failure rates for the major subassemblies of the 5 cm and 30 cm thrusters are tabulated in Table 9. As indicated previously, the total failure rate of each subassembly was obtained by determining the failure rates of the parts (or of similar parts) which make up the subassembly and the length or number of TIG welds, spot welds, electron beam welds, and brazes required for assembly. The failure modes considered were:

1. Structural weld failure
2. Sealing weld failures

Table 7. Thruster and Reservoir Early Failure Modes

FAILURE MODE	SYMPTOMS	PROBABLE CAUSES	DIAGNOSTIC PROCEDURES
1. Propellant System — (Includes Propellant reservoir, Feedlines, Vaporizers, Isolators, Propellant Ducting, Discharge Chamber Seals)			
Propellant leak, liquid	Low propellant utilization Abnormal overcurrent frequency Liquid mercury near thruster	Cracked propellant line Propellant line fitting loose	Thruster operation/ inspection
	Vaporizer control characteristics abnormal Isolator shorted	Liquid Hg intrusion of vaporizer Vaporizer plug cracked/ wetted	Thruster operation Weigh isolator vaporizer subassembly Inspect for liquid Hg
Propellant leak, vapor	Vaporizer control characteristics abnormal. Abnormal overcurrent frequency. Low propellant utilization.	Cracked isolator (Cracked weld or braze in vapor ducting) (Poor mechanical seal between subassemblies)	Thruster operation/ calibration Leak test subassemblies
Propellant blockage	Abnormal vaporizer-discharge characteristics No propellant flow	Propellant line pinched Vaporizer contaminated and obstructed Mercury frozen in propellant lines	Thruster operation/ calibration
Loss of propellant reservoir pressure	Thruster inoperative Thruster operates only at low beam	Cracked weld in reservoir leaking seal	Pressure check before thruster operation
2. Cathodes — (Includes Thruster Cathode, Neutralizer Cathode and Associated Keeper Electrodes)			
Cannot ignite keeper discharge	Orifice closed Barium exhausted	Contaminated by exposure to atmosphere or dirt	Inspection Prevention care in handling
	Keeper spacing	Damage in handling	
Keeper discharge abnormal	Keeper alignment Barium depletion Orifice dimensions	Mechanical damage Improper operation Exposure to contaminants	Inspection Prevention care in handling
3. Heaters — (Includes Cathode Tip Heaters, Isolator Heaters, Vaporizer Heaters)			
Open circuit	No heater current	Broken wire Burned out heater	Electrical checkout
Short circuit	No heater voltage	Incorrect wiring Internal short in heater	Electrical checkout
Poor thermal contact	Abnormal voltage-current characteristics	Mechanical contact of heater and insulator destroyed	Thruster operation/ calibration
Partially shorted heater	Abnormal power/control characteristics	Insulator deterioration	Ohmmeter checkout Thruster operation/ calibration
4. Electrical Integrity — (Includes Wiring Harness, Ceramic Insulators, Connectors, Electrodes, etc.)			
Open circuit	Cannot draw current	Improper wiring Broken wire/connector	Ohmmeter check
Short Circuits - low voltage	Cannot apply voltage	Improper wiring Damaged insulation Grid spacing shorted by metallic flakes	Ohmmeter check
Short-circuits-high voltage	Application of normal voltage Results in breakdown/ overcurrents	Insulator surfaces contaminated Spacing tolerances between Electrically stressed components insufficient as a result of distortion or metallic flakes	High voltage "Megger" measurement

3. Insulator failures
4. Structural failures
5. Heater failures

2. Reservoir Failure Rates

The reservoir failure rate was calculated for a single spherical tank as a function of its capacity in pounds of mercury. The system includes expulsion systems, filling values, and welded joints and assumes a safe operating pressure of four atmospheres with a 43% safety factor. The failure rates for component parts were obtained from data in the sources listed in the previous section on thruster failure rates. Again the unit failure rates were adjusted by a K factor of 1/200 as recommended by the Martin Company's "Reliability Data Handbook" to convert data from airborne to space environment.

A listing of the components and construction features, along with their associated failure rates, is given in Table 10. Total estimated reservoir system failure rate versus mercury propellant capacity is shown in Fig. 2.

Table 8. Thruster Wearout Failure Modes

Component	Wearout Failure Mode
Discharge Chamber Cathode	1. Orifice Erosion
	2. Barium Depletion
Neutralizer Cathode	1. Orifice Erosion
	2. Barium Depletion
Accelerator Grid	1. Charge Exchange Erosion
Screen/Accelerator Grid System	1. Short due to Metal Flakes from Discharge Chamber Erosion

Table 9. Thruster Failure Rates

Subassembly	Unit Failure Rate, Failure/10^9 hour	
	8 cm Thruster	30 cm Thruster
Thruster Shell and Structure	2475	5988
Optics and Supports	2698	3300
Cathode-Isolator-Vaporizer	2995	3178
Neutralizer-Vaporizer	2408	2525
Main Isolator-Vaporizer	--	1356
TOTAL	10,576	16,347

3. Power Processor Unit Failure Rates

The power processor unit is the collection of power circuitry and control logic required to operate and control an ion thruster from a solar panel bus, for example. At the present time, power processor units for both the 8-cm and 30-cm mercury-bombardment thrusters are under development. Availability of flight versions of these units is scheduled for late 1974.

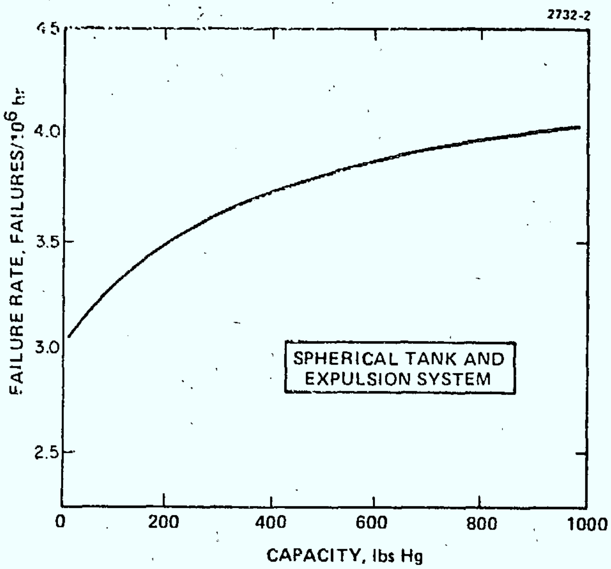


Fig. 2. Mercury propellant reservoir failure rates.

Functional block diagrams of the 8-cm and 30-cm ion thruster power processor units are shown in Figs. 3 and 4, respectively. From these diagrams, the various power supplies and controls required to run an ion thruster can be seen. As an example of the redundancy techniques employed in these PPU's, note the incorporation of a standby screen inverter in the power supply which provides dc power to the ion beam. Because of the modularization of the screen supply, partial redundancy can be used which results in minimal weight penalties. As indicated, standby redundancy is also proposed for the discharge supply. A reliability block diagram of the 30-cm ion thruster PPU is presented in Fig. 5, showing the results of a failure rate analysis of each of the blocks for 10,000 hours continuous operation.

The reliabilities of the 30-cm and 8-cm ion thruster PPU are given in Figs. 6 and 7. Figure 6 presents the 30-cm thruster PPU reliability with the screen and discharge inverter redundancy as indicated in Fig. 4. Figure 7 on the other hand, shows the reliability of the 8-cm thruster PPU with no redundancy (i.e., the reliability based simply on parts count). However, also shown is the impressively large increase in system reliability (from a circuit standpoint) by the simple addition of a low-weight redundant control module.

The results of the above analyses of ion thruster PPU units have shown that by the use of partial redundancy techniques relatively high subsystem reliability can be obtained with nominal weight penalties.

V. Effect of Reliability Considerations on Propulsion System Designs

As indicated above, the failure modes associated with early and wearout failures of the major subsystems of an ion propulsion system are at this time fairly well understood and thus predictable. This understanding allows for the design of burn-in testing procedures which can eliminate the concern of infant mortality of flight-qualified systems. In addition, by proper design of both mission and propulsion system, required operating times can always be made less than the wearout life limitations of any of the system components. Furthermore, the chance failure rates and, thus, reliabilities of the power processor subsystem (because it is generally made up of statistically tested electronic components and its circuitry is amenable to failure mode analysis) and the reservoir subsystem (because it is relatively simple and passive) are obtainable with adequate accuracy. The major remaining uncertainty in evaluating total ion propulsion system reliabilities is the exact value of the random or chance failure rates of ion thrusters. Since the establishment of thruster reliability at high confidence levels via testing would be both time consuming and costly (especially since 5,000 to 20,000 hours of operation is required for most space missions), other methods for increasing total propulsion system reliability such as redundancy must be considered. Redundancy of course, can lead to severe system mass penalties. Since increased mass is undesirable in any space system, it is important that these penalties be minimized.

Solar-Electric (Primary) Propulsion

An approach or methodology,[30] based on extensive analyses and system design considerations, has been developed for the purpose of designing prime solar electric propulsion (SEP) systems for space vehicles. It is based on the design concept of seeking a minimum mass system while maintaining the system reliability at or above a given level. This methodology has been used extensively in SEP spacecraft design studies. A brief description of this design approach and an example of its effect on system design and predicted reliability levels will be given below.

General Design Considerations

Where redundant components must be employed to increase system reliability, an effective means of reducing the concomitant system mass penalty is the use of a modularized system concept: the replacement of a single large component with a number of smaller subsystem components. In such a subsystem the incorporation of a redundant component for reliability purposes in general, will result in a relatively small mass penalty. For modularization to be considered, this reduction in mass penalty must compensate for the increase in initial system mass which normally results when a system is modularized.

A unique factor in the design of a solar-electric propulsion system is the nature of the output characteristics of the solar panel power source. First, the I-V characteristic and, thus, the maximum amount of power available from the panel, in general, is a function of time. Second, the maximum power available will be delivered only when the ion engine load is properly matched to the solar panel characteristic. Thus, it is apparent that the ion engine system load must be continually and properly programmed during the flight. (The switching of thrusters during a mission can have a major effect on system reliability, e.g., in an application in which power decreases operating thrusters become standby redundant.)

Reference 30 shows that the optimum procedure for programming an ion engine load is a combination of varying the ion beam current (throttling propellant flowrate) at constant beam voltage (constant specific impulse) and switching ion engine modules. In general, the throttling of ion thrusters can introduce a decrease in thruster efficiency. The degree to which a thruster must be throttled for a given application is a function of the number of thruster modules employed. Thus, the penalty associated with power matching requirements is also determined by the degree of thruster subsystem modularization.

The above discussion identifies two basic tradeoffs involved in the design of a solar-electric propulsion system: system reliability versus system mass, and power matching versus system performance. (The latter can be effectively related to system mass.) In each case, the number of modules employed in the system becomes the major design variable. Therefore, to minimize the system weight for a given application, the optimum number of modules must be determined. Since the total power level is specified by the flight dynamics analysis, definition of the number of modules also results in specification of optimum thruster size.

Since 1965 numerous interplanetary and geosynchronous application studies of primary solar-electric propulsion systems have been carried out and optimum (from a reliability-weight standpoint) SEP designs evolved. In most cases, a 30-cm mercury-bombardment ion thruster was found to be the optimum (or near optimum) module size. Thus, in the example discussed below this thruster size will be assumed.

In designing a modularized propulsion system, there are several ways in which the major subsystems (thrusters, power conditioners, and reservoirs) can be integrated. For example, each thruster could have its own reservoir and power conditioning and control system. In the other limit, by incorporating the proper cabling and switching matrix and manifolding and valving system, an individual thruster could be operated by any power conditioning panel and could be supplied with propellant by any reservoir. Between these two limiting designs several other possible configurations exist.

The general configuration shown in Fig. 8 has been shown to be optimum (i.e., minimizes system weight) for most solar-electric spacecraft studies to date.[31] In this case, there are a number of operating and standby redundant thruster modules and power processing units (PPU's). (The PPU's include the individual thruster control systems as well as the power circuitry.) The PPU's are electrically attached to the thrusters by means of electrical cabling through a switching matrix. Each "operating" thruster has its own PPU. Interswitching of operating thrusters and PPU's is not provided for. However, in the event of a thruster or PPU failure, any PPU can be switched to a standby thruster and a "standby" PPU can be switched to any thruster. Furthermore, a reservoir system is employed which feeds propellant to a common manifold through which propellant is distributed to the feed lives of all engines. Actual introduction and control of propellant into the thrust chamber of an operating engine is achieved by applying power to its associated vaporizer. Because of the extreme penalty for the additional propellant involved, redundancy in the reservoir system is undesirable (although multiple reservoirs might be employed because of capacity, structural, or packaging considerations). Thus, it is assumed that the reliability of the reservoir subsystem is increased internally without reservoir redundancy and does not become part of the tradeoff studies discussed above.

Propulsion System Designs and Reliabilities

Examples of the application of the above reliability considerations to a number of solar electric propulsion system designs are provided in Ref. 32. In this study eight potential SEP spacecraft missions were evaluated and optimum propulsion system designs for each mission were generated. The optimization procedure (as formulated in the EPSTOP computer program) used as inputs the low thrust trajectory data[33] and provided as outputs specification of the optimum numbers (from a reliability-weight-power matching standpoint) of initially operating and standby thruster and PPU modules, the manner in which these modules should be switched on and off, and the required degree of thruster throttling. Since it was recognized that the chance failure rate of the ion thruster is not established, the sensitivity of the system reliability to a wide range of assumed thruster

Table 10. Reservoir (Gas Pressurized, Bladder Type) Failure Rates

Component	Unit Failure Rate, $\lambda/10^6$ Hour	Number Used	Total Failure Rate, $\lambda/10^6$ Hour
1. Gasket	0.035	1	0.035
2. Fittings, Welded	0.005/in.	10 in.	0.050
3. Bladder	0.130	1	0.130
4. Flanges	0.205	2	0.410
5. Tubing, Metal	0.090	2	0.180
6. Heater and Control	0.220	1	0.220
7. Valve, Filler	0.500	1	0.500
8. Valve, Solenoid	0.890	1	0.890
9. Tank, High Press Gas	0.065	1	0.065
10. Ends, Hemispherical and Flange	0.050	2	0.100
11. Welded Joints	0.005/in.		
a. 10 lb Hg Reservoir		41	0.205
b. 100 lb Hg Reservoir		66	0.330
c. 250 lb Hg Reservoir		83	0.415
d. 500 lb Hg Reservoir		101	0.505
e. 750 lb Hg Reservoir		112	0.560
f. 1000 lb Hg Reservoir		121	0.605
12. Screws	0.025		
a. 10 lb Hg Reservoir		12	0.300
b. 100 lb Hg Reservoir		18	0.450
c. 250 lb Hg Reservoir		24	0.600
d. 500 lb Hg Reservoir		30	0.750
e. 750 lb Hg Reservoir		36	0.900
f. 1000 lb Hg Reservoir		36	0.900
13. Totals — Reservoir systems			
a. 10 lb Hg Reservoir			3.085
b. 100 lb Hg Reservoir			3.360
c. 250 lb Hg Reservoir			3.595
d. 500 lb Hg Reservoir			3.835
e. 750 lb Hg Reservoir			4.040
f. 1000 lb Hg Reservoir			4.085

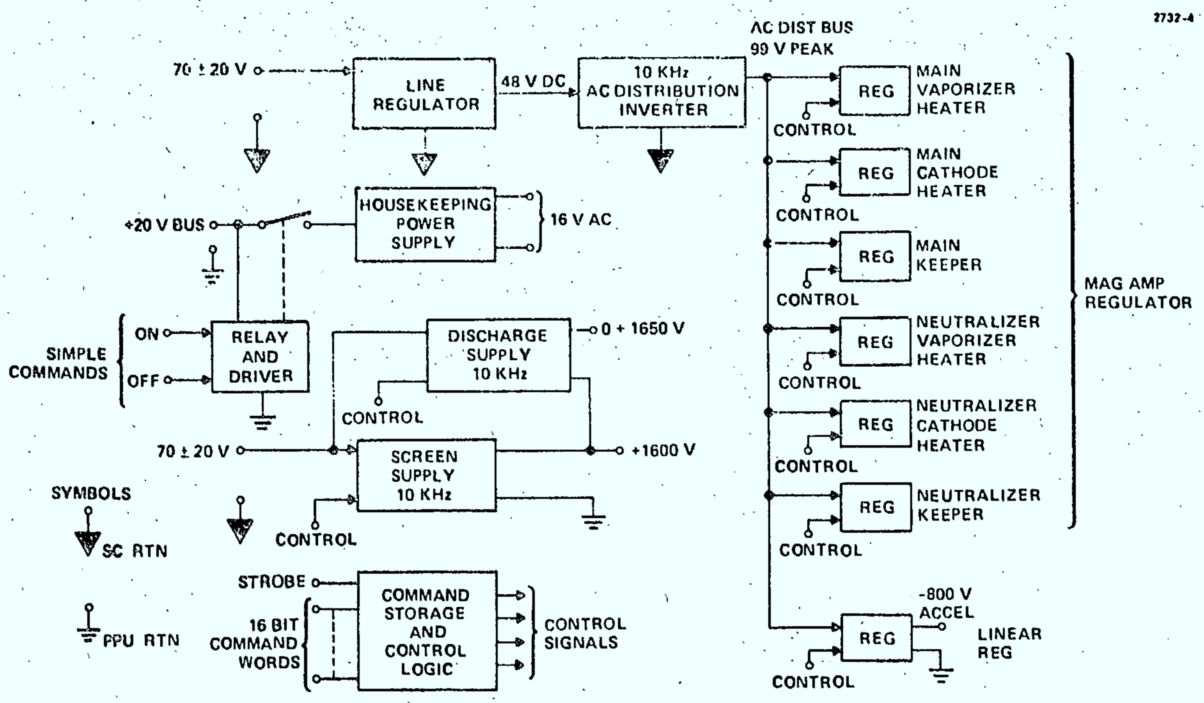


Fig. 3. 8 cm thruster power processor function block diagram.

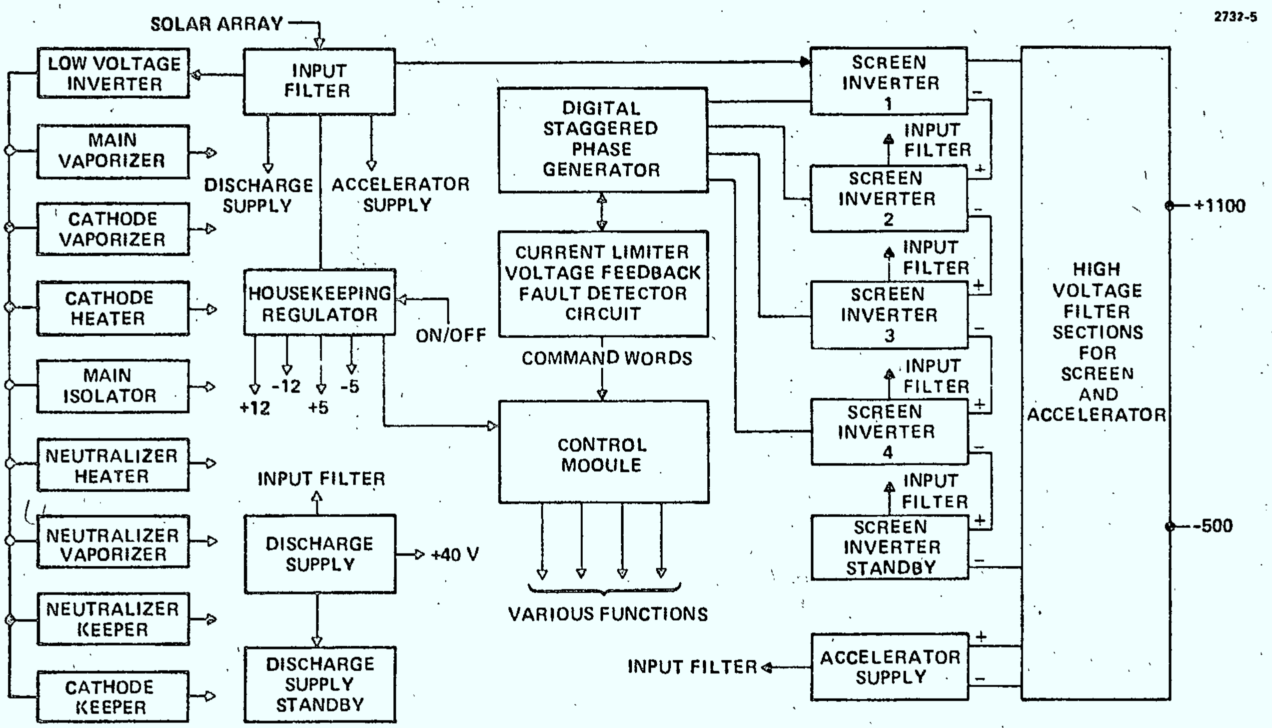
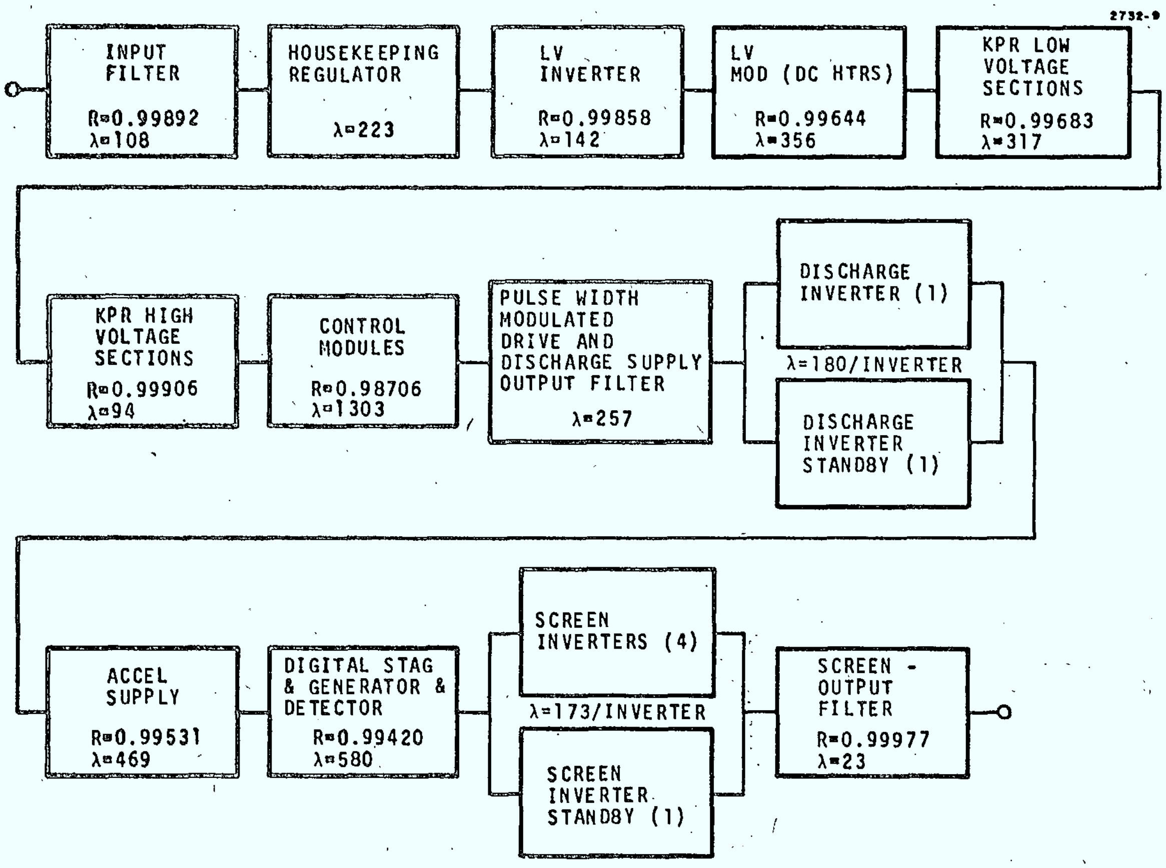


Fig. 4. 30 cm power processor functional block diagram.



*Numerical reliability values are based on 10,000 hrs continuous operation.

Fig. 5. 30 cm thruster power processor reliability block diagram.* (Failure rates, 10^{-9} F/hour).

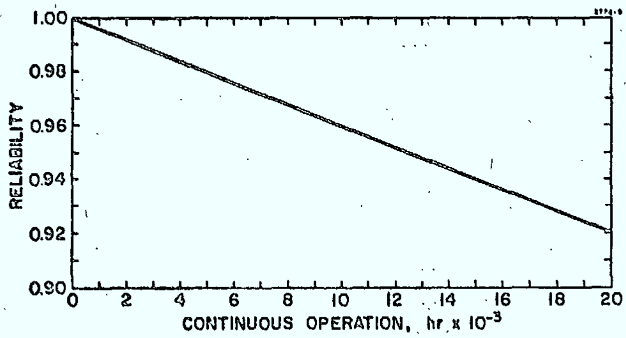


Fig. 6. 30 cm thruster power processor reliability.

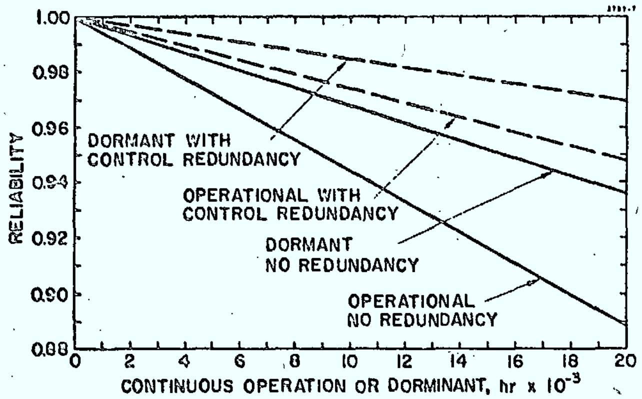


Fig. 7. 8 cm thruster power processor reliability (For operational and dormant conditions).

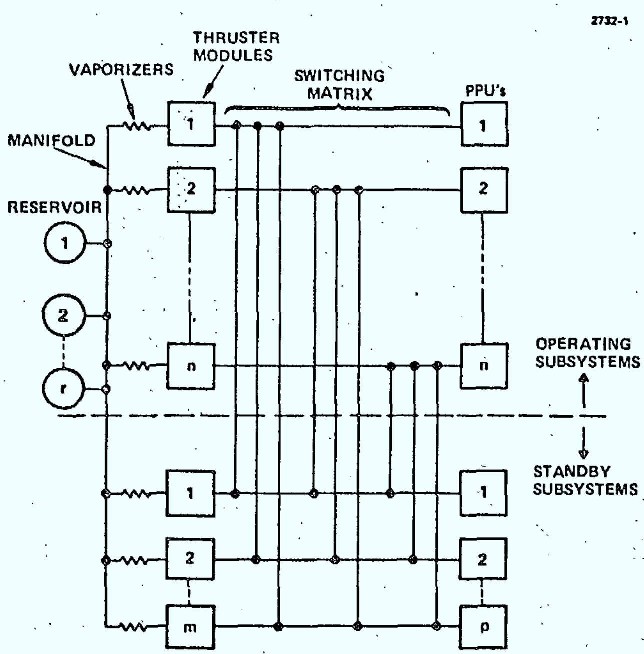


Fig. 8. General block diagram of electric thrust subsystem.

module failure was evaluated. The thruster module and PPU failure rates assumed for this study are given in Table 11. The thruster failure rates are presumed constant throughout their lifetime. Three different failure rates are indicated for the thrusters. The lowest failure rate represents the minimum which is expected based on the studies in Ref. 34. The largest failure rate is estimated to be an upper bound on the failure rate which might be experienced.

Table 11. Component and Subsystem Failure Rates (Failures in 10^9 Hours)

Thruster Failure Rate	30 cm Thruster Module
Minimum	6,400
Intermediate	25,600
Maximum	64,000
Power Processor Unit Failure Rate	$1{,}670 + 70 \left(\frac{T}{10{,}000}\right)$ where T = Thrust Time in Hours

As indicated, the overall failure rate of the power conditioning panels is lower than that of the thrusters because of internal subsystem redundancy. This same redundancy also leads to a power conditioning system failure rate which is not constant but increasing in time.

The recommended baseline modularized thruster and PPU system designs for the trajectory profiles given in Ref. 33 are shown in Table 12. It should be noted that these designs provide a resonably high probability of successful mission completion even if the maximum assumed thruster failure rate is experienced. In addition, these designs represent the minimum mass systems for the individual missions under consideration. Incorporation of additional thrusters or power conditioning panels for other reasons (e.g., commonality of designs between missions) would obviously be acceptable from a reliability standpoint.

The thruster switching times and thruster throttling range requirements are shown in Table 13. The degree of thruster and power conditioning panel redundancy associated with the recommended designs can be determined by comparing the number of thrusters and the number of power conditioning panels noted in Table 12 with the maximum number of operating thrusters (as well as the manner in which they are programmed) given in Table 13. Finally, it is important to note that although system operating times in excess of 1000 days are involved in two of the missions, no individual thruster is required to operate for more than 10,000 hours.

A major objective of the study cited above was to find a limited number of baseline system designs (i.e., thruster/PPU combinations) which could adequately perform all the missions under consideration. Inspection of the optimum designs (e.g., see Table 12) suggest the definition of the three baseline designs defined in Table 14.

Table 14 shows the reliabilities of the selected thruster array and PPU systems for each of the eight missions. Comparison of data in Tables 12 and 14, show that in some cases the baseline system has a higher or lower reliability than the corresponding "optimum." In these cases more or less redundancy (and, therefore, weight) is incorporated in an attempt to gain commonality.

To completely specify overall propulsion system reliability, consideration must now be given to the circuitry required to switch the various thrusters and PPU's and to the liquid mercury propellant storage system. The switching of thruster modules and power conditioning panels during a mission is performed by a logical controller and a power switching matrix. The general operation of these subsystems is as follows. At the beginning of a mission the programmer assigns a power conditioning panel to each operating thruster (i.e., closes the appropriate switches in the power switching matrix), and commands the power conditioning panels to turn on the thrusters. The start-up sequence is provided by the panels. Panels and thrusters are then directed to switch on and off, or to throttle thrust as required during the mission. If an operating thruster fails, the failure monitor sends an operation interrupt signal to the programmer which in turn shuts off the power conditioning panel for the failed thruster and opens the power switches. A standby panel and thruster are then activated or the panel previously in operation can be transferred to a standby thruster and brought on line. Power conditioning panel failures are also monitored and replacement panels, if available, are substituted for failed ones.

The weight and failure rate of the switching circuitry will be principally that of the power switching matrix. The logical controller will involve digital circuits that are complex but are inherently lightweight and can be made highly reliable through the use of redundancy.

The weight and reliability of the power switching matrix were estimated for all the missions in the three baseline designs. As a typical example of the switching requirements consider propulsion system baseline design No. 1. In accordance with the data in Table 14, the switching matrix couples eight thrusters with five power conditioning panels. Each of these missions begins with four operating thrusters and four standby thrusters. The four initially operating panels are connected to their respective operating thrusters, but also have the capability of operating any of the standby thrusters. One standby panel has the capability of operating any of the eight thrusters. These switching capabilities were assumed in the calculation of the thruster and power conditioning reliabilities. It should be noted that the switched off thrusters can be operated by any of the five panels but a switched off panel can only operate five of the eight thrusters. Therefore, the switched off panels are somewhat limited as standbys. This limitation is justified by the high panel reliabilities (they have internal redundancy) and by the increased switching complexity that would be required for its removal.

Table 12. Optimum Thruster and PPU System Designs

Missions	No. of Thrusters	No. of PC Panels	Thruster and PC Panel Mass ~ KG	System Reliability		
				Thruster Failure Rate		
				Min.	Intermediate	Max.
Eros Rendezvous	6	5	105	1.000	0.989	0.885
Encke Rendezvous	8	5	119	0.998	0.995	0.913
Mars Orbit and Return	12	5	147	0.994	0.991	0.845
Jupiter Flyby/ Saturn Probe	5	4	85	0.987	0.974	0.914
Saturn Orbiter	8	6	125	0.985	0.983	0.959
0.1 A.U. Solar Probe	13	9	204	0.996	0.993	0.857
Mercury Orbiter	15	8	206	0.991	0.990	0.896
Geosynchronous Orbit and Return	8	7	144	0.997	0.995	0.942

Table 13. Switching Times and Throttling Requirements for Optimum System Design

Mission	Switching Times		Maximum Throttling $P/P_{Nominal}$
	No. of Operating Thrusters	T-Days	
Eros Rendezvous	4→3 3→2	172 260	0.67
Encke Rendezvous	4→3 3→2 2→1 1→2 2→3	73 115 225 836 920	0.50
Mars Orbit and Return	4→3 3→0 0→3 3→4	130 350 470 912	0.69
Jupiter Flyby/Saturn Probe	4→3 3→2 2→1 1→0	74 112 155 550	0.21
Saturn Orbiter	6→5 5→4 4→3 3→2 2→1	66 89 116 165 250	0.26
0.1 A.U. Solar Probe	8→0 0→8 8→0 0→8 8→7 7→8	79 86 124 133 171 241	0.88
Mercury Orbiter	7	–	1.00
Geosynchronous Orbit and Return	7→6 6→5 5→4	2 14 93	0.80

Each of the switches actually represents a parallel combination of nine relays. It can be shown that there are 28 of these basic switching groups for each of the three missions in propulsion system baseline design No. 1. The number of groups in the various missions for baseline designs No. 2 and No. 3 have been calculated and can be shown to vary from 18 for the Saturn Orbiter to 54 for the Mercury Orbiter and Geosynchronous Orbiter. The resulting reliabilities (tabulated in Table 14) were calculated using a single operation reliability conservatively estimated at 0.999999, and an average total number of relay operations is twice the total number of relays.

As stated previously, a propellant reservoir weight-reliability optimization based on modularization is not as appropriate as it is for the thruster and power conditioning subsystems. The optimum reservoir system configuration (assuming no other constraints), therefore, appears to consist of a single tank with redundancy in its expulsion system. Furthermore, the number of valves should be kept to a minimum (possibly one) to carry out system operation (although redundant valves may be employed). Thus, in each of the baseline propulsion systems a single reservoir was employed and sized to the specific propellant requirement of the particular mission. The reliabilities of these various sized reservoirs were obtained from Fig. 2.

A summary of the weights and reliabilities of the ion propulsion systems for the eight baseline missions is given in Table 14. (Statements concerning items that have been included in these estimates should be carefully noted.) These results indicate that through redundancy techniques relatively high system reliabilities can be achieved for most SEP missions, even though long operating times are involved.

Typical of the implementation of a modularized primary propulsion system is the 21 kW thrust subsystem[35] shown in Fig. 9. This SEP thrust subsystem consists of eight 30-cm thruster modules and eight power processor units. A maximum of seven thrusters is required for the mission set for which it was designed providing a redundant thruster and PPU. Three mercury propellant reservoirs are chosen so that complete reservoir units could be removed, depending on specific mission requirements. The integration of this thrust subsystem into a SEP spacecraft[36] is depicted in Fig. 10.

Table 14. Summary of Baseline Propulsion Systems Characteristics

Prop. System Baseline Design	Mission	Total Power, kW	Prop. Weight(1) kg	Thruster Array		PPU System	Thruster Array and PPU System Reliability(2)	Reservoir		Switching Matrix	Total System(3) (includ. prop.)	
				Total No.	Max No. Oper.	No. of PPU Panels		No.	Reliability	Reliability	Weight, kg	Reliability
#1	Eros	12	506	8	4	5	0.999	1	0.983	0.99950	648	0.982
	Encke	12	502	8	4	5	0.995	1	0.956	0.99950	644	0.951
	Mars Orbit	12	706	8	4	5	0.962	1	0.954	0.99950	857	0.917
#2	Jupiter Flyby	18	443	8	4	6	0.984	1	0.974	0.99935	599	0.958
	Saturn Orbiter	18	576	8	6	6	0.983	1	0.968	0.99968	737	0.951
#3	.1 au Solar Probe	21	822	12	8	8	0.898	1	0.987	0.99927	1042	0.886
	Mercury Orbiter	21	1584	12	7	8	0.923	1	0.977	0.99913	1830	0.901
	Geosyn-chronous Orbit	18	1584	12	7	8	~1.0	1	0.985	0.99913	1830	0.984

(1) Propellant weights include 5% contingency

(2) Reliability based on intermediate thruster failure rate

(3) System weights and reliabilities do not include cabling or mechanisms.

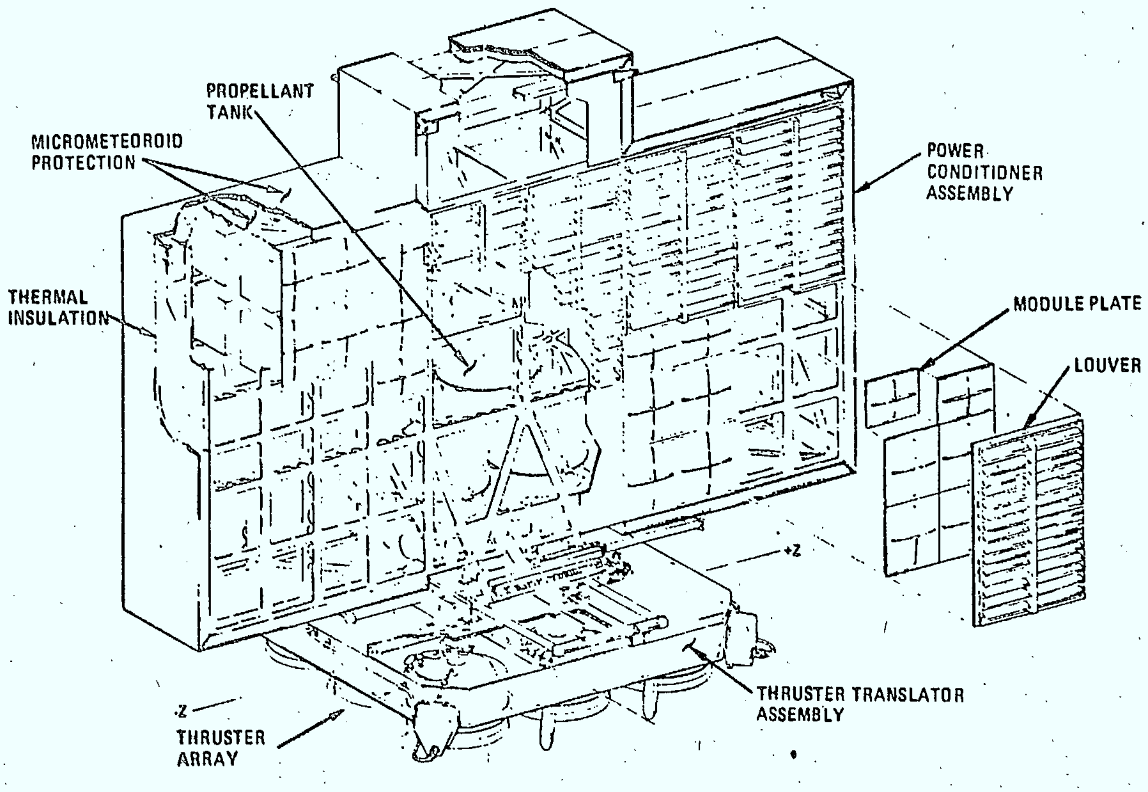


Fig. 9. Modularized thrust subsystem configuration concept.

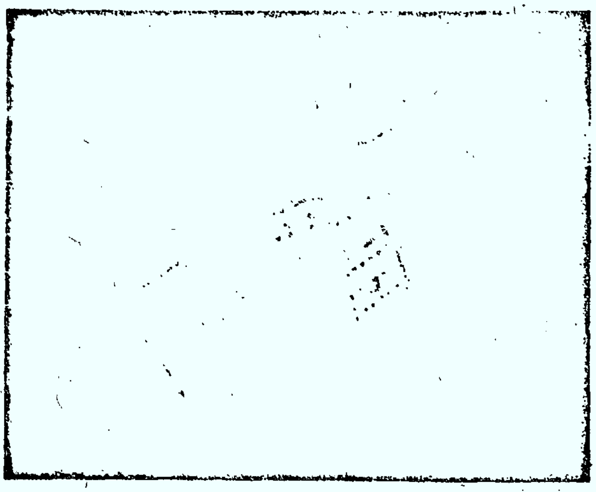


Fig. 10. Typical solar electric propulsion (SEP) spacecraft.

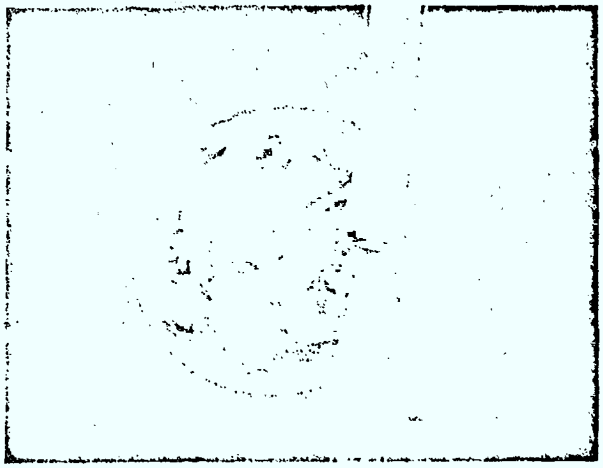


Fig. 11. Synchronous communication satellite (using ion propulsion).

Satellite Control (Auxiliary) Propulsion

As in the case of SEP, numerous design studies on ion engine satellite control systems have been carried out. In general, the satellite control applications require system operating times up to twice that of the SEP missions (e.g., as much as 20,000 hours). For this reason reliability considerations are especially critical in the design of these systems.

Typical of ion propulsion satellite control applications is the attitude control and stationkeeping of synchronous satellites. For example, consider the design of a synchronous satellite attitude control and stationkeeping (AC/SK) system based on deflectable beam ion thrusters in combination with reaction wheels.[37] One control system configuration option which would satisfy all of the requirements of a synchronous satellite AC/SK system consists of three thruster stations (e.g., see Fig. 11 for artists rendition of such a system on a high-power communication satellite). Two of these stations would be mounted at the base of the satellite with their nominal thrust vectors (parallel to the satellite longitudinal axis but offset from the center of mass) pointed north (or south) while the remaining station would be mounted on the body of the satellite so that it would nominally thrust through the center of mass in a west (or east) direction. North-south stationkeeping is unidirectional and performed during a 60° sector of the orbit at the time of maximum correction influence. This correction mode requires the two N-S thruster stations to operate in a continuous manner four hours per day. Thus, these thrusters must be turned on and off 365 times per year during which time they will accumulate 1460 hours of steady-state operation.

While providing the stationkeeping function, beam vectoring is used to produce three-axis control torques to allow reaction wheel energy to be reduced. Periodically during the orbit (but not during N-S stationkeeping), the E-W thruster is operated to compensate for solar pressure and earth triaxiality effects. This correction would occur five times per day for a total of three hours of operation.

Because of this cycling requirement, as well as the long duration of typical satellite control missions, reliability considerations have dictated that each thruster station be composed of three thrusters in a cluster. In this way, each control function has at least threefold thruster redundancy. The power conditioning unit associated with each thruster station is designed to operate a single thrust unit. In the event of a thruster failure, switching is provided to transfer the conditioned power to a standby. Power conditioning reliability is increased to the desired level by internal redundancy, rather than by separate standby systems.

A block diagram showing the attitude control and stationkeeping thruster subsystem and its points of electrical interface with other spacecraft subsystems is presented in Fig. 12.

The thruster power processing unit (PPU) consists of three integrated units which share common housekeeping bias supplies, sequential circuitry and drive electronics in order to reduce total power conditioning weight. Between the PPU and the three thruster stations is a switching matrix mechanized with latching relays. This allows rerouting of power to a redundant thruster within a given thruster station and provides a redundant station power conditioning capability (only a maximum of 2 of 3 power conditioners are required to operate at a given time) in the event of a partial failure in the PPU. To minimize thruster power cabling weight, the thruster select switching would be located near its associated thruster station and the station select switching would be located with the PPU.

Individual thruster on/off commands and switch-state controls are issued from the command distribution circuitry. Thruster beam vector control signals from the attitude control electronics provide two-axis deflection analog reference signals for the closed-loop electrostatic deflection power supplies in the PPU.

As indicated, a total of three thruster stations are provided for attitude control and stationkeeping. Each station (see Fig. 13) is equipped with three separate ion thruster units which share a common liquid mercury feed system. Only one thruster per station is needed to fulfill mission thrust requirements, while two redundant thrusters per station ensure high propulsion reliability with minimal mass penalty since the mercury reservoir constitutes the major mass contribution of the thruster system.

VI. Conclusion

Based on the successes of SERT II, both the flight qualification and the long duration operation in space of ion propulsion systems have been established. Based on the extensive, developmental, laboratory life testing of ion propulsion systems, subsystems, and components, early failure modes are well understood (and preflight checkout procedures defined) and wearout lifetime projections of 15,000 to 20,000 hours are justified. Thus, the major remaining question appears to be that of system reliability during the required operating life of an ion propulsion system (i. e., an assessment of chance failure modes).

Because of the lack of statistical testing of fully developed systems, other approaches to determining chance failure rates must be considered at this time. Reasonable estimates of the chance failure rates of the three major ion propulsion subsystems — thruster, reservoir, and power processor unit — can be obtained by an evaluation of component part failure rates and with the use of accepted analytical techniques. For example, the overall failure rate of the power processor subsystem can be readily obtained because these units are made up of well documented space qualified electronic components and reliability analysis of electronic circuitry is well established. Similarly, the propellant reservoir subsystem failure rate can be determined with good accuracy because these units are relatively simple in design and passive in operation. Whereas, ion thruster failure rates are more difficult to access, failure mode analyses, comparison with similar devices and components, and knowledge of construction features do lead to reasonable estimates.

Furthermore, using the chance failure rates obtained for the major ion propulsion subsystems, it has been shown that through the judicious use of redundancy, relatively high system reliabilities can be obtained for most primary and auxiliary ion propulsion applications without major weight penalties.

Thus, while statistical testing of fully developed, flight qualified, ion propulsion systems should be encouraged and while additional flight tests are desirable, relatively high system reliabilities can be predicted with confidence based on the use of good system design practices, available test data, and accepted analytical techniques.

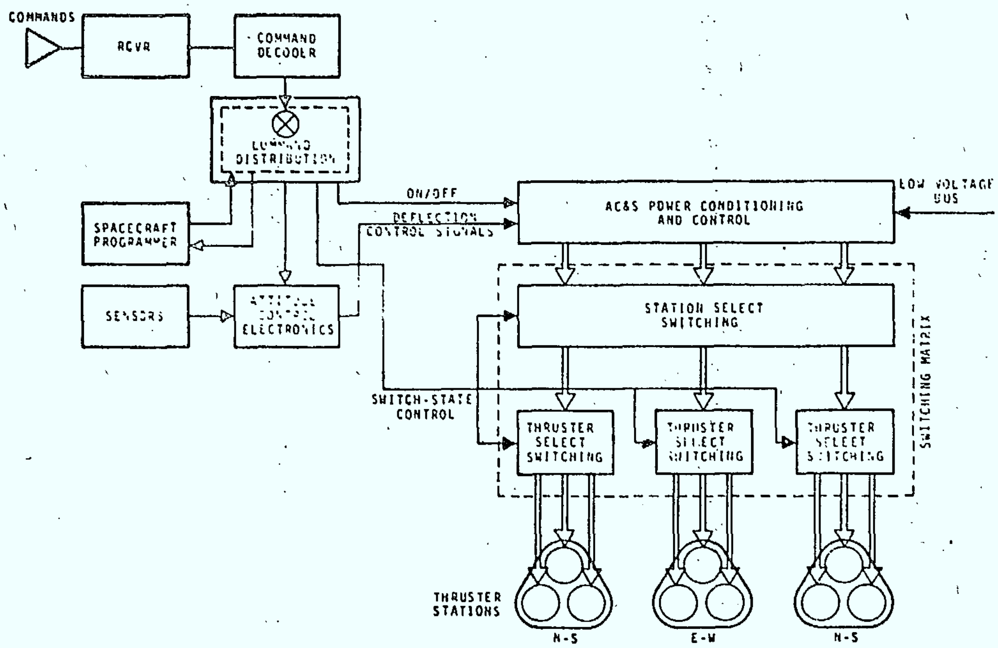


Fig. 12. Attitude control and stationkeeping thruster subsystem Interface with spacecraft.

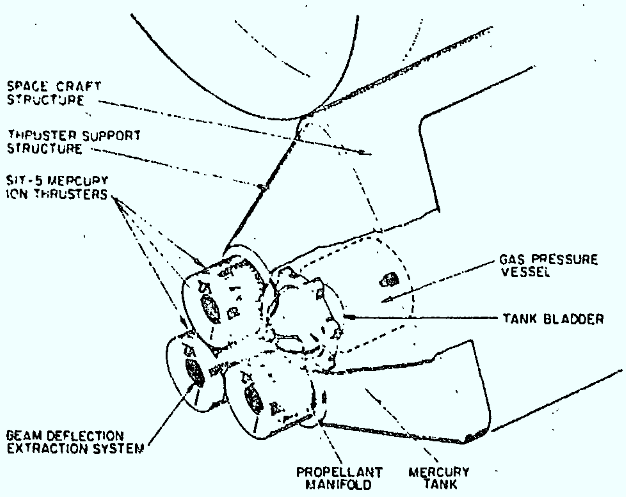


Fig. 13. AC/SK thruster station

Acknowledgment

Much of the information in this paper was supplied by members of the staff of a number of organizations including the NASA Lewis Research Center, Jet Propulsion Laboratory, Xerox Corporation, Lockheed Missiles and Space Company, and Hughes Aircraft Company. The help of these individuals is gratefully acknowledged and appreciated.

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AMERICAN INSTITUTE OF AERONAUTICS AND ASTRONAUTICS

MEMORANDUM

TO: Distribution

FROM: William Kerslake, Chairman, AIAA Electric Propulsion Technical Committee

SUBJECT: Compilation of Poll on Auxiliary Electric Propulsion Sent to Builders of Spacecraft

1. The attached poll was sent to the eleven following aerospace companies: Boeing Aerospace, Fairchild Space and Electronics, General Electric (Phila., Pa.), Grumman Aerospace, Hughes Aircraft, Lockheed Missiles & Space, Martin Marietta, MIT Lincoln Labs, Philco Ford, RCA Astro Electronics, and TRW Systems. The poll was directed to the man who advises or makes the decision to proceed with new spacecraft programs. In about half the cases the poll was routed downward in the organization for answering by a design group supervisor or senior staff engineer. The poll was not directed to electric propulsion technologists, but rather to designers, planners, or builders of spacecraft.

2. The poll was returned by 10 of the 11 companies polled. The company not returning the poll was Martin Marietta. They said that in the past they had conducted spacecraft design studies using primary electric propulsion, but that at present their company had no plans to build communication satellites.

3. A copy of a summary of the responses to the poll is also attached. Except for question 3, the number in front of a line indicates the number of times that a responder checked a reply. For question 3, there are 10 columns (one for each response) and the number in the column refers to the order of propulsion system preference. Question 3 shows that most (6) companies would design chemical auxiliary propulsion into their next satellites. Two responders, however, would use resistojets and one would use mercury bombardment thrusters. One responder indicated no preference of thruster type and stated that the choice would depend on reliability.

4. Looking at the response to questions 5, 6, and 9, the poll indicates that spacecraft designers would use auxiliary electric propulsion in the future if a system is proven reliable and if it is shown to be cost effective. Question 9 addressed reliability or user acceptance. Six responders (of 10 total) felt that a full mission cyclic system life test on the ground was necessary. Of those six only two would fly electric thrusters with no flight demonstration test. Eight of ten called for a successful flight test before using electric propulsion on one of their future satellites.

Bill Kerslake

William Kerslake
Chairman, AIAA Electric
Propulsion Technical Committee

AMERICAN INSTITUTE OF AERONAUTICS AND ASTRONAUTICS

OPINION POLL

On Auxiliary Space Electric Propulsion Use

1) How familiar are you with auxiliary electric propulsion? (Check one.)

- 0 Never heard of it (please pass the poll along to someone else in your organization).
- 0 Have heard of it, but don't know much about it.
- 9 Fairly familiar with it.
- 1 Very familiar with all aspects of it.

2) How often do you think auxiliary electric propulsion will actually be used for N-S station keeping of geosynchronous satellites launched within the next 20 years? (Check one.)

- 0 Never.
- 2 Only a few time as an experiment.
- 6 A moderate number of times.
- 2 Most of the time.

3) What variety of auxiliary propulsion do you personally advocate for North-South station keeping of a typical 7-year mission satellite launched in 1979? (Rank in order of preference with 1 denoting the most preferred.)

9	4	No					Cold gas
8	1	1	1	1	2	1	Chemical propulsion
5	2				1		Resistojet
3	3	3		4		2	Pulsed plasma
7	3	3					Colloid
4	3	2					Cesium contact
2	3	2		2	4		Cesium electron bombardment
1	3	2	2	3	3		Mercury electron bombardment
6	3						Mercury r-f discharge

4) How do you assess the comparison of risks to advantages of using electric propulsion? (Check one.)

- 4 Risks far outweigh the advantages.
- 2 Risks slightly outweigh the advantages.
- 0 Risks equal the advantages.
- 4 Advantages slightly outweigh the risks.
- 0 Advantages far outweigh the risks.

5) Do you plan to use auxiliary electric propulsion for North-South station keeping on

Your next satellite	0	Yes	10	No
Some future satellite	8	Yes	2	No

6) Under what circumstances would you perform North-South station keeping with electric propulsion? (Check as many as desired to indicate a positive reply.)

1 If used functionally (no backup chemical system).
6 If used as an experiment.
6 If Government pays for an experiment.
2 If Government pays for functional use (no backup chemical system).
5 If Government pays for functional use (no backup chemical system and Government will pay full anticipated losses in the event that electric propulsion system fails.
0 Under no circumstances.
4 Other (please describe) Responses were: would use if system received adequate development including successful flight demo.; if directed by a customer; if Goverment pays for demonstration flight.

7) How much effort should NASA devote to development of auxiliary electric propulsion? (Check one.)

0 None.
0 Much less than present.
1 Slightly less than present.
0 Same as present.
4 Slightly more than present.
5 Much more than at present.
0 No opinion.

8) How influential are you about determining the choice of auxiliary propulsion systems?

0 Not at all influential.
1 Slightly influential.
5 Moderately influential.
3 Very influential.
1 I make the final decision.

9) Under what circumstances for a future flight would you use auxiliary electric propulsion for North-South station keeping? (Check as many as apply.)

0 Under no circumstances.
6 If a full mission cyclic life test has been demonstrated on the ground using a flight-type thruster system.
6 If NAS performed a successful flight demonstration.
3 Other (describe) If weight and $'s of E.P. are less than chem. prop., all factors included; if wt. constraints require its use and if successful flight demo.; if cost effective and of proven reliability.
4 If a competitor successfully used it on a flight.

Signed (optional) ______
Title ______
Company ______
Address ______
Phone ______

APPENDIX B

RCS, HAMILTON STANDARD PROPOSAL

HSPC75R15

MULTIPURPOSE BUS SATELLITE

REACTION CONTROL SUBSYSTEM TECHNICAL PROGRAM PROPOSAL

APRIL 30,1975

Prepared for:

SPAR Aerospace Products Ltd.
Toronto,Ontario,Canada

Hamilton Standard
DIVISION OF UNITED AIRCRAFT CORP.

United Aircraft
OF CANADA LIMITED

CONTENTS

1.0 INTRODUCTION

United Aircraft of Canada, Ltd. and Hamilton Standard Division of United Aircraft Corporation are pleased to offer SPAR Aerospace Products Limited our full support of the Multipurpose Bus (MPB) program. Accordingly, we hereby indicate our availability and willingness to furnish the Reaction Control Subsystem and required ground support equipment.

This volume describes the technical and programmatic aspects of the proposed RCS program. Specific emphasis is placed on those features of the program which are critical to success, i.e., thorough technical understanding and detail program planning.

In the Technical Description section of this volume, the following topics are discussed:

- Technical Overview - An overview presentation of the basic requirements critical to mission success.
- RCS Subsystems - Description of the mechanical and electrical subsystems, packaging, and hardware weight and power.
- Mission Analysis - Detail analysis of mission requirements in terms of critical performance parameters and propellant weight.
- Components - Description of high and low thrust engines and other subsystem components.
- Reliability - Analysis of subsystem based on reliability considerations.
- Thermal Management - Description of thermal management aspects of subsystem design.

In the Program Plan section of this volume, the following topics are discussed:

- Management - Organization and key personnel who will implement the MPB RCS program.
- Schedule - Milestones for accomplishing the MPB RCS program in a cost effective manner.
- Deliverable Items - Description of the subsystems and ground support equipment to be furnished.
- Test Program - Definition of the tests to demonstrate the capability of the RCS to meet mission requirements.

1.0 continued

As can be concluded by the scope of this volume, particularly considering the relatively short time permitted for its preparation, United Aircraft of Canada, Ltd. and Hamilton Standard are deeply committed and ready to support the Multipurpose Bus (MPB) program. Our capability to provide needed mission analysis support, total propulsion subsystem design and manufacture (mechanical and electronic), leadership in the field of proven catalytic hydrazine rocket engine technology, demonstrated rocket engines in the thrust classes specifically required for MPB make United Aircraft of Canada, Ltd. and Hamilton Standard eminently qualified as the supplier of the Multipurpose Bus Reaction Control Subsystem.

Furthermore, we are prepared to commit all of the talents, facilities, and resources necessary for the development, fabrication, and support of the MPB program. Assurance that the necessary support and resources will be made available during the performance of the program is provided by the following commitments:

A) Assignment of the management/technical team that was instrumental to the success of the Communications Technology Satellite RCS program. Specifically, Mr. Harry Garfinkel as MPB RCS Program Manager and Mr. Vincent J. Sansevero as MPB RCS Engineering Manager.

B) Commitment of substantial precontract effort to support SPAR during the forthcoming spacecraft design definition.

C) Priority handling of MPB RCS program equipment throughout all sequences of manufacturing and assembly, thereby assuring capability to meet program schedule.

2.0 TECHNICAL DESCRIPTION

This section contains the technical description of the proposed monopropellant catalytic hydrazine reaction control subsystem (RCS) for the Multipurpose Satellite Bus. A technical overview is presented followed by a mechanical description of the subsystem including a layout of the proposed RCS and its weight breakdown.

Subsections are also presented which describe:

A) Electrical subsystem

B) Mission analyses

C) Selected components history and capability

D) Reliability assessment for eight year mission

E) Thermal management considerations

2.1 Technical Overview

The purpose of the RCS is to perform the following functions:

A) Precession and despin while the spacecraft is in the spinning mode.

B) Three-axis limit cycle, wheel spin-up (pitch axis) and on-board capture (offset operation), while the spacecraft is in the attitude acquisition mode.

C) North-south and east-west station keeping, pitch momentum dumping and whecon control (offset operation) while the spacecraft is in the three-axis stabilized mode.

The maneuver sequence for a six and an eight year mission along with propellant requirements is presented in Tables 2.4-I and 2.4-II respectively. The total propellant and pressurant requirements including all thrust inefficiencies, loading tolerances and expulsion efficiencies, and a 5% growth contingency are 194.6 pounds and 241.4 pounds for a six and an eight year mission, respectively.

The proposed RCS has full engine redundancy for all functions. A reliability assessment for the eight year mission presented in subsection 2.6 herein, yields a mission probability of success of .9545. This assessment includes the unreliability associated with the proposed qualified latching valve module which contains only six (6) latching valves. The arrangement of the six (6) latching valves

2.1 continued

gives full protection for any single engine failure and offers a weight saving of 2.29 pounds over the eight (8) latching valves suggested in the RCS specification. The maximum dry weight and wet weight for the proposed RCS compares favorably with the specification design goal requirements and can be seen below.

RCS WEIGHT SUMMARY

	Design Goal (pounds)	Proposed Maximum (pounds)
Dry Weight	70.0	64.07
Wet Weight (6 yrs.)	265.0	258.69
Wet Weight (8 yrs.)	310.0	305.44

All components proposed for the RCS have been fully developed and most have been qualified on similar applications. A delta qualification test is required for the propellant tank because of modifications to the mounting approach and to the low thrust engine because of increased pulsing and steady state life requirements as well as a modification of the capillary tube inside diameter on the offset engines to accommodate lower impulse bit requirements. The Electrical Control Unit has five (5) of the printed circuit boards which are identical to those qualified for the CTS program. Some modifications have been made to accommodate the MPB power conditioning and additional temperature sensor and heater group requirements. Most of the circuit designs are identical to those qualified on CTS.

In summary, the proposed RCS meets all MPB mission requirements for the six and eight year missions with a weight saving of 6.31 lbs and 4.56 lbs and utilizes proven and qualified components.

2.2 Mechanical Subsystem Description

2.2.1 Schematic - The Hydrazine Reaction Control System (RCS) is a mass expulsion hydrazine propulsion system which uses nitrogen pressurant operating in a blowdown mode to supply propellant to the monopropellant thrusters upon command. The RCS is schematically depicted in Figure 2.2-1. The propellant and pressurant is stored in four (4) identical spherical pressure vessels which utilize a passive expulsion device to supply propellant at the tank outlet without gas injestion. This device has been qualified for use on the SATCOM program which will be launched in December 1975. Through the use of the passive propellant management device rather than the normal elastomeric diaphragm, a 14 lb_m system weight saving is realized. A further benefit in terms of reliability and life potential is achieved since the materials of construction are fully compatible metallics rather than life limited elastomers. Each of the individual tanks has its own fill and vent valve to enhance balancing in both the spinning and 3-axis control mode. Each pair of tanks has a propellant fill and drain valve for propellant loading and off-loading. The propellant flowing from each pair of tanks is filtered by a large capacity, low micron rating, etched disc filter. Immediately downstream of the filters, pressure transducers are incorporated to monitor system status.

After the propellant has been filtered, it then passes through a latching valve which functions as an isolation valve. This valve can isolate a pair of propellant tanks from the distribution system either to prevent a tank failure from incapacitating the entire RCS or to isolate the backup tank pair from the system as necessary.

The propellant then flows into four (4) latching valves which function as isolation valves for the various groups of engines. The number and function of the engines in each group has been calculated to achieve the required mission reliability of at least 0.95.

The system includes two (2) 5 lb_f high thrust engines (HTEs) for vehicle precession control during spin mode and 16 low thrust engines (LTEs); i.e., twelve 0.2 lb_f LTEs for vehicle despin and attitude acquisition, station acquisition, momentum dump and station keeping during the 3-axis stabilized mode and four (4) 0.05 lb_f LTEs for capture and roll-yaw control during the 3-axis mode.

In each engine the propellant is catalytically decomposed, upon command, to provide the impulse required for the various spacecraft maneuvers. Each engine includes a catalyst bed heater located on the thrust chamber wall and a platinum resistance chamber temperature sensor. Surrounding the thrusters is a thermal shield to minimize heat loss and improve performance.

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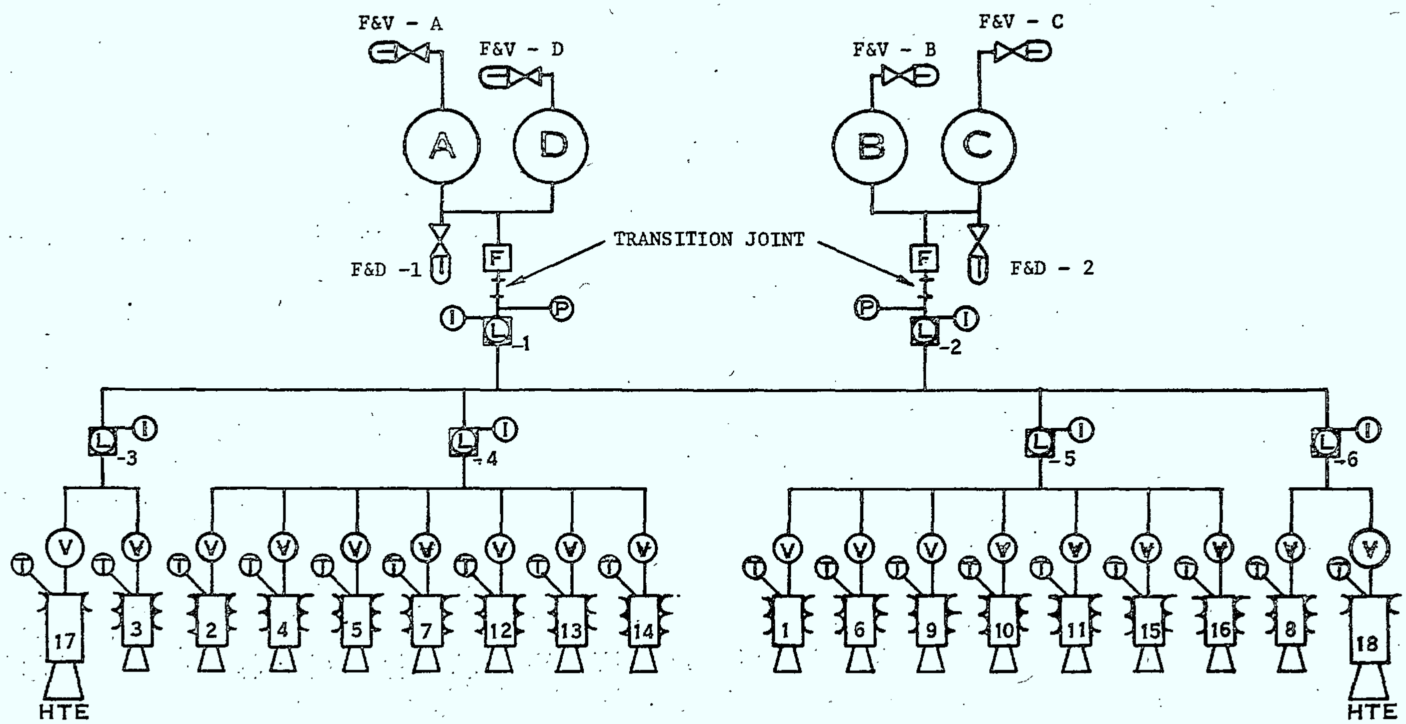

KEY
T - TEMPERATURE INDICATOR
— FILL AND DRAIN VALVE-CAPPED
V — THRUSTER SOLENOID VALVE
— ELECTRIC COMPONENT HEATER-REDUNDANT
- ELECTRIC COMPONENT HEATER-SINGLE
P — PRESSURE INDICATOR
L LATCHING VALVE WITH POSITION INDICATOR
— TRANSITION JOINT

FIGURE 2.2-1. MPB RCS SCHEMATIC

2.2.1 continued

The proposed schematic differs from the schematic suggested by SPAR in the area of the thrust chamber group isolation valves. The suggested schematic uses six (6) thruster isolation valves while the HS proposed schematic utilizes only four (4) thruster isolation valves. A study of the two arrangements shows that the HS approach meets the RCS subsystem reliability requirement while providing 2.29 lb_m weight savings. This weight savings coupled with an acceptable reliability assessment and a lower cost design, dictated the selection of the four (4) thruster isolation valve arrangement.

2.2.2 Design - The RCS has been arranged to fit within the specified envelope, perform the specified functions, and be easily maintained and serviced. The experience gained in the design and manufacture of hydrazine propulsion systems including RAE-B, CTS, BSE, and IUE, has been utilized to the fullest in the proposed offering. The various components have been grouped into functional modules to minimize structural interface requirements and reduce overall assembly complexity. The modular arrangement has been successfully used on the CTS reaction control system and is shown in Figure 2.2-2 -- note the neatness and accessibility of the various modules. The modules in the MPB RCS include six (6) rocket engine modules and a latching valve module. In addition, the RCS includes the four (4) propellant tanks, an Electrical Control Unit, individual rocket engine mounts, fluid lines, bracketry, and tank support structure. The preliminary design layout is shown in Figure 2.2-3.

The four (4) propellant tanks are located in accordance with the SPAR configuration drawing. The support structure for the tanks is configured as suggested by SPAR. A preliminary load/stress analysis was performed on the structure which indicates that the arrangement is feasible with reasonable size and weight struts to withstand the vibration loads with fully loaded (8 year mission propellant load) tanks. This type of structure has been used on the Hamilton Standard IUE propulsion system.

The propellant tank propellant outlet port is located approximately one (1) inch radially outboard of the tank polar mount center line. This provides a minimum of 40 pounds of fuel for the spinning portion of the six year mission without uncovering the outlet port. Even if the 5% contingency fuel is off-loaded prior to launch, there will be a minimum of 24.4 pounds of propellant available in the spin mode prior to the port uncovering and after the completion of all precession maneuvers and despin. The location of the propellant outlet port at this point allows essentially complete off-loading of propellant when the spacecraft is in the normal launch orientation. It should also be noted that the propellant management device may provide sufficient expulsion force to partially overcome centrifugal

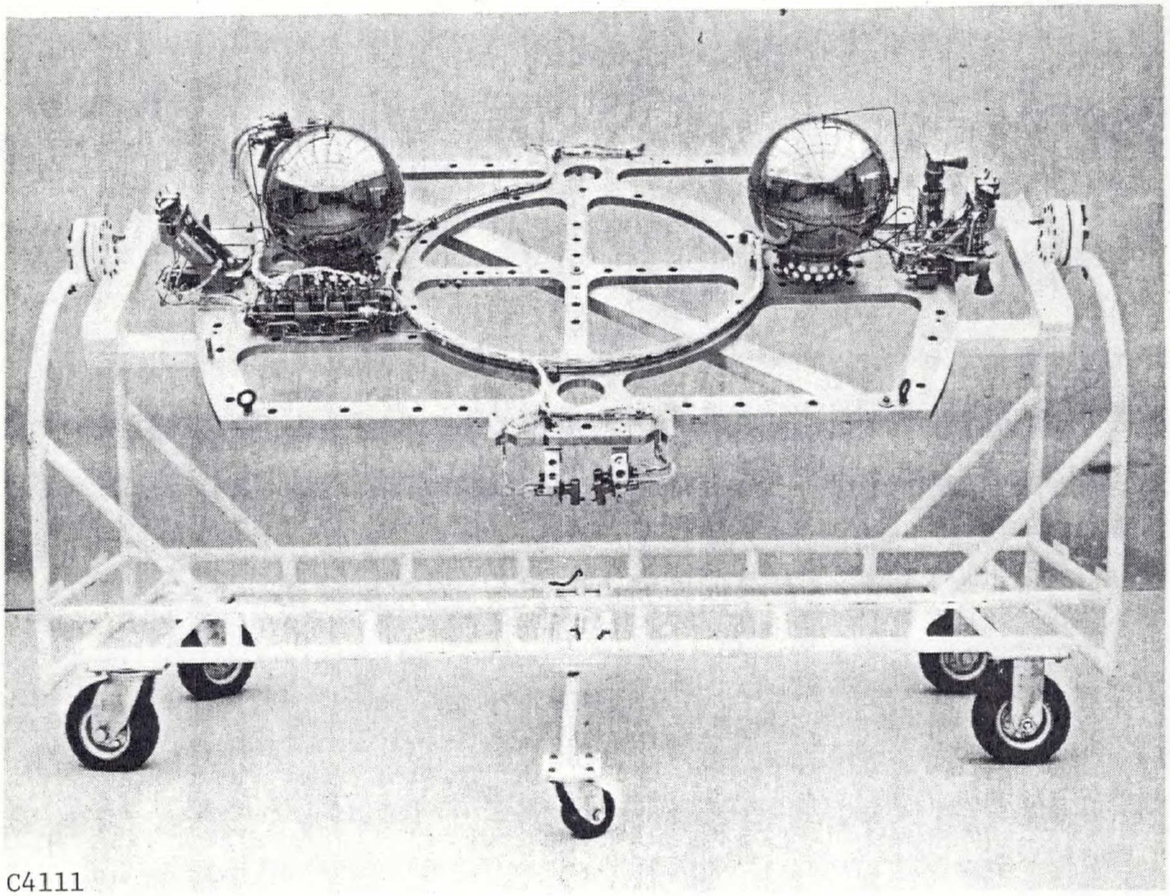

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FIGURE 2.2-2. CTS REACTION CONTROL SUBSYSTEM

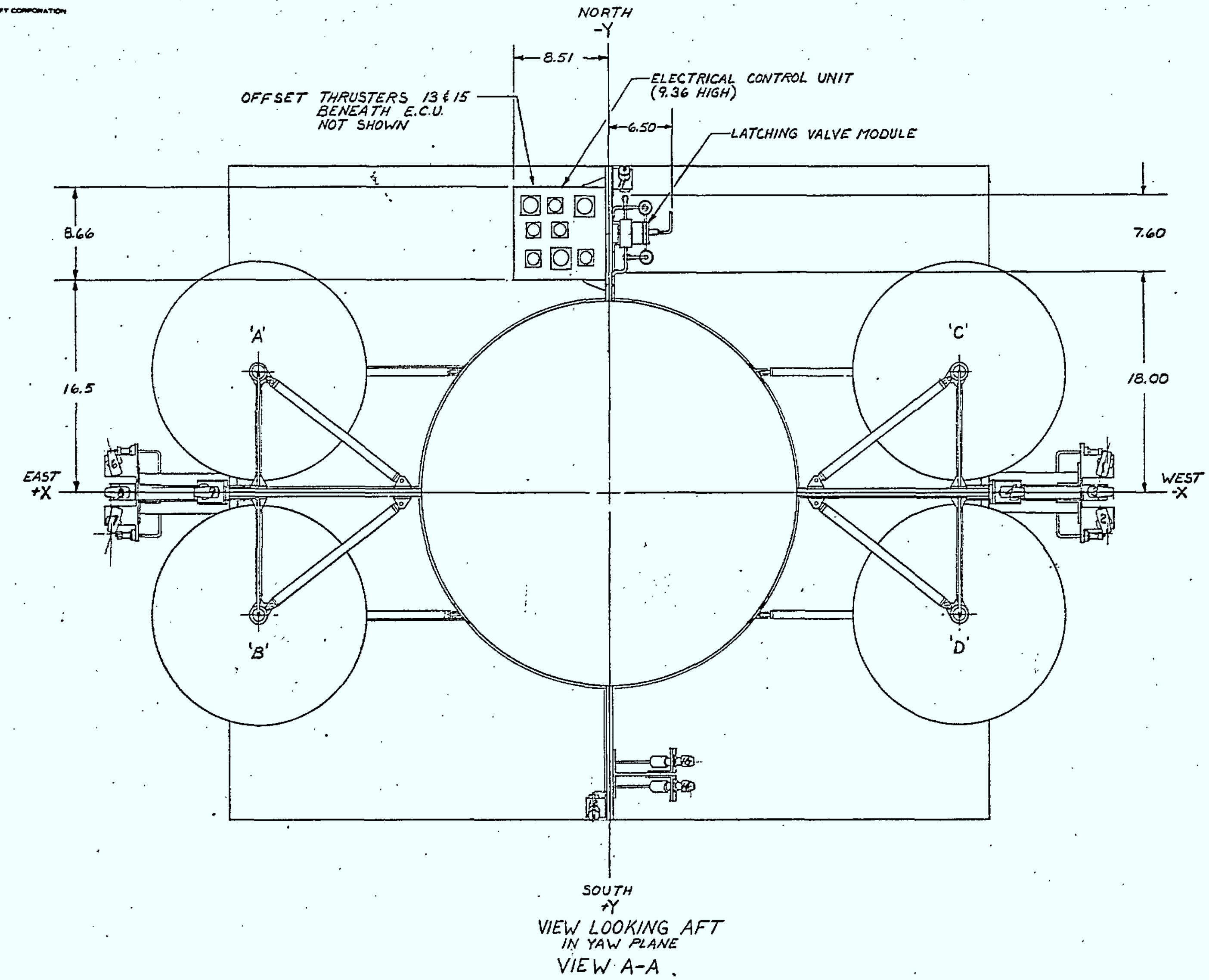

FIGURE 2.2-3. REACTION CONTROL SUBSYSTEM LAYOUT

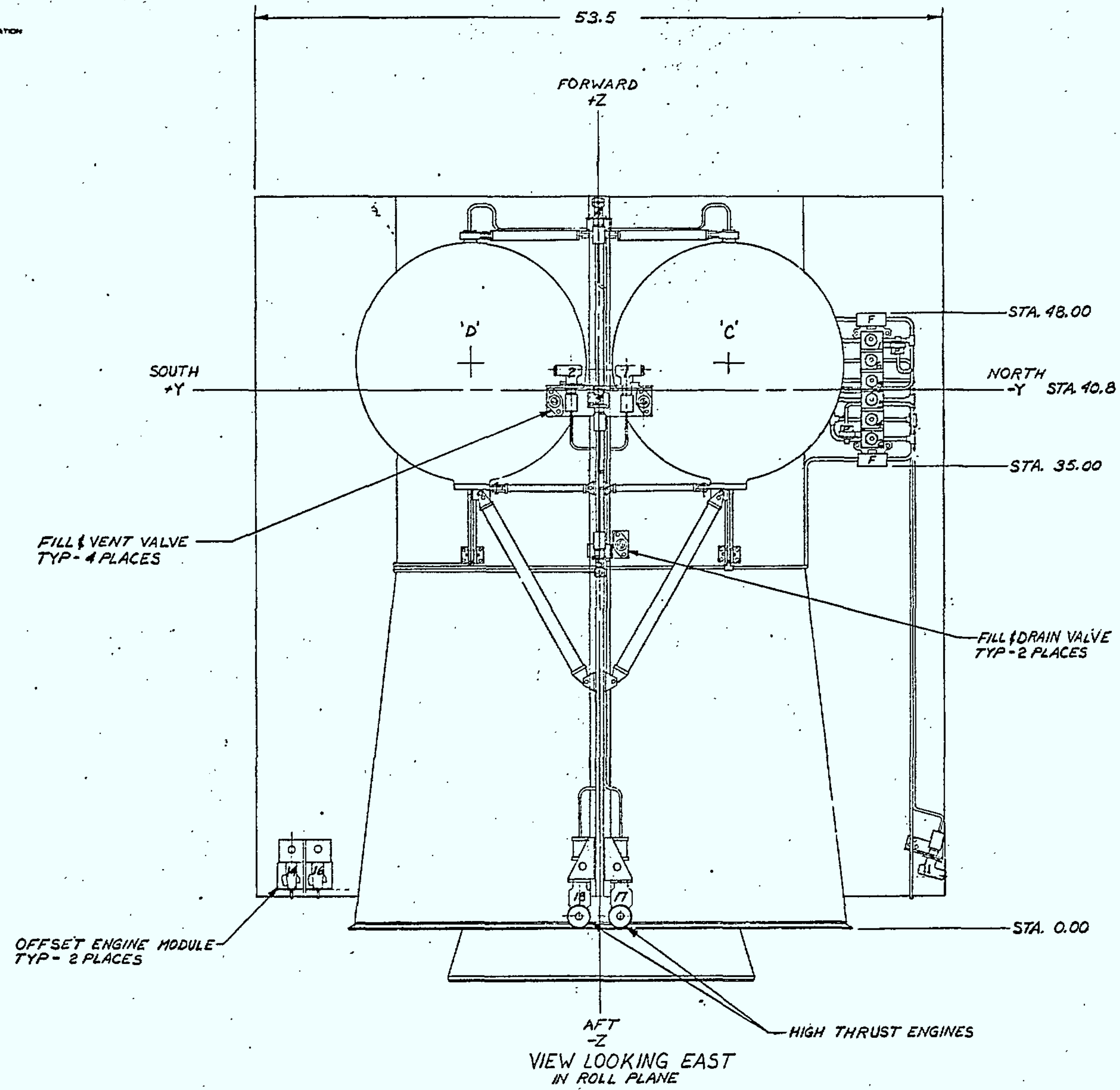

FIGURE 2.2-3. REACTION CONTROL SUBSYSTEM LAYOUT

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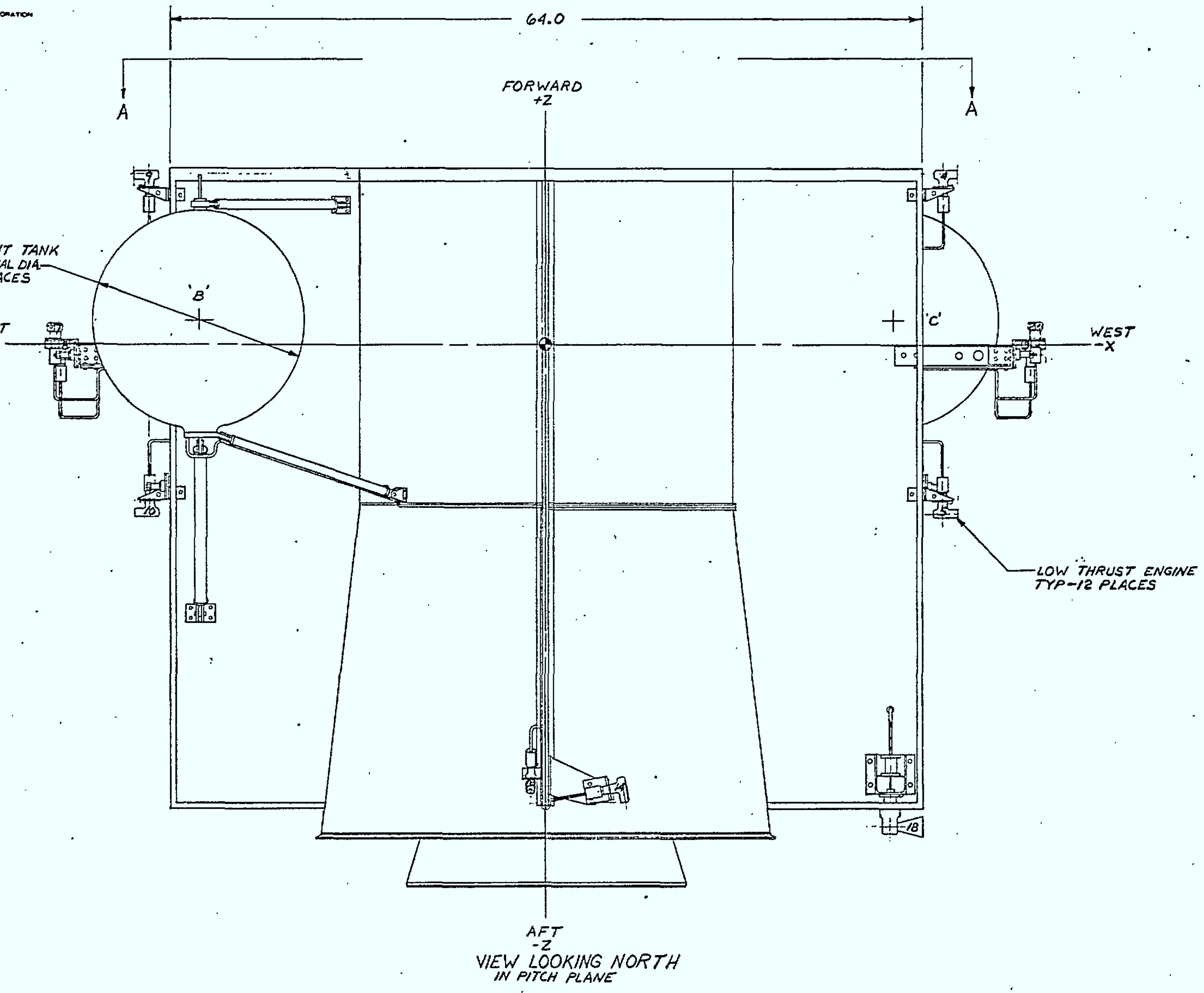

FIGURE 2.2-3. REACTION CONTROL SUBSYSTEM LAYOUT

2.2.2 continued

forces while spinning and thus provide considerably more than 40 pounds usable propellant in the spin mode for the six year mission.

The latching valve module consists of the six (6) system latching valves, the two (2) tank pressure transducers, and the two (2) system filters with their associated plumbing and all mounted on a common mounting plate. The result is a low weight, minimum volume package which offers minimum vehicle interface requirements, convenient manufacturing and handling and ready access for servicing and maintenance. Figure 2.2-4 shows a latching valve module from the CTS RCS. The proposed unit would be similar except for the elimination of one (1) latching valve. This specific six (6) valve module configuration has been qualified for use on BSE. The latching valve module is located on the north-south panel. Located on the opposite side of the panel is the Electrical Control Unit. Figure 2.2-4 shows the electrical junction box from the CTS system. The proposed ECU would be similar in design differing only to satisfy the new electrical requirements which necessitate the addition of two (2) printed circuit board assemblies. The ECU location minimizes electrical wiring weight by reducing latching valve and pressure sensor lead lengths. Although the system drawing orients the ECU connectors in the +Z direction the connectors can be located convenient to vehicle interfacing.

The Rocket Engine Modules for engines 1, 2, 3, 6, 7 and 8 are identical units which not only locate and support three (3) LTEs in their proper orientation but also incorporate provisions for two fill and vent valves, i.e., the REM that supports LTEs 1, 2 and 3 also contains the fill and vent valves for the two (2) tanks, C and D, on its side of the vehicle. Similarly the other REM has the fill and vent valves for tanks A and B. Location of the fill and vent valves at the REMs provide easy access for servicing the system with pressurant.

The fill and drain valves are located on the brackets that are provided for LTEs 5 and 10, one on each side of the vehicle. The fill and drain valves are located below the four (4) tanks to allow gravity draining of the system. Only small amounts of propellant will remain in the feed lines below the fill and drain valves requiring vacuum or purging.

The other type of REM is the two (2) engine offset package. This module is identically used in the two required positions. All modules are machined from aluminum to achieve maximum precision in locating mounting surfaces to the desired angles and positions. The module concept utilizes standard extruded 6061-T6 aluminum sections to minimize overall machining with a buildup of riveted sections, machined as necessary for minimum weight and functional

2.2.2 continued

reasons, to arrive at a truly low cost, minimum risk module structure. Figures 2.2-4 and 2.2-5 show the CTS east and west REMs. Fully visible in these photographs is the multi-function advantage of the HS modular arrangement. Not only do the modules position the rocket engines but they also support other components (principally fill and drain valves). Additionally, both high and low thrust engines can be intermixed. As can be seen in the figures the result is a light weight, protective structure that reduces interface requirements (number of fasteners) to a minimum.

The plumbing arrangement and manufacturing techniques are typical of Hamilton Standard propulsion systems, including RAE-B, CTS, IUE and BSE programs. Minimum weight tubing, well supported by tube clamps, is welded with an automatic Astro-Arc TIG welder. The pressurant fill lines are fabricated from 1/8 O.D. x .016 wall titanium tubing. The propellant distribution lines down to the filter are fabricated from 3/16 O.D. x .016 wall titanium tubing. Downstream of the filter at the transition joint the system material changes from titanium to CRES. All of the main propellant distribution lines are 3/16 O.D. x .016 wall 304L CRES while lines to the individual LTEs are 1/8 O.D. x .016 wall 304L CRES. Figure 2.2-6 is provided to more easily trace the tube routing.

The functional requirements of the engines and the vehicle structure make it desirable to assemble the RCS on the actual vehicle. Since the two (2) honey comb partitions are the major support structure for the RCS, all modules and tank support structures must be attached to these partitions. Specific definition of the interface requirements would be generated early in the program so that SPAR could incorporate these requirements into the structure during fabrication. At present specific locations and types have not been worked out; however, volumetric requirements for the various modules are given in Figure 2.2-3.

2.2.3 Weight - Preliminary system dry and wet weights have been calculated. The maximum estimated system dry weight is 64.07 lb_m which is 5.93 lb_m under the allowable dry weight limit of 70 lb_m. The six year orbit wet weight has been calculated at 258.69 lb_m which is a savings of 6.31 lb_m as compared to the allowable limit of 265 lb_m. The eight year orbit calculated wet weight is 305.44 lb_m which is 4.56 lb_m under the weight allowance of 310 lb_m. Table 2.2-I is a weight breakdown of the various system elements and includes justification data for the weight values. Weight control, as practiced by Hamilton Standard, is a very real effort and has resulted in bettering the weight targets in four (4) hydrazine system programs; RAE-B, CTS, BSE and IUE.

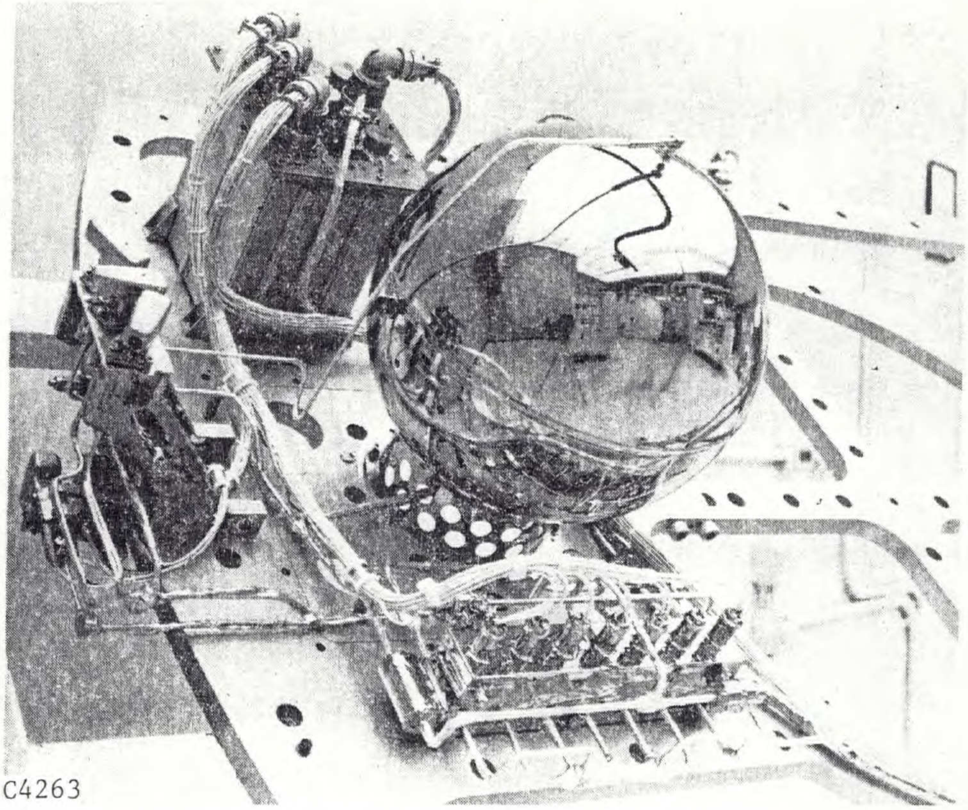

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FIGURE 2.2-4. CTS RCS - EAST REM, EJB, LATCH VALVE MODULE

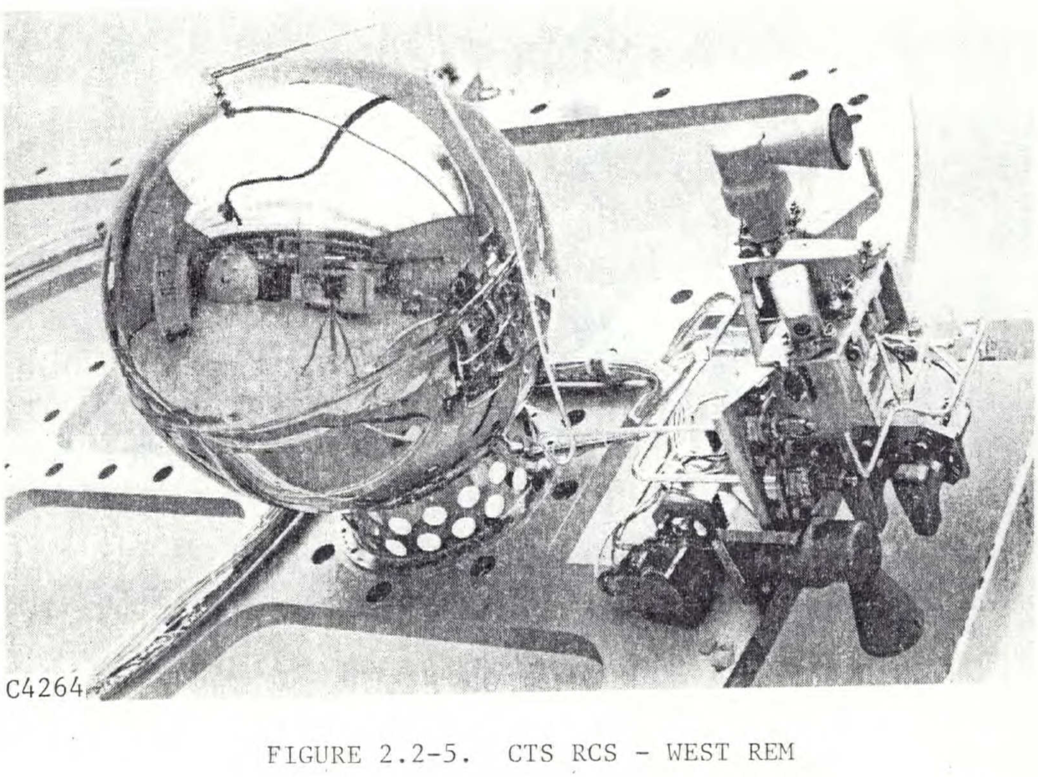

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FIGURE 2.2-5. CTS RCS - WEST REM

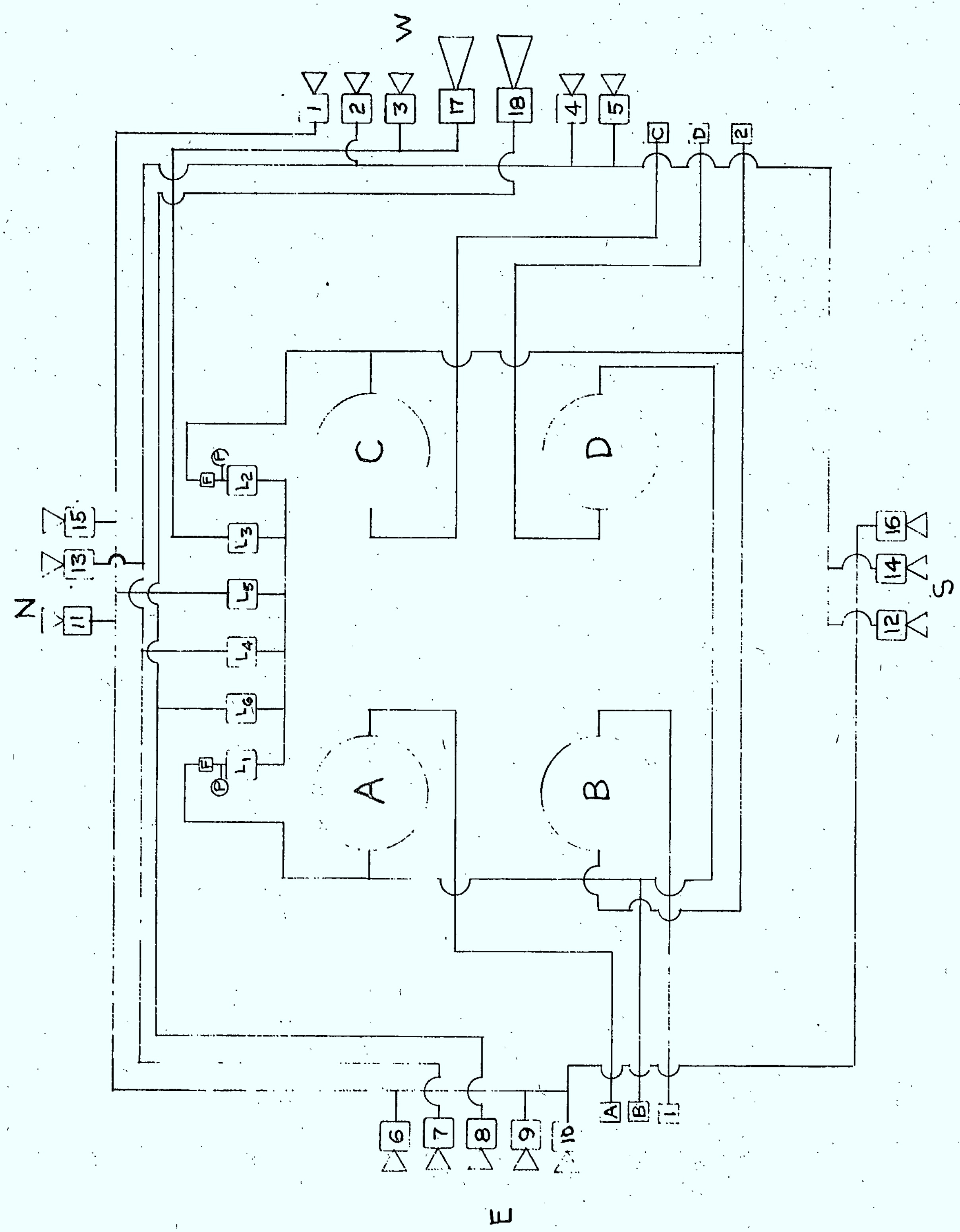

FIGURE 2.2-6. MECHANICAL ARRANGEMENT SCHEMATIC

TABLE 2.2-I

RCS WEIGHT SUMMARY

COMPONENT	NUMBER REQUIRED	NOMINAL WEIGHT (lb_m)	JUSTIFICATION
Propellant Tank	4	22.92	Modified Actual Weight
Fill & Drain/Vent Valve	6	0.78	Actual Value
Filter	2	0.44	Actual Value
Pressure Sensor	2	0.36	Actual Value
Latching Valve	6	3.12	Actual Value
HTE - TCA	2	0.76	Actual Value
HTE Valve	2	0.57	Actual Value
HTE Chamber Heater	2	0.22	Actual Value
HTE Thermal Shield	2	0.14	Actual Value
Chamber Temp. Sensor	18	1.12	Modified Actual Weight
LTE - TCA	16	1.75	Actual Value
LTE Valve	16	2.59	Actual Value
LTE Chamber Heater	16	0.29	Actual Value
LTE Thermal Shield	16	0.20	Actual Value
Electrical Control Unit	1	7.96	Estimated, Based on CTS Unit
Tubing	AR	1.93	Estimated, Based on Layout & CTS Unit
Wiring	AR	2.85	Estimated, Based on Layout & CTS Unit
Structure	AR	11.95	Estimated, Based on Layout & IUE Unit
Fasteners and Miscellaneous Hardware	AR	1.80	Estimated, Based on Layout & CTS Unit
Nominal Estimated Dry Weight		61.75 lb_m	
$\pm 2\sigma$ Allowance		2.32	
Maximum Estimated Dry Weight		64.07 lb_m	

WET WEIGHT SUMMARY:

	6 YEAR MISSION	8 YEAR MISSION
Maximum Estimated Dry Weight	64.07 lb_m	64.07 lb_m
Propellant	190.03	238.10
Pressurant	4.59	3.27
Maximum Estimated Wet Weight	258.69 lb_m	305.44 lb_m

2.3 Electrical Subsystem

The electrical subsystem provides all necessary power conditioning, power switching, and component interconnections to operate the RCS in response to spacecraft command signals. Signal conditioning of sensor signals for latching valve position, thruster catalyst bed temperatures, and propellant tank pressures is also provided for spacecraft telemetry input.

Each of three electrical interfaces is implemented with a separate electrical connector. These three interfaces are:

1. Command Signals - 18 thrusters, 6 propellant isolation latching valves, and 10 heater actuation drivers.

2. Spacecraft Power - 27.5 VDC for valves and heaters. Internal power conditioning is provided for signal conditioning.

3. Telemetry Signals - Conditioned data for 6 propellant latching valve positions (open/closed), 18 thruster temperatures, 2 tank pressures and 27.5 V current.

All electrical components are connected to the Electrical Control Unit (ECU) through electrical cables and connectors. The ECU contains all power switching and conditioning, signal conditioning and interconnections between the RCS components and the spacecraft. Figure 2.3-1 is a functional block diagram of the proposed ECU.

2.3.1 Electrical Control Unit (ECU) - The electrical control unit is an aluminum box mounted to the spacecraft structure through low-resistance mounting faces. The box has a removable cover to facilitate assembly, ground maintenance, and checkout. The cover includes a combination environmental/EMI gasket to provide an EMI tight enclosure and an environmental seal. The ECU is identical in construction to the Communications Technology Satellite (CTS) Electrical Junction Box. The ECU contains the following elements:

A. Means for termination and interconnection of the component cables. The electrical system components are connected to the unit with twisted shielded cables consistent with EMC requirements. Entry points into the box will be through four electrical connectors with the following functional separation:

Connector J4 - Thruster valve wiring
Connector J5 - Latching valve wiring including position indicators
Connector J6 - Sensor wiring
Connector J7 - Heater wiring

2.3.1 continued

B. 18 thruster valve drivers, 6 propellant latching valve drivers, and 10 heater drivers which operate solenoid valves and heater groups from the spacecraft 27.5 VDC power in response to spacecraft command signals.

C. Spacecraft power conditioning consisting of EMI suppression, reverse polarity protection and redundant power distribution. In addition a DC/DC converter is used to generate $\pm$ 12 VDC secondary power for signal conditioning functions.

D. Signal conditioning for valve position indicators, temperature sensors, pressure transducers, and 27.5 V input current.

E. Means for interface connections with the spacecraft through three interface connectors for power in, command, and telemetry functions. In addition, a test connector is provided for valve current verification.

2.3.1.1 Electronic Packaging - The unit consists of 7 printed circuit boards and a DC/DC converter. Each printed circuit board is plugged directly into a connector and is held in place by side guides. The use of internal connectors does not significantly increase the weight of the box but obviously permits easy repairability and replaceability of the circuit boards. This feature is highly desirable in an assembly of this complexity. The remaining portion of the unit is a DC/DC converter with regulators to provide $\pm$ 12 volts. R.F. filtering is provided. See Figures 2.3-2 and 2.3-3.

Commonality of circuits allows the use of five (5) CTS printed circuit boards with no changes. Additional and modified circuits will be incorporated on two (2) new printed circuit boards.

Consistent with meeting EMC requirements and eliminating backshells on the RCS connectors interfacing with the ECU, we propose to use feed-through connectors (Bendix FJT series) for all terminations except power in. The power in connector must be conventional to avoid a high current surge when power is applied. Appropriate protection against reverse polarity, transients, and interference generaged by the converter will be provided.

2.3.1.2 Electrical Driver Circuits Description - Solid state driver circuits form a major part of the ECU. The driver circuits must provide solenoid and heater drive power in response to logic level DC command signals from the spacecraft.

Thruster and latch valve driver circuits are identical to those used in CTS. All switching circuits are solid state with slew rate

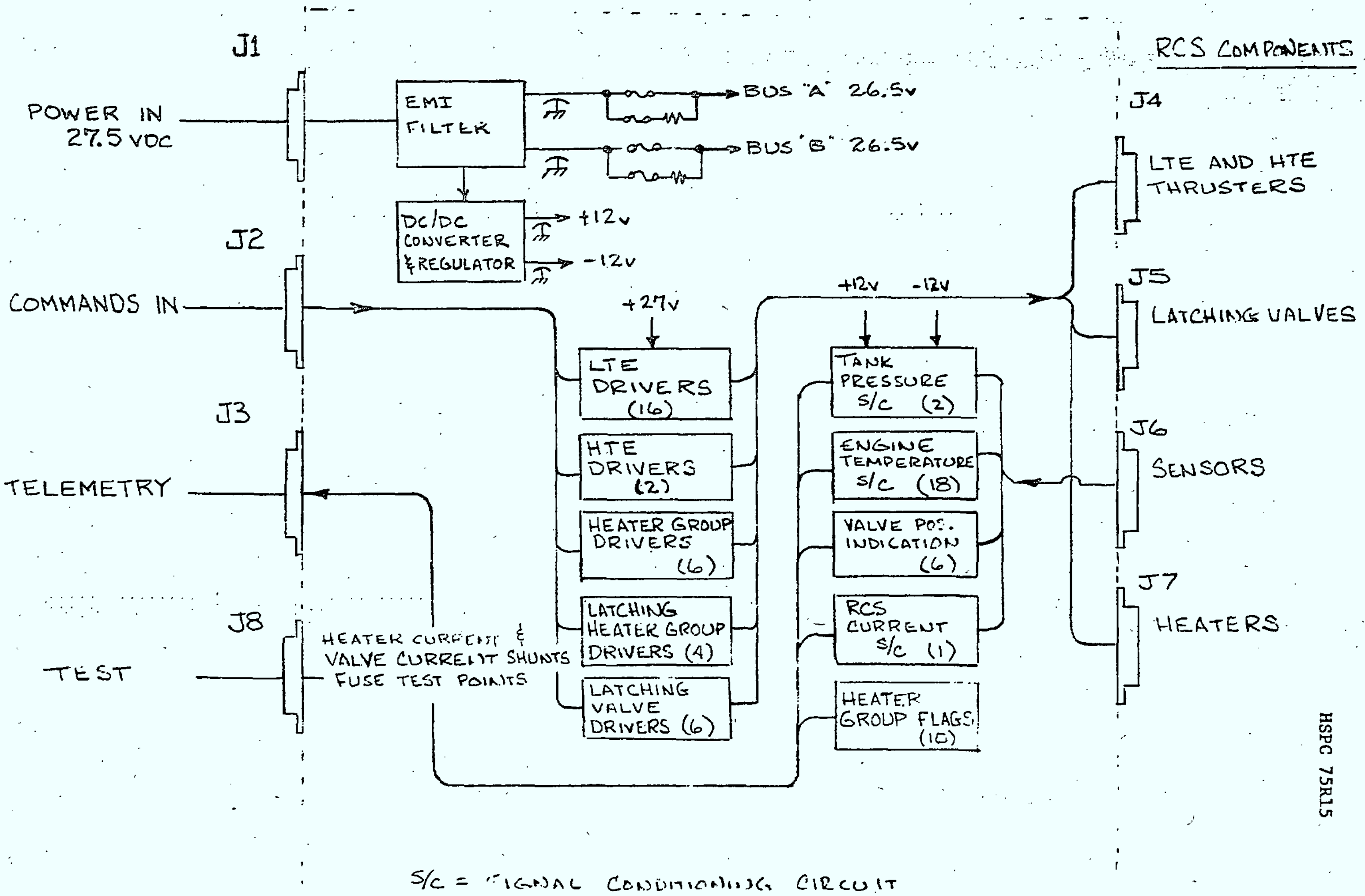

FIGURE 2.3-1. ECU BLOCK DIAGRAM

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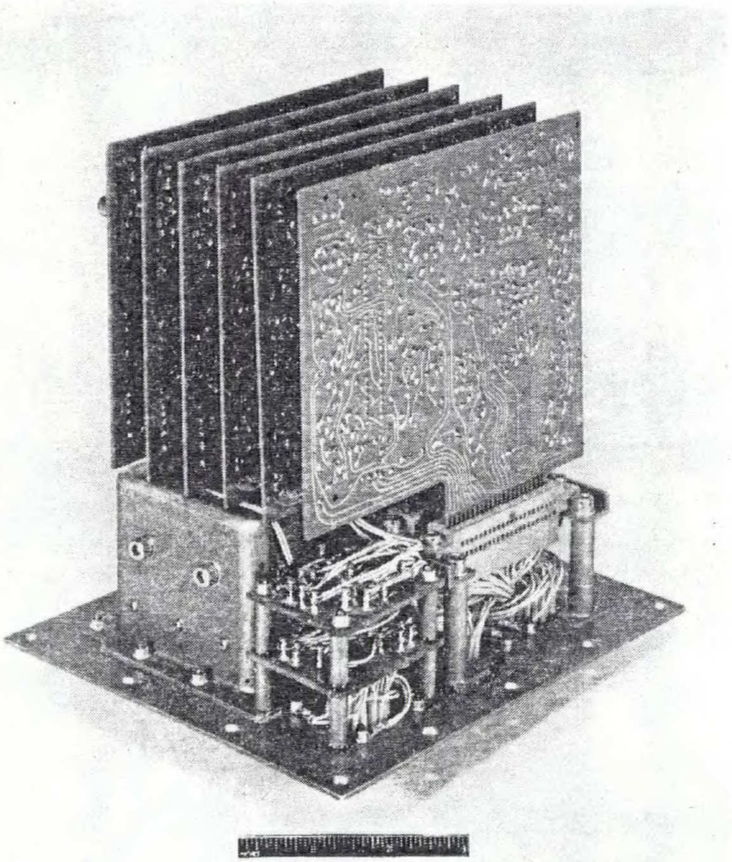

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FIGURE 2.3-2. ELECTRICAL JUNCTION BOX

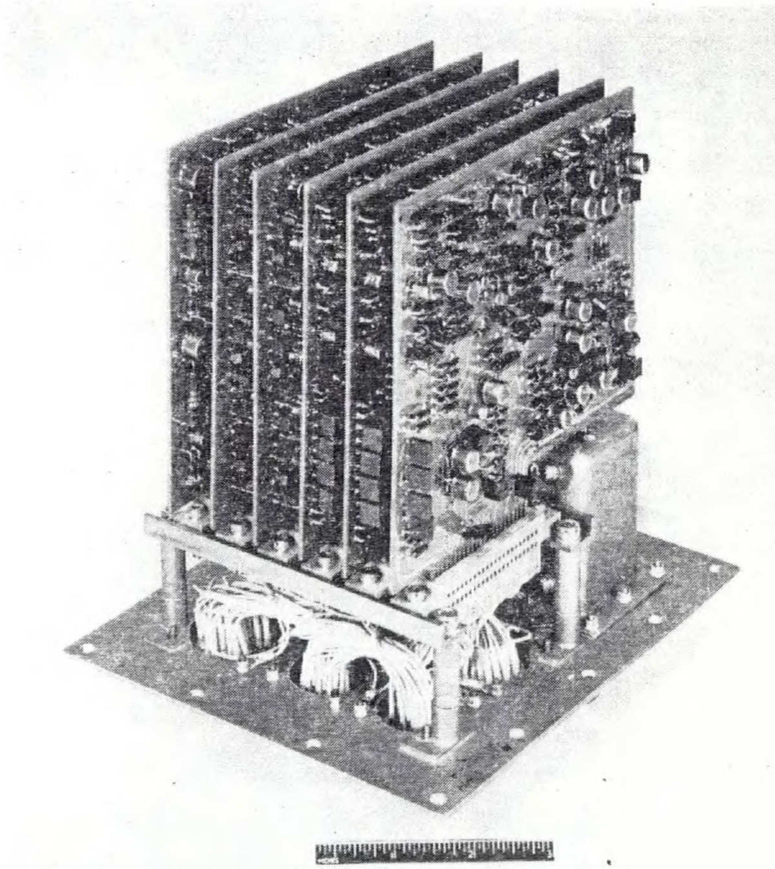

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FIGURE 2.3-3. CTS ELECTRICAL JUNCTION BOX

2.3.1.2 continued

control and suppression diodes selected to achieve desired turn-off response time. Thruster driver circuits utilize a series combination of switching transistors such that a single electronic failure cannot cause a failed "open" condition for the valve. Suppression diodes are redundant and are protected using solid state fuses such that the failed "open" valve condition cannot occur.

Heater driver circuits are slew-rate controlled using capacitors. Electronic latch provisions are incorporated for four (4) groups which are powered for long periods of time. CMOS logic elements operating from the internally generated +12 volts are proposed for this function.

2.3.1.3 <u>Signal Conditioning</u> - Signal conditioning circuitry is supplied within the ECU.

The analog conditioners operate from plus and minus 12 volts DC generated internally.

The telemetry signals provided by the RCS are 0-5 VDC analog voltages for pressure, temperature, and 27.5 V current parameters. Resistance-to-voltage converter circuits are used to precondition the thrust chamber sensors. The probes are each connected in the feedback loop of an operational amplifier circuit. The input to each amplifier is connected through a fixed precision resistor to a voltage reference (Vr) source thus setting up the constant current. By adjusting the value of this input resistor and another resistor on the non-inverting input, the gain and bias point of each buffer circuit is established. The amplifiers are of the low power type with adequate filtering to reduce the noise pickup. The buffer amplifier is fed to an isolated amplifier for final scaling to the 0 to 5 volt DC level.

The potentiometric pressure transducer is excited with a constant precision voltage generated by an operational amplifier circuit. The wiper voltage is applied to an isolation amplifier for final conditioning. Zero volt returns are provided for high accuracy.

Input current from the 27.5 volt bus is converted to a millivolt level voltage by a series shunt in the ground return line. The shunt resistance is small to minimize power dissipation. The shunt voltage is amplified in a differential mode using two operational amplifiers which are configured to give an extremely high common mode rejection ratio. Scaling to the 0-5 VDC level is done by an isolation amplifier.

Each of the buffer amplifiers requires an accurate voltage reference (Vr). A redundant reference source was selected to provide this reference with an extremely high reliability.

2.3.1.3 continued

Latching valve position indicator output is signal conditioned to discrete 0/5 VDC outputs by the ECU.

2.3.1.4 Power - The quiescent power required from the 27.5 VDC source is 2.5 watts. This power includes the excitation currents and voltages applied to each sensing element and inefficiencies of the DC/DC conversion.

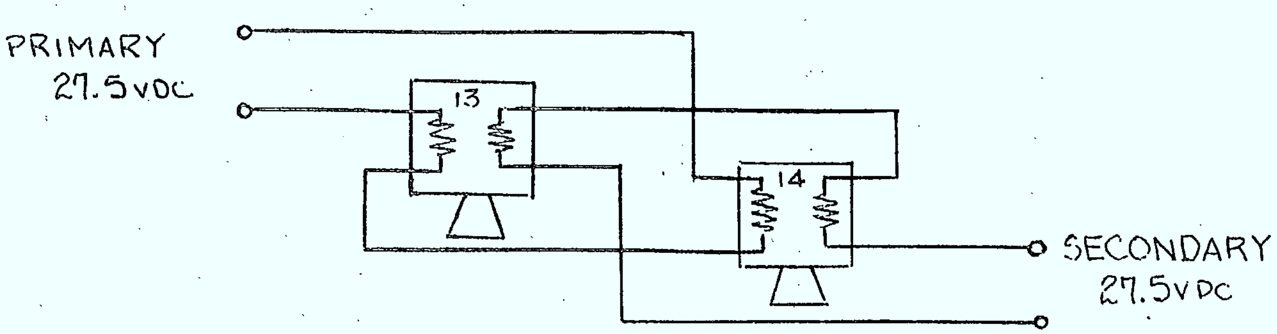
The maximum and minimum power required by each of the RCS components is listed below:

Component	Power at 28 VDC (watts) Maximum	Minimum
HTE Valve	12.35	11.97
LTE Valve	5.06	4.90
Latching Valve	9.22	-
Pressure Transducer	.0053	.0047
HTE Chamber Heater	2.45	2.27
LTE Chamber Heater per element	1.15	1.04

The HTE and LTE chamber heaters have been arranged in the following heater groups:

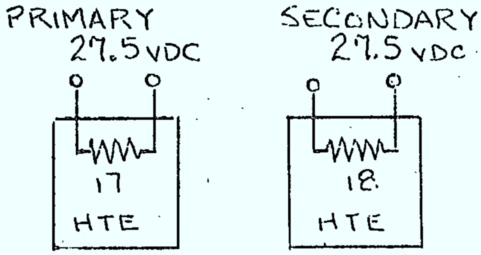
Heater Group	Engines	Nominal Power at 28 VDC (watts)	Heater Type
1	[13, 14]	2.2	LTE Primary
2	[15, 16]	2.2	LTE Primary
3	[2, 7] [5, 4] [12, 8]	6.6	LTE Primary
4	[1, 6] [9, 10] [11, 3]	6.6	LTE Primary
5	17	2.4	HTE Primary
6	[13, 14]	2.2	LTE Secondary
7	[15, 16]	2.2	LTE Secondary
8	[2, 7] [5, 4] [12, 8]	6.6	LTE Secondary
9	[1, 6] [9, 10] [11, 3]	6.6	LTE Secondary
10	18	2.4	HTE Secondary

LTE HEATER WIRING
(typ. 8 places)



2.3.1.4 continued

HTE HEATER WIRING



2.3.1.5 Electromagnetic Compatibility - The RCS is designed to meet Communications Research Centre Document EV 01-02 Issue A, Class 5. Under this classification the propulsion subsystem is designed to meet the following test requirements:

	Test Title	Test Document and Procedure
1.	Conducted Emissions 30 Hz to 20 kHz Signal and Power Leads	MIL-STD-462A CE-01 & CE-02
2.	Conducted Emissions to kHz to 100 mHz Signal and Power Leads	MIL-STD-462A CE-03 & CE-04
3.	Radiated Emissions 14 kHz to 1 GHz	MIL-STD-462A RE-02
4.	Conducted Susceptibility 30 Hz to 400 mHz	MIL-STD-462A CS-01 & CS-02
5.	Electric Field Susceptibility 15 kHz to 20 GHz	MSFC - Spec 279 Para. 4.2.1.11
6.	Spikes and Conducted Transient Emissions	CTS EV 01-02 Para. 4.2.1
7.	Spike Susceptibility	MIL-STD-462A CS-06

We have designed and tested systems of this type to these specifications for the CTS and Shuttle Environmental Control Systems.

2.3.1.6 Power Switching - The ECU as proposed does not include provisions for power switching of the 27.5 VDC bus in response to command signals. Incorporation of this function within the ECU cannot be accomplished totally since power is required to operate this power switch itself. Voltage drop considerations make the use of solid state switching inferior to using an electromechanical relay. This in turn raises reliability and redundancy questions. We recommend this power switching function be provided external to the RCS.

2.4 Mission Analysis

The mission sequences defined in Table I of the RCS specification for six and eight year missions have been analyzed and the propellant required for each maneuver computed. The results of this analysis are presented in Table 2.4-I and 2.4-II.

The total propellant required for the six year mission is 190.03 pounds and for the eight year mission is 238.10 pounds. This includes all specified contingencies and thrust inefficiencies. In addition, 4.59 and 3.27 pounds of N_2 pressurant are required for the six and eight year missions respectively.

2.4.1 Pressure Schedule - The blowdown pressure schedules for the six and eight year missions are shown in the last two columns of Table 2.4-I and 2.4-II respectively. The tankage has been selected to allow a nominal blowdown ratio of 3/1 for the eight year mission. With a nominal initial pressure of 350 psia at 70°F and a maximum initial pressure of 383 psia at 120°F, the final pressure is 168.3 psia for six years and 120.8 psia for eight years.

The larger tank ullage associated with the six year mission requires 1.32 pounds of additional pressurant, if the initial pressure of 350 psia is maintained. This additional pressurant could be eliminated if the final pressure for the six year mission were set equal to the final pressure for the eight year mission. The pressurant weight savings would however be more than offset by the increase in propellant associated with the lower engine performance at lower average supply pressures. As a result it is recommended that initial tank pressure be 350 psia for both the six and eight year missions.

2.4.2 HTE Performance - The pulsing performance for the high thrust engine for the MPB duty cycle of .135 on and .875 off is well documented. This duty cycle is the same as that required for CTS. Hamilton Standard Qualification Test Report SVHSER 6280 for the High Thrust Engine Model REA 16-7 contains detail performance in rotational format for this duty cycle of the MPB range of propellant supply pressures and temperatures. The data from this report has been used to predict the performance values shown in Table 2.4-I and 2.4-II for the precession maneuvers.

2.4.3 LTE Performance - For steady state or very dense pulsing associated with despin, north-south and east-west station keeping the performance prediction of the LTE engine is based upon statistical firing data for twelve (12) IUE LTEs firing 60 second steady state burns each at three different supply pressures. The 3 sigma repeatability of this data is ±4.0%, ±4.7%, ±3.5% at 350, 255, and 155 psia supply pressure respectively. A linear regression analysis of the data yields the following expression:

2.4.3 continued

$$I_{SP} = 162.6 \, (P_S)^{.06107}$$

where: I_{SP} is the cumulative specific impulse in lb_m sec/lb_f

P_S is the supply pressure in psia

Since most MPB steady state firing maneuvers will be in excess of 60 seconds the performance predicted by the equation is slightly conservative. The above equation was used to determine the LTE steady state performance values shown in Table 2.4-I and 2.4-II.

Except for offset operation the pulsing performance values for the LTEs used in the MPB mission tables are based upon CTS measured performance values described in detail in Hamilton Standard Qualification Test Report SVHSER 6281 for the Low Thrust Engine Model REA 10-15. The CTS LTE offset engines have been modified for use on the MPB program to accommodate the MPB smaller torque impulse bit requirement. This modification includes a reduction of the capillary tube I.D. for .010 inches to .006 inches and a reduction in the nozzle throat diameter. As a result, the engine maximum I_{BIT} at 383 psia inlet pressure and both 1.1 watt heaters on is 1.8×10^{-3} lb_f seconds. With a two (2) foot moment arm this assures a stable offset limit cycle with no additional momentum wheel weight penalty. The average offset I_{BIT} for the six year mission is 1.2×10^{-3} lb_f seconds and for the eight year mission 1.0×10^{-3} lb_f seconds. The performance for these impulse bits with both heater elements on is 121 and 120 lb_f seconds/lb_m respectively.

2.4.4 Thrust Inefficiencies - There are three forms of thrust inefficiency which have been considered in preparing the MPB mission Tables 2.4-I and 2.4-II. The first is the loss in delivered impulse during the HTE precession maneuvers. This loss is a result from cancellation of some of the delivered impulse while spacecraft is spinning. The rotational efficiency measured by test for the HTE for MPB duty cycles is 95.5% or greater. This value has been used in computing the MPB precession maneuver propellant requirements.

The north-south engines are vectored 15% outboard. This vectoring reduces the north-south impulse by 3.4%. In addition, it is estimated that the plume drag of the north-south engines on the solar panels further reduces the north-south impulse by 2.5%. These inefficiencies have been used in computing the north-south station keeping propellant requirements noted in Table 2.4-I and 2.4-II. Appendix A of this proposal contains a general analysis for computing plume drag losses for the proposed HTE and LTE engines.

2.4.5 Engine Locations - The suggested engine locations in the RCS specification minimize plume drag forces on the solar panels while allowing for full redundancy of engines and provide maximum moment arms for critical torque maneuvers. However, if the solar panel axis were moved to coincide with the spacecraft c.g. or if the solar panel orientation were fixed over the c.g. during north-south station keeping, it may be possible to further reduce north-south thrust inefficiencies by as much as 50%. This would require moving the north-south engines and clustering them around the c.g. on the north-south faces. A propellant saving of approximately 3.8 pounds for the six year mission and 5.0 pounds for the eight year mission could be realized. It should be cautioned that these savings are very much a function of the geometry of the lower support structure for the solar panels and may not be easily achieved.

TABLE 2.4-I

MULTIPURPOSE BUS RCS MISSION ANALYSIS (6 YEARS)

MANEUVER	ENGINE NUMBER	DUTY CYCLE t_{on}/t_{off} (seconds)	PULSE TRAIN LENGTH	ANGULAR IMPULSE (ft lb_f sec)	MOMENTUM ARM (ft)	VELOCITY CHANGE (ft/sec)	LINEAR IMPULSE (lb_f sec)	MAX I_{BIT} (lb_f sec)	AVE I_{BIT} (lb_f sec)	TOTAL PULSES OR TOTAL HOURS	MAX. PULSES PER ENGINE OR HOURS PER ENGINE	AVE I_{SP} (lb_f sec/lb_m)	PROPELLANT REQUIRED (lb_m)	TANK PRESSURE INITIAL	TANK PRESSURE FINAL
Pre Apogee Precession	17	.135/.875	400	2807	3.333	-	842.2*	.800	.682	1,235	1,235	234	3.60	350	342.6
Post Apogee Precession	17	.135/.875	200	1337	3.333	-	401.1*	.770	.660	608	608	230	1.74	342.6	339.2
Despin	1, 7	Continuous	-	954	3.308	-	288.4	-	-	.29 Hrs.	.29 Hrs.	232	1.24	339.2	336.3
Attitude Acquisition A) Wheel Spin-Up	5 or 9	.007/.816	-	60	1.167	-	51.4	6.0×10^{-3}	5.0×10^{-3}	10,280	10,280	140	.36	336.3	335.7
B) Limit Cycle	1,2,4,5, 6,7,11,12	Low	-	80	1.167/ 3.308	-	35.7	-	-	-	-	100	.36	335.7	335.0
C) Capture	13, 14	Low	-	40	2.000	-	20.0	2.0×10^{-3}	1.6×10^{-3}	12,500	6,250	125**	.16	335.0	334.7
Station Acquisition A) In Plane	4, 5, 9, 10	High	-	-	-	80.0	2497.1	-	-	2.57 Hrs.	2.57 Hrs.	231	10.81	334.7	315.2
B) Out-Of-Plane	1, 6, 11, 2, 7, 12	High	-	-	-	80.0	2615.1***	-	-	2.91 Hrs.	1.45 Hrs.	230	11.37	315.2	297.0
On-Board Roll-Yaw Control	13, 14	Low	Continuous	1885	2.000	-	942.5	2.0×10^{-3}	1.2×10^{-3}	785,416	392,708	121**	7.80	297.0	168.3
Pitch Momentum Dumping	5 or 9 4 or 10	.007/.560	252	104	1.167	-	89.1	6.0×10^{-3}	3.4×10^{-3}	26,205	13,103	130	.67	297.0	168.3
East-West Station Keeping	4, 5, 9, 10	High	-	-	-	84.0	2374.4	-	-	3.2 Hrs.	3.2 Hrs.	227	10.46	297.0	168.3
North-South Station Keeping	1, 6, 11 2, 7, 12	High	-	-	-	999.0	29,900.4***	-	-	40.5 Hrs.	20.3 Hrs.	227	131.72	297.0	168.3
													180.29		

* Uses rotational efficiency - 95.5%.
** Uses small tube engine and both heaters on.
*** Allows for 15° cant of N/S engines, 2.5% plume loss.

6 yr. propellant req.	180.29 lb_m
Exp. efficiency (.998)	.36 lb_m
Loading tolerance (±.2%)	.36 lb_m
Propellant contingency 5%	9.02 lb_m
TOTAL PROPELLANT	190.03 lb_m
TOTAL PRESSURANT	4.59 lb_m
TOTAL	194.62 lb_m

TABLE 2.4-II

MULTIPURPOSE BUS RCS MISSION ANALYSIS (8 YEARS)

MANEUVER	ENGINE NUMBER	DUTY CYCLE t_{on}/t_{off} (seconds)	PULSE TRAIN LENGTH	ANGULAR IMPULSE (ft lb_f sec)	MOMENTUM ARM (ft)	VELOCITY CHANGE (ft/sec)	LINEAR IMPULSE (lb_f sec)	MAX I_{BIT} (lb_f sec)	AVE I_{BIT} (lb_f sec)	TOTAL PULSES OR TOTAL HOURS	MAX. PULSES PER ENGINE OR HOURS PER ENGINE	AVE I_{SP} (lb_f sec/lb_m)	PROPELLANT REQUIRED (lb_m)	TANK PRESSURE INITIAL	TANK PRESSURE FINAL
Pre Apogee Precession	17	.135/.875	400	2807	3.333	-	842.2*	.800	.682	1,235	1,235	235	3.60	350	339.8
Post Apogee Precession	17	.135/.875	200	1337	3.333	-	401.1*	.770	.660	608	608	230	1.74	339.8	335.0
Despin	1, 7	Continuous	-	954	3.308	-	288.4	-	-	.29 Hrs.	.29 Hrs.	232	1.24	335.0	331.1
Attitude Acquisition A) Wheel Spin-Up	5 or 9	.007/.816	-	60	1.167	-	51.4	$6.0x10^{-3}$	$5.0x10^{-3}$	10,280	10,280	140	.36	331.1	330.2
B) Limit Cycle	1,2,4,5, 6,7,11,12	Low	-	80	1.167/ 3.308	-	35.7	-	-	-	-	100	.36	330.2	329.3
C) Capture	13, 14	Low	-	40	2.000	-	20.0	$2.0x10^{-3}$	$1.6x10^{-3}$	12,500	6,250	125**	.16	329.3	328.9
Station Acquisition A) In-Plane	4,5,9,10	High	-	-	-	80.0	2497.1	-	-	2.57 Hrs.	2.57 Hrs.	231	10.81	328.9	303.1
B) Out-Of-Plane	1, 6, 11 2, 7, 12	High	-	-	-	80.0	2615.1***	-	-	2.91 Hrs.	1.45 Hrs.	230	11.37	303.1	279.9
On-Board Roll-Yaw Control	13, 14	Low	Continuous	2515	2.000	-	1257.5	$2.0x10^{-3}$	$1.0x10^{-3}$	1,257,500	628,750	120**	10.48	279.9	120.8
Pitch Momentum Dumping	5 or 9 4 or 10	.007/.493	286	139	1.167	-	119.1	$6.0x10^{-3}$	$3.0x10^{-3}$	39,700	19.850	130	.92	279.9	120.8
East-West Station Keeping	4,5,9,10	High	-	-	-	112	3084.8	-	-	5.0 Hrs.	5.0 Hrs.	225	13.71	279.9	120.8
North-South Station Keeping	1, 6, 11 2, 7, 12	High	-	-	-	1320	38,506.5***	-	-	62.9 Hrs.	31.5 Hrs.	225	171.14	279.9	120.8
													225.89		

* Uses rotational efficiency - 95.5%.
** Uses small tube engine and both heaters on.
*** Allows for 15° cant of N/S engines, 2.5% plume loss.

8 yr. propellant req.	225.89 lb_m
Exp. efficiency (.998)	.45 lb_m
Loading tolerance ±.2%	.45 lb_m
Propellant contingency 5%	11.30 lb_m
TOTAL PROPELLANT	238.10 lb_m
TOTAL PRESSURANT	3.27 lb_m
TOTAL	241.37 lb_m

2.5 Components

The philosophy in the selection of components for use on the MPB RCS is to use only proven hardware - proven not just by development testing but by qualification testing or flight. Table 2.5-I lists the selected RCS components including a description of each item, the supplier's name, the unit's flight and/or qualification history and a comment on verification testing. Of all the components selected, only three components (propellant tank, offset engine and temperature sensor) have not been fully qualified on flight hydrazine systems designed by Hamilton Standard. In addition to these three components, only the HTE valve and the two thrust chamber heaters were not used on the CTS RCS. A more detailed description of the three components requiring design modification is as follows:

Propellant Tank - The tank pressure vessel is titanium which, of course, has been used by Hamilton Standard in previous systems, including CTS. The proposed MPB tank mounting provisions have been fully executed on the IUE program. Passive propellant management techniques have been under development for years, flown on other types of systems, and has been qualified for the SATCOM hydrazine control system. The proposed MPB RCS tank is based on this qualified configuration. Through the use of this passive expulsion technique a lighter weight system is possible (approximately 14 lb_m), cleaning, handling and servicing are simplified as compared to an elastomeric diaphragm expulsion tank.

Offset Engines - The offset engine is a design modification of the low thrust engines used in other locations. This engine uses the same valve, thrust chamber, heater, temperature sensor and heat shield as the other LTEs. It differs only in the diameter of the injector tube (0.006 versus 0.010 dia.) and the size of the nozzle. Thus, commonality of parts, manufacturing techniques, tooling and fixturing and test procedures has been maintained where possible, minimizing cost, schedule and the risks involved.

Chamber Temperature Sensor - The sensor will consist of a housing with the same configuration of the temperature sensors as uded on the IUE program. However, the sensing element will be a platinum resistance element as used on the CTS and BSE programs rather than the chromel-alumel thermocouples used on IUE. Thus, although the component is new, the elements that make up the item have been previously qualified thereby reducing the overall risk.

TABLE 2.5-I

MPB COMPONENT SUMMARY

RCS Component	Component Subassembly	Description	Supplier	Flight History	Qualified Applications	Verification Testing at Component Level for MPB
Fill and Drain/Vent Valve	-	The component is used to fill and vent the pressurant, fill and drain the propellant, and to clean and flush the system. The valve is manually operated, with torquing of the adjusting cap effecting primary sealing of a CRES poppet against a titanium seat. Redundant sealing and non-interchangeability features are provided.	Pyronetics	Apollo	CTS	No
Propellant Tank	-	This spherical titanium tank is machined from closed die forgings. The passive expulsion device has been developed and has been qualified for the SATCOM program. The basic tank shell has been modified to interface with the MPB structure. Polar boss mounts and welded tube ports provide an efficient structural and reliable tank package.	Fansteel	-	SATCOM	Yes - Qualification program to verify tank mount changes.
Pressure Transducer	-	The unit consists of a capsule type bellows sensing element coupled to a potentiometer wiper within a hermetically sealed and welded package. With a 5 VDC excitation voltage, a 0 to 5 VDC analog voltage output, which is proportional to tank pressure, is obtained.	Bourns, Inc.	Saturn	CTS BSE IUE	No
Filter	-	The filter uses stacked, etched discs for the filtering elements and filters down to 10 micron absolute.	Vacco Industries	Intelsat IV RAE-B	CTS BSE IUE	No
Latching Valve	-	A short D.C. pulse applied to either the opening or closing coil of the torque motor actuator causes the valve's poppet to move to the selected position. Permanent magnets "latch" the poppet in position. Valve poppet position indication is provided. Sealing is accomplished with an AF-E-102 elastomer poppet insert against a CRES seat.	Hydraulic Research & Manufacturing Co.	RAE-B SMS	CTS BSE IUE	No
High Thrust Engine	Thrust Chamber Assembly	The monopropellant hydrazine thruster uses Shell 405 spontaneous catalyst and provides 5 $lb_f \pm 5\%$ thrust at 350 psia inlet pressure. Six 0.015 in. I.D. injector tubes with penetrating diffusers inject into a 30-35 ABSG upstream/14-18 ABSG downstream catalyst bed. A spherical mid-bed retainer helps provide repeatable performance throughout life. A 59:1 area ratio right angle bell nozzle is used to produce 237 lb_f-sec/lb_m steady state specific impulse.	Hamilton Standard	ATS III IDCSP/A Skynet II NATO II NATO III	CTS BSE IUE NRL-MSD	No

TABLE 2.5-I (continued)

RCS Component	Component Subassembly	Description	Supplier	Flight History	Qualified Applications	Verification Testing at Component Level for MPB
High Thrust Engine (continued)	Thrust Chamber Valve	The valve is a solenoid operated unit. Application of a D.C. signal actuated the valve to the open position. Removal of the D.C. signal causes the spring and pressure force to close the valve. Sealing is effected by the AFE-102 elastomeric seal trapped in the plunger/poppet seating against a CRES seat.	Wright Components		IUE NRL-MSD	No
Low Thrust Engine	Thrust Chamber Assembly	The monopropellant hydrazine thruster uses Shell 405 spontaneous catalyst and provides 0.29 lb_f. at an inlet pressure of 350 psia. A single 0.010 in. I.D. injector tube with a penetrating diffuser injects into a 30-35 ABSG catalyst bed. The thruster uses a 55:1 conical nozzle and produces 225 lb_f-sec/lb_m steady state specific impulse.	Hamilton Standard	Solrad X	CTS BSE IUE NRL-MSD	Yes
	Thrust Chamber Valve	The valve operation is identical to the HTE valve.	Wright Components	Solrad X	CTS BSE IUE NRL-MSD	
Offset Engine	Thrust Chamber Assembly	The thruster is the same as the Low Thrust chamber described above except for the substitution of a 0.006 in. I.D. injector tube and a smaller diameter nozzle. These changes allow much smaller I_{BIT} pulses.	Hamilton Standard	-	-	Yes
	Thrust Chamber Valve	This valve is the same as used on the Low Thrust Engine.	Wright Components	-	CTS BSE IUE NRL-MSD	
Thrust Chamber Heater - HTE	-	The heater clamps around the thrust chamber. A Nichrome V resistance wire element wound on an alumina mandrel and encapsulated in Inconel 600 allows use up to 1600°F.	Tayco Engineering, Inc.	-	BSE IUE NATO III CTS*	No

*Similar configurations qualified.

TABLE 2.5-I (continued)

RCS Component	Component Subassembly	Description	Supplier	Flight History	Qualified Applications	Verification Testing at Component Level for MPB
Thrust Chamber Heater - LTE	-	The heater consists of Tophet A resistance wire wound on an alumina mandrel. Each of the redundant elements is then encapsulated in Inconel 600 and sealed.	Tayco Engineering, Inc.	-	IUE CTS* BSE*	No
Thrust Chamber Temperature Sensor	-	The sensing element is a Platinum resistance element insulated in Magnesium Oxide and sheathed in Inconel.	Tayco Engineering, Inc.	-	CTS* BSE*	No

*Similar configurations qualified.

2.6 Reliability Prediction

A preliminary reliability prediction has been made based on "Mission Analysis" presented herein.

Failure rate estimates were the same as for the Communications Technology Satellite, except for electrical connectors and heaters, for which explanatory notes are included. Reliability data on very long steady burns is not extensive enough in itself to provide a failure rate estimate. The assumption has been made that the failure rate is the same as for a thrust chamber continuously pulsing on a one-second duty cycle, which is a typical pulse train rate. The steady-state would involve more propellant throughput than a pulsing engine, but this is an area which can be demonstrated by qualification. On the other hand, the dynamic and varying thermal influence of pulsing would be absent from the steady burn, thus favoring reliability.

The model is predicated on engine wearout distribution being beyond 635,000 pulses or beyond 33 1/2 hours steady burn per engine. Pulsing life is supported by extensive test background. Steady burn life is estimated by analysis and extrapolation of test. The tank expulsion feature is included in the prediction on the same design basis as the CTS.

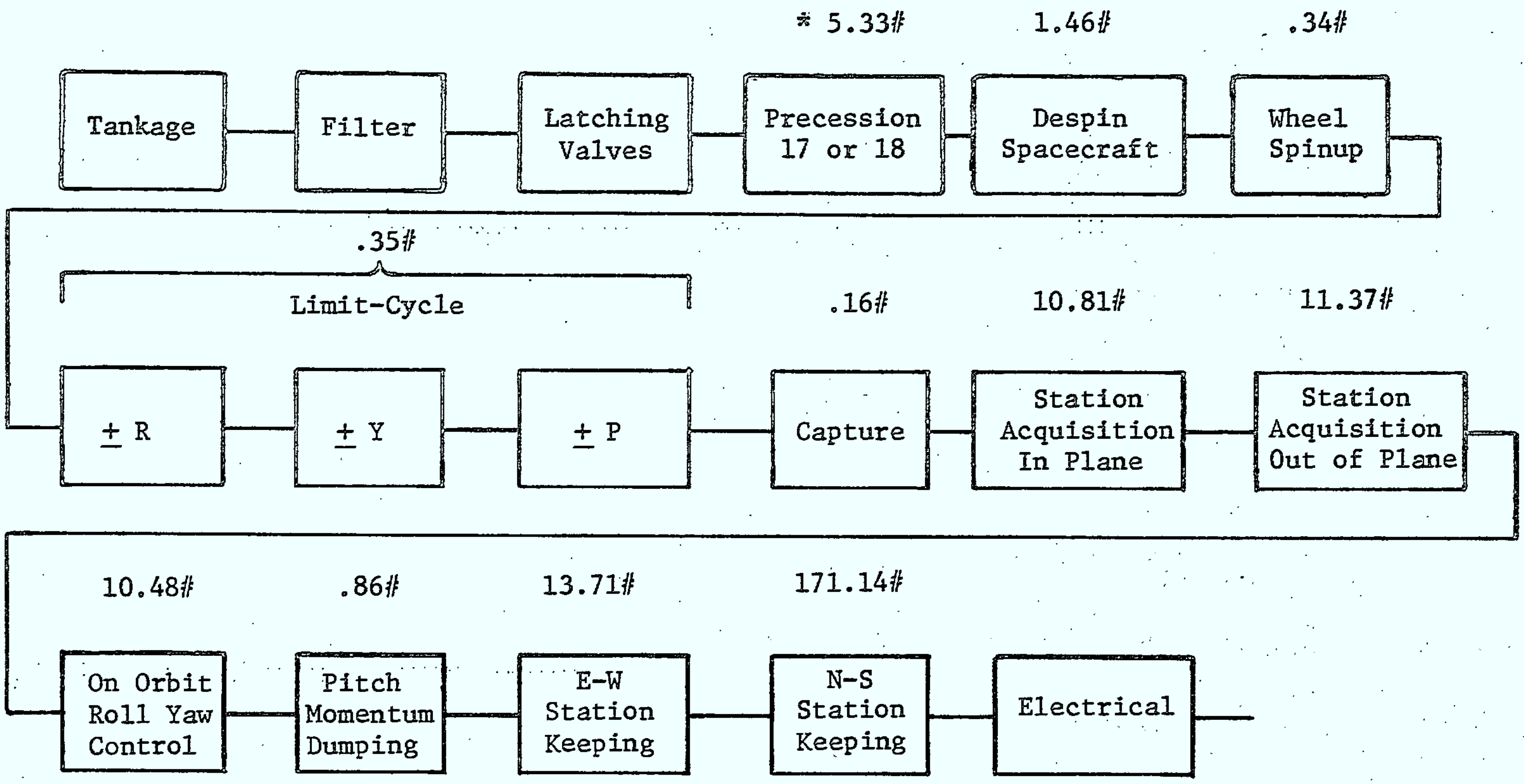
Simplifying assumptions were made in the model, wherein engine pulses required for some of the earlier maneuvers (which require relatively smaller running) have been lumped with a somewhat differing model for the principal uses of the same engines. Also certain heaters and heater drivers have been included in the models for two or more different maneuvers, violating the requirement for independence in a probability sense. We believe these were reasonable assumptions.

The reliability prediction for eight (8) years is 0.95. The apparent governing factor is the offset engines, because of the very large number of engine pulses required. The apparent next most demanding items are the electrical connectors and printed circuit board contacts. We feel that if more were known, lower failure rates would be justified for mated connectors in an unmanned orbiting environment.

SERIES RELIABILITY ELEMENTS
OF
MULTIPURPOSE BUS STUDY
(8 years)

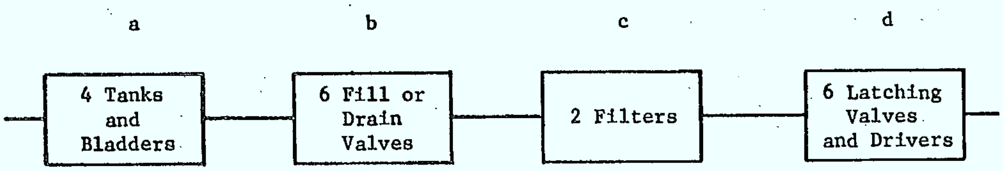
Feed System and Latching Valves	0.9972
Precession Engines, Drivers and Heaters (including capture)	0.9999+
Offset Engines, Drivers and Heaters (capture and on-orbit, roll-yaw)	0.9643
Station Acquisition (in-plane) and E-W Station Keeping, plus Wheel spin-up, Momentum dumping, Limit cycle ± P	0.9999+
Station Acquisition (out-of-plane) and N/S Station Keeping, plus Despin, Limit cycle ± Y, Limit cycle ∓ R	0.9998
Electrical Connectors, Contacts, etc.	0.9930
Overall Reliability	0.9545

RELIABILITY MODEL - MULTIPURPOSE SATELLITE BUS, RCS



*weight of propellant

FEED SYSTEM AND LATCHING VALVES



a	b	c	d
* λ = .011 x 10^{-6}/hr/ tank; plus λ = 17.5 x 10^{-6}/ expulsion	λ = 0	λ = 0	λ fail closed = .005x 10^{-6}/hr λ failure to close = .05 x 10^{-6}/hr λ failure to open = (trivial unrel.; opens at beginning of mission) λ_{LV} driver = .005 x 10^{-6}/hr

$$R_a = e^{-.044 \times 10^{-6} \times 70,080} \times e^{-70 \times 10^{-6}}$$

$$R_a = 0.9968514471$$

R_d = The probability that no L.V. closed inadvertently and (1 - [probability that one or more engines fail open AND the required LV fails to close on command])

$$R_d = (e^{-6 \times .005 \times 10^{-6} \times 70,080}) \underbrace{\left(1 - \underbrace{\left[1 - e^{-.08 \times 10^{-6} \times 1.318503 \times 10^{-6}}\right]}_{.1001077455} \underbrace{\left[1 - e^{-.10 \times 10^{-6} \times 70,080}\right]}_{.0069835012}\right)}_{.9993008974}$$

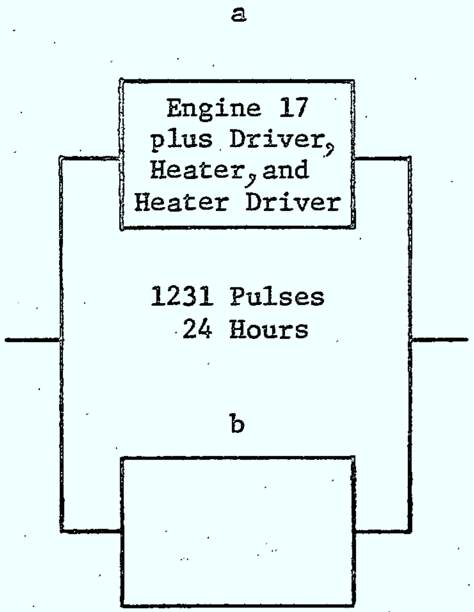
$$R_d = .9978998085 \times$$

$$R_{abcd} = .9972021742$$

* λ - failure rate

R = reliability

PRECESSION



$\lambda_{engine} = 0.15 \times 10^{-6}/\text{pulse}$

$\lambda_{valve\ driver} = 0.0009 \times 10^{-6}/\text{hr}$

$\lambda_{heater} = 1.0 \times 10^{-6}/\text{hr}$

$\lambda_{heater\ driver} = 0.022 \times 10^{-6}/\text{hr}$

$\lambda_{hourly} = 1.0229$

$$r_a = e^{-.15 \times 10^{-6} \times 1231} \times e^{-1.0229 \times 10^{-6} \times 24}$$

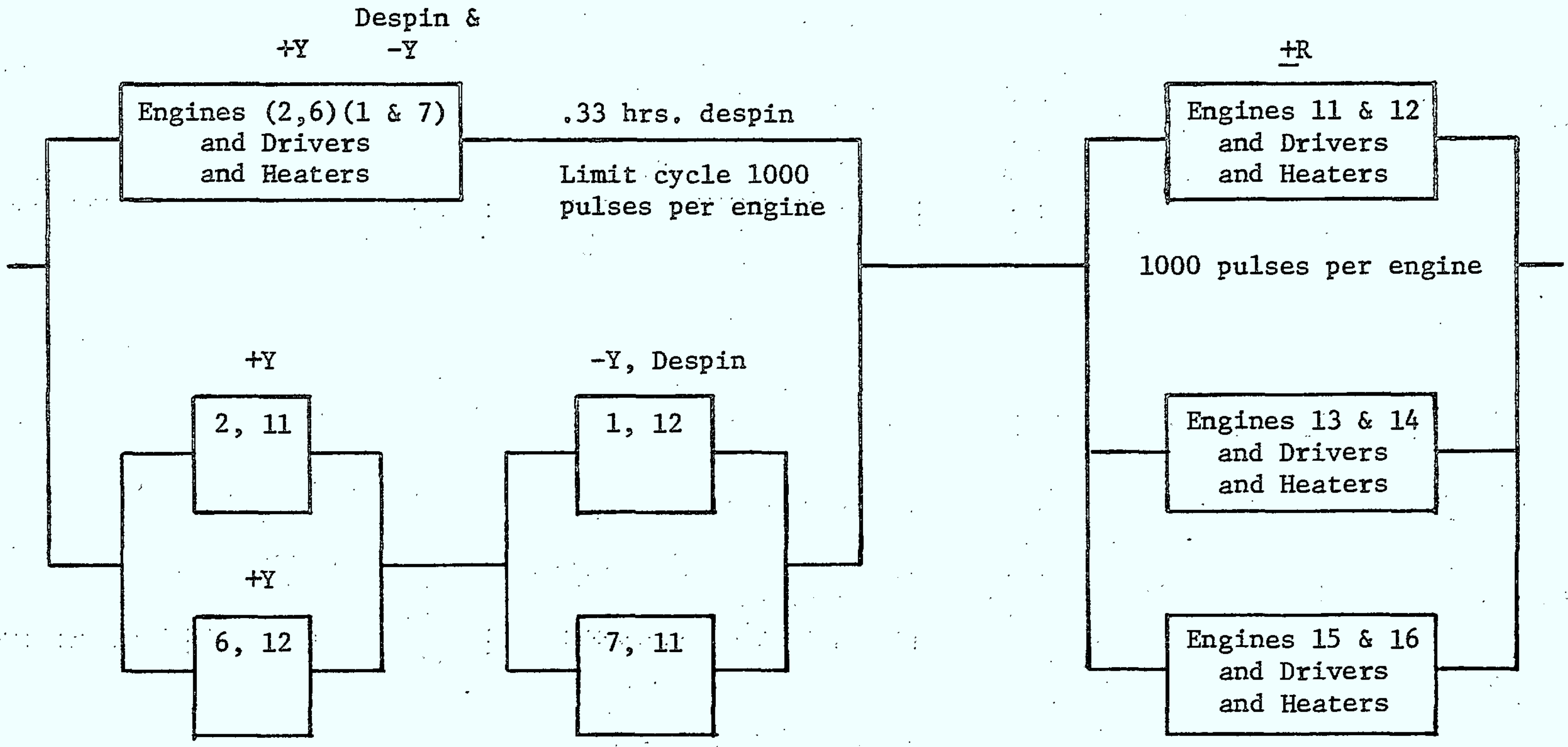
$$r_a = .9997908223 = r_b$$

$$r_{total} = 2r - r^2 =$$

$$r_{total} = .9999999562$$

HSPC 75R15

DESPIN AND ± Y AND ± R

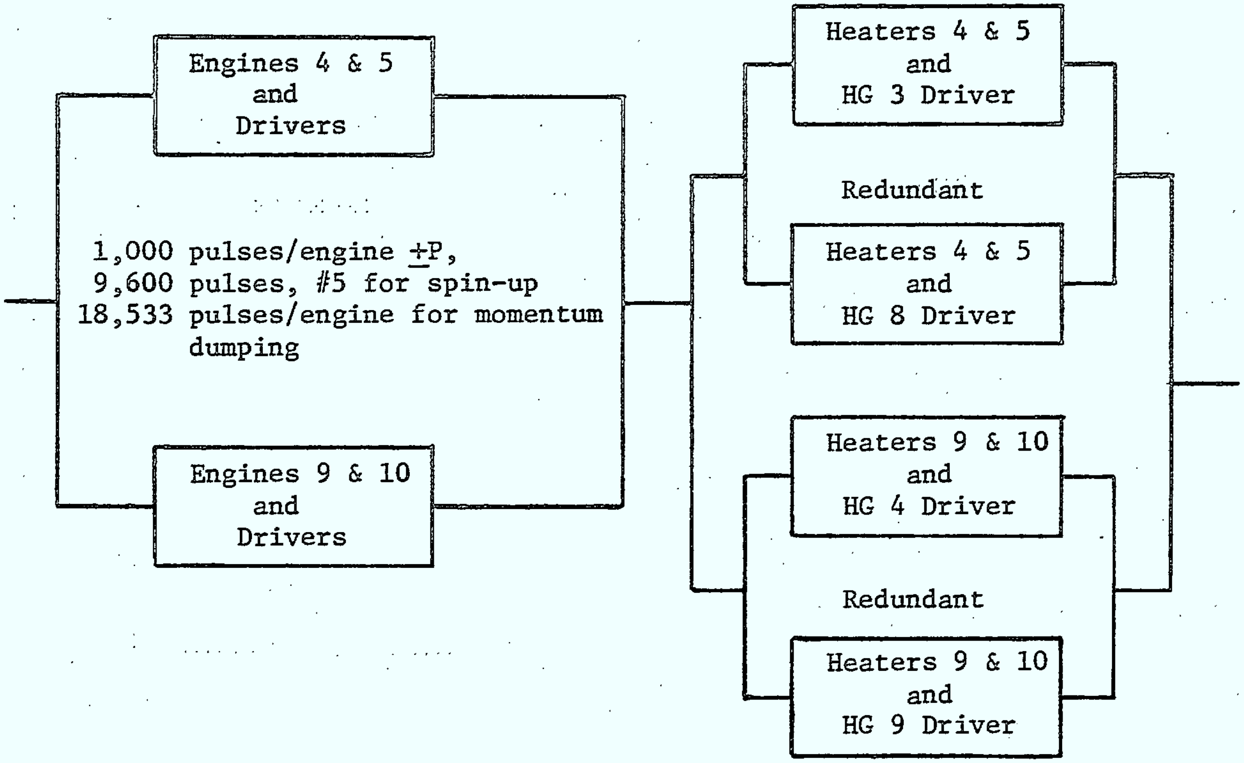


NOTE:

1. Add the normal usage here to the model for N/S station keeping, which uses the same engines (except 13, 14, 15, 16 we ignore in the above model) for much greater usage.

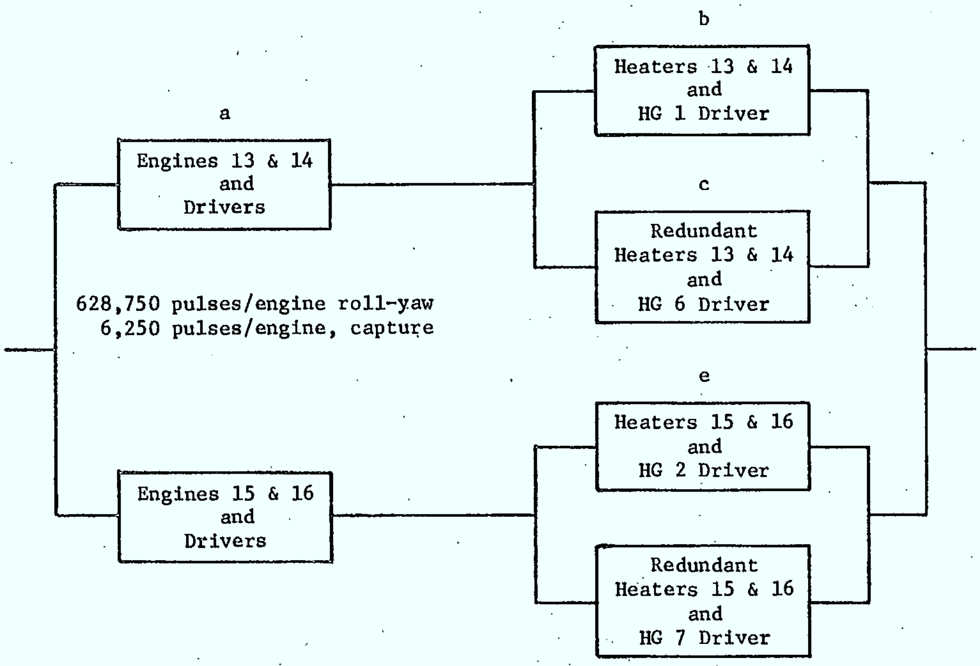
2. Because of the approximation to be employed, the above model does not block diagram the heaters.

WHEEL SPIN-UP, ±P LIMIT CYCLE, AND MOMENTUM DUMPING



NOTE: Absorb this requirement in in-plane station acquisition and E-W station keeping.

ON-BOARD CAPTURE AND ON-ORBIT ROLL-YAW CONTROL



$$R_a = e^{-0.15 \times 10^{-6} \times 2 \times 628,750} \times e^{-0.018 \times 10^{-6} \times 70,080}$$

$$= .8255037772$$

$$R_b = e^{-2.0 \times 10^{-6} \times 70,080} \times e^{-.022 \times 10^{-6} \times 70,080}$$

$$R_b = .8678800544 = R_c$$

$$R_{bc} = 2R - R^2$$

$$R_{bc} = .98254432$$

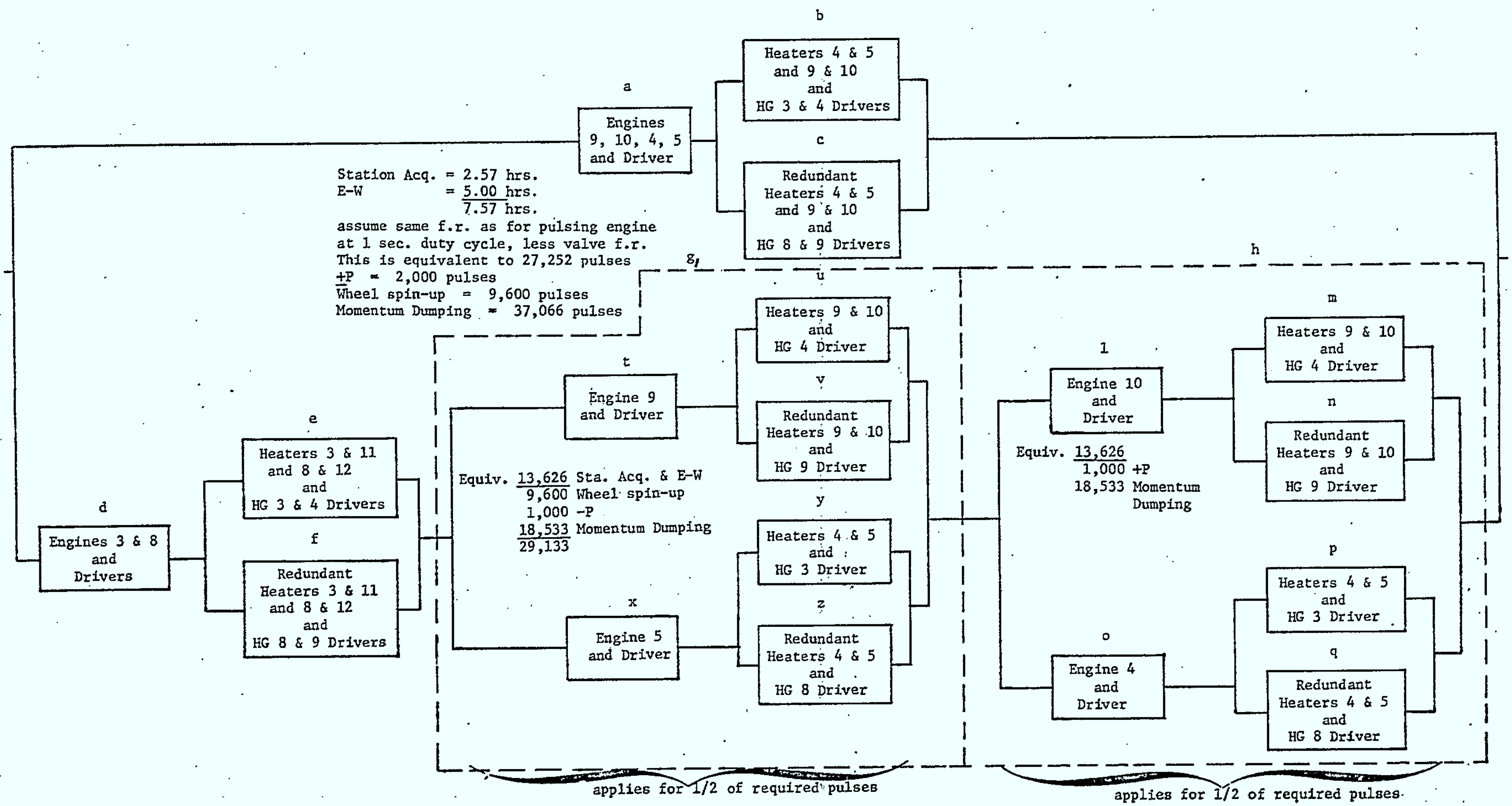
$$R_{abc} = .8110940474 = R_{def}$$

$$R_{abcdef} = 2R - R^2$$

$$R_{abcdef} = .9643145411$$

STATION ACQUISITION IN PLANE AND EAST-WEST STATION KEEPING
PLUS WHEEL SPIN-UP, ±P LIMIT CYCLE AND MOMENTUM DUMPING

Heater Time = 13.24 hrs. E-W
2.57 } hrs. Sta. Acq.
.50 }
24.00 Limit cycle
74.64 hrs. mom. dumping
114.95 hrs.



$R_a = e^{-0.05 \times 10^{-6} \times 27,252} \times e^{-.15 \times 10^{-6} \times \underbrace{(2000 + 9,600 + 37,066)}_{48,666}} \times e^{-.009 \times 10^{-6} \times 4 \times 70,080}$

$R_a = .9888769438$

$R_b = e^{-4.0 \times 10^{-6} \times 114.95} \times e^{-.044 \times 10^{-6} \times 70,080}$

$R_b = .9964629502 = R_c$

$R_{bc} = 2R - R^2 = .9999874893$

$R_{abc} = .9888645722$

$Q_{abc} = .0111354278$

$R_d = e^{-0.05 \times 10^{-6} \times 27,252} \times e^{-.15 \times 10^{-6} \times 37,066}$

$R_d = .9931014053$

$R_{ef} = R_{bc} = .9999874893$

$R_t = e^{(-0.05 \times 10^{-6} \times 13,626)} \times e^{(-0.15 \times 10^{-6} \times 29,133)} \times e^{(-.009 \times 10^{-6} \times 70,080)}$

$R_t = .9943341419$

$R_u = R_v = e^{-2.0 \times 10^{-6} \times 57.47} \times e^{-.022 \times 10^{-6} \times 35,040}$

$R_u = R_v = .9991145722$

$R_{uv} = 2R - R^2 = .999999216$

$R_{tuv} = R_{xyz} = .9943333623$

$R_{tuvxyz} = 2R - R^2 =$

$R_{tuvxyz} = .9999678892 = R_g$

$R_l = e^{0.05 \times 10^{-6} \times 13,626} \times e^{-0.15 \times 10^{-6} \times 19,533} \times e^{-.009 \times 10^{-6} \times 70,080}$

$R_l = .9829601783$

$R_{mn} = uv = .999999216$

$R_{lmn} = .9829594077 = R_{opq}$

$R_{lmnopq} = 2R - R^2$

$R_{lmnopq} = .9997096182 = R_h$

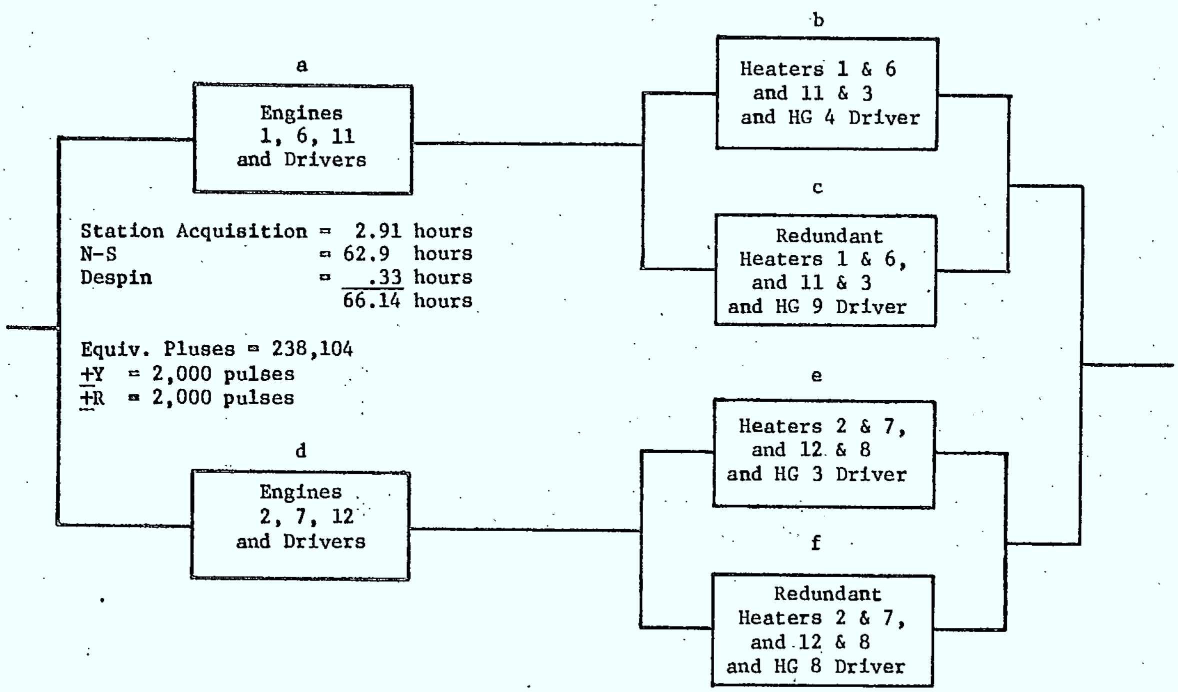
$R_{defgh} = .9927687263$

$q_{defgh} = .0072312737$

$R_{total} = R_{abc} R_{defgh} + R_{abc} Q_{defgh} + R_{defgh} \times Q_{abc}$

$R_{total} = .9999194767$

STATION ACQUISITION OUT-OF-PLANE AND N/S STATION KEEPING PLUS VEHICLE DESPIN, ±Y AND ±R LIMIT CYCLE



$$R_a = e^{-0.05 \times 10^{-6} \times 238{,}104} \times e^{-0.15 \times 10^{-6} \times 4000} \times e^{-.009 \times 10^{-6} \times 3 \times 70{,}080}$$

$$R_a = .9857057864$$

$$R_b = e^{-4.0 \times 10^{-6} \times 160.31} \times e^{-0.022 \times 10^{-6} \times 70{,}080}$$

$$R_b = R_c = .9978206183$$

$$R_{bc} = 2R - R^2$$

$$R_{bc} = .9999952503$$

$$R_{abc} = .9857011046$$

$$R_{abcdef} = 2R - R^2$$

$$R_{abcdef} = .9997955416$$

Heater time =	135.48	N-S
	24.00	Limit Cycle
	.83	Despin
	160.31	

ELECTRICAL, GENERAL

8 Connectors	Contact Points	Busses	Isolation Capacitors	28 V, EMI	5 V, EMI	28V Shunt
λ_{conn} = 0.009 x 10^{-6}/hr	0.0879	0.0016	0.0013	0	0	0

$$R = e^{-(0.009 + 0.0879 + 0.0016 + 0.0013) \times 10^{-6} \times 70{,}080}$$

$$R = e^{-.0998 \times 10^{-6} \times 70{,}080}$$

$$R = .993030417$$

TIMES FOR HEATERS

Assume heaters on for 1/2 hour before each firing.

Assume zero failure rate when "off".

TIMES FOR E-W

26 mins per burn, per VJS
+30 mins preheat

$$56 \times 1 \times \frac{62.9 \text{ hrs. burn}}{26} = 135.48 \text{ hours}$$

TIMES FOR N/S

18.2 mins per burn

$$\frac{30}{48.2} \times \frac{1}{60} \times \frac{5 \times 60}{18.2} = 13.24 \text{ hours}$$

FAILURE RATES

Heater (MIL217a Low Population Parts)		= 1.0×10^{-6} failures/hr
TCV (fail open) from CTS		= $.08 \times 10^{-6}$/cycle
TCV (fail open or closed)	from CTS	= $.10 \times 10^{-6}$/cycle
TC		= $.05 \times 10^{-6}$/pulse
LV Driver		= $.005 \times 10^{-6}$/hour
Heater Group Driver		= $.022 \times 10^{-6}$/hour
HTE Engine driver		= $.0009 \times 10^{-6}$/hour
LTE Engine driver		= $.009 \times 10^{-6}$/hour
Connectors, each	See special explanation from CTS	= $.009 \times 10^{-6}$/hour
Contacts, all		= $.0879 \times 10^{-6}$/hour
Busses, all		= $.0016 \times 10^{-6}$/hour
Isolation Capacitors, all		= $.0013 \times 10^{-6}$/hour
28 V EMI		= 0
5 V EMI		= 0
28 V Shunt		= 0

RELIABILITY ESTIMATE FOR ELECTRICAL CONNECTORS

For the purpose of this prediction the numbers and types of electrical connectors (including numbers of active pins) and printed circuit board contacts are the same as for the Communications Technology Satellite. Data on these was included in Hamilton Standard ACS-AR-165, CTS-RCS Reliability Analysis, Rev. B, 12/27/72 pages 2-15, 4-4, 6-1 and 6-2. The failure rates for connectors for CTS were derived from the RADC Notebook. These failure rates are not being used for MSB because they would result in a reliability of about 0.97 for 8 years for connectors and contacts. This would not meet the MSB reliability goal.

It is suspected that application of an hourly failure rate continuously for 8 years for connectors not being mated or unmated nor subject to vibration or corrosive environment is unnecessarily conservative. Therefore, available literature has been resurveyed to see if this view could be supported.

The following two references on long life hardware did not include coverage of connectors. This suggests the lack of a life-time problem for connectors in space application, although the evidence is only inferred.

a) Volumes I-IV, NASA CR-128908 Long Life Assurance Studies for Manned Spacecraft Long Life Hardware, December 1972

b) Volumes 1 and 2, Proceedings of the Symposium on Long Life Hardware for Space, Marshall Space Flight Center, March 17-19, 1972.

Report RADC-TR-73-171 June, 1973 "Reliability Study of Circular Electrical Connectors", by Hughes Aircraft for Rome Air Development Center indicates a generic rate for MIL-C-38999 connectors of 0.0087 failures/10^6 hrs for test and 0.018 for field. Furthermore, Figure 3.2-2 indicates questionnaire results which show that only about 15% of the experienced connector failures occur in field use and maintenance as contrasted to initial fabrication, test, assembly, etc. Thus, if ground failures are excluded, the 0.018 value would decrease to 0.018 x .15 = 0.003.

Report JPL (Jet Propulsion Laboratory) 32-1544, 12/1/71 reports on the space experience for the MARS/MARINER spaceflights. It reports 2,454,644 hours of "miniature connector" usage, with zero failures.

Lockheed LMSC/D154080 SSD Electronics Parts Orbital Failure Rates (Active and Dormant Operations), for HiRel Coaxial Connectors, lists a failure rate of 0.010 failures/10^6 hrs for active connectors and 0.001 failures/10^6 hours for dormant connectors,based on proprietary data.

Martin-Denver QR-3701010 Rev. 003 Sept. 30, 1969, "Viking Lander System and Projects Integration, Supplier Reliability Requirements" lists an active failure rate of 0.0001 failures/10^6 hours per pin. Since CTS had about 250 pins, or an average of 32 per connector, the failure rate for a connector would be 0.0032 failures/10^6 hrs.

The RADC-TR-73-171 indicated that the data therein would be used to update the RADC Notebook. On the strength of that study and the other supportive facts above, a failure rate 0.009 failures/10^6 hours per connector was chosen for this study. This assumes that field usage will be as mild as lab test and that the highest reliability connectors will be used in the design. Also, special care must be given to the care and handling of connectors, such as limitation and control of matings and de-matings, use of mating tools, etc. The CTS estimates for printed circuit board contacts were used "as is". Certainly, controls similar to those for the circular connectors should be used for the insertion, checkout, removal, etc. of printed circuit boards.

RELIABILITY ESTIMATE FOR HEATERS

The failure rate for heaters was taken directly from MIL-HDBK-217A, Table VII-XXVI. Problems experienced during the CTS program led us to use the standard MIL-HDBK value. Although those problems were solved and corrected, a sufficiently large body of usage has not yet accumulated statistically to justify adjusting the handbook failure rate downward, as was done for CTS.

2.7 Thermal Management

The spacecraft thermal control subsystem maintains the RCS components between 40°F and 130°F except for the propellant tanks which are maintained between 40°F and 120°F. In order to improve the specific impulse and impulse repeatability, performance heaters are located on all LTEs and HTEs.

2.7.1 HTE Thermal Control - Each HTE has a single element 2.36 watt heater. This heater produces a minimum 200°F chamber temperature when mounted to a 40°F spacecraft structure and radiating to sunless space. When HTE heater is turned off, the heat leak to space from the HTE is a maximum of 2.0 BTU/hr assuming a 40°F spacecraft structure mount temperature.

2.7.2 LTE Thermal Control - Each LTE has two independently powered 1.1 watt heater elements. One heater element produces a minimum chamber temperature of 240°F at minimum power and when mounted to a 40°F spacecraft structure and radiating to sunless space. Under the same conditions, two heater elements produce 426°F minimum chamber temperature. If both heater elements are turned off under these conditions, the maximum heat leak to space if .5 BTU/hr. When the LTE is in the sun and heaters are operating at maximum power, the maximum chamber temperature produced is 420°F for one heater element and 602°F for two heater elements. The mean temperature between the hot and cold extremes is then 330°F for one heater element and 514°F for two heater elements. It is recommended that the one LTE heater element on the engine to be fired be turned on 60 minutes prior to any maneuver except for the offset engines. The smaller capillary tube in the offset engines allows repeatable operation at higher chamber temperatures. It is therefore recommended that during offset operation that both heater elements on the LTE offset engines be turned on. This improves the specific impulse for this mode of operation.

3.0 PROGRAM PLAN

This section describes the management approach to the Multipurpose Bus RCS program and includes a description of program schedule, deliverable hardware and test program.

United Aircraft of Canada, Ltd. and Hamilton Standard bring to the MPB program an aerospace-oriented and proven program management system based on twelve years of experience as prime contractor and subcontractor on such flight programs as Applications Technology Satellite (ATS), Interim Defense Communications Satellite (IDCSP/A), Skynet, Solar Radiation Satellite X, Sandia Roll Rate Control System, RAE-B Velocity Control Propulsion System, Broadcast Satellite Experimental (BSE) Secondary Propulsion Subsystem, International Ultraviolet Explorer (IUE) Hydrazine Auxiliary Propulsion System, as well as Communications Technology Satellite (CTS) Reaction Control Subsystem for the Communications Research Centre. Figures 3.0-1 thru 3.0-5 graphically present a summary of the Hamilton Standard hydrazine systems experience.

3.1 Management

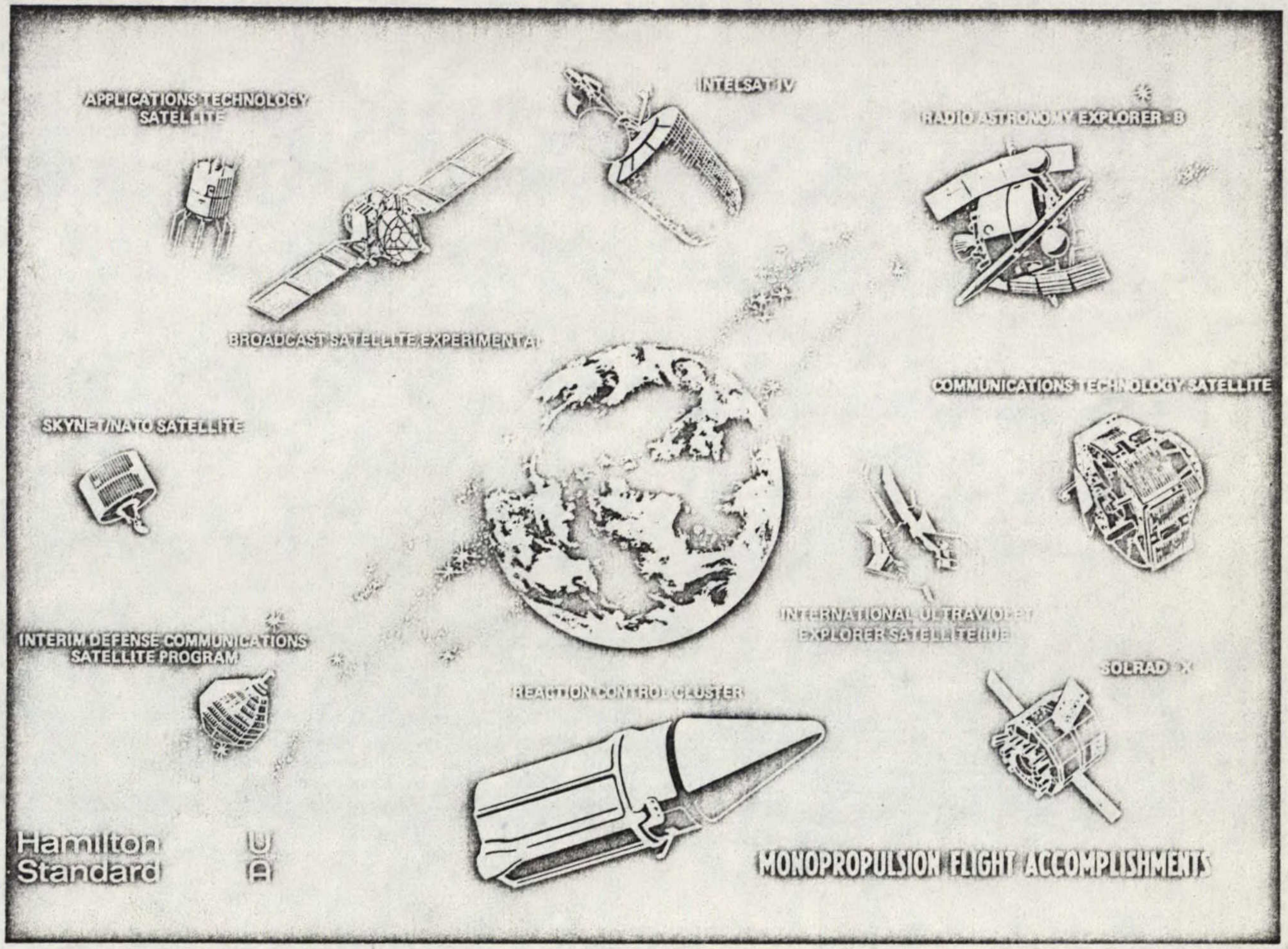
The principal sphere of endeavor of United Aircraft Corporation and its subsidiary United Aircraft of Canada, Ltd. is PROPULSION both within the earth's atmosphere and beyond. Our propulsion products include turbojet and turbofan aircraft engines, liquid propellant rocket engines, industrial and marine gas turbines, solid propellant rocket boosters, storable liquid and hybrid propulsion systems, turbine powered passenger trains and boats, and reaction control systems.

Hamilton Standard is one of five operating divisions of United Aircraft Corporation (UAC) and will have prime responsibility for all program, cost, and technical aspects of the MPB RCS program.

United Aircraft of Canada, Ltd. (UACL) will participate actively in the MPB program and will be responsible for the administration of the RCS contract as well as other discrete program tasks such as design and manufacture of the Electrical Service Cart. The major subsidiary corporations and divisional organizations which constitute United Aircraft Corporation are shown in Figure 3.1-1.

3.1.1 Division Management Structure - Hamilton Standard is diversified, autonomous operating division of the United Aircraft Corporation. For 55 years, Hamilton Standard has provided products of superior quality and performance for existing aerospace and industrial needs. Current product lines and advanced activities include propulsion systems, environmental control and life support systems, jet engine fuel controls and air inlet controls, propellers, electronic and

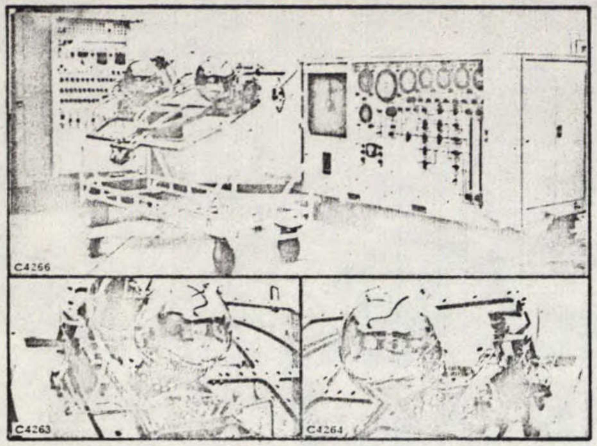
PROGRAM	LAUNCH DATE (S)
APPLICATION TECHNOLOGY SATELLITE (ATS)	III NOVEMBER 1967 IV (D) AUGUST 1968 V (E) AUGUST 1969
INTERM DEFENSE COMMUNICATIONS SATELLITE PROGRAM (IDCSP/A)	NOVEMBER 1969 SECOND LAUNCH DATE CLASSIFIED
NATO SATELLITE	MARCH 1970 JAN 1971
SKYNET	1972
INTELSAT IV	JANUARY 1971 DECEMBER 1971 JANUARY 1971 JULY 1972
SOLAR RADIATION SATELLITE (SOLRAD X)	JULY 1971
REACTION CONTROL CLUSTERS	MAY 1974
RADIO ASTRONOMY EX-PLORER -B (RAE-B)	JUNE 1973
COMMUNICATIONS TECHNOLOGY SATELLITE (CTS)	SCHEDULED 1975
BROADCAST SATELLITE EXPERIMENTAL	SCHEDULED 1976
INTERNATIONAL ULTRAVIOLET EXPLORER	SCHEDULED 1977



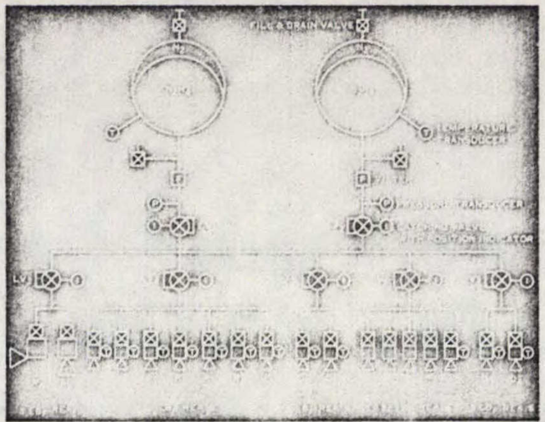
SPACE SYSTEMS DEPARTMENT
ACS ACCOMPLISHMENTS

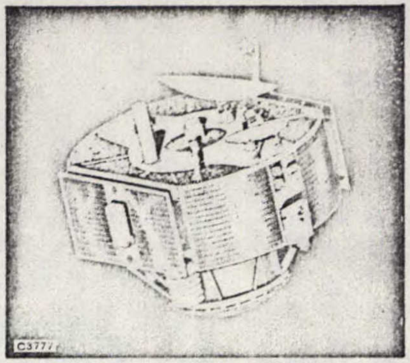
FIGURE 3.0-1

REACTION CONTROL SUBSYSTEM FOR CANADIAN COMMUNICATIONS TECHNOLOGY SATELLITE



CTS REACTION CONTROL SUBSYSTEM





COMMUNICATIONS TECHNOLOGY SATELLITE(CTS)

SYSTEM DESCRIPTION

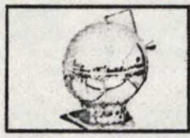
Hamilton Standard has designed, developed and qualified for the Canadian government, an integrated propulsion system for the Canadian Communication Technology Satellite. The propulsion system includes 2 positive expulsion type propellant tanks, 0.1 lb_f REA's and 5 lb_f REA's, isolation valves, instrumentation for diagnostic and status purposes and associated equipment such as fill and drain valves, filters etc. The system incorporates both active and passive thermal control techniques for thermal management. The electrical subsystem includes driver circuits for control of engine valves. The system includes 18 engines arranged in four rocket engine modules (REM's). Two of the four REM's include four 0.1 lb_f REA's and the other two REM's include four 0.1 lb_f REA's plus one 5 lb_f REA.

The system has been designed specifically for the Whecon type attitude control system used on the CTS spacecraft. This imposes tight control on minimum impulse bit sizes (0.002 lb-sec) on the 0.1 lb_f REA's with a demonstrated life of 375,000 pulses.

The program included provisions for the design and delivery of a propellant servicing cart, an electrical checkout cart, and other items of ground support equipment.

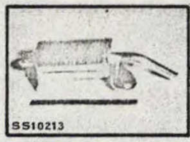
CHARACTERISTICS

87 lbm (wet weight
8,335 lb_f-sec (total impulse)
(2) 5 lb_f high thrust engines
(16) 0.22 lb_f low thrust engines
Spin/non spin application
13" diameter tanks



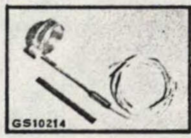
TANK PSI

LTE HS



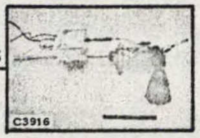
FILTER
VACCO

EJB HS

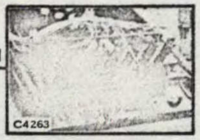


HEATERS
TSI

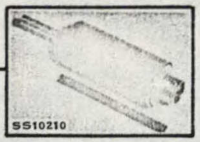
HTE HS



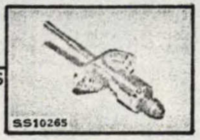
L. V. HR&M



PRESSURE
X–DUCER
BOURN



F&D
VALVES
PYRONETICS



TEST SEQUENCES

ACCEPTANCE

EXAMINATION OF PRODUCT → DRY WEIGHT CHECK → PROOF TEST
FUNCTIONAL AND ELECTRICAL CHECK → VIBRATION TEST ACCEPTANCE LEVEL
FLUSH AND DRY → EXTERNAL LEAKAGE TEST
INTERNAL LEAKAGE TEST → PASSIVATION TEST → PROPELLANT LOAD
REFERENCE PERFORMANCE TEST → FLUSH AND DRY → EXTERNAL LEAKAGE TEST
FUNCTIONAL & ELECTRICAL CHECK → POST TEST INSPECTION
PREPARATION FOR DELIVERY → FINAL INSPECTION

QUALIFICATION

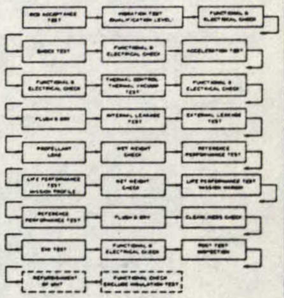
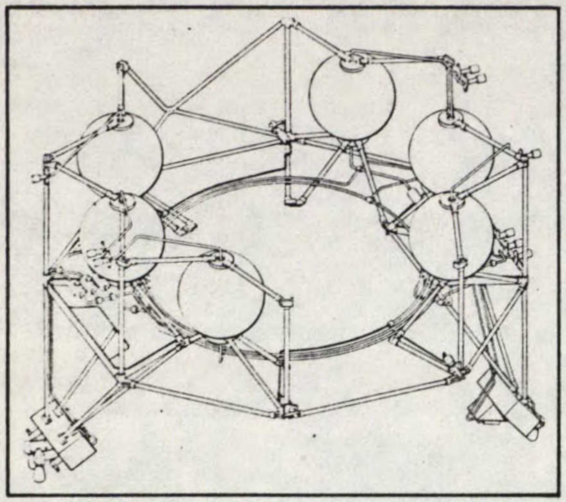
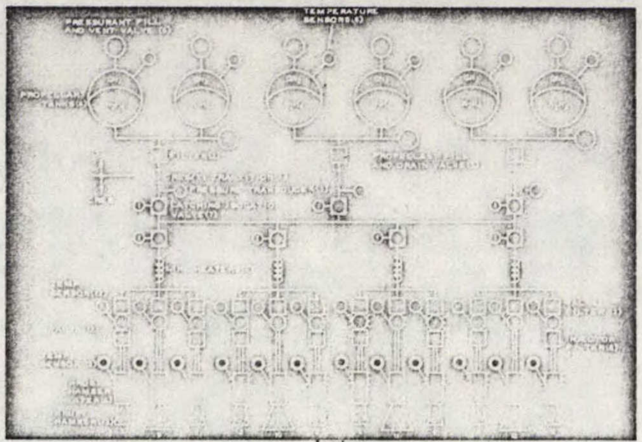


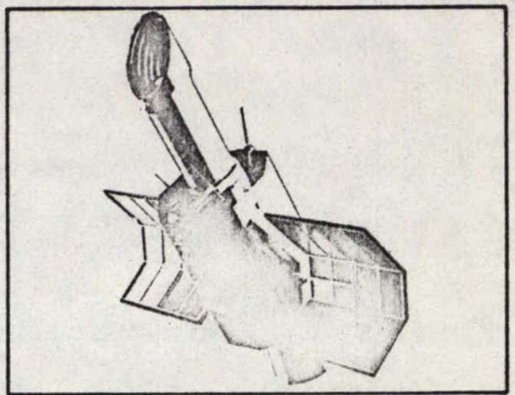
FIGURE 3.0-2

HYDRAZINE AUXILIARY PROPULSION SYSTEM FOR INTERNATIONAL ULTRAVIOLET EXPLORER (IUE)



IUE HYDRAZINE AUXILIARY PROPULSION SYSTEM





INTERNATIONAL ULTRAVIOLET EXPLORER SATELLITE (IUE)

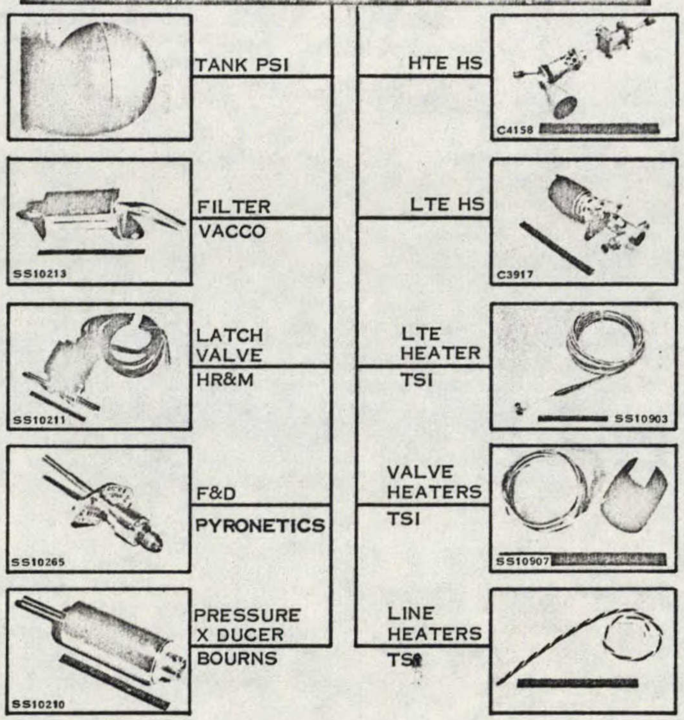
SYSTEM DESCRIPTION

The Hydrazine Auxiliary Propulsion System together with momentum wheels provides total control of the NASA Goddard Space Flight Center,s IUE no spin and 3 axis modes from 3rd stage separation to the end of the 5 year vehicle life. It consists of six spherical titanium tanks with AFE-332 position. Expulsion diaphrams, two trays of feed system components, and four thrusters supported by an aluminum strut structure which forms the IUE propulsion bay. Multilayer blanket insulation of selected surface properties provides possive thermal control of this entire bay. Two identical modules of four engines each are supported on booms below the spacecraft, to provide advantageous moment arms. Valve and line heaters are coupled with balanced radiative and absorbtive surfaces to provide module thermal control.

The program includes provisions for propellant and pressurant servicing equipment and a complete electrical test console

CHARACTERISTICS

Weight: 60 lb dry, 110 lb wet
$>$10,000 lb_f-sec total impulse
Five year mission life
Six tanks; 9.6 in dia, EPT-10 Diaphragms
All TIG welded system
Four 5 lb_f thrusters
Eight 0.1 lb_f thrusters
Full educational redundancy
Mission average Isp $>$ 210 sec
Structurally independent
Thermally independent
Full electrical system including junction box



TEST SEQUENCES

QUALIFICATION

EXAMINATION OF PRODUCT – MASS PROPERTIES – FUNCTIONAL – PROOF PRESSURE
LEAKAGE TEST – GAS FLOW TEST – FUEL COMPATIBILITY – CONTAMINATION
BASE POINT FIRING TEST – ALIGNMENT CHECK – QUALIFICATION VIBRATION
ACCELERATION TEST – ALIGNMENT CHECK – LEAKAGE TEST – GAS FLOW
THERMAL BALANCE & THERMAL VACUUM – LEAKAGE TEST – GAS FLOW
SPIN FIRING – FUNCTIONAL – LEAKAGE TEST – GAS FLOW – ALIGNMENT CHECK
CONTAMINATION – 5 LB THRUSTER BASE POINT – 0,1 LB THRUSTER BASE POINT

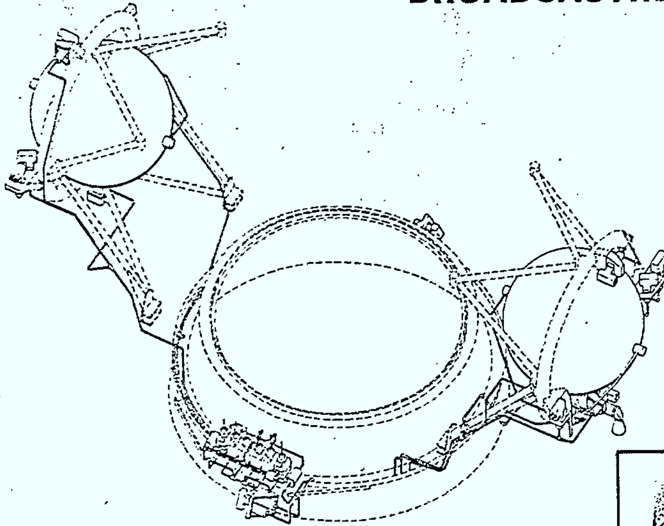
ACCEPTANCE

EXAMINATION OF PRODUCT – MASS PROPERTIES – PROOF PRESSURE
LEAKAGE TEST – GAS FLOW TEST – FUNCTIONAL – ALIGNMENT CHECK
VIBRATION TEST – ALIGNMENT CHECK – THERMAL VACUUM TEST
LEAKAGE TESTS – GAS FLOW TEST – ELECTRICAL PERFORMANCE
POST TEST INSPECTION

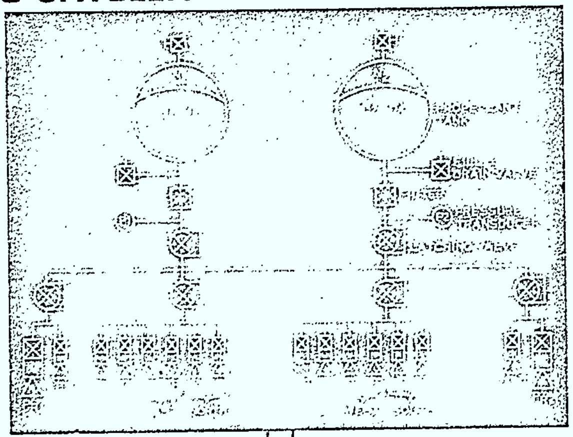
FIGURE 3.0-3

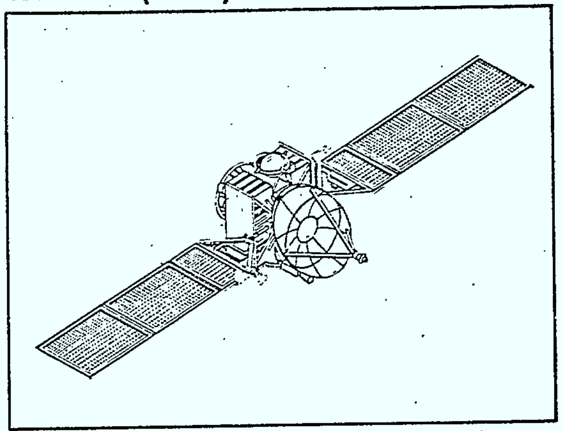
Hamilton Standard

SECONDARY PROPULSION SYSTEM FOR MEDIUM SCALE BROADCASTING SATELLITE FOR EXPERIMENTAL PURPOSE (BSE)



BSE SECONDARY PROPULSION SYSTEM





BROADCAST SATELLITE, EXPERIMENTAL

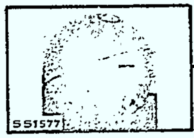
SYSTEM DESCRIPTION

The secondary propulsion system, together with three axis reaction wheels, provides total control of General Electric Space Systems Division's Broadcast Satellite. Being developed under contract from Tokyo Shibaura Electric of Japan/NASDA. After launch by a Thor-Delta vehicle, it provides precession and orientation during transfer orbit, reaction wheel unloading and $\pm 0.1^\circ$ North-South and East-West station keeping over the 5 to 7 year vehicle lifetime. The spacecraft will be the highest powered communications satellite to be launched in this decade with North-South station keeping.

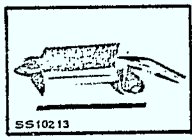
The propulsion system consists of eight clusters of 1 to 3 engines, a line component module and two spherical titanium tanks, together with the associated TIG welded plumbing and Raychem wiring harness. Heaters and thermal shields are incorporated with each engine. Instrumentation includes: engine temperature sensers, latch valve position indicatars, tank pressure and tank temperature measurements.

CHARACTERISTICS

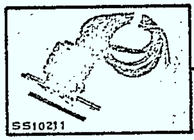
- Weight: 40 lb dry, 115 lb wet
- 17,000 lb_f-sec total impulse
- 7 year mission life
- 2 tanks, 18 in. dia., EPT-10 diaphragms
- All TIG welded system
- Two 5lb_f - thrusters
- Fourteen 0.1 lb_f thrusters
- Full functional redundancy
- Mission average $I_{sp} > 220$ sec
- Active thruster thermal control



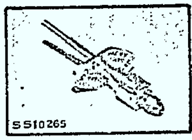
TANK PSI



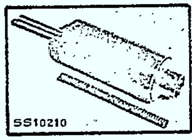
FILTER
VACCO



LATCH VALVE
HR&M

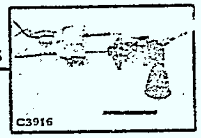


F&D
PYRONETICS

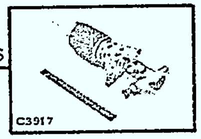


PRESSURE X DUCER
BOURNS

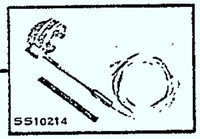
HTE HS



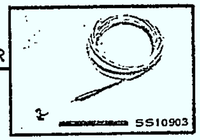
LTE HS



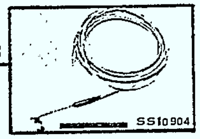
HTE HEATER
TAYCO



LTE HEATER
TSI



TEMP. SENSOR
TSI



TEST SEQUENCES

QUALIFICATION

COMPONENT ACCEPTANCE TEST → SPS ACCEPTANCE TEST → BASELINE PERFORMANCE → LEAKAGE → INTERNAL LEAKAGE → MISSION SIMULATION → VIBRATION & THERMAL VACUUM → POST. TEST INSPECTION

NOTE: GE SPECIFIED ELECTRICAL PERFORMANCE, WT., PROOF AND LEAKAGE ALL VERIFIED DURING SPS ACCEPTANCE

ACCEPTANCE

SECONDARY PROPULSION SUBSYSTEM (SPS)

EXAMINATION OF PRODUCT → ELECTRICAL PERFORMANCE → WT → PROOF PRESSURE

ELECTRICAL PERFORMANCE:
- HEATERS
- INSTRUMENTATION
- VALVES
- HARNESS & J-BOX

LEAKAGE → LEAKAGE → CLEANLINESS → VIBRATION

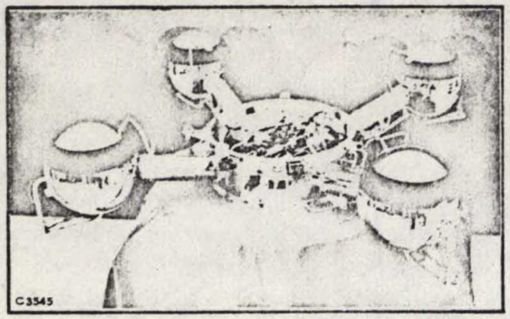
LEAKAGE:
- EXTERNAL

LEAKAGE:
- INTERNAL
- EXTERNAL

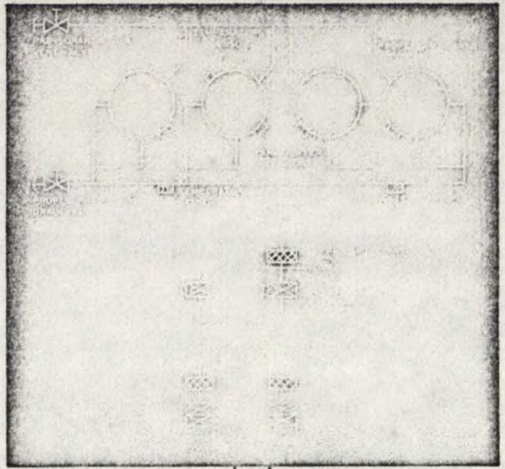
CLEANLINESS:
- PARTICULATE COUNT
- FLUSH
- DRY

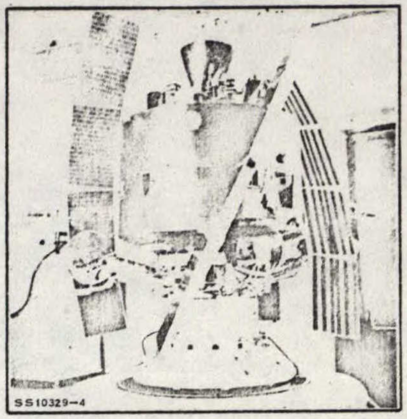
FIGURE 3.0-4

VELOCITY CONTROL PROPULSION SYSTEM FOR RADIO ASTRONOMY EXPLORER-B



C3545





SS10329-4

SYSTEM DESCRIPTION

The system consists of four spherical propellant tanks, parallel redundant latching solenoid valves, two 5 lbf Rocket Engine Assemblies (REA's), fill and drain/vent valves for loading and unloading propellant and pressurant, a filter upstream of each propellant control valve, a filter upstream of the parallel isolation valves and the instrumentation required for status and diagnostic purposes.

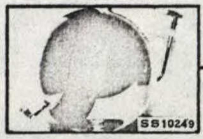
A combination of passive and active thermal control is included to provide thermal management. Each propellant line is equipped with parallel redundant heaters. The four line heaters are controlled by quad redundant thermostats. The line heaters have been sized (total wattage) to provide at 12 VDC the minimum power necessary to prevent any portion of the line from freezing. Active thermal control on the component tray is provided by two heaters wired in parallel, and controlled by two parallel thermostats.

CHARACTERISTICS

- 63 lb_f (wet weight)
- 10,000 lb_f-sec (total impulse)
- 2 5 lb_f REA's
- Spinning Satellite
- 9.85" Diameter Tanks
- 2 Year Storage Life
- 30 Days Operational Life

SIGNIFICANT ACHIEVEMENTS

Launch	6-10-73	10:13 AM		
Trans Lunar Burn	6-11-73	11:30 AM	Duration ΔV Desired ΔV Achieved	- 260 Seconds - 37.0 Meters/Sec - 37.5 Meters/Sec
Lunar Orbit Trim Burn	6-16-73		Duration ΔV Desired ΔV Achieved	- 33 Seconds - 5.7 Meters/Sec - 5.65 Meters/Sec
Lunar Circularization	6-19-73		Duration ΔV Desired ΔV Achieved	- 192 Seconds - 30.4 Meters/Sec - 32.0 Meters/Sec
Lunar Eccentricity Correction	7-19-73	4:00 PM	Duration Eccentricity Desired Eccentricity Achieved	- 12 Seconds - 0.002000 - 0.001995
Jettison VCPS	7-20-73	1:00 PM		



SS10249

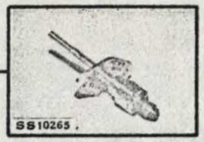
PROPELLANT TANK
ARDE

C2922

REA
H.S.

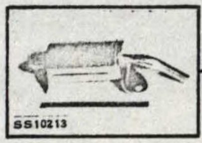
SS10211

LATCHING VALVE
HR&M



SS10265

F & D VALVE
VACCO

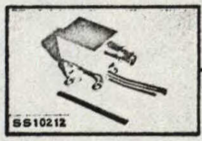


SS10213

FILTER
VACCO

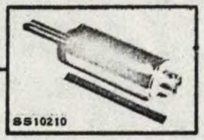
SS10214

TC HEATER
TSI



SS10212

T.C.V.
HR&M



SS10210

PRESS X–DUCER
BOURNS

TEST SEQUENCES

ACCEPTANCE

EXAMINATION OF PRODUCT — ELECTRICAL CHECK — DRY WEIGHT — PROOF PRESSURE — INTERNAL LEAKAGE — EXTERNAL LEAKAGE — POST TEST INSPECTION

QUALIFICATION

BASEPOINT — DECONTAMINATION AND CONTAMINATION CHECK — LEAKAGE — ELECTRICAL — MASS PROPERTIES — SHOCK — ACCELERATION — VIBRATION — ENGINE ALIGNMENT CHECK — PROOF PRESSURE — THERMAL VERIFICATION — THERMAL VACUUM — WET WEIGHT — MISSION PROFILE — SPIN FIRINGS — EXTREME TEMPERATURE FIRINGS — ENGINE STARTS — ENGINE DECONTAMINATION AND CONTAMINATION CHECK — POST TEST INSPECTION

FIGURE 3.0-5

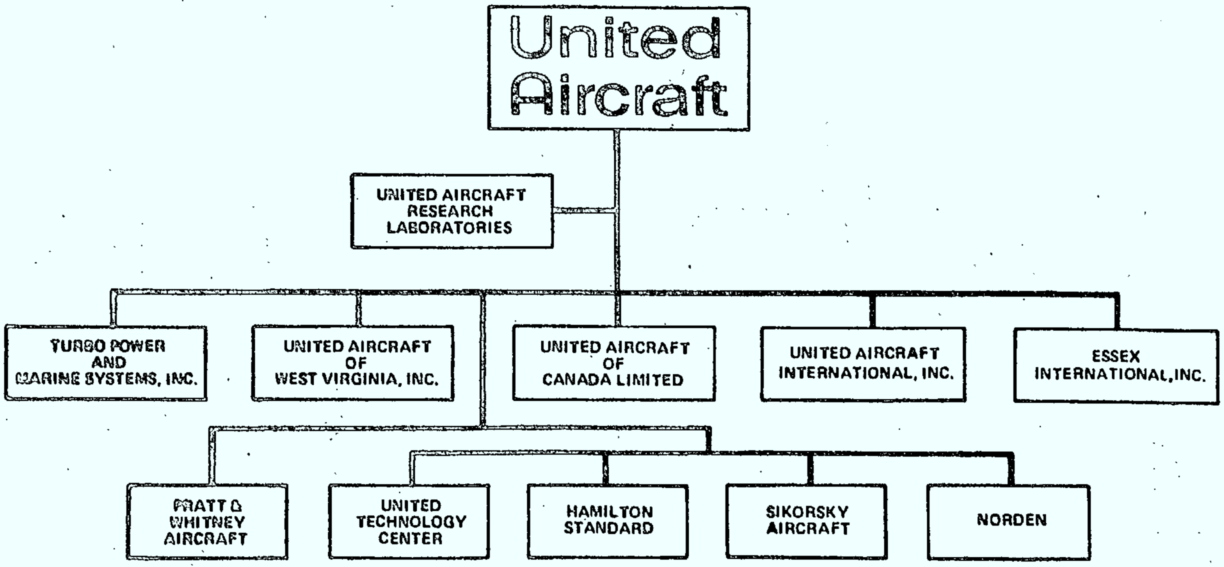


FIGURE 3.1-1. UAC ORGANIZATION

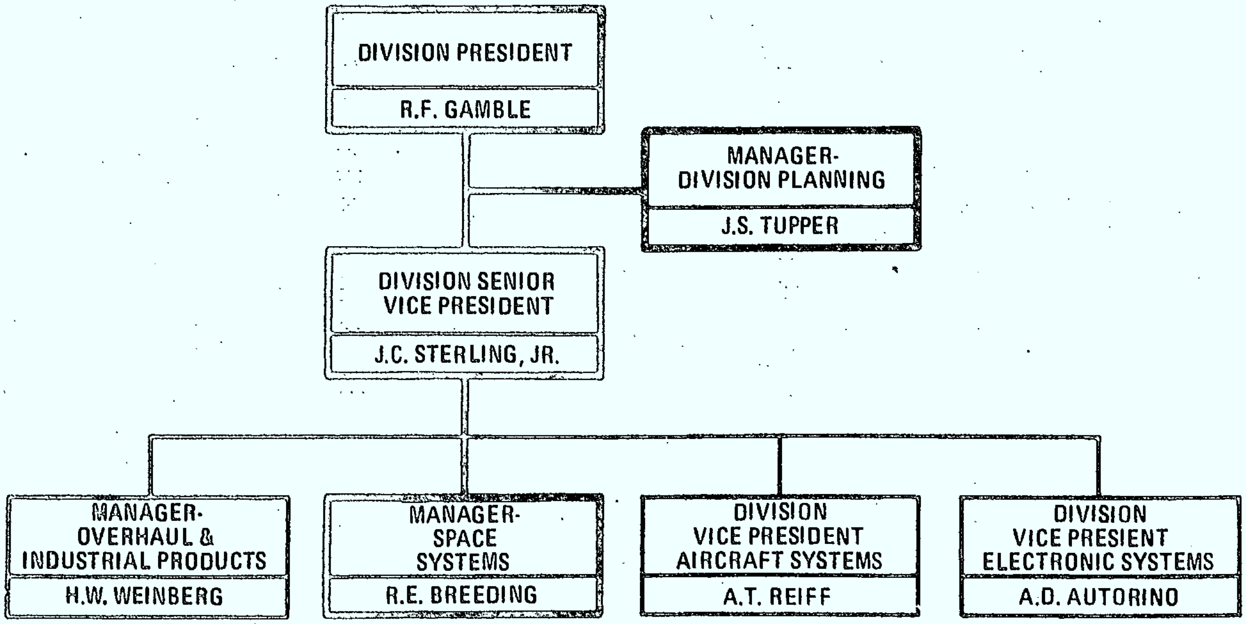


FIGURE 3.1-2. DIVISION ORGANIZATION

3.1.1 continued

micro-miniaturization equipment, electron beam welding machines, digital data systems, inertial guidance systems, and aerospace ground support equipment. Hamilton Standard is a complete operating organization, encompassing research, engineering, testing, production, reliability, quality control, sales, purchasing, service, finance and accounting, and personnel functions all in its own facilities. The division organization is shown in Figure 3.1-2.

A total of 5,500 administrative, engineering, and manufacturing personnel are involved in the development and fabrication of these products. The precision nature of Hamilton Standard products requires a highly skilled and diversified complement of production manpower which currently numbers approximately 2,500. The average experience of production personnel at Hamilton Standard is 12 years. Approximately 1,000 engineering and scientific personnel are employed at Hamilton Standard. Our engineering skills cover virtually every basic technical discipline with an average personnel experience level of 11 years.

3.1.2 Space Systems Department Organization - The Space Systems Department (SSD) is one of Hamilton Standard's semi-autonomous product departments as shown above. SSD is responsible for all space and missile products related to hydrazine propulsion and life support systems and will be responsible for the conduct of the MPB RCS program. The department organization is shown in Figure 3.1-3.

Mr. Robert E. Breeding, Manager Space Systems Department, is responsible for all the activities of the Space Systems Department and reports to the Division Senior Vice President, Mr. J. C. Sterling. Mr. Breeding has successfully directed the activities of the 300-man plus department on numerous hydrazine system and engine programs including the Sandia Roll Rate Control System, RAE-B velocity control propulsion system, the Canadian Communications Technology Satellite propulsion system, and a host of other hydrazine engine and technology programs including NATO III, NRL/MSD and Solrad X. In the environmental control system area, his responsibilities included the Apollo Portable Life Support System, the Lunar Module (LM) Environmental Control System, and the Space Station Prototype Environmental and thermal control system, and the current Space Shuttle Atmospheric Revitalization and the Freon Coolant Subsystems. Mr. Breeding has acquired over 20 years experience in the aerospace industry in all facets of Hamilton Standard's product and technical areas of cognizance.

The Space Systems Department utilizes the program management system concept to manage and staff each of its programs and is fundamentally composed of two basic organizational categories:

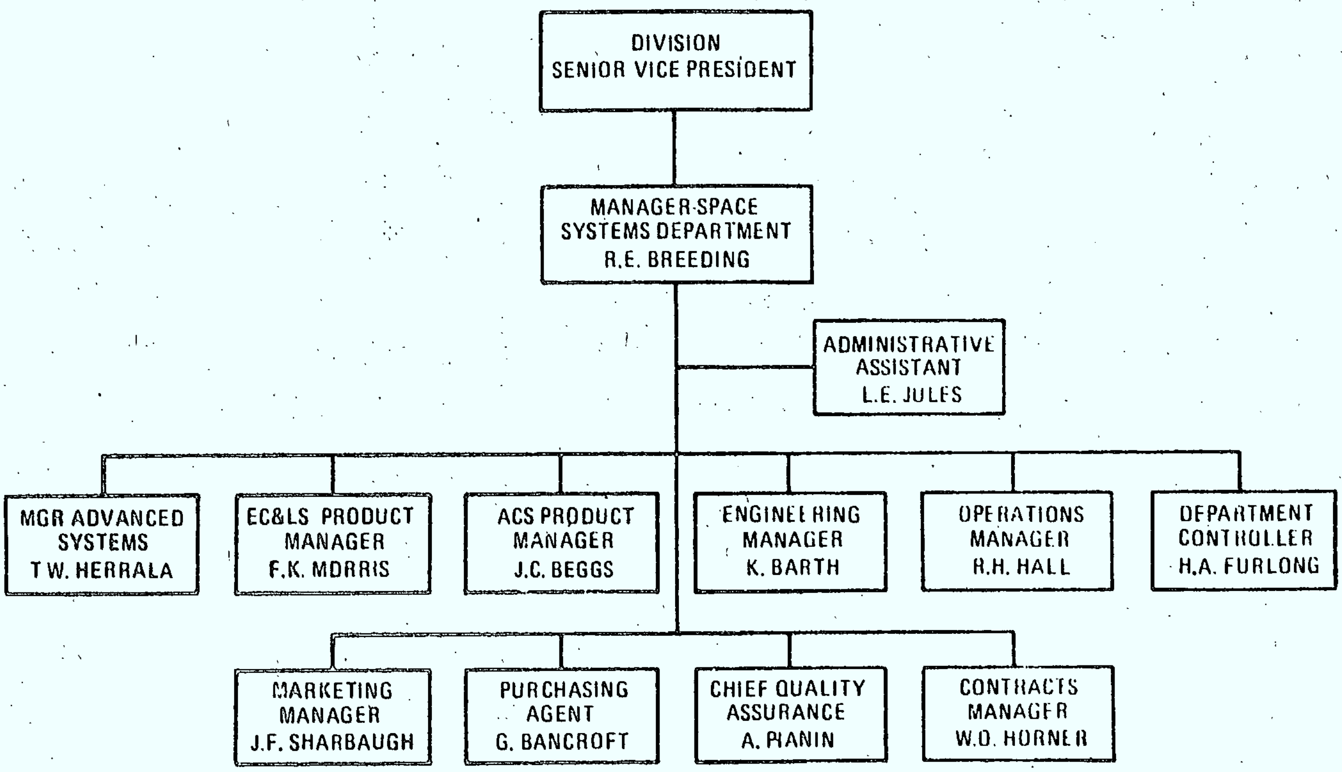


FIGURE 3.1–3. SPACE SYSTEMS DEPARTMENT ORGANIZATION

3.1.2 continued

- Program Management groups incorporating overall responsibility for the planning, management, and success of specific programs.

- Functional organizations (Engineering, Operations, Purchasing, Quality Assurance, Contracts Administration and Controller) responsible for providing each program with the necessary personnel, facilities, special skills, and other resources to be directed by the Program Manager. On each program, a full staff of specialists is assigned to the program and they are totally dedicated to that program. They receive these directions from the Program Manager and form the team to accomplish that program.

3.1.3 MPB RCS Program Organization - The MPB RCS program will be directed by a full time Program Manager who will have responsibility and resources for meeting all technical, cost and schedule objectives of the program. The Program Manager will be assisted by a program team which consists of functional managers who are responsible to the Program Manager for the execution of tasks within each of the major functional organizations - Engineering, Operations, Quality Assurance, Program Control and Contracts Administration. Together with the Program Manager, these functional managers form the Program Management Office (PMO) and are physically located in a PMO complex to enhance communication between personnel and to clearly identify them to all Hamilton Standard, UACL and SPAR personnel. The proposed MPB RCS organization is shown in Figure 3.1-4.

Our most experienced personnel have been assigned to the program team and are intimately knowledgeable of the management, mission and subsystem requirements of the MPB RCS. Specifically, the RCS Program Manager, Mr. Harry Garfinkel and the RCS Engineering Manager, Mr. Vincent J. Sansevero were instrumental in the success of the CTS RCS program which was accomplished for the Communications Research Centre, Department of Communications, Ottawa, Ontario. This same team is currently directing the International Ultraviolet Explorer Hydrazine Auxiliary Propulsion System program for the National Aeronautics and Space Administration, Goddard Space Flight Center and will be available for MPB RCS.

3.2 Program Schedule

The proposed RCS program schedule is shown in Figure 3.2-1 and represents a cost effective approach to accomplishing program requirements. The schedule reflects the current availability of designs without modification for the majority of RCS components

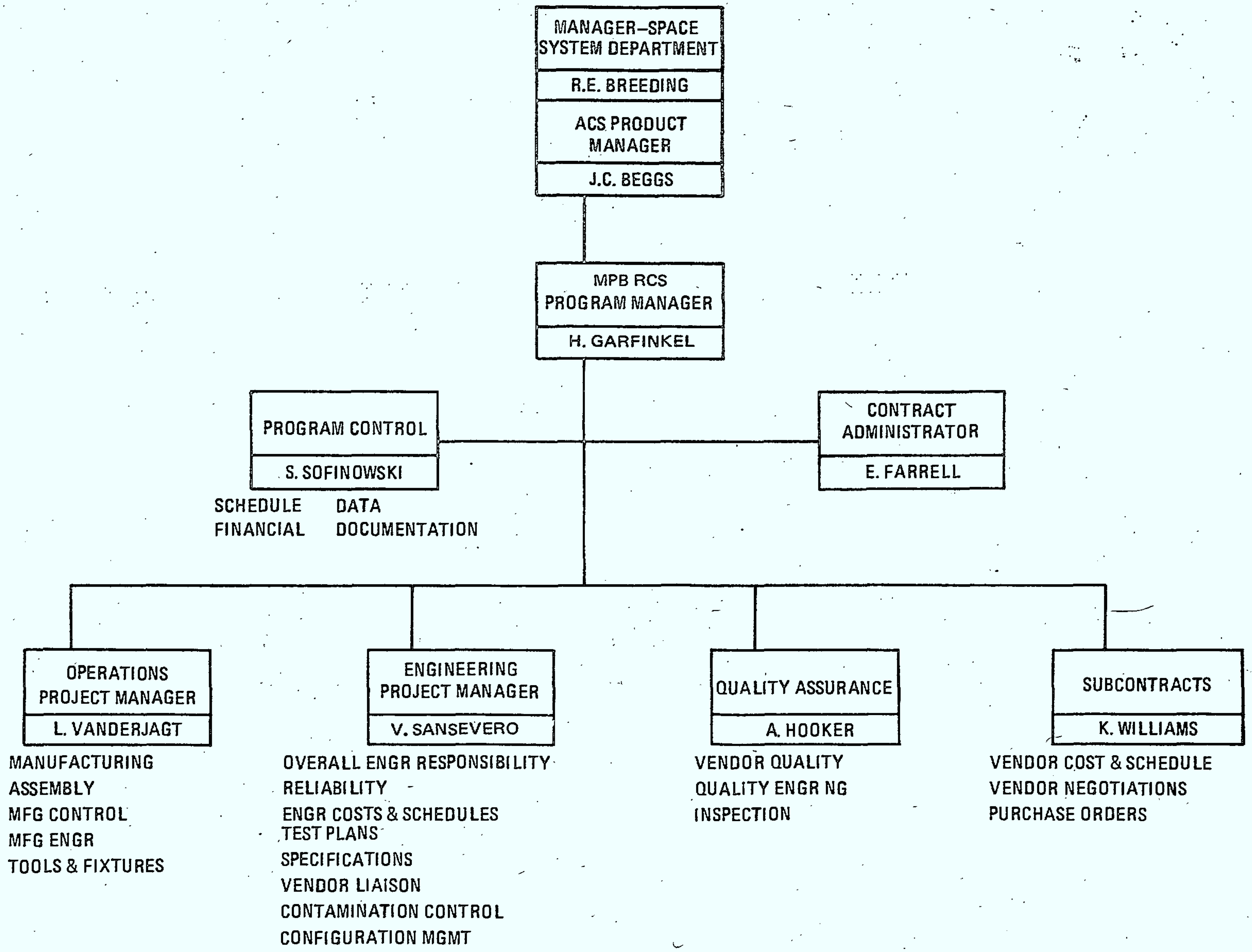


FIGURE 3. 1-4. ORGANIZATION CHART

MULTIPURPOSE BUS REACTION CONTROL SUBSYSTEM PROGRAM SCHEDULE

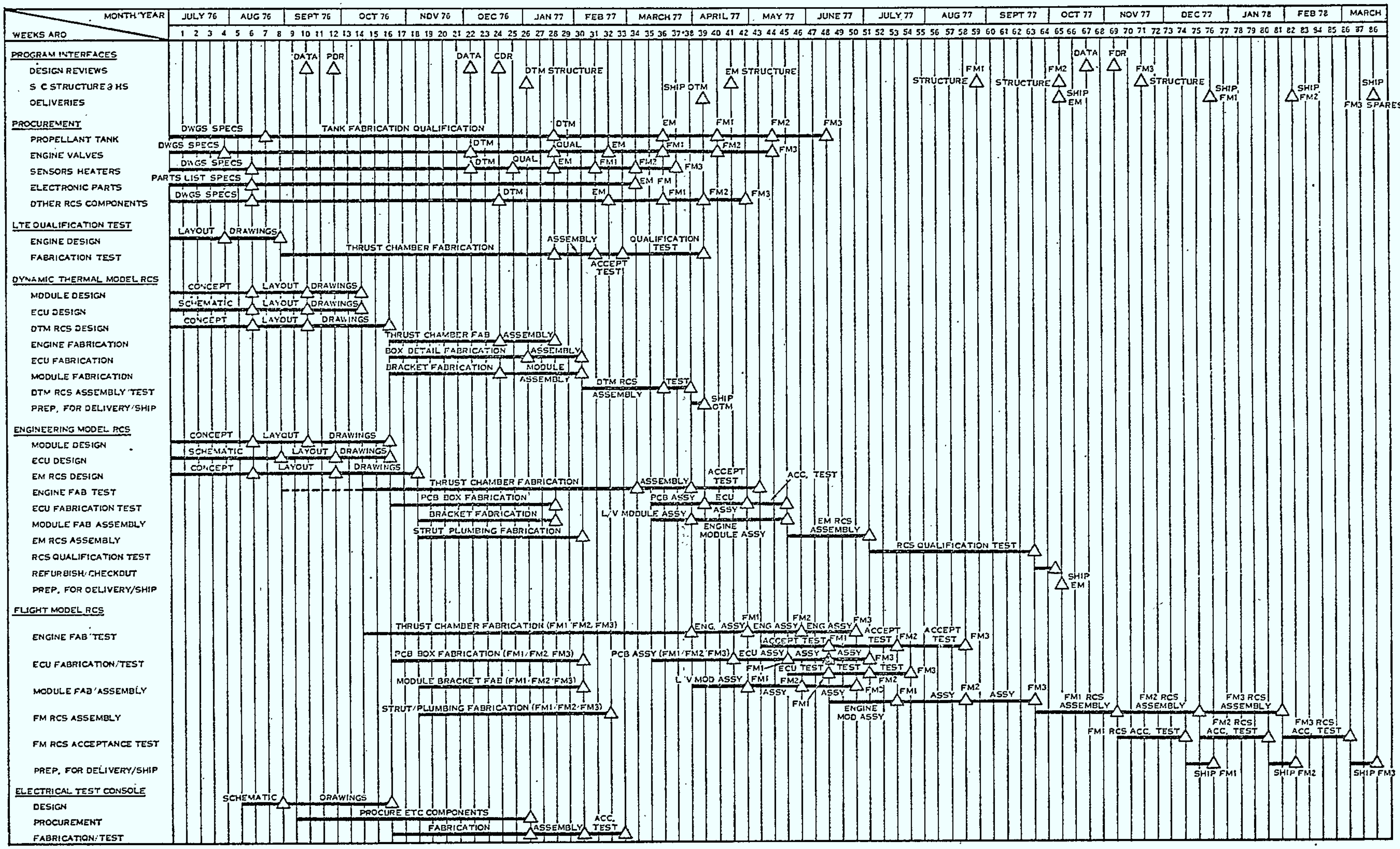


FIGURE 3.2-1

3.2 continued

including latching valve and latching valve module, fill and drain/vent valves, filter, pressure transducer, heaters, and heat shields and various printed circuit board assemblies for the Electrical Control Unit (ECU). Relatively minor modifications are required for the remainder of system components. Such modifications are described in Section 2.0 of this proposal.

Of particular importance to program cost is the schedule approach utilized for the manufacture and assembly of Engineering Model (EM) and Flight Model (FM) subsystems. It should be noted that fabrication of EM and FM RCS details are accomplished concurrently. Assembly of engines, modules and subsystems follow in a sequential pattern as does acceptance testing of FM subsystems. This schedule permits maximized utilization of learning processes and skilled personnel.

As indicated by the schedule, design reviews are scheduled for Week 12 - PDR, Week 24 - CDR, and Week 69 - FDR. The design, including drawings, of the low thrust engine will be presented at the PDR. Additionally, RCS and ECU concept drawings and schematics and released component procurement specifications will be reviewed. All required subsystem design, drawings and analyses will be presented at the CDR. During the FDR, the results of engine, tank and RCS qualification testing will be reviewed.

In order to maintain the proposed subsystem delivery schedule, spacecraft structures are required to be furnished on Weeks 26, 41, 59, 65 and 71.

3.3 Deliverable Items Description

Hardware and documentation deliveries will be in accordance with the RCS statement of work. Of particular interest is the approach Hamilton Standard has taken with regard to hardware for the Dynamic Thermal Model subsystem and Electrical Servicing Cart.

3.3.1 Dynamic Thermal Model RCS - In general, the Dynamic Thermal Model (DTM) RCS will include all provisions required to permit the conduct of a meaningful spacecraft dynamic and thermal test program.

The DTM will simulate the mass, geometry, center of mass and vibrational, structural and thermal characteristics of the final RCS design as nearly as possible. To accomplish this degree of simulation, actual RCS components will be used, as in the case of low thrust (LTE) and high thrust (HTE) engine thrust chamber valves, latching valves, system filters, fill and drain valves, and heaters and temperature sensors. The engines themselves, tankage, ECU and

3.3.1 continued

pressure transducers will simulate flight hardware, except for slight changes which have been evaluated as not affecting the intent of the spacecraft dynamic and thermal test programs. These changes have been made primarily for economic reasons and include omission of thrust chamber injection tubes, LTE catalyst, modified tank welding and fabrication processes, and final ECU contents. The ECU will include the required terminal boards and wiring to permit actuation of thrust chamber and latching valves using the ECU interface connectors.

Subsystem pressurant and propellant lines will be fabricated using processes planned for the EM and FM subsystems. By employing such processes the DTM will not only reflect the final design but will provide the opportunity to develop critical tube bending and welding processes. Similarly, electrical wiring routing will be photographed and subsequently used for the final design.

Bracketry and mounting provisions for the DTM will be based on the designs generated for FM subsystems. The DTM will be capable of being pressurized to 200 psia and storing the flight mass of water and the flight volume of alcohol. Additionally, the subsystem will be capable of isolating these referee fluids upstream of the latching valves and, upon valve actuation permit flow down to the HTE and/or LTE thrust chamber valves.

3.3.2 Electrical Servicing Cart - The Electrical Servicing Cart (ESC) for the MPB RCS will be designed and fabricated by UACL as in the CTS RCS program. See Figure 3.3-1. The MPB ESC will incorporate various improvements which have been built into the Electrical Test Console for the IUE HAPS as shown in Figure 3.3-2. These improvements include:

- Desk top work surface
- Digital readout in engineering units of temperature sensors and pressure transducers
- Programmable pulsing inputs of thruster valve on time and off time from 1 to 9999 msec and pulse train length from 1 to 9999 pulses

3.4 Test Program

The MPB RCS test program will be in accordance with the requirements of specification SPAR-SG.350. Of particular significance are the tests which will be conducted to demonstrate north-south station keeping and offset operation capabilities of the low thrust engine (LTE) and the mission simulation test of the Engineering Model RCS.

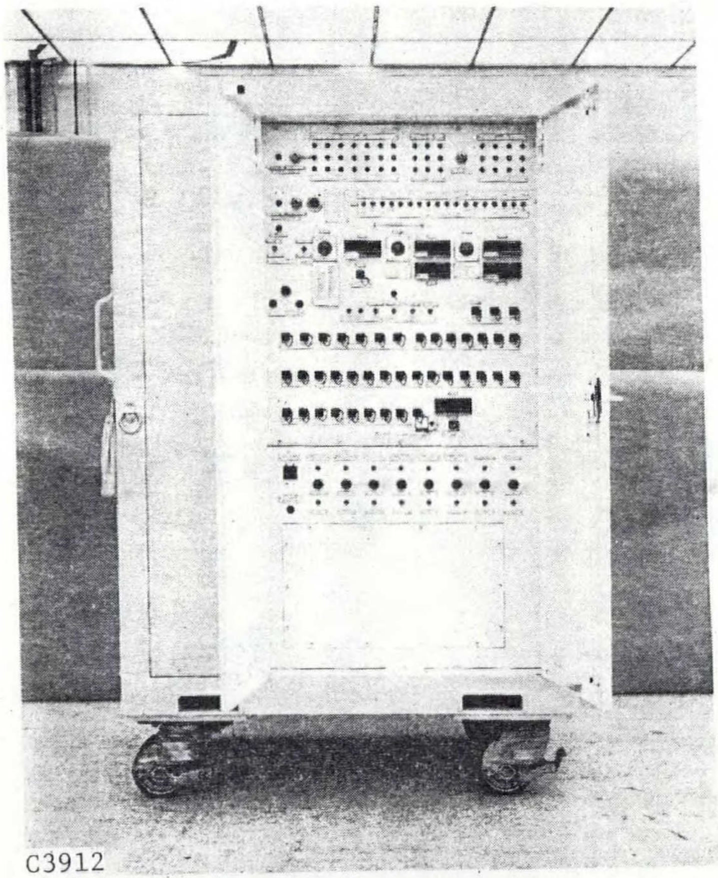


FIGURE 3.3-1. ELECTRICAL SERVICE CART - CTS RCS

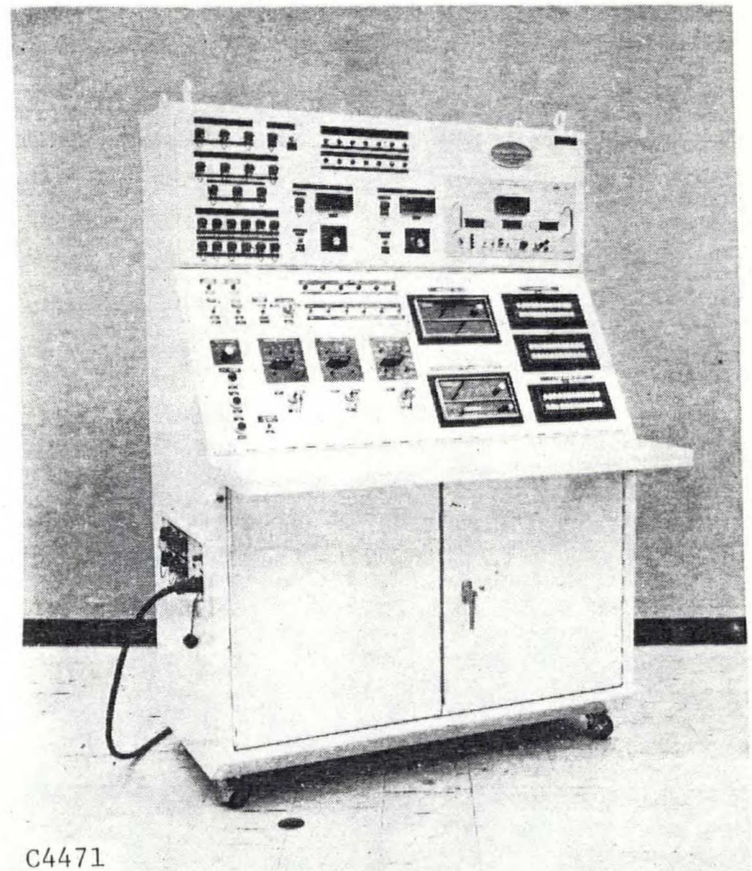


FIGURE 3.3-2. ELECTRICAL TEST CONSOLE - IUE HAPS

3.4.1 North-South Station Keeping - One (1) LTE will be subjected to the following test sequence:

A) Vibration Test

B) Vibration - qualification level

C) Gas Flow/Leakage/Functional and Electrical

D) Firing Base Point

E) Performance Map - six (6) duty cycles and three (3) inlet pressures

F) Life Test - forty (40) hours steady state with firing basepoint every five (5) hours

G) Gas Flow/Leakage/Functional and Electrical

H) Examination of Product

3.4.2 Offset Operation - One (1) LTE will be subjected to the following test sequence:

A) Acceptance Test

B) Vibration - qualification level

C) Gas Flow/Leakage/Functional and Electrical

D) Firing Base Point

E) Performance Map - four (4) duty cycles, three (3) temperatures and three (3) inlet pressures

F) Life Test - 800,000 pulses at duty cycle of .007 sec on/.789 sec off with firing basepoint every 100,000 pulses

G) Gas Flow/Leakage/Functional and Electrical

H) Examination of Product

3.4.3 MPB RCS Mission Simulation - The Engineering Model RCS will be subjected to the test series defined by Table 3.4-I and includes the basepoint, mission simulation and basepoint firing sequence required by specification SPAR-SG.350 paragraph 4.2.2 (o), (p), and (q).

TABLE 3.4-I

MPB ENGINEERING MODEL RCS
FIRING TEST SEQUENCE

SEQUENCE NUMBER		ENGINE NUMBERS	MANEUVER NAME	MANEUVER FREQUENCY	TOTAL RUNS	DUTY CYCLE	PULSES OR HOURS
BASE POINT	1	3, 8, 9, 10, 13, 14, 15, 16	Base Point	Hi/Lo Press.	2	.007/100 60 sec SS	10 Pulses 60 sec SS
	2	1, 2, 4, 5, 6, 7, 11, 12	Base Point	Hi/Lo Press.	2	.007/100 60 sec SS	10 Pulses 60 sec SS
	3	17	Base Point	Hi Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
	4	18	Base Point	Hi Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
	5	17	Base Point	Lo Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
	6	18	Base Point	Lo Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
BLOWDOWN MISSION SIMULATION	7	17	Pre-Apogee	Once	1	.135/.875	1231
	8	17	Post-Apogee	Once	1	.135/.875	606
	9	1, 7	Despin	Once	1	SS	.165 hrs.
	10	5	Wheel Spin-Up	Once	1	.007/.816	9600
	11	1, 2, 4, 5, 6, 7, 11, 12	Limit Cycle	Once	1	.007/1.993	3600
	12	13, 14	Capture	Once	1	.007/2.297	6250
	13	9, 10	In-Plane	Once	1	SS	1.285 hrs.
	14	1, 6	Out-of-Plane	Once	1	SS	1.455 hrs.
	15*	13, 14	Roll-Yaw Control	Continuous	1	.007/.789	628,750
	16*	5, 10	Pitch Dumping	1/60 min.	139	.007/.493	134 Pulses
	17*	4, 9	East-West Sta. Keep.	1/60 min.	139	SS	65 seconds
	18*	1, 6	North-South Sta. Keep.	1/80 min.	104	SS	1090 seconds
BASE POINT	19	3, 8, 9, 10, 13, 14, 15, 16	Base Point	Hi/Lo Press.	2	.007/100 60 sec SS	10 Pulses 60 sec SS
	20	1, 2, 4, 5, 6, 7, 11, 12	Base Point	Hi/Lo Press.	2	.007/100 60 sec SS	10 Pulses 60 sec SS
	21	17	Base Point	Hi Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
	22	18	Base Point	Hi Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
	23	17	Base Point	Lo Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS
	24	18	Base Point	LO Press.	1	.135/.875 60 sec SS	100 Pulses 60 sec SS

* Note: These runs are to be run concurrently.

APPENDIX A

PLUME ANALYSIS

FOR

HYDRAZINE MONOPROPELLANT ENGINES

Prepared By: V. J. Sansevero, Jr.

28 April 1975
Date

PLUME ANALYSIS
FOR
HYDRAZINE MONOPROPELLANT ENGINES

Plume impingement forces and heating rates are functions of the plume density profile. The following analysis outlines the methods to be used to determine plume density profiles and the forces and heating rates they produce.

The plume density profile is a function of axial (X) and radial (Y) distances from the nozzle exit, the nozzle throat diameter (d), the plume density parameter (β) and the plume spreading parameter (δ). The locus for points defining constant density ratio (ρ_o/ρ) can be computed as follows (Reference 1):

1) $$X_A = d\left(\beta\, \rho_o/\rho\right)^{\frac{1}{2}}\left[\cos\theta_A\right] e^{\left\{-\frac{\delta^2}{2}\left[1-\cos\theta_A\right]^2\right\}}$$

where:

X_A is axial distance from nozzle exit measured in (inches)

d is nozzle throat diameter measured in (inches)

β is plume density profile parameter defined below

θ_A is arc tan of (Y_A/X_A)

Y_A is radial distance from nozzle center line measured in (inches)

ρ_o/ρ is the plume density ratio and represents the gas products density in the engine chamber prior to expansion divided by the density at the given point (X, Y) in the plume.

δ is the plume spreading parameter defined below.

The plume density profile parameter can be computed as follows (Reference 1):

2) $$\beta = \frac{0.2}{1-\cos\left[\theta_\alpha - \theta_{Me} + \theta_e\right]}$$

where:

θ_∞ is the Prandl-Meyer turning angle for MACH ∞

θ_{Me} is the Prandl-Meyer turning angle at the nozzle exit MACH number

θ_e is the nozzle exit half angle

To determine the plume spreading parameter (δ) it is first necessary to compute the velocity ratio (V_e/V_{max}) and the thrust coefficient ratio ($C_F/C_{F_{max.}}$) (Reference 2):

$$3) \quad \frac{V_e}{V_{MAX}} = M_e \left[\frac{\gamma-1}{2} \left(1 + \frac{\gamma-1}{2} M_e^2 \right)^{-1} \right]^{1/2}$$

where:

$\frac{V_e}{V_{max}}$ is the exit velocity of the nozzle divided by the maximum velocity computed assuming complete conversion of the gas thermal energy to kenetic energy; i.e., gas static temperature is (0^oR). Note in equation 1 it has been assumed that at any point in the plume the direction of the velocity vector is along a line joining the point to the center of the nozzle exit plane and its magnitude is V_{max}.

$$4) \quad \frac{C_F}{C_{F_{MAX}}} = \frac{1}{2} \left(1 + \cos \theta_e \right) \left(\frac{V_e}{V_{MAX}} \right) \left[1 + \left(\gamma M_e^2 \right)^{-1} \right]$$

where:

$\frac{C_F}{C_{F_{max}}}$ is the nozzle thrust coefficient divided by the maximum nozzle thrust coefficient assuming all gas leaves the nozzle parallel to the nozzle axis and at velocity equal to V_{max}.

The value of ($C_F/C_{F_{max}}$) can be used directly to compute the plume spreading parameter (δ) as follows (Reference 2).

$$5) \quad \delta = \left[\sqrt{\pi} \left(1 - \frac{C_F}{C_{F_{MAX}}} \right) \right]^{-1}$$

SAMPLE PLUME PROFILE PROBLEM

Given CTS High Thrust and Low Thrust Engines compute plume profile.

PARAMETER	HTE	LTE	SOURCE
A/A^*	60	55	Given
θ_e	8.5°	15°	Given
γ	1.3	1.3	Given
M_e	5.3	5.2	From gas tables
θ_∞	159.2°	159.2°	$\theta_\infty = 90^\circ \left[\sqrt{\frac{\gamma+1}{\gamma-1}} - 1\right]$
θ_{Me}	92.2°	91.2°	From gas tables
β	.2668	.2277	From Equation 2
V_e/V_{max}	.8990	.8956	From Equation 3
$C_F/C_{F_{max}}$	.9185	.9053	From Equation 4
δ	6.9226	5.9576	From Equation 5
d	.180"	.0305"	Given
X @ $\rho_o/\rho = 10^5$ $\theta_A = 0^\circ$ $\theta_A = 30^\circ$	 29.4" 16.6"	 4.6" 2.9"	From Equation 1

Figures 1 and 2 illustrate plume profiles for the CTS High Thrust and Low Thrust Engines computed as noted above.

REFERENCES: PLUME PROFILE

1. Silbulkin, M. and W. H. Gallaher, "Far Field Approximation for Nozzle Exhausting into a Vacuum", AIAA Journal, Volume 1, No. 6, June 1963, PP 1452 - 1453.

2. Brook, John W., "Far Field Approximation for Nozzle Exhausting into a Vacuum", J. Spacecraft and Rockets, Volume 6, No. 5, May 1969, PP 626 - 628.

PLUME FORCE ANALYSIS

If it is assumed that the plume impinges on some arbitrarily oriented flat surface and that a portion (η_1) of the molecules are trapped and move off with negligible velocity and that the remaining portion (η_2) of the molecules are reflected with perfectly elastic collisions, the plume forces on the surface can be computed as follows using the density profiles previously developed.

Assume a right-handed rectangular coordinate system with origin at the C.G. of the spacecraft is given by (X, Y, Z). Assume the surface, in the spacecraft coordinate system, is defined by the equation:

1) $$\ell X + m Y + n Z = P$$

Where (λ, m, n) are direction cosines of a line normal to the surface and (P) is the perpendicular distance from the surface to the spacecraft origin.

The differential normal force on a differential element of surface area, assuming that the molecules which are stopped (η_1) (i.e., leave with negligible velocity) impart all their normal momentum to the surface element and that the molecules which are reflected (η_2) impart twice their normal momentum to the surface, can be defined by the equation:

2) $$dF_N = \eta_1 V_N \, d\dot{M} + 2 \eta_2 V_N \, d\dot{M}$$

where:

dF_N	is the differential normal force in pounds on a differential area of surface
V_N	is the velocity of the molecules normal to the surface element
$d\dot{M}$	is differential mass flow rate impinging on the surface element

The differential tangential force on a differential element of surface area, assuming that the molecules which are stopped (η_1) (i.e., leave with negligible velocity) impart all their tangential momentum to the surface element and that the molecules which are reflected (η_2) impart none of their tangential momentum, can be defined by the equation:

3) $$dF_T = \eta_1 V_T \, d\dot{M}$$

where:

V_T is the velocity of the molecules tangential to the surface element

The differential mass flow rate can be computed as follows:

$$4) \quad d\dot{M} = \frac{\rho}{g} V_N \, dA$$

where:

dA Differential surface area element

ρ is the free stream density of the gas just prior to impact with the surface element in lb_m/in^3

g is the gravity constant 386 $lb_m \; in/lb_f \; sec^2$

If the location of the area element is at point (X, Y, Z) in the spacecraft, coordinate system and if the engine nozzle exit plane center is located at point (X_o, Y_o, Z_o) in the spacecraft coordinate system then the direction cosines of the gas velocity vector can be computed by the following equations:

$$5) \quad l_v = \frac{X - X_o}{\sqrt{(X - X_o)^2 + (Y - Y_o)^2 + (Z - Z_o)^2}}$$

$$6) \quad m_v = \frac{Y - Y_o}{\sqrt{(X - X_o)^2 + (Y - Y_o)^2 + (Z - Z_o)^2}}$$

$$7) \quad n_v = \frac{Z - Z_o}{\sqrt{(X - X_o)^2 + (Y - Y_o)^2 + (Z - Z_o)^2}}$$

$$8) \quad \vec{V} = l_v \vec{i} + m_v \vec{j} + n_v \vec{k}$$

where:

$\vec{V}$ is a unit velocity vector with components (ℓ_V, m_V, n_V)

9) $\vec{N} = \ell\,\vec{i} + m\,\vec{j} + n\,\vec{k}$

where:

$\vec{N}$ is a unit normal vector to the surface element

10) $\vec{V} \circ \vec{N} = |\vec{V}|\,|\vec{N}| \cos \theta_N$

where:

θ_N is the angle between the unit velocity vector and the unit normal vector

11) $\therefore \cos \theta_N = \ell \ell_V + m m_V + n n_V$

If the magnitude of the velocity of the gas is a constant V_{max} then:

12) $V_N = V_{max} \cos \theta_N$

13) $V_T = V_{max} \sin \theta_N$

Substituting 4, 15, 16 in 2 and 3

14) $dF_N = (\eta_1 + 2\eta_2) \dfrac{\rho V_{MAX}^2}{g} (\cos^2 \theta_N)\, dA$

15) $dF_T = \eta_1 \dfrac{\rho V_{MAX}^2}{g} (\cos \theta_N)(\sin \theta_N)\, dA$

It can be shown that:

16) $\dfrac{\rho V_{MAX}^2}{g} = \dfrac{2 \gamma P_c}{(\gamma - 1)(\rho_0 / \rho)}$

where:

P_c is the engine chamber pressure in psia

γ is the specific heat ratio

ρ_0/ρ is the local plume density ratio

17) $$d F_N = \frac{2 \gamma P_c (\eta_1 + 2\eta) \cos^2 \theta_N}{(\gamma - 1)(\rho_o / \rho)} dA$$

18) $$d F_T = \frac{2 \gamma P_c \eta_1 (\cos \theta_N)(\sin \theta_N)}{(\gamma - 1)(\rho_o / \rho)} dA$$

The differential normal and tangential components of force given by equations 17 and 18 must be broken into orthogonal components prior to addition or integration. The following equations can be used:

19) $d F_{NX} = \ell \; d F_N$

20) $d F_{NY} = m \; d F_N$

21) $d F_{NZ} = m \; d F_N$

22) $$d F_{TX} = \frac{(\ell_v - \ell \cos \theta_N) \, d F_T}{\sin \theta_N}$$

23) $$d F_{TY} = \frac{(m_v - m \cos \theta_N) \, d F_T}{\sin \theta_N}$$

24) $$d F_{TZ} = \frac{(n_v - m \cos \theta_N) \, d F_T}{\sin \theta_N}$$

PLUME DENSITY PROFILE COORDINATE TRANSFORMATION

The objective of this analysis is to determine the plume density ratio at a point (X, Y, Z) in the spacecraft coordinate system for an engine whose nozzle exit plane center is located at point (X_o, Y_o, Z_o) and whose nozzle axis has direction cosines (ℓ_A, m_A, n_A).

The cosine of the angle (θ_A) between the nozzle axis and the line joining the nozzle exit plane center and the point (X, Y, Z) can be computed as follows:

$$1)\quad \cos\theta_A = \ell_V \ell_A + m_V m_A + n_V n_A$$

where: (ℓ_V, m_V, n_V) are direction cosines of the gas velocity vector see plume force analysis equations 5, 6, and 7. (ℓ_A, m_A, n_A) are direction cosines of the nozzle axis in spacecraft coordinate system.

The distance X_A measured along the nozzle axis from the nozzle exit plane center to a plane perpendicular to the nozzle center line which includes the point (X, Y, Z) can be computed as follows:

$$2)\quad X_A = \cos\theta_A \sqrt{(X - X_o)^2 + (Y - X_o)^2 + (Z - Z_o)^2}$$

Substituting equations 1 and 2 in the plume analysis Equation 1 the following plume density ratio equation results.

$$3)\quad \boxed{\frac{\rho_o}{\rho} = \frac{(X - X_o)^2 + (Y - Y_o)^2 + (Z - Z_o)^2}{\beta\, d^2\, e^{-\delta(1 - \ell_V \ell_A - m_V m_A - n_V n_A)}}}$$

PLUME HEATING RATES

Figure 3 illustrates plume heating rates as a function of plume density ratio for impingement on a 3 foot diameter cylinder.

Hamilton Standard

DIVISION OF UNITED AIRCRAFT CORPORATION

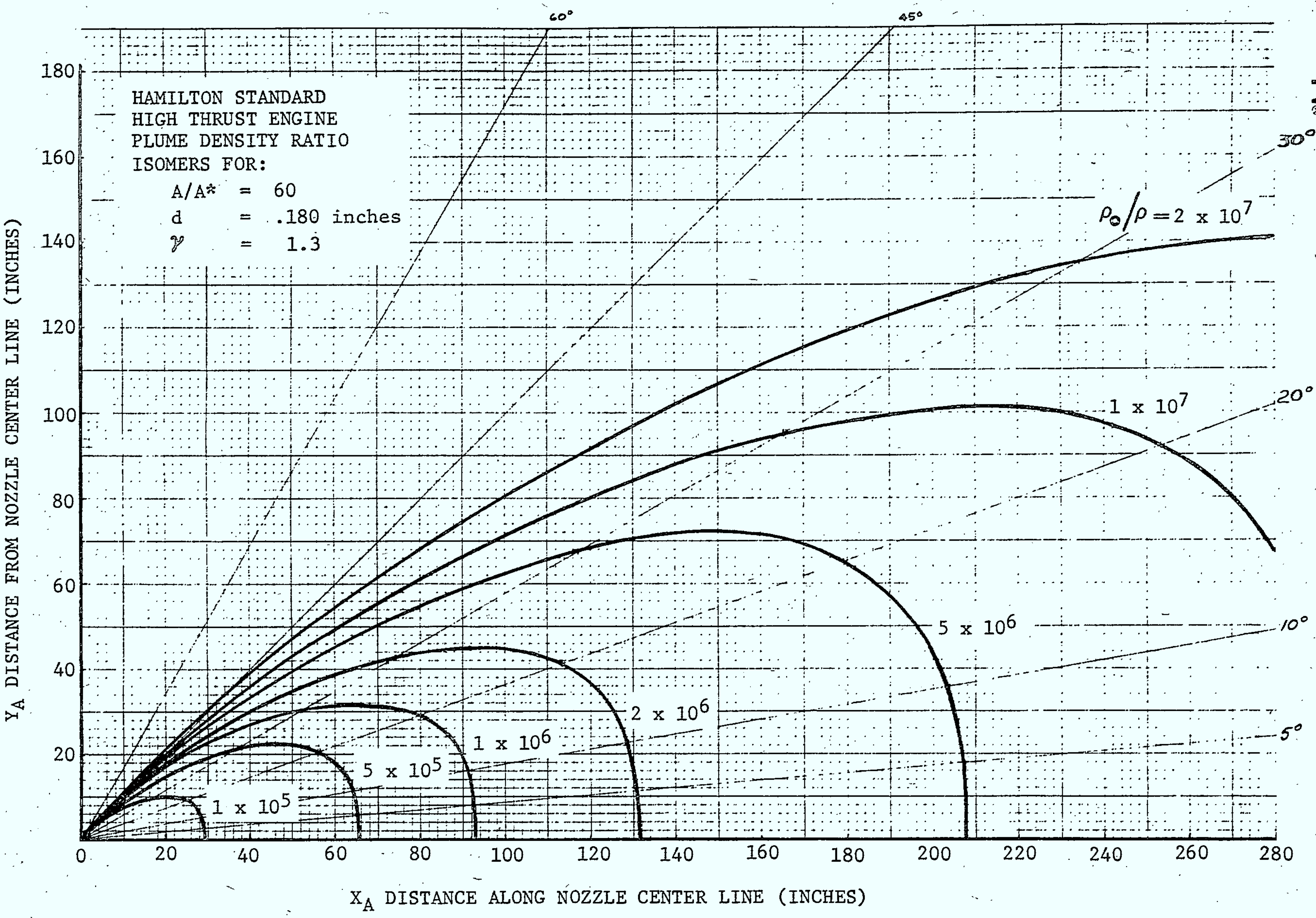


FIGURE - 1A

HSPC 75R15

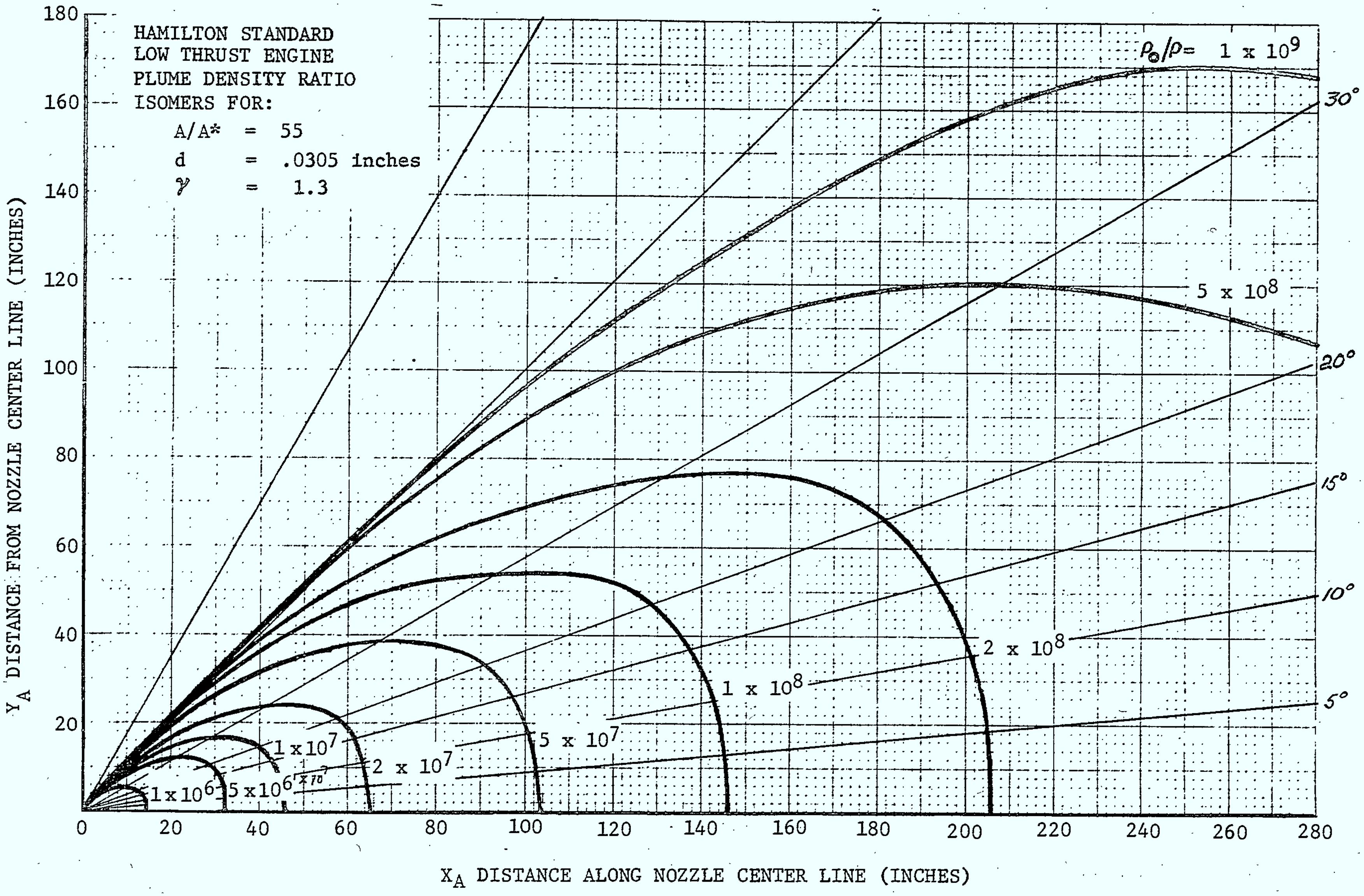

HAMILTON STANDARD
LOW THRUST ENGINE
PLUME DENSITY RATIO
ISOMERS FOR:
A/A* = 55
d = .0305 inches
γ = 1.3
ρ_o/ρ = 1 x 10^9
5 x 10^8
2 x 10^8
1 x 10^8
5 x 10^7
2 x 10^7
1 x 10^7
5 x 10^6
1 x 10^6
30°
20°
15°
10°
5°
Y_A DISTANCE FROM NOZZLE CENTER LINE (INCHES)
X_A DISTANCE ALONG NOZZLE CENTER LINE (INCHES)
0
20
40
60
80
100
120
140
160
180
200
220
240
260
280

FIGURE - 2A

HTE IMPINGEMENT

HEAT FLUX

CYLINDER DIAMETER = 3 FT.

$\gamma = 1.3$

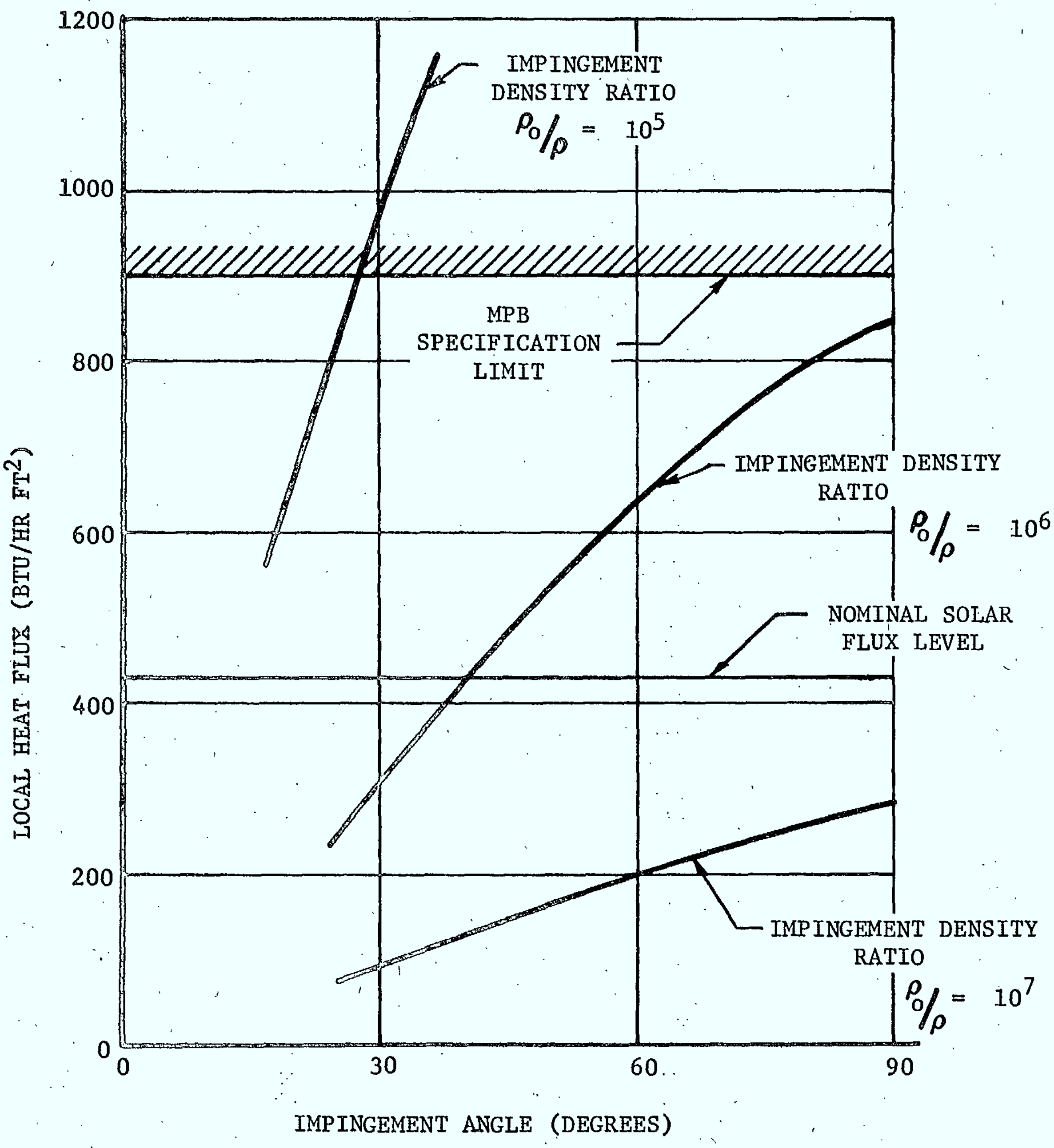


FIGURE - 3A

15 May, 1975

Hamilton Standard Response to Questions
Posed by Spar Subsequent to Receipt of Their Proposal

ATTACHMENT

Part II Technical Questions

1. Question: Will the 27.5 volt current telemetry be a single valued function?

   Answer: Yes.

2. Question: Does the test connector check the LTE driver fuses?

   Answer: No, it checks only the main power fuses.

3. Question: Are the CMOS logic elements used in the ECU essential? They may be too radiation sensitive.

   Answer: TTL logic could be used in place of the CMOS elements if the radiation sensitivity is too high with no cost or weight impact.

4. Question: Do both reference voltages generated within the ECU feed all points?

   Answer: No, each reference voltage feeds one-half of the system.

5. Question: Could the value of the two reference voltages be made a telemetry parameter?

   Answer: Yes, this could be done with negligible impact and could be very useful.

6. Question: What life testing has been performed on BSE and IUE programs?

   Answer: A Design Verification Test (DVT) was performed for the BSE program on one low thrust engine and included the firing tests presented in Table I. The following summarizes the life accumulated on this engine:

Number of firings	- 74
Number of pulses	- 128,140
Total impulse	- 6,535 lb_f-sec
Total "on-time"	- 6.8 hours

   No life tests have been performed for IUE.

7. Question: What is the HTE heater element material on BSE and IUE?

   Answer: The resistance wire element material for the HTE thrust chamber heater for both BSE and IUE is Tophet A. Table 2.5-I of HSPC 75R15 should be corrected accordingly.

8. Question: Provide backup analyses for offset engine performance.

   Answer: Attached are pertinent analyses performed, i.e., LTE chamber temperature, I_{bit} and capillary tube sizing.

9. Question: Provide backup analyses for tank strut sizing.

   Answer: Attached herewith are tank strut analyses and a weight breakdown of the RCS structure.

10. Question: Do the "simplifying assumptions" noted on page 43 of HSPC 75R15 have a conservative effect on the reliability prediction?

    Answer: The assumption concerning lumping engine pulses is slightly conservative, whereas the heater assumption is slightly optimistic. Neither assumption should significantly alter the results of the reliability prediction analyses.

11. Question: Why is the reliability prediction for the MPB RCS for an eight-year mission the same as for the CTS RCS which is based on a two-year mission?

    Answer: Although similar failure rates have been used for MPB and CTS in most cases, there are two significant differences, i.e., LTE heater redundancy and more realistic connector failure rates (reference HSPC 75R15, page 57).

TABLE I

FIRING TESTS AND INITIAL CONDITIONS

(S/N 00007 LTE)

	Run No.	Duty Cycle On/Off (secs)	Pulse Train Length	Nominal Inlet Pressure (psia)	Nominal Chamber Throat Temp. (°F)
Production Acceptance Test Performance	2836	.125/60	30	340	Note (1)
		60/-	1	340	Hot
	2837	.125/60	30	130	Note (1)
		60/-	1	130	Hot
DVT Reference Performance	2891	.125/60	30	340	340
		60/-	1	340	Hot
	2892	.125/60	30	130	340
		60/-	1	130	Hot
	2893	.007/100	10	300	340
DVT Duty Cycle Characterization	2894	1800/-	1	340	340
	2895	8/-	1	340	340
	2896	2/6	20	340	340
	2897	8/300	10	340	340
	2898	6/1800	10	340	340
	2899	.125/60	10	340	340
	2900	.125/.875	30	340	340
	2901	.007/300	10	340	340
	2902	.007/1800	10	340	340
	2903	.007/300	10	340	285
	2904	.007/1800	10	340	285
	2905	.007/300	10	340	400
	2906	.007/1800	10	340	400
	2907	1800/-	1	270	340
	2908	8/-	1	270	340
	2909	2/6	20	270	340
	2910	8/300	10	270	340
	2911	6/1800	10	270	340
	2913	.125/60	10	270	340
	2914	.125/.875	30	270	340
	2915	.007/300	10	270	340
	2916	.007/1800	10	270	340
	2917	.007/300	10	270	285
	2918	.007/1800	10	270	285
	2919	.007/300	10	270	400
	2920	.007/1800	10	270	400
	2921	1800/-	1	200	340
	2922	8/1	1	200	340
	2923	2/6	20	200	340
	2924	8/300	10	200	340
	2925	6/1800	10	200	340
	2926	.125/60	10	200	340
	2927	.125/.875	30	200	340
	2928	.007/300	10	200	340
	2929	.007/1800	10	200	340
	2930	.007/300	10	200	285
	2931	.007/1800	10	200	285
	2932	.007/300	10	200	400
	2933	.007/1800	10	200	400

TABLE I — continued

	Run No.	Duty Cycle On/Off (secs)	Pulse Train Length	Nominal Inlet Pressure (psia)	Nominal Chamber Throat Temp. (°F)
DVT Life	2935	1200/-	1	270	340
	2936	1200/-	1	270	340
	2937	.125/.250	5000	270	340
	2938	.007/.250	7500	270	340
	2939	.110/.015	30000	270	340
	2940	1200/-	1	270	340
	2941	.125/.250	5000	270	340
	2942	.007/.250	7500	270	340
	2943	.110/.015	30000	270	340
	2944	1200/-	1	270	340
	2945	.125/.250	5000	270	340
	2946	.007/.250	7500	270	340
	2947	.110/.015	30000	270	340
	2948	1200/-	1	270	340
DVT Reference Performance	2950	.125/60	30	340	340
		60/-	1	340	Hot
	2951	.125/60	30	130	340
		60/-	1	130	Hot
	2952	.007/100	10	300	340
Special (Note 2)	2953	.007/100	10	300	340
	2954	.007/100	10	300	340

Note (1): Chamber (throat) temperature for these tests obtained by setting chamber heater voltage at 14 ± 1 VDC. The 285, 340, and 400°F chamber (throat) temperatures in other tests were achieved by varying heater voltage.

Note (2): These tests were added to the basic DVT program and were conducted at varying valve voltage conditions (28.6, 27.6, and 26.8 VDC) to determine its influence on impulse bits of a 7 ms pulse.

Note (3): Runs 2835, 2890, and 2949 were 40 second steady state firings; and 2912 and 2934 were 30 second steady state firings. These tests were used to verify test facility setup and data acquisition readiness prior to formal tests.

May 13, 1975
V. Sansevero

ESTIMATE MPB LTE CHAMBER TEMPERATURE RANGE

From CTS Analysis Report ACS-AR-163 Addendum A

Worst Case Cold

Power (watts)	$T_{chamber}$
.828	227°F
.855	233°F

The CTS Analysis is based upon a minimum mount temperature of 17°F. However, MPB minimum mount temperature is 40°F, i.e., 23°F warmer. CTS spacecraft level test data indicate lower temperatures then predicated by the analysis by 20 to 30°F. It will be assumed that these two effects cancel and that direct logrithmic extrapolation of the CTS Analysis power temperature curve for the cold case using minimum MPB expected power will produce a fair estimate of the minimum MPB temperatures:

Number of Heater Elements	MPB Minimum Power (watts)	$T_{chamber}$ (Cold Case)
one	.897	240°F
two	1.793	426°F

The CTS Analysis data for the worst hot case from ACS-AR-163 Addendum A gives:

Power (watts)	$T_{chamber}$
1.052	401
1.017	394

Direct logrithmic extrapolation of this data for MPB maximum expected power yields the following maximum MPB temperatures:

Number of Heater Elements	MPB Maximum Power (watts)	$T_{chamber}$ (Hot Case)
one	1.150	420°F
two	2.300	602°F

May 13, 1975
V. Sansevero

ESTIMATE OF MPB OFFSET ENGINE MAXIMUM I_{BIT} SIZE FOR .007 SECOND PULSE WIDTH

From CTS LET Qual Report SVHS 6281 Appendix D, Figure 3D (NOTE: scale on figure must be adjusted to 10°F for five divisions starting with 200°F at the origin to correct for typing error), the I_{bit} at 602°F, 514°F and 426°F for CTS LTE can be estimated as follows:

$T_{chamber}$	CTS I_{bit} at 305 psia (lb_f sec)
426°F	3.69×10^{-3}
514°F	3.83×10^{-3}
602°F	3.97×10^{-3}

Scaling the CTS data to the maximum MPB pressure using $I_{bit} \propto (P_t)^{.50}$ we get:

$T_{chamber}$	CTS I_{bit} at 383 psia (lb_f sec)
426°F	4.14×10^{-3}
514°F	4.29×10^{-3}
602°F	4.45×10^{-3}

Increasing the I_{bit} for a worst case plus 15% of 3σ I_{bit} repeatability and allowing for a 3/1 reduction in surge flow for the MPB offset engines we get:

$T_{chamber}$	Max MPB I_{bit} for Offset Engine at 383 psia (lb_f sec)
426°F	1.59×10^{-3}
514°F	1.64×10^{-3}
602°F	1.71×10^{-3}

Reducing this I_{bit} by 15% at the nominal 514°F MPB offset chamber temperature and scaling for the average six and eight year MPB roll-yaw supply pressures of 233 and 200 psia we get:

Maneuver	P_t (psia)	Nominal MPB I_{bit} for Offset Engine at 514°F (lb_f sec)
6-year roll-yaw	233	1.12×10^{-3}
8-year roll-yaw	200	1.03×10^{-3}

May 13, 1975
V. Sansevero

SIZE OF MPB OFFSET ENGINE CAPILLARY TUBE I.D.

The objective of this analysis is to estimate the size of the MPB offset engine capillary tube I.D. which will reduce surge flow by 3/1 ratio from that obtained using the present CTS LTE. At a given supply pressure and for a given tube length

$$\dot{W}_{surge} \propto d_t^{2.71}$$

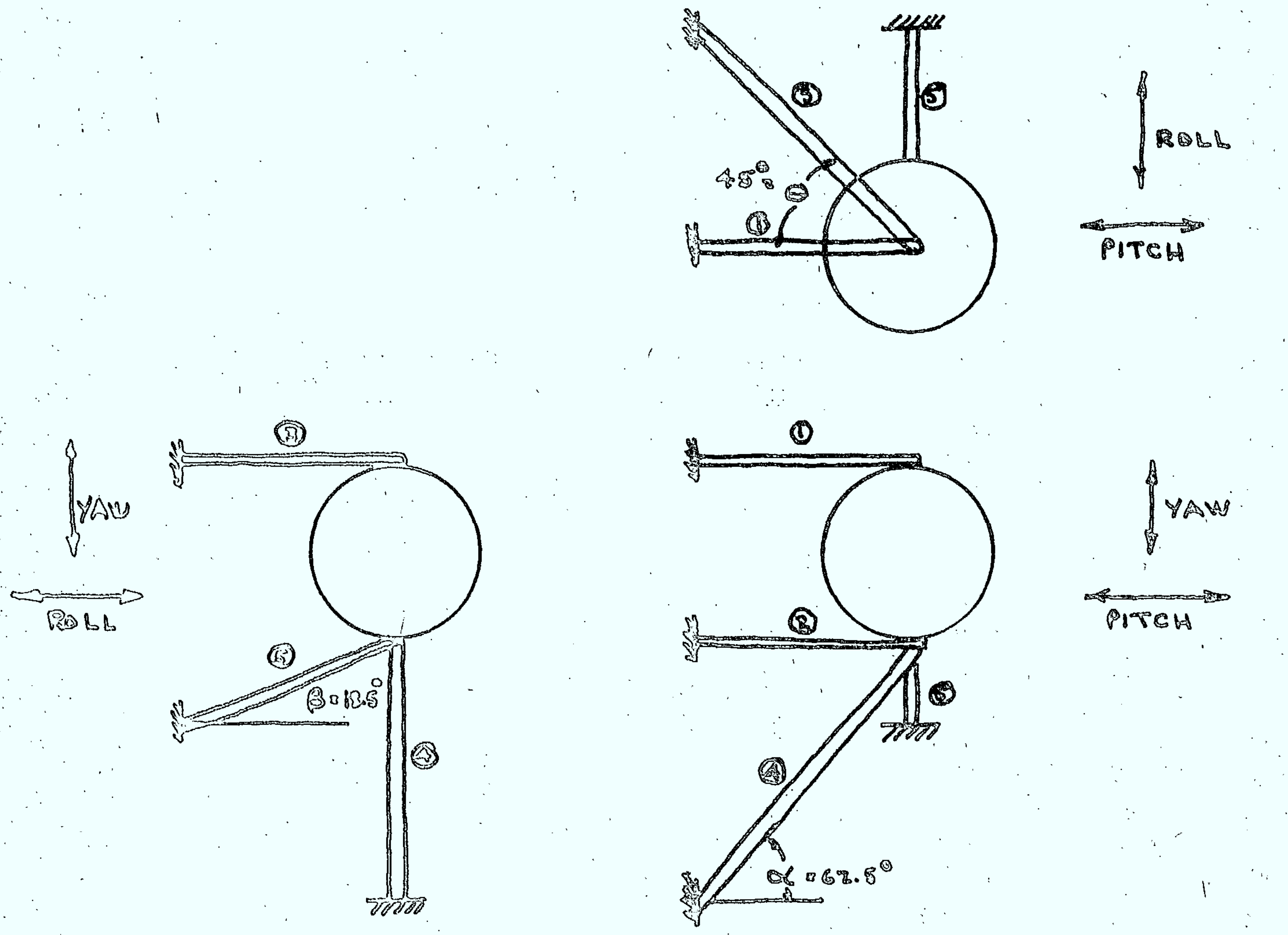
$$\therefore \quad d_{T_{MPB}} = d_{T_{CTS}} (1/3)^{.369}$$

$$d_{T_{MPB}} = .007 \text{ inches}$$

$$\therefore \quad \text{Use } .0065 \pm .0005 \text{ inches}$$

MPB TANK STRUT SIZING

SKETCH -



ASSUMPTIONS -

1. Strut Material = 6061-T6 Aluminum Alloy.
2. Maximum weight of a fully loaded tank is 66.4 lb.
3. Design 'G' level is 45 g's.
4. Fatigue limit stress for 10^4 cycles = 31,000 PSI
5. Allowable stress limit = 20,000 PSI (to allow for F.S. & unknowns).
6. Assume column buckling with ~~pinned~~ FIXED ends ~~can occur~~, therefore column ~~must be~~ analyzed for PINNED END Euler buckling WILL HAVE F.S. = 4
7. Assume that column lengths and angles can be scaled from drawing.

Pitch Axis

In this axis struts ① and ② are the main supports with the other struts only taking small loads to satisfy $\Sigma F = 0$ & $\Sigma M = 0$. $L_1 = L_2 = 9.69$ in.

Therefore, $R_1 = R_2 = \frac{45(66.4)}{2} = \underline{\underline{1494 \text{ lb.}}}$

Tube cross-sectional area $= \frac{1494}{20{,}000} = \underline{0.0747 \text{ in}^2 \text{ minimum}}$

For a ½ O.D. x .065 wall tube

$A_{MIN} = 0.0794 \text{ in}^2$ (O.K.)

$L'/\rho = 62.2$

$P_{CR} = (0.0794)(19{,}600) = 1557$ lb (O.K.).

$W = 0.1066$ lb/ft (nominal)

Since all conditions have been met struts ① & ② will be $\underline{\underline{1/2 \text{ O.D.} \times .065 \text{ wall}}}$

Yaw Axis

In this axis only strut ④ is the prime load carrying member while the other members take a small amount of bending and whatever small loads required to balance the system.

The length of the strut is 17.5 in.

$$R_4 = \frac{W_E}{\sin\theta} = \frac{45(66.4)}{\sin 62.5^\circ} = \underline{\underline{3369 \text{ lb.}}}$$

Required tube cross-sectional area $= \frac{3369}{20{,}000} = \underline{0.1685 \text{ in}^2 \text{ minimum}}$

Trials

Tube Size	A min	L'/ρ	σ_{CR}	P_{CR}	W	
3/4 x .095	0.1754	74.7	16,200	2842	0.2346	Does not meet column buckling.
3/4 x .120	0.21511	77.0	15,800	3399	0.2850	Possible
1.0 x .065	0.1702	59	22,600	3846	0.2291	Lighter than above tube.

To save weight the strut will be $\underline{\underline{1.0 \text{ O.D.} \times 0.065}}$ wall tubing

Roll Axis.

In this axis struts ③ and ⑤ are the main supports with the other struts taking some bending and whatever small loads required to balance the system.

The length of strut ③ is 16 in.

The length of strut ⑤ is 16.4 in.

$$R_3 = \frac{45(66.4)}{2\cos\theta} = \frac{45(66.4)}{2\sin 45^\circ} = \underline{\underline{2113\text{ lb.}}} \qquad A_{REQD} = \frac{2113}{20{,}000} = \underline{\underline{0.1057\text{ in}^2}}$$

$$R_5 = \frac{45(66.4)}{2\cos 18.5^\circ} = \underline{\underline{1575\text{ lb.}}} \qquad A_{REQD} = \frac{1575}{20{,}000} = \underline{\underline{0.0788\text{ in}^2}}$$

Tube Size	A_{MIN}	L/ρ	σ_{CR}	P_{CR}	W	
1/2 x .065	.0794	105	9000	715	.1066	none of these meet column buckling criteria for strut ⑤
1/2 x .083	.0987	109	8300	819	.1305	
5/8 x .049	.0792	81	14,800	1172	.1064	
5/8 x .065	.1019	83	14,000	1426	.1372	
3/4 x .049	.0965	66	19,000	1833	.1295	← use for strut ⑤
3/4 x .058	.1127	65	19,200	2164	.1513	← use for strut ③

Therefore: Strut ③ will be $\underline{\underline{3/4 \text{ O.D.} \times .058 \text{ wall}}}$ tubing

Strut ⑤ will be $\underline{\underline{3/4 \text{ O.D.} \times .049 \text{ wall}}}$ tubing

5/13/75
J. WOJNAROWSKI

MPB - RCS
Structural Weight Breakdown

Tank mounting struts		= 3.05 lb. (nominal)
Strut end fittings (ON STRUTS)		= 2.25
Strut mounting brackets (ON VEHICLE)		= 2.40
Latching Valve module bracket		= 0.23
East/West REM	(2 @ 1.0 lb each)	= 2.00
North/South Offset REM	(2 @ 0.35 lb each)	= 0.70
North/South Roll REM	(2 @ 0.10 lb. each)	= 0.20
East/West Pitch REM	(4 @ 0.10 lb. each)	= 0.40
H.T.E. REM	(2 @ 0.36 lb. each)	= 0.72
		W = 11.95 lb. (nominal).

APPENDIX C

APOGEE MOTOR, THIOKOL PROPOSAL

Thiokol

THIOKOL/ELKTON DIVISION

EP315-75
MAY 22, 1975

A PROPOSAL TO
SPAR AEROSPACE PRODUCTS LTD.
TORONTO, ONTARIO M6B 3X8, CANADA

FOR THE
MULTIPURPOSE SATELLITE
APOGEE KICK MOTOR

VOLUME I
TECHNICAL PROPOSAL

A DIVISION OF THIOKOL CORPORATION

P.O. Box 241, Elkton, Maryland 21921
301/398-3000

Thiokol / ELKTON DIVISION

In Reply Refer To:
EP315-75

May 22, 1975

SPAR Aerospace Products Ltd.
825 Caledonia Road
Toronto, Ontario M6B 3X8
Canada

Attention: Mr. Ron F. Rich
Subcontracts Administrator

Subject: Budgetary Proposal for the Multipurpose Satellite Apogee Kick Motor

Reference:
1) SPAR RFP No. 5498, dated May 9, 1975
2) SPAR - SG.356 Multipurpose Bus Study Specification Requirements (Preliminary), dated May 6, 1975

Gentlemen:

Thiokol welcomes this opportunity to propose on the Apogee Kick Motor for the SPAR Multipurpose Satellite. In response to the referenced RFQ, we can offer both a low-risk design that meets all the requirements of the SPAR SG.356 specification and a program plan that will meet or exceed all the time-phasing requirements of your schedule.

The proposed STAR 30 design contains many features of other spherical rocket motors we have developed and qualified for similar applications. Two prototype STAR 30 motors have already been built: one has been successfully static tested; the second is scheduled to be environmentally tested and static-fired this year, making the proposed program a minimum-risk, low-cost approach to your AKM needs. An alternative design is presented which will save about 1-1/2 pounds in the AKM case weight, offering a potential spacecraft weight savings.

Apogee kick motors are a major Thiokol/Elkton product line, and we have developed and flown more space motors than all other manufacturers combined. We want to produce this apogee motor for SPAR Aerospace. We have the know-how, experience, personnel, and facilities to do the job. Our management is dedicated to the philosophy of excellence; and by using our demonstrated technology, materials, and components, we can assure a reliable, low-cost, lightweight motor for a successful SPAR flight program.

We are proud of the fine relationship that we have established with the cognizant Canadian personnel at the Communications Research Centre at Ottawa, Ontario, in the development of the Communication Technology Satellite Apogee Motor. We look forward to establishing a similar relationship with the SPAR personnel.

We invite you to contact spacecraft contractors such as the Communications Research Centre, Hughes, TRW, Boeing, Rockwell International, and General Electric, and U.S. Government agencies such as SAMSO and NASA-Goddard, who have procured Thiokol STAR motors for their applications, to verify the performance, quality, and reliability of our space motors. We also cordially invite you to visit our plant, see our products, and discuss your requirements with our personnel.

Mr. Oren Phillips, Manager of Space Systems, and Mr. James Pletz, Program Manager, will be pleased to respond to any requests you may have for additional information.

Very truly yours,

THIOKOL CORPORATION
ELKTON DIVISION

T. M. Davis
General Manager

MDR/cm
Enclosures: 1) EP315-75 (3 copies)
2) Drawing E28386
3) STAR 30 Brochure (3)
4) Facilities Brochure (1)

EP315-75

THIOKOL CORPORATION
ELKTON DIVISION
ELKTON, MARYLAND

A PROPOSAL FOR THE MULTIPURPOSE SATELLITE APOGEE KICK MOTOR

VOLUME I
TECHNICAL PROPOSAL

SUBMITTED TO

SPAR AEROSPACE PRODUCTS LTD.
825 CALEDONIA ROAD
TORONTO, ONTARIO M6B 3X8
CANADA

MAY 22, 1975

T. M. Davis
General Manager

Proposal Coordinator:
J. A. Pletz

1174138/012

STAR 30 ROCKET MOTOR

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1.0 INTRODUCTION AND SUMMARY

This volume presents technical data and background information on the proposed Apogee Kick Motor for the SPAR Aerospace Products Multipurpose Satellite. It has been prepared in response to SPAR RFP No. 5498 dated May 9, 1975, and the preliminary SPAR SG. 356 Multipurpose Bus Study specification requirements.

We offer a reliable Thiokol STAR 30 motor design that meets the SPAR SG. 356 requirements, combined with a program plan that will meet or better your schedule requirements. The basic motor design (Figure 1, Drawing E28386) provides a total impulse of 267,180 lbf/sec in vacuum at 20° C (68° F) which will, in turn, provide the required nominal velocity increment of 6,024 ft/sec to a spacecraft that has an initial pad lift-off weight of 1925 pounds, including the apogee motor. The STAR 30 motor (including remote S&A) for this application weighs 957.9 pounds, which is 2.1 pounds under the 960-pound SPAR SG. 356 specification limit.

We have also provided technical information on design features that permit the desired capability for a 10-percent increase in propellant weight and have presented an alternative case attachment flange design that will save approximately 1-1/2 pounds in AKM case weight as well as provide potential spacecraft weight savings.

Our program plan includes the manufacture and testing of two qualification motors plus the delivery of three flight motors, one empty inert motor, and one loaded inert motor for use as a Dynamic and Thermal Model. The qualification motor tests will be completed within 15 months; flight quality motors will be available for shipment 17 months after contract go-ahead.

In preparing our response, we have included the following requested information:

1) A weight estimate and mechanical schematic.

2) A reliability assessment.

3) Total impulse and thrust time curves of the unit proposed.

We have consulted with the Canadian consulate and with the Canadian Government Liaison Office in Philadelphia to seek potential sources of motor components and tooling. The Canadian trade index and the Department on Industry's Canadian Defense Commodity volume were also used. Request for quotations have been sent to Canadian firms; however, because of the small quantities involved and

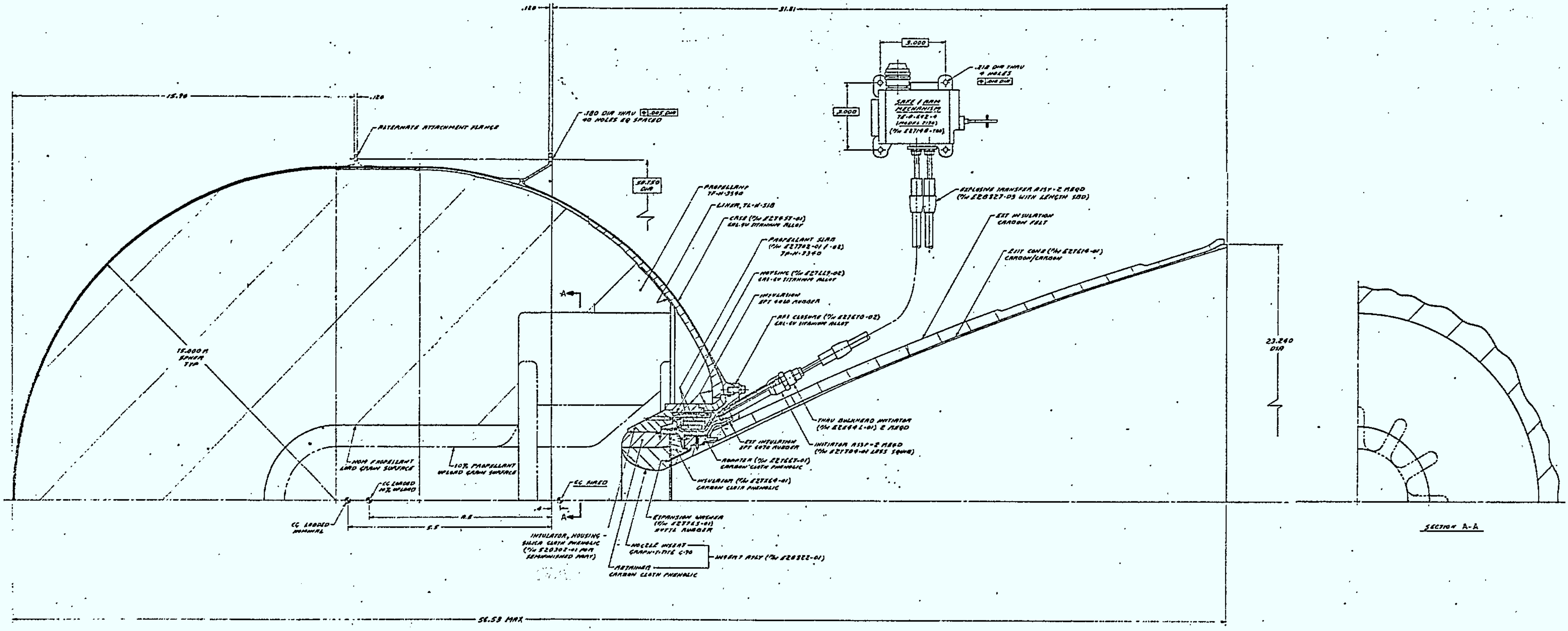

FIGURE 1. MULTIPURPOSE SATELLITE APOGEE KICK MOTOR

MOTOR ASSY
EP315-75
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E26386

the relatively few firms worldwide that have the manufacturing capability to process titanium cases and lightweight nozzle hardware, we have had few positive responses for fabrication of these items to meet drawing and specification requirements. Thiokol is constantly looking for competent fabricators who can deliver a quality product on schedule. We would welcome any suggestions that SPAR may have in this regard for the Multipurpose Satellite AKM.

Volume II of this proposal presents our budgetary fixed price quotation for the qualification and delivery of STAR 30 rocket motors for your application. As specified, our budgetary estimate and schedule have been predicated on a start date of July 1, 1976.

Space Motor Experience

Thiokol/Elkton pioneered the design and manufacture of high performance, highly reliable rocket motors for space and upper stage use starting in 1959. We have developed and produced the 34 designs listed in Table I for the 21 different customers listed. The SARV retro rocket was used in the first U.S. recovery of an orbiting vehicle and is still in use. The Mercury retro rocket provided recovery propulsion for the first manned space vehicles. Starting with the Gemini retro and Surveyor retro motors, Elkton developed and produced spherical space motor designs from 5 to 37 inches in diameter, with propellant weights ranging from 4 to 2500 pounds. Thiokol/Elkton's experience in the design and testing of motors with high mass fractions and unmatched proven experience in over 1200 space flights to date will be utilized in meeting the goals of this program.

Attention — at all levels — to assuring product reliability has developed the craftsmanship and provided the facilities Thiokol will use to build quality into the SPAR motor. Thiokol puts our customer's need for reliable space propulsion first in the order of priorities. The results of the flights listed in Table I demonstrate a reliability of 0.998 at 50% confidence. Improvements in design, processing, and system effectiveness in recent years assure higher reliability for flights of current or future designs.

The proposed motor is an off-loaded version of the current Thiokol STAR 30 motor. The STAR 30 contains many features of the other spherical rocket motors we have developed and qualified for similar AKM applications. Our goal is to establish a design that meets or exceeds all your technical requirements and to deliver motors that will perform reliably.

The variety of Elkton space motors of similar design to the proposed AKM is shown in Figure 2 and Table II. These programs illustrate the extent of the background and capability of Thiokol to assure success on the SPAR AKM program.

TABLE I

THIOKOL/ELKTON SPACE MOTOR EXPERIENCE SUMMARY

Motor	Model Number	Propellant Wt., lb	Customer	User	Contract Start	Flights
Sarv Retro	TE-M-236B	40.3	GE	USAF	1958	>325
Cygnus 20	TE-M-521	253	NASA	NASA	1958	1
Mercury Retro	TE-M-316	67	McDonnell	NASA	1959	27
Titan II Retro/Titan III Retro	TE-M-344	4.6	Martin & Hill AFB	USAF	1960	>450
Titan II Vernier	TE-M-345	83.6	Martin	USAF	1960	146
SynCom Apogee	TE-M-375	77.6	Hughes	NASA	1961	1
Surveyor Main Retro (STAR 37)	TE-M-364-1 & -5	1250 & 1300	Hughes	NASA	1961	7
Hitchhiker Apogee	TE-M-345-11	81.5	Lockheed	USAF	1962	2
Gemini Retro	TE-M-385	67.4	McDonnell	NASA	1962	40
STRYPI Upper Stage (STAR 26)	TE-M-442	516.6	Sandia	AEC	1960	6
LES Apogee	TE-M-444	88.15	MIT	USAF	1964	2
A-IMP Deceleration (STAR 13)	TE-M-458	68.3	Goddard	NASA	1964	2
Burner II (STAR 37B)	TE-M-364-2	1440	Boeing & SAMSO	USAF	1965	21
Cygnus 5 Reentry Motor	TE-M-500	3.7	Goddard	NASA	1965	4
Cygnus 15, Trailblazer Fourth Stage	TE-M-456	99.5	Goddard	NASA	1965	10
RAE Apogee (STAR 17)	TE-M-479	153	Goddard	NASA	1966	2
SESP A and 70-2 Apogee (STAR 13A)	TE-M-516	73	Boeing & General Dynamics	USAF	1966	2
Delta Third Stage (STAR 37D)	TE-M-364-3	1440	Douglas/Goddard	NASA	1966	17
Skynet I Apogee (STAR 17A)	TE-M-521	247	Philco-Ford	USAF	1967	4
SESP 68-1 Motors (STAR 13A)	TE-M-444-1	73	Boeing	USAF	1968	1
	TE-M-538	52.9				1
	TE-M-537	73				1
Drag Make-Up (STAR 6)	TE-M-541 & 542	13.2	Lockheed	USAF	1968	152
Burner IIA Apogee (STAR 26B)	TE-M-442-1	524	Boeing	USAF	1969	4
Extended Delta Third Stage (STAR 37E)	TE-M-364-4	2290	Goddard	NASA	1968	14
Skynet II Apogee (STAR 24)	TE-M-604-1	437	Philco-Ford	U. K.	1970	2
IMP H and J Apogee (STAR 17A)	TE-M-521-5	247	Goddard	NASA	1971	2
CTS Apogee (STAR 27)	TE-M-616-1	734	CRC	Canada	1972	(Production)
Altair III (STAR 20)	TE-M-640	606	LTV	NASA	1972	3
Block 5D (STAR 37S)	TE-M-364-15	1450	SAMSO	USAF	1972	(Production)
FltSatCom (STAR 37F)	TE-M-364-19	1863	TRW	Navy	1973	(Qualification)
Japanese-N (STAR 37N)	TE-M-364-14	1230	Nissan Motors	Japan	1973	(Production)
BSE Apogee (STAR 27)	TE-M-616-4	700	GE	Japan	1974	(Production)
GMS Apogee (STAR 27)	TE-M-616-5	690	Hughes	Japan	1974	(Qualification)
GPS Apogee (STAR 27)	TE-M-616-8	550	Rockwell	USAF	1975	(Qualification)
Pioneer-Venus OIM (STAR 24)	TE-M-604-2	390	Hughes	NASA	1975	(Prod/Qual)
IUE (STAR 24)	TE-M-604-4	467	NASA-GSFC	NASA	1975	(Production)
LAGEOS (STAR 24)	TE-M-604-3	440	NASA-GSFC	NASA	1975	(Production)
						>1,249

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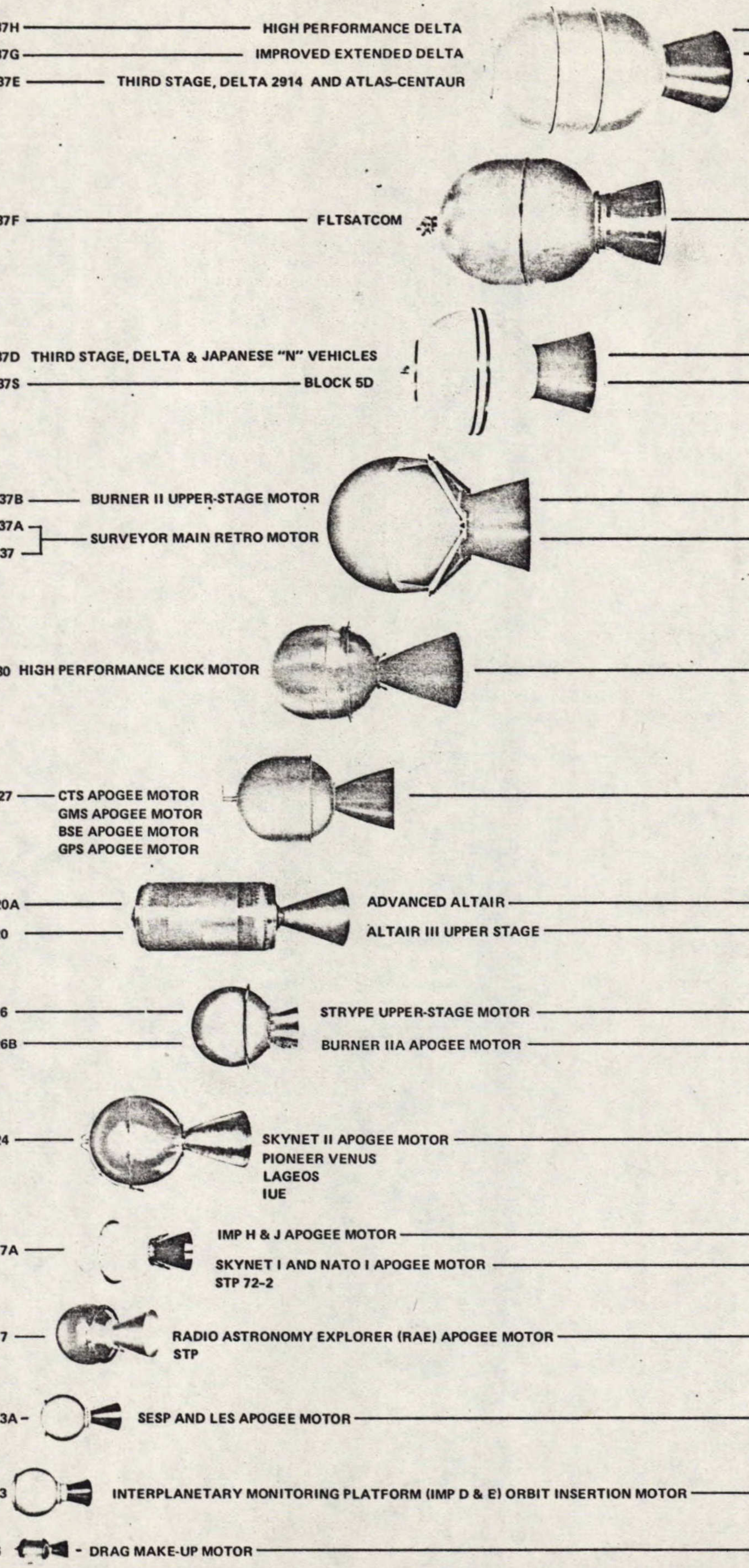

MOTOR WEIGHT, LBM	MOTOR	APPLICATION	RECORD AS OF MAY 1975		
			TESTED	DEL'D	FLOWN
2558	STAR 37H	HIGH PERFORMANCE DELTA	1	–	–
2518	STAR 37G	IMPROVED EXTENDED DELTA	4	–	–
2473	STAR 37E	THIRD STAGE, DELTA 2914 AND ATLAS-CENTAUR	12	27	14
2019	STAR 37F	FLTSATCOM	3	In Qual	–
1582	STAR 37D	THIRD STAGE, DELTA & JAPANESE "N" VEHICLES	13	21	17
1560	STAR 37S	BLOCK 5D	1	4	–
1583	STAR 37B	BURNER II UPPER-STAGE MOTOR	1	22	21
1442	STAR 37A	SURVEYOR MAIN RETRO MOTOR	50	7	7
1339	STAR 37				
1062	STAR 30	HIGH PERFORMANCE KICK MOTOR	1	–	–
799	STAR 27	CTS APOGEE MOTOR GMS APOGEE MOTOR BSE APOGEE MOTOR GPS APOGEE MOTOR	5	Qual Complete	–
694	STAR 20A	ADVANCED ALTAIR	1	–	–
664	STAR 20	ALTAIR III UPPER STAGE	5	10	3
592	STAR 26	STRYPE UPPER-STAGE MOTOR	4	10	6
576	STAR 26B	BURNER IIA APOGEE MOTOR	1	6	4
476	STAR 24	SKYNET II APOGEE MOTOR PIONEER VENUS LAGEOS IUE	6	4	2
274	STAR 17A	IMP H & J APOGEE MOTOR	3	5	2
		SKYNET I AND NATO I APOGEE MOTOR STP 72-2	5	5	5
174	STAR 17	RADIO ASTRONOMY EXPLORER (RAE) APOGEE MOTOR STP	6	6	2
84	STAR 13A	SESP AND LES APOGEE MOTOR	4	7	7
79	STAR 13	INTERPLANETARY MONITORING PLATFORM (IMP D & E) ORBIT INSERTION MOTOR	6	4	2
13.7	STAR 6	DRAG MAKE-UP MOTOR	47	203	153

FIGURE 2. STAR SPACE MOTOR EXPERIENCE SUMMARY

TABLE II

STAR MOTOR DESIGN AND PERFORMANCE SUMMARY

STAR DESIGNATION	MODEL NUMBER	NOMINAL DIAMETER, INCHES	TOTAL IMPULSE, LBF-SEC	PROPELLANT WEIGHT, LBM	PROPELLANT MASS FRACTION
STAR 6	TE-M-541	6.2	3,080	10.7	0.793
STAR 13	TE-M-458	13.5	18,800	68.3	0.869
STAR 13A	TE-M-516	13.5	21,050	73	0.869
STAR 17	TE-M-479	17.4	44,500	153.4	0.881
STAR 17A	TE-M-521	17.4	71,600	247	0.900
STAR 20	TE-M-640-1	19.7	173,000	604	0.910
STAR 24	TE-M-604	24.5	125,785	440	0.919
STAR 26	TE-M-442	26.1	140,500	516	0.872
STAR 26B	TE-M-442-1	26.1	142,700	524	0.910
STAR 27	TE-M-616	27.3	215,200	736	0.925
STAR 30	TE-M-700	30.0	299,200	1,014.5	0.944
STAR 37	TE-M-364-1	36.8	356,255	1,230	0.900
STAR 37A	TE-M-364-5	36.8	376,500	1,300	0.901
STAR 37B	TE-M-364-2	36.8	418,100	1,440	0.910
STAR 37D	TE-M-364-3	36.8	418,100	1,440	0.910
STAR 37E	TE-M-364-4	36.8	654,400	2,290	0.926
STAR 37F	TE-M-364-19	36.8	536,100	1,863	0.926
STAR 37G	TE-M-364-11	36.8	675,400	2,333	0.927
STAR 37N	TE-M-36414	36.8	356,255	1,230	0.900
STAR 37S	TE-M-364-15	36.8	420,430	1,450	0.925

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2.0 TECHNICAL DESCRIPTION

We have selected our STAR 30 (TE-M-700) rocket motor (Figure 3) for the SPAR Apogee Kick Motor application. The first development test of this motor was successfully conducted under sea level conditions at Thiokol/Elkton Division on December 19, 1974. One additional development motor is being fabricated for environmental testing, including temperature cycling and vibration. Following the completion of environmental tests, the motor will be fired under altitude conditions in September 1975.

The STAR 30 is a 30-inch-diameter, elongated spherical rocket motor with an overall length of 56.6 inches. Fully loaded, the motor provides a total impulse of 302,170 lbf-sec. The STAR 30 may be off-loaded as much as 25 percent by a combination of casting less propellant (Figure 4) and removing propellant by machining. Additional impulse may be gained by lengthening the cylindrical section of the motor case. Total impulse versus motor weight for varying configurations is shown in Figure 5.

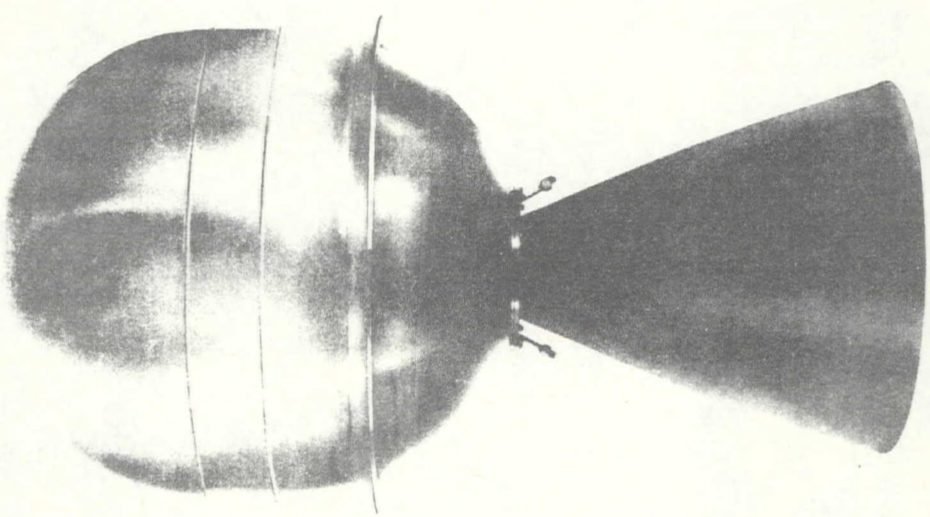

FIGURE 3. STAR 30 ROCKET MOTOR

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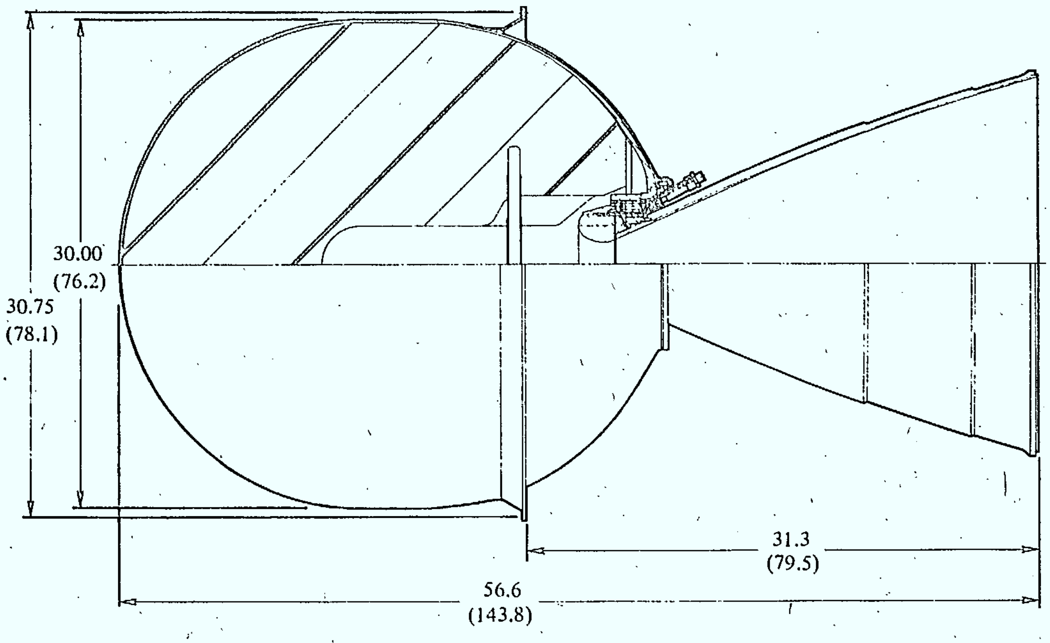

STAR 30 MOTOR ASSEMBLY

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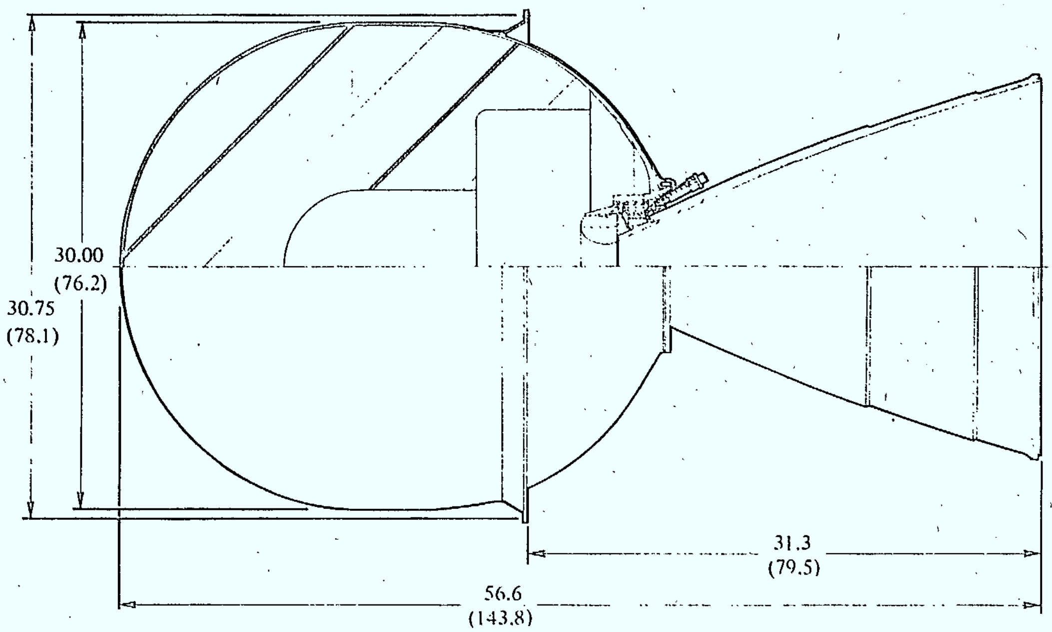

STAR 30 OFF-LOADED MOTOR ASSEMBLY

FIGURE 4. STAR 30 MOTOR CONFIGURATIONS

2.1 Motor Description

The STAR 30 rocket motor assembly consists of an elongated titanium case, high expansion carbon/carbon nozzle, internal insulation of ethylene propylene rubber, propellant grain of 89 percent total solids case-bonded hydroxyl-terminated polybutadiene propellant, an aft-end internal pyrogen ignition system, and a remote electromechanical safe-and-arm device.

The individual characteristics of the selected design concepts and materials are summarized in Table III. This chart shows how all features have been carefully chosen to maximize those interrelationships which result in low total motor weight and reliable performance. The STAR 30 motor is a logical combination of high performance components, materials, and design concepts proven in ground and flight tests during the past 15 years. The development geneology for these features is shown in Figure 6. The characteristics of the STAR 30 motor reflect the best use of the present state-of-the-art in upper stage and apogee rocket motors.

2.1.1 Motor Case. The STAR 30 uses the titanium alloy (6Al-4V) case material that has been flight-proven in hundreds of STAR motors of 12 different designs, including the STAR 17, STAR 27, and STAR 37 motors. The lightweight, highly reliable metal case retains its structural integrity after operation. It eliminates the

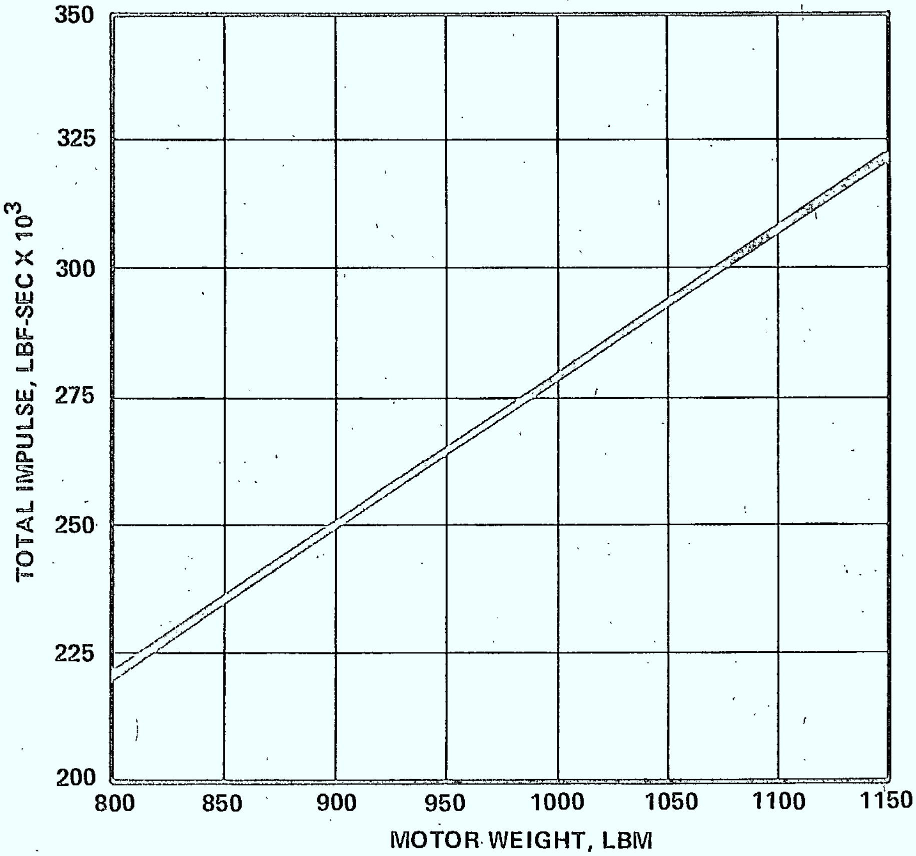

FIGURE 5. TOTAL IMPULSE VS MOTOR WEIGHT

TABLE III
CHARACTERISTICS OF SPECIFIC DESIGN CONCEPTS AND MATERIALS

HTPB Propellant	Carbon-Carbon Nozzle	Grain Design	6Al-4V Titanium	EPDM Case Insulation	Toroidal Igniter
Characteristics Leading to Low Total Motor Weight					
• High $I_{sp_{eff}}$ and excellent physical properties and space storability • Excellent bonding characteristics simplify liner system and reduce liner weight. • Low burn rate reduces throat and nozzle size and increases expansion ratio	• Structure of cone not adversely affected by high temperatures; thus thickness is controlled by structural requirements and not by insulation requirements. • Above, combined with excellent erosion resistance, means nozzle weight is practically insensitive to burn time and thus lower burn rates can be used. • Both of above mean that very high expansion ratios become advantageous. • Graphite throat of G90 means low erosion thus high average expansion ratio and greater neutrality.	• Low average surface area leads to small throat area and higher expansion ratio and lower nozzle weight. • Neutral surface area history minimizes MEOP while maintaining pressure for good combustion efficiency and thus reduces case weight. • Head-end web affords thermal protection to case and minimizes insulation weight.	• Low deflection due to pressurization means reduced propellant stresses and improved thrust vector alignment. • High allowable operating temperatures and heat sink capability lead to low insulation weight. • High allowable operating temperatures permit more effective use of low burn rates (i.e., permit longer burn times) and lead to higher expansion ratios and/or smaller nozzles.	• Low density plus high thermal effectiveness per pound.	• Sharing structure with nozzle reduces weight. • Special attachment bosses not required.
Other Characteristics					
• No volatile plasticizers that might degrade in the space environment or require special seals. • Proven processibility in production batches. • Demonstrated capability for slot machining and total impulse adjustment. • All ingredients readily available in quantity.	• All design concepts and features proven in JPL low thrust motor.	• Easily processed with proven casting, pressure curing, and slot machining techniques. • Easily off-loaded without impact on insulation design or throat diameter.	• Attachment flange can be located at almost any point on case. Can be located at any point in cylindrical section without weight or cost penalty. • Cases are reusable. • Strength level proven in production programs • Fabrication techniques proven. • Excellent static and dynamic balance characteristics.	• Proven processing and fabrication techniques in several programs, including STAR motors, C-4, and MX. • Proven bonding capability to selected propellant • Available production item.	• Positive, reliable space ignition of either fully loaded or off-loaded configurations. • Easy accessibility after installation in spacecraft.

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STAR Series of Production Motors, 1961 — 1975

- Titanium strength level evaluation
- Case fabrication methods
- Basic nozzle structural concept
- Pressure curing of propellant
- Definition of material characteristics, which insures survival in space, and qualification of materials
- Spin effects evaluation and characterization
- AKM application experience: balancing, nozzle alignment, S&A device integration, impulse control, off-loading techniques
- Precision techniques for titanium case structural evaluation
- Neutral trace designs for case weight optimization

Low Thrust End Burner for NASA-JPL
W_p = 242.2 lb
tested in 1971 at sea level and 1972 in alt. chamber

- Advanced grain design (case bonded end burner, pressure cured)
- Low propellant burn rate to facilitate high expansion ratio
- Carbon-carbon exit cone, $\epsilon_i = 80:1$
- EPDM insulation (Kirk-Gard 4030)
- Carbon felt external insulation for exit cone

STAR 20A Advanced Altair for NASA-Langley
W_p = 629 lb
Mass Fraction = 0.912
Tested 1974 at 180 to 290 rpm

- TP-H-3335 HTPB/HMX/AP/Al propellant (90% solids)
- Low propellant burn rate to facilitate high expansion ratio
- Slot machining of TP-H-3335 propellant
- Stress relief by slots and pressure cure (no relief flaps)
- TL-H-318 liner

Improved STAR 37
W_p = 2302 lb
Tested 1974 at 100 rpm

- HTPB/AP/Al propellant, 89% solids
- EPDM insulation (Kirk-Gard 4030)
- TL-H-318 liner
- Pressure cure

STAR 30 AKM for Thiokol IR&D
W_p = 1014.5 lb
Mass Fraction = 0.944
Tested 1974 at 100 rpm

- Advanced grain design (84% web, pressure cured grain, machined slots)
- HTPB/AP/Al propellant, 89% solids
- EPDM case insulation (Kirk-Gard 4030)
- Carbon-carbon exit cone, $\epsilon_i = 75:1$
- Toroidal igniter
- 175,000 psi minimum ultimate strength design level for 6Al-4V titanium
- TL-H-318 liner
- Carbon felt external insulation for exit cone

FIGURE 6. GENEOLOGY OF STAR 30 DESIGN

possibility of outgassing and can be finished to meet the most stringent surface emissivity requirements. The motor is attached to the spacecraft by an attachment flange having a basic bolt hole diameter of 30.75 inches (see Figure 7). Variation in the bolt hole diameter and axial position of the flange is easily accomplished since the flange is made of a separate forging of titanium, machined and welded in place.

The basic design is fully compliant with the allowable envelope shown in Figure 1 of specification SPAR SG.356. To permit a reduction of approximately 1-1/2 pounds in case weight, we recommend that the attachment flange location specified in the RFP (31.31 inches from the end of the nozzle) be modified to permit attachment at the center of the case as shown in Figure 1 (Drawing E28386). This attachment arrangement may also result in a weight reduction in the mating spacecraft attachment fittings. The weight saving trade study is shown in Figure 8. This alternative attachment location also places the motor CG in the same general area.

The motor case is fabricated by conventional machining techniques from solution heat-treated and aged 6Al-4V titanium with a minimum design ultimate tensile strength of 175 ksi. The case wall thickness is contoured to achieve, as nearly as possible, a constant effective stress. In this manner a high degree of material utilization is achieved. Each hemispherical dome is machined from a closed die hemispherical forging and the cylindrical section from a rolled ring forging. The sections of the case are joined by circumferential TIG welds after being machined to final thickness and then subjected to a final aging cycle and finish-machined. A single opening is machined in the aft hemisphere to admit the bolt-on nozzle igniter assembly. The closure section of the nozzle, which contains the integral igniter assembly, is also fabricated of 6Al-4V titanium. It provides machined interfaces for positive nozzle alignment.

2.1.2 Nozzle Assembly. The STAR 30 nozzle consists of a lightweight carbon/carbon exit cone that is insulated on the exterior with carbon-felt material. The felt insulation limits the external temperature of the hot nozzle to 1000°F. The nozzle is attached to the motor case and the main nozzle structural support by a titanium ring. The ring is protected on the motor chamber side with insulation of ethylene propylene rubber (EPDM). The nozzle throat is made of Graph-I-Tite G-90 graphite to control erosion and thrust alignment.

The nozzle assembly is structurally and aerodynamically the same as the STAR 27 nozzle. The nozzle exit cone is fabricated of carbon/carbon rather than carbon phenolic material.

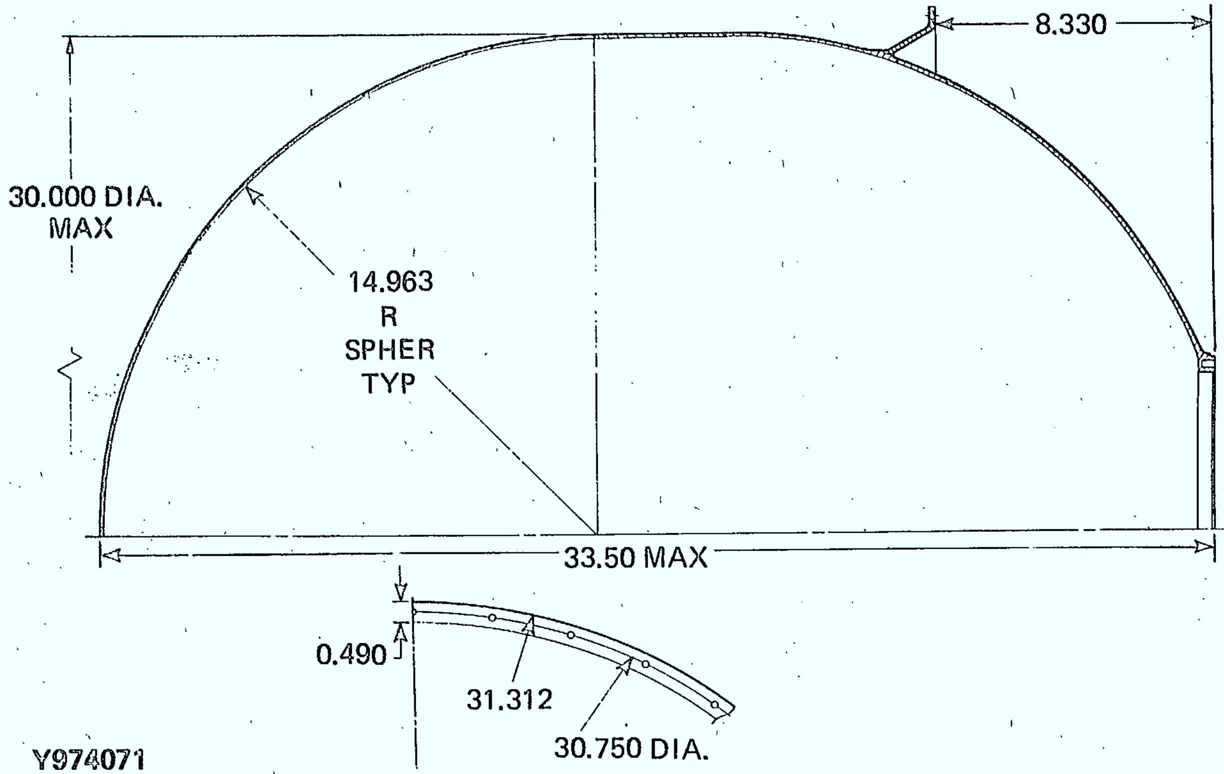


FIGURE 7. STAR 30 CASE

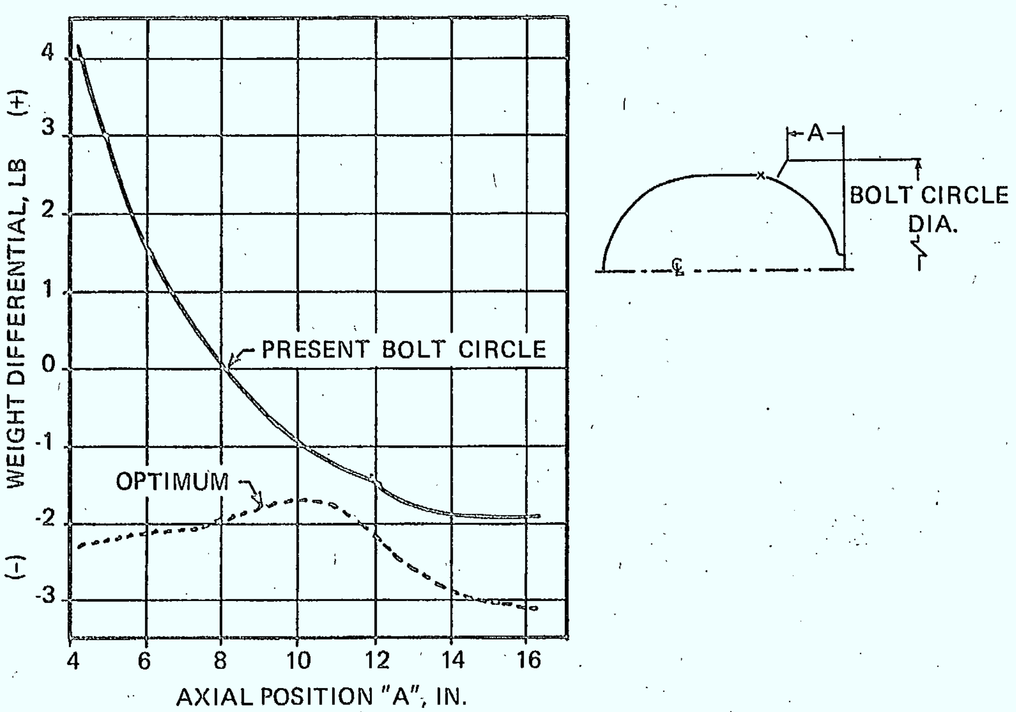


FIGURE 8. CASE WEIGHT SAVINGS

The "hot nozzle" concept permits the main structural and aerodynamic elements of the nozzle to reach their natural equilibrium temperature with the exhaust gases; i.e., to become very hot, rather than to limit the temperature of these elements with various forms of thermal insulators. Materials are available in the form of various fiber-reinforced carbons and graphites or carbon/carbon composites. These materials are a special form of carbon with its associated high-temperature characteristics. Nozzle parts made from these materials require no thermal insulation other than that required to protect adjacent components from the hot nozzle.

Thiokol/Elkton has been in the forefront of hot nozzle development for several years. Our design experience has included seven firings (three at simulated altitude), of which five were of flightweight designs (three different designs). An eighth firing (at altitude) will be performed this year in the second STAR 30 test. Expansion ratios have been as high as 85:1 and burn times as long as 150 seconds and the propellants have included beryllium propellants and 89 percent solids HTPB.

2.1.3 <u>Case Insulation.</u> The interior of the motor case is insulated with EPT 4030 ethylene propylene rubber. This lightweight insulation material maintains the case temperature below 150°F during motor operation and below 700°F during the post-operation heat soak. There is a weight penalty in additional insulation if the lower burnout temperature limit of 500°F is required. The current design is most weight effective within the existing envelope.

The insulation is fabricated using standard layup techniques in female molds. After curing in an autoclave, the insulators are removed from the molds, inspected, and bonded into the case. Ultrasonic inspection of the insulated cases is performed to assure that the insulation is properly bonded. The EPT 4030 insulation fabrication techniques are well established, and the adhesion characteristics are proven with this material for the propellant/liner/insulation bond system. In addition, the EPT 4030 rubber has superior insulating properties and lower density than buna-N or polyisoprene insulation materials.

2.1.4 <u>Propellant Grain.</u> The STAR 30 propellant grain provides efficient propellant packaging, simplicity in off-loading, and low propellant structural loading. Off-loading of the motor is easily accomplished by machining, eliminating the need for new casting equipment as requirements change. The 89% total solids/18% aluminum hydroxyl-terminated polybutadiene (HTPB) propellant, TP-H-3340, provides high vacuum impulse and good storage and space aging properties. The propellant was chosen based on mechanical property data and ballistic and structural results from the first STAR 30 motor test and from the test of a 2340-pound grain in the STAR 37H motor.

TL-H-318 liner is used to provide a strong bonding interface between the TP-H-3340 propellant and the EPT 4030 insulation.

2.1.5 Ignition System. The STAR family of motors all utilize pyrogen igniters for motor ignition. The STAR 30 pyrogen igniter is located in the aft end to provide flexibility in propellant grain weight selection without modification of the ignition system. The pyrogen uses the titanium nozzle structural member as an internal pressure chamber rather than a separate pyrogen case, thus reducing weight and motor imbalance. The igniter propellant is TP-H-3340, the same as the motor. Sea level and vacuum ignition tests have proven the proposed design. Aft end pyrogen igniters have been successfully used on the STAR 12, STAR 13, STAR 17, and STAR 26 motors.

2.1.6 Safe-and-Arm System. Qualified for numerous apogee kick motors, the Thiokol Model TE-O-642-4, 1-amp/1-watt no-fire remote electromechanical safe-and-arm explosive transfer assembly and through bulkhead initiators (TBI's) are used to provide the firing impulse to the motor ignition system. This safe-and-arm system, which meets Eastern and Western Test Range safety requirements, provides for ease in access, counter balancing by selective mounting to save spacecraft weight, and redundancy.

The explosive transfer assembly and the TE-O-642-4 S&A are shown in Figures 9 and 10. The selected S&A is currently being used on four STAR AKM's and is the lightest S&A currently in use.

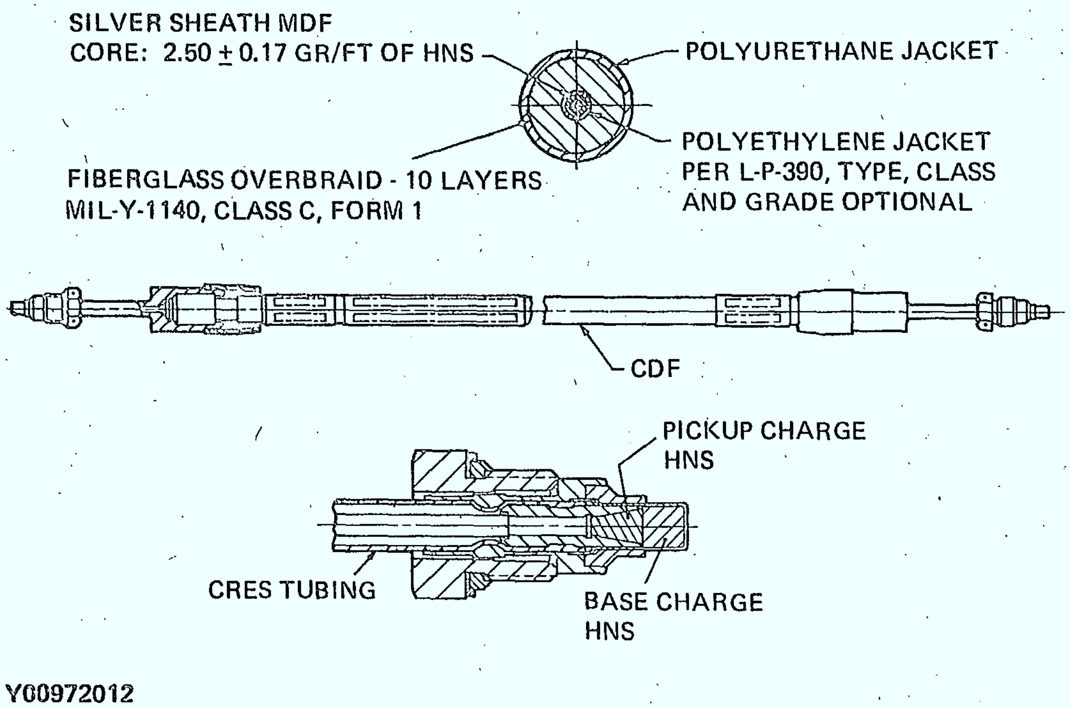
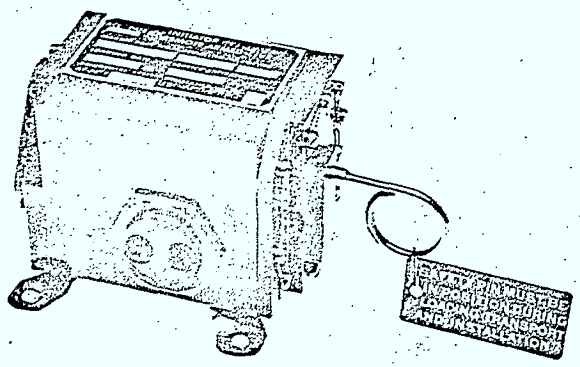


FIGURE 9. EXPLOSIVE TRANSFER ASSEMBLY



The TE-O-642-4 is a non-fragmenting, remotely mounted, remotely actuated electromechanical Safe and Arm Initiator Device. Because of the non-fragmenting feature, the device can be located on spacecraft without damage to nearby equipment.

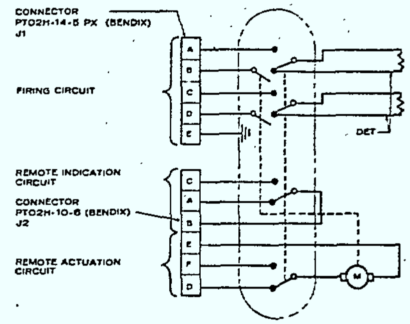
The motive power for the unit is furnished by a 28-volt reversible DC motor with an integral planetary gear speed reduction unit. The rotational power of the DC motor is transmitted to the shafts by spur gears, helix gears, and a friction clutch.

On the rotor are the visual indication flag, the firing train leads, and the rotary switches used for electrical circuitry control, including remotely armed-disarmed position indication. When the safety pin is manually removed and the arming current is applied, the motor rotates the rotor 90 degrees to the fully armed position. If arming power is applied with the safety pin in place, the motor operates and the slip clutch prevents any damage to the unit.

The Safe and Arm Device contains, in the explosive output area (adapter area), a boost charge of MDF and PETN contained in stainless steel cups. The function of the boost charge is to transfer the detonation wave from the safe and arm internal firing train to the explosive leads connecting the safe and arm device to the Through-Bulkhead Initiators. Redundant firing circuits and firing trains are provided to assure the highest reliability. The firing train is initiated with a 1 amp 1 watt detonator which meets the requirements of MIL-I-23659.

DETONATOR CHARACTERISTICS

Unit Weight	1.27 kg (2.8 lb)
Bridgewire Resistance	1.0 + 0.20 - 0.10 ohm
Maximum "No-Fire" Current	1.0 amp for 5 min.
Minimum "All-Fire" Current	3.50 amps
Firing Time at 5 amps	9 milliseconds
Firing Time at 10 amps	less than 1 millisecond



SCHEMATIC DIAGRAM

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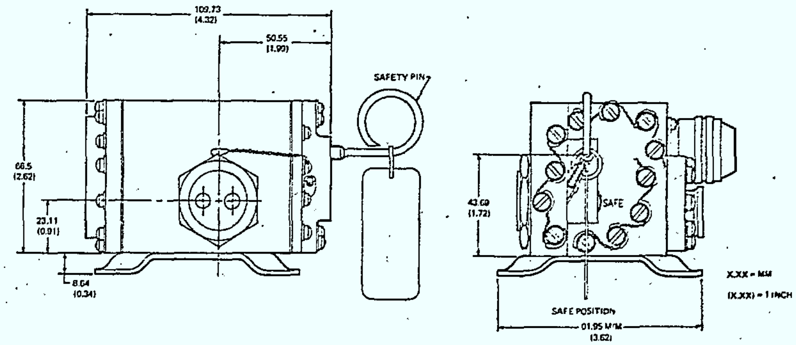

PERFORMANCE FEATURES

The unit is non-fragmenting.

Remote arming or disarming is done electrically.

Firing circuits and actuation and remote indication circuits are in separate connectors.

Unit can be manually disarmed but cannot be manually armed.

An indicator flag shows safe or armed status.

Remote monitoring of safe or armed status provided by normally closed circuits within the unit.

In the safe position, the detonators are shunted and the firing circuits open.

Safety pin prevents accidental arming of the unit during transportation, handling, or checkout.

Safety pin non-removable when arming power is applied.

The unit is hand safe if inadvertently fired in the safe position.

Arming or disarming time is less than 0.5 sec. at 28 vdc from -37.2°C to 71.1°C (-35°F to 160°F). Unit operable from 18 vdc to 32 vdc. Motor operating current at 28 vdc is 150 ma.

Mechanical and electrical systems are inseparable whether the device is operated electrically or manually.

FIGURE 10. TE-O-642-4 SAFE-AND-ARM DEVICE

2.2 Ballistic Performance

The ballistic performance for the SPAR AKM baseline motor was calculated for a motor providing a vacuum total impulse of 267,180 lbf-sec at 68°F. Motor performance for this configuration is summarized in Table IV and Figure 11. Nominal performance for the 10% uploaded motor is provided for comparison in Table V and Figure 12. Performance data at firing conditions of 20, 68, and 100°F are shown.

TABLE IV

PERFORMANCE SUMMARY - BASELINE DESIGN

(Nominal Calculated at Vacuum, Propellant Weight = 895.3 lb)

Temperature Conditioned, °F	20	68	100
Pressure, Maximum, psia	511	539	553
Pressure, Average, psia	448	473	485
Thrust, Maximum, lbf	5870	6195	6357
Thrust, Average, lbf	5220	5510	5660
Time, Burn, sec	50.8	48.3	47.1
Time, Action, sec	51.8	49.3	48.1
Total Impulse, lbf-sec	266,520	267,180	267,370
Specific Impulse, Propellant, lbf-sec/lbm	297.7	298.4	298.6
Specific Impulse, Effective,* lbf-sec/lbm	295.6	296.4	296.6

*Based on 901.5 lbs expended weight (895.3 lbs propellant + 6.2 lbs inerts expended)

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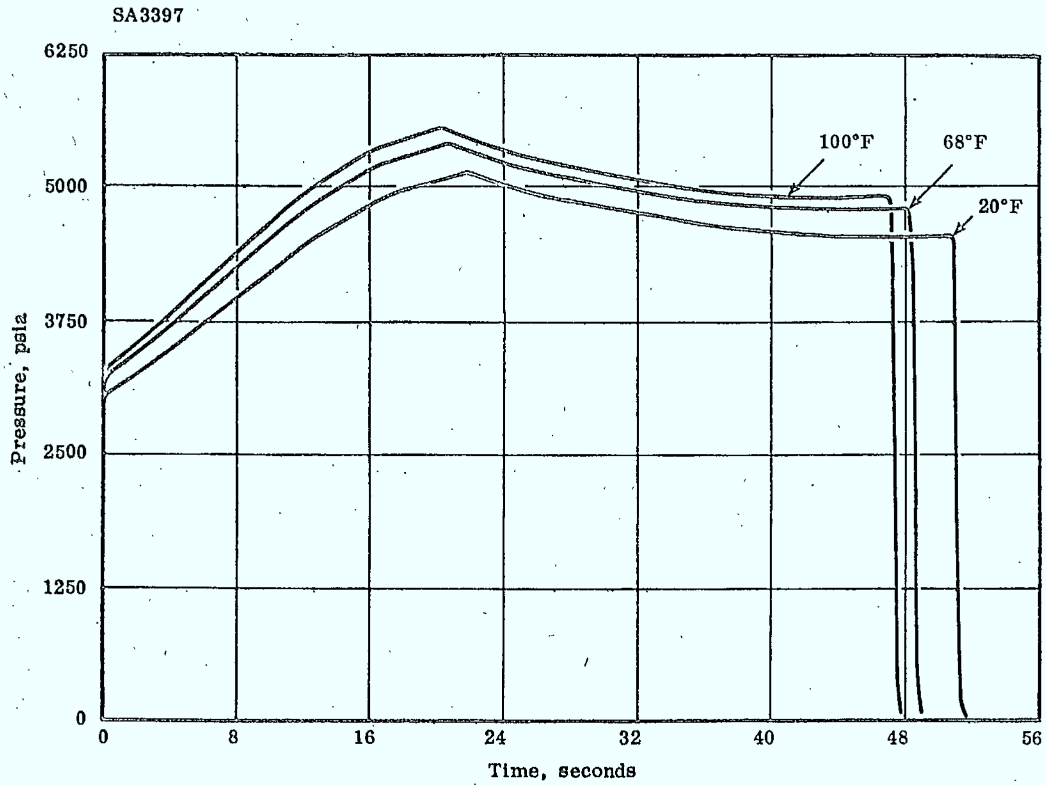
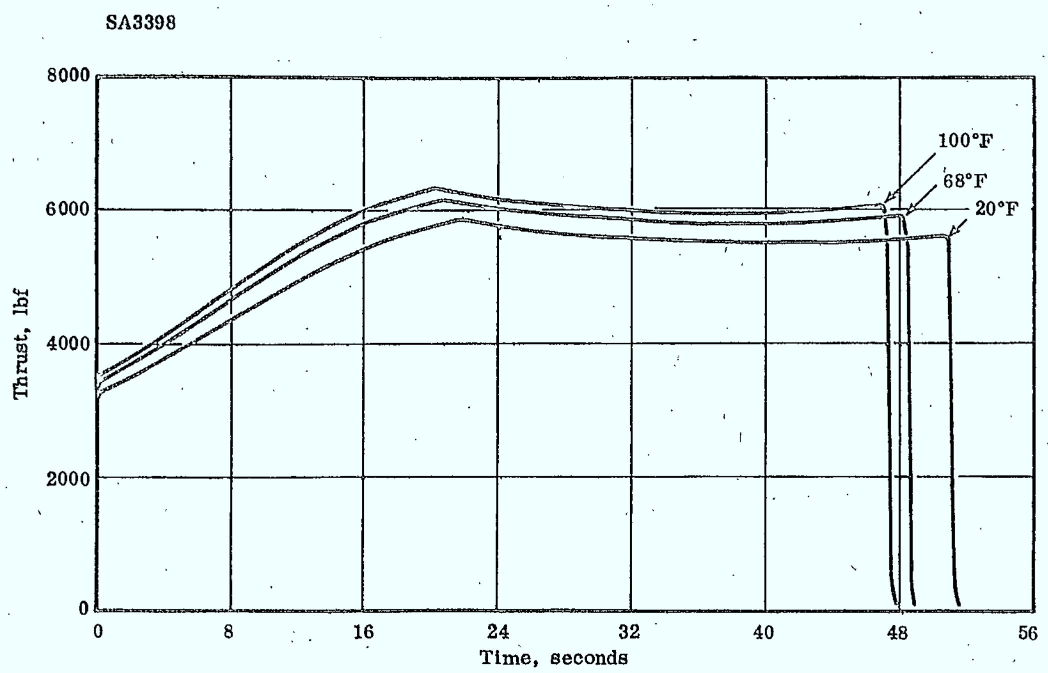


PRESSURE VERSUS TIME



THRUST VERSUS TIME

FIGURE 11. PRESSURE AND THRUST VERSUS TIME — BASELINE DESIGN

TABLE V

PERFORMANCE SUMMARY — 10% UPLOAD DESIGN

Nominal Calculated Performance
(TE-M-700 Loaded to 984.8 lbs Propellant)

Temperature, °F	20	68	100
Pressure, Maximum, psia	568	600	614
Pressure, Average, psia	499	526	539
Thrust, Maximum, lbf	6640	7008	7190
Thrust, Average, lbf	5100	5268	5387
Time, Burn, sec	57.3	55.6	54.4
Time, Action, sec	58.1	56.6	55.4
Impulse, Total, lbf-sec	293,155	293,903	294,090
Impulse, Specific, Propellant, lbf-sec/lbm	297.7	298.4	298.6
Impulse, Specific, Effective,* lbf-sec/lbm	296.0	296.8	297.0

*Based on 990.3 lbs expended weight (984.8 lbs propellant + 5.5 lbs inerts expended)

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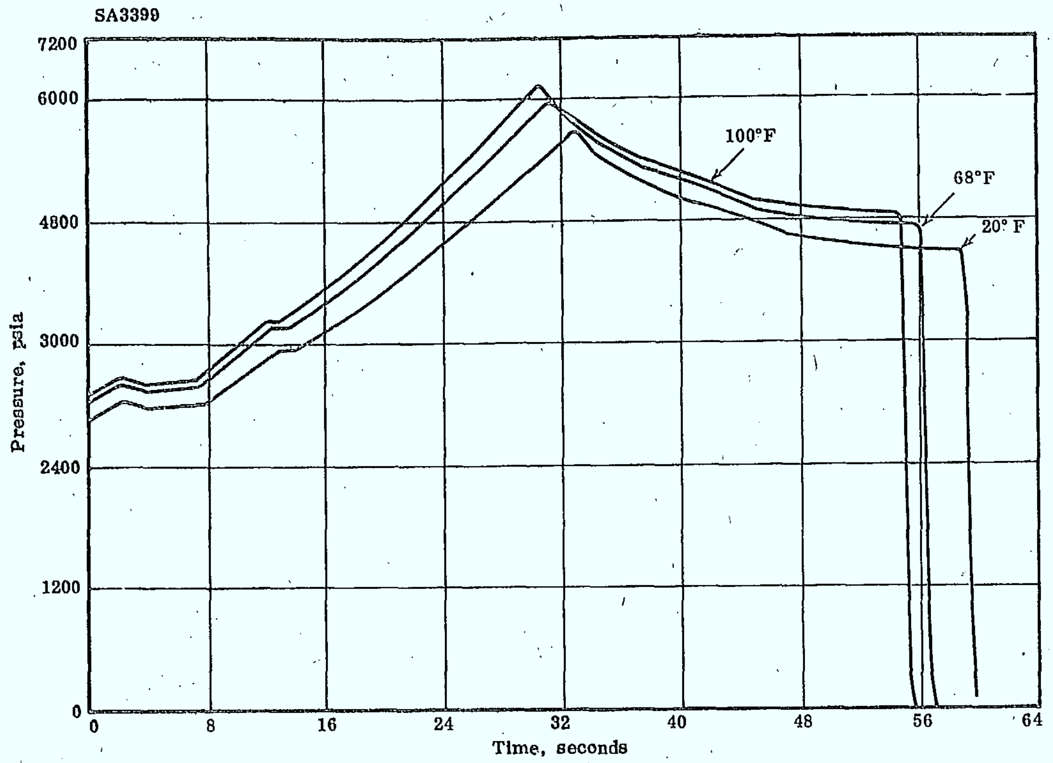
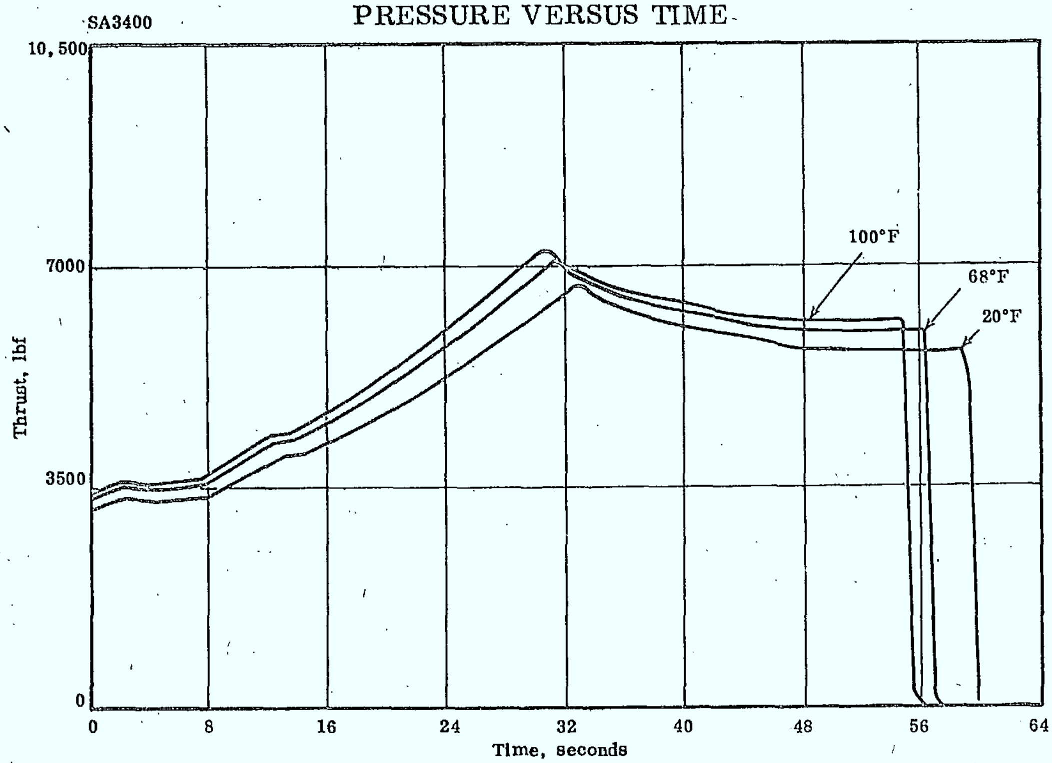


PRESSURE VERSUS TIME



THRUST VERSUS TIME

FIGURE 12. PRESSURE AND THRUST VERSUS TIME — 10% UPLOAD DESIGN

2.3 Mass Properties

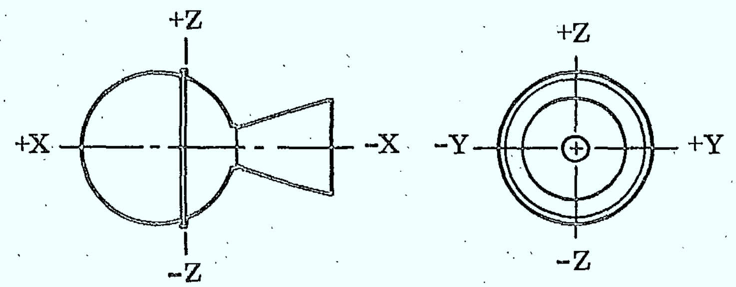
The mass properties calculated for the two motor configurations are shown in Table VI and Figure 13.

TABLE VI

STAR 30 MOTOR WEIGHTS

Weights, lbs	Baseline Design I_T = 267,180	10% Upload I_T = 293,903
Motor Total Weight*	957.9	1047.4
Propellant Weight	895.3	984.8
Motor Inert Weight	62.6	62.6
Motor Burnout Weight	56.4	57.1

*Includes 2.8 lbm for remote safe-and-arm device and 0.8 lbm for dual ETA lines.



Motor Condition(1)	Weight, lb	CG Location, inches			Mass Moment of Inertia About CG, lb-in.^{2}		
		X	Y	Z	I_{ox} (Roll)	I_{oy} (Pitch)	I_{oz} (Yaw)
Baseline Design I_T = 267,180 lb-sec							
Loaded Motor	954.3	9.5	0	0	98,800	99,210	99,200
Fired Motor	52.8	-0.4	0	0	6,000	10,310	10,300
10% Upload I_T = 293,903 lb-sec							
Loaded Motor	1043.8	8.5	0	0	102,400	112,210	112,200
Fired Motor	53.5	-0.4	0	0	6,020	10,410	10,400

(1) Moments and CG do not include remotely mounted S&A and ETA lines.

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FIGURE 13. STAR 30 MASS PROPERTIES

2.4 Specification Compliance

It is suggested that the following clarifications or revisions to the SPAR SG. 356 preliminary specification be incorporated to reduce program costs or to simplify the use of existing hardware designs without further modification. We suggest that a technical interchange meeting be held with SPAR to resolve these items prior to release of the final procurement specification.

SG. 356 Reference		Comment
3.3.8	Maximum Case External Temperature	Delete the sentence - "The case external temperature shall not exceed a temperature which would cause insulation debonding." While insulation debonding after motor operation is undesirable from an imbalance viewpoint, it is a subjective criterion which should be examined in light of any effect on spacecraft performance (see section 3.5.1.3).
3.5.1.3	Static and Dynamic Balance	The values stated in this paragraph are too restrictive for this size rocket motor. Request revision as follows:

	Before Firing		
	Unloaded	Loaded	After Firing
Static, lb-in.	2.0	8.0	6.0
Dynamic, lb-in.^{2}	5.0	60.0	50.0

3.5.5	Factors of Safety	Thiokol standard design criterion for STAR series motors with respect to minimum burst pressure is 1.25 times MEOP. A 1.3 factor will result in an increase in motor weight.
4.4.4	Motor X-ray	Thiokol has well-established procedures for radiographic inspection of STAR rocket motors using a 25 MEV betatron and two particular film types that were selected to optimize exposure for interface and through-web conditions at a quality level of 2 - 2T. To continue this standard practice, a specification revision is required to extend the density range to 1.8 to 3.5 H&B and a quality level of 2 - 2T.

SG. 356 Reference	Comment
4.4.11 Leak Test	Thiokol considers the Halogen leak test to be unnecessarily restrictive for the assurance of seal integrity and uses a pressure decay criterion for many of its STAR motors. We therefore request revision of this paragraph to permit this alternative method.

3.0 RELIABILITY ASSESSMENT

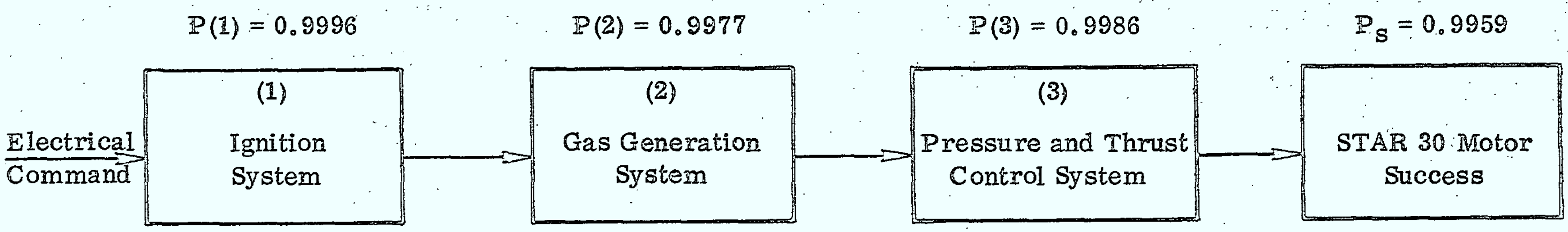
The predicted reliability of the SPAR AKM exceeds the 0.995 requirement established in Section 3.7 of SG. 356.

Thiokol drawings, specifications, and manufacturing procedures are reviewed by the Reliability Engineer for compliance with requirements and will be compared with documentation from previous successful space motor programs. The reviews will include confirmation that such reliability improvements as pressure curing, moisture control, and insulation grain orientation that have been incorporated in recent flight-proven Thiokol motors are carried out in accordance with methods thoroughly proven on previous STAR motor programs. Reliability personnel will participate in formal design reviews and report reliability status in regular progress reports as specified.

The high level of reliability achieved in design will be maintained during manufacturing and assembly by comprehensive quality control and unscheduled reliability audits. The details of the quality control and manufacturing assurance methods will be provided in the quality plan.

The motor qualification test program will provide confidence that the inherent reliability has not been changed by the design modifications required for compliance with SPAR specifications. The test program will confirm the performance capability and confidence in Thiokol ballistic predictions and in structural and thermal analyses.

A preliminary reliability prediction based on past history of spherical space motors indicates a motor reliability of at least 0.9959 with a 0.50 confidence level. A block diagram of the motor showing the major components and assemblies with their reliability is presented as Figure 14. All elements are in series except the initiation system, which consists of parallel elements of detonating cord and through-bulkhead initiators.



(1) Ignition System

Provides chemical energy necessary for main propellant ignition.

Provides positive safe and arming.

(2) Gas Generation System

Propellant provides chemical energy necessary for conversion to kinetic energy.

No debonding of insulation and liner.

No loss of structural integrity.

(3) Pressure and Thrust Control System

Provides expansion of combination gases.

No loss of structural integrity.

Contains combustion gases for maximum thrust generation.

$$P_S = P(1),\ P(2),\ P(3) = (0.9996)\ (0.9977)\ (0.9986) = 0.9959$$

FIGURE 14. RELIABILITY BLOCK DIAGRAM (FOR 50% CONFIDENCE LEVEL)

4.0 PROGRAM PLAN

The proposed Apogee Kick Motor (AKM) program consists of the following effort:

- Design of the Apogee Kick Motor to meet the SPAR technical requirements
- Procurement and fabrication of AKM hardware. The longest lead time articles are the motor case and nozzle exit cone
- Qualification of the AKM by environmental and static tests
- Manufacture and delivery of a loaded inert model
- Deliver an empty fired unit
- Delivery of three flight motors
- Program management to meet all technical, schedule, quality, reliability, and administrative requirements of the contract

The program schedule, based on the specified contract start date of July 1, 1976, is shown in Figure 15.

4.1 Motor Design Finalization

The design finalization activities for the proposed program will be accomplished during the first 2 months. A coordination meeting to review all performance and motor design features, physical and electrical interfaces, handling procedures, and test plans will be held before motor loading is initiated.

The planned program makes maximum use of the designs, analyses, tests, and other data existing for the STAR 30 motor and the data obtained on other STAR AKM motor programs. Procurement, planning, design, and analysis work will begin immediately upon receipt of authorization to proceed. The design will be reviewed to incorporate any changes resulting from contract negotiations and to add any revisions introduced in the final released copy of the SPAR SG.356 specification. Sufficient time is available to incorporate minor design iteration in the fabricated hardware.

4.2 Procurement and Component Fabrication

We will initiate major item procurement in advance of program contractual go-ahead in order to meet your early loaded inert motor (dynamic model) delivery with the use of fired qualification test hardware. This approach saves the cost of new hardware for an item which will not see flight use. However, in order to accomplish this, we will need to be notified that the Multipurpose Satellite will utilize the STAR 30 motor as the AKM. This notification is required by February 1976.

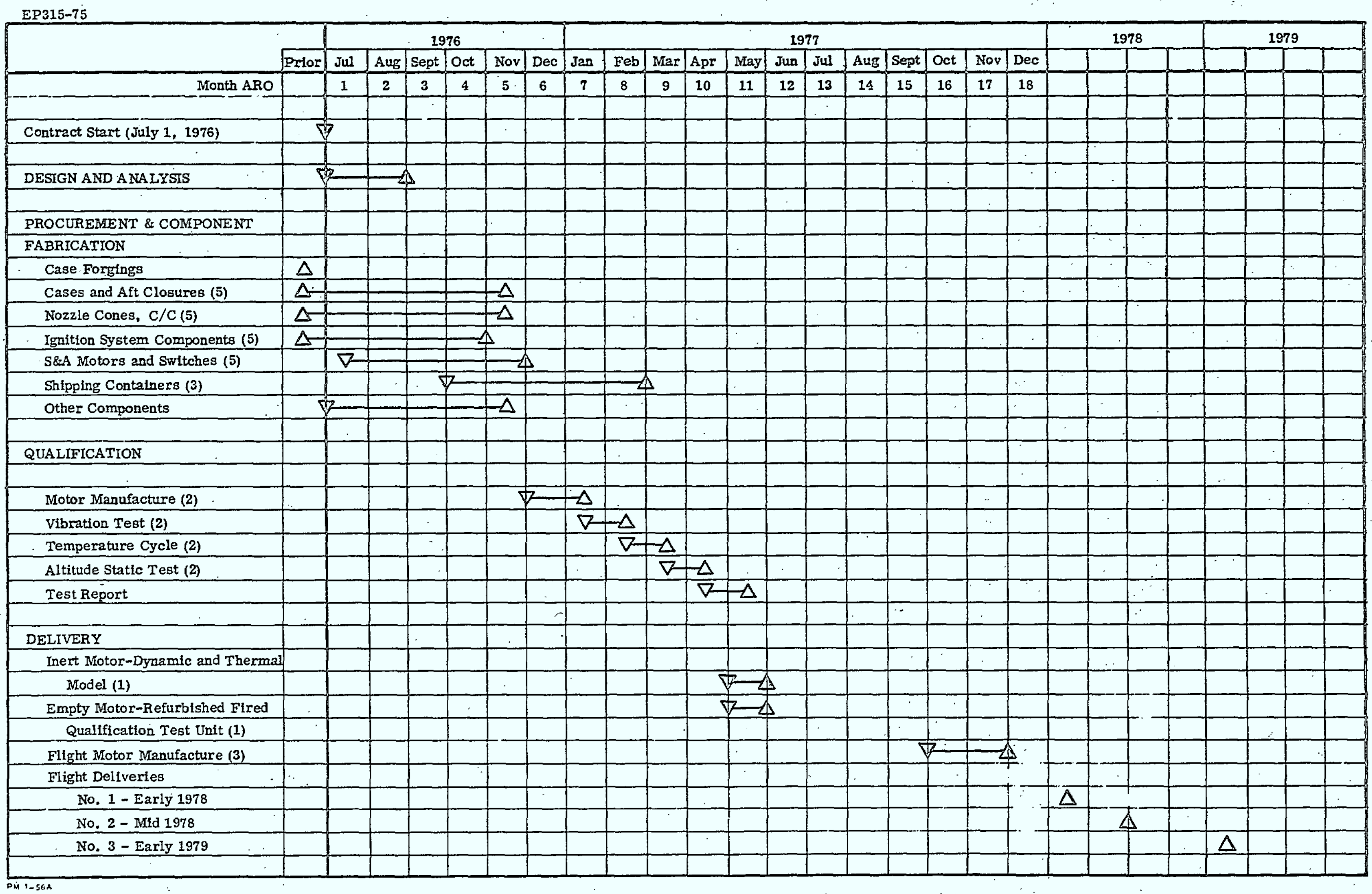


FIGURE 15. PROGRAM SCHEDULE

4.3 Qualification

Under Thiokol funded STAR 30 development program a motor case will be subjected to a pseudo burst and load testing before the SPAR Qualification motor manufacture starts during the 7th month. Simulated thrust loads will be applied to the test unit during pseudo burst to demonstrate conformance with the factor of safety requirements of SG.356, par. 3.5.5.

The pseudo burst test article will consist of a case, closure, and adapter assembled with flight-type fasteners and O-ring. The assembly will be supported in a test ring at the spacecraft mounting flange. Motor thrust loads will be simulated on the test article during case pressurization. Data evaluation of the test will include comparisons of test results with predictions. The pseudo burst test will use fired, refurbished test motor hardware and all data will be provided to SPAR Aerospace Products Ltd.

Each motor case will be hydrotested as part of the standard inspection requirements.

To present the most economical program we plan to utilize a residual fired case from Thiokol's development program for one of the two qualification motors. This motor case will be refurbished, rehydrotested,and reinspected to assure its suitability for use. The second qualification motor case will use new hardware. The motor cases received will be inspected, cleaned, and insulated with EPT 4030 insulation Motors will be assembled in the empty condition, and weights and longitudinal CG's will be determined. The empty motors will be spin balanced at 150 to 175 rpm and then disassembled and ballasted (if needed). Our recent experience has shown that motors of this type normally meet the balance requirements in the as-built condition and require no ballast. A photograph of the Gisholt vertical balance machine being used to check a Skynet STAR 24 motor is shown in Figure 16. The insulated cases will be assembled for casting, lined with TL-H-318 liner, loaded with TP-H-3340 propellant, and pressure-cured. The desired final propellant grain configuration will be obtained by machining the loaded propellant assembly in a vertical turret lathe (see Figure 17).

After radiographic inspection has verified the propellant grain integrity of the loaded cases, the motors will be fully assembled, the center of gravity will be measured, and the units will be returned for final spin-balance determinations. To date, no loaded motor manufactured by Thiokol/Elkton has required propellant removal to adjust the imbalance to meet specification requirements.

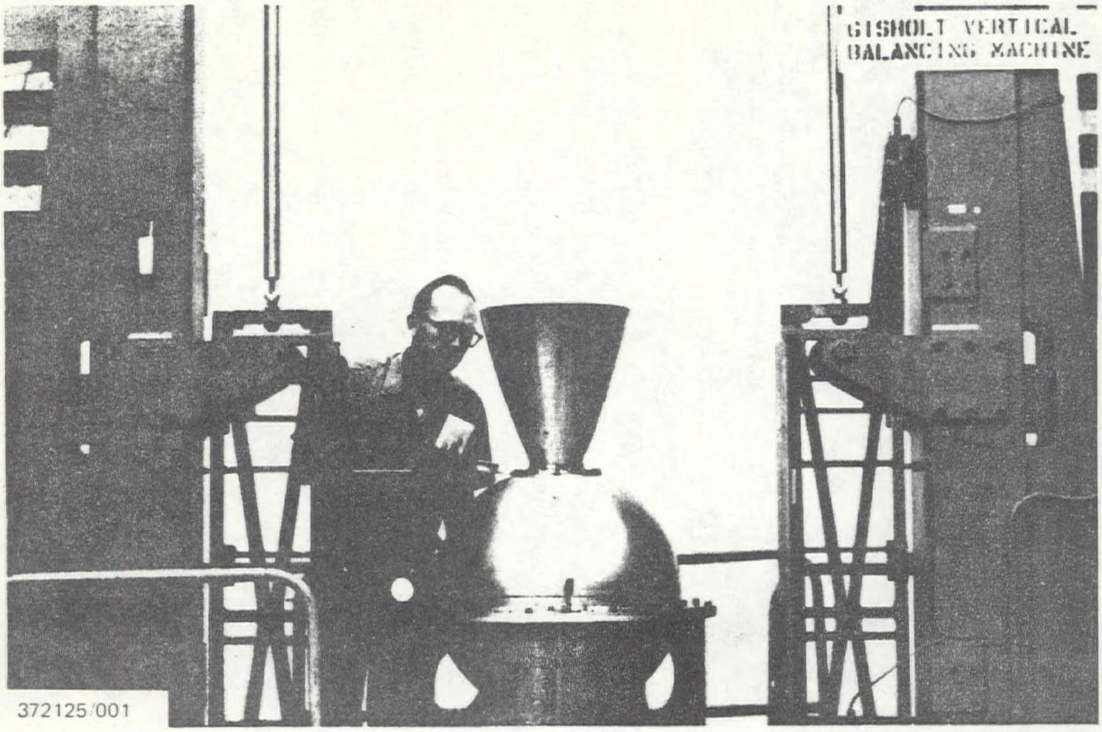


FIGURE 16. GISHOLT VERTICAL BALANCING MACHINE,
WALLOPS ISLAND STATION

Manufacture of the two Qualification motors will be completed during the 7th program month. The Qualification motors will then be subjected to vibration testing and temperature-cycling exposure tests. The vibration test will be conducted to confirm compliance with the sinusoidal and random vibration levels of SPAR SG. 356, par. 3.4.7. The temperature cycling tests will be performed from +20 (+0, -10)° F to +100 (+10, -0)° F in accordance with SPAR SG. 356, par. 4.2.1. Upon completion of the vibration and thermal cycling tests, the motor will be visually and radiographically inspected.

Vacuum performance of the Qualification motors will be evaluated in tests conducted in the T-3 Cell (Figure 18) at the Arnold Engineering Development Center (AEDC). The test conditions will include a cold nozzle test verification with the aft 6 inches of the exit cone at -197° F to demonstrate compliance with S.G. 356, par. 4.5.5.4. and 3.4.2. An altitude test will be conducted with the motor exposed to a simulated altitude of at least 100,000 feet for 5 minutes prior to firing. The pressure will be maintained at a minimum of 50,000 feet throughout tailoff, or until blowback occurs.

The motor will be mounted on a fixture which will permit rotation about its spin axis. The motor will be rotated at a minimum speed of 150 rpm for at least 5

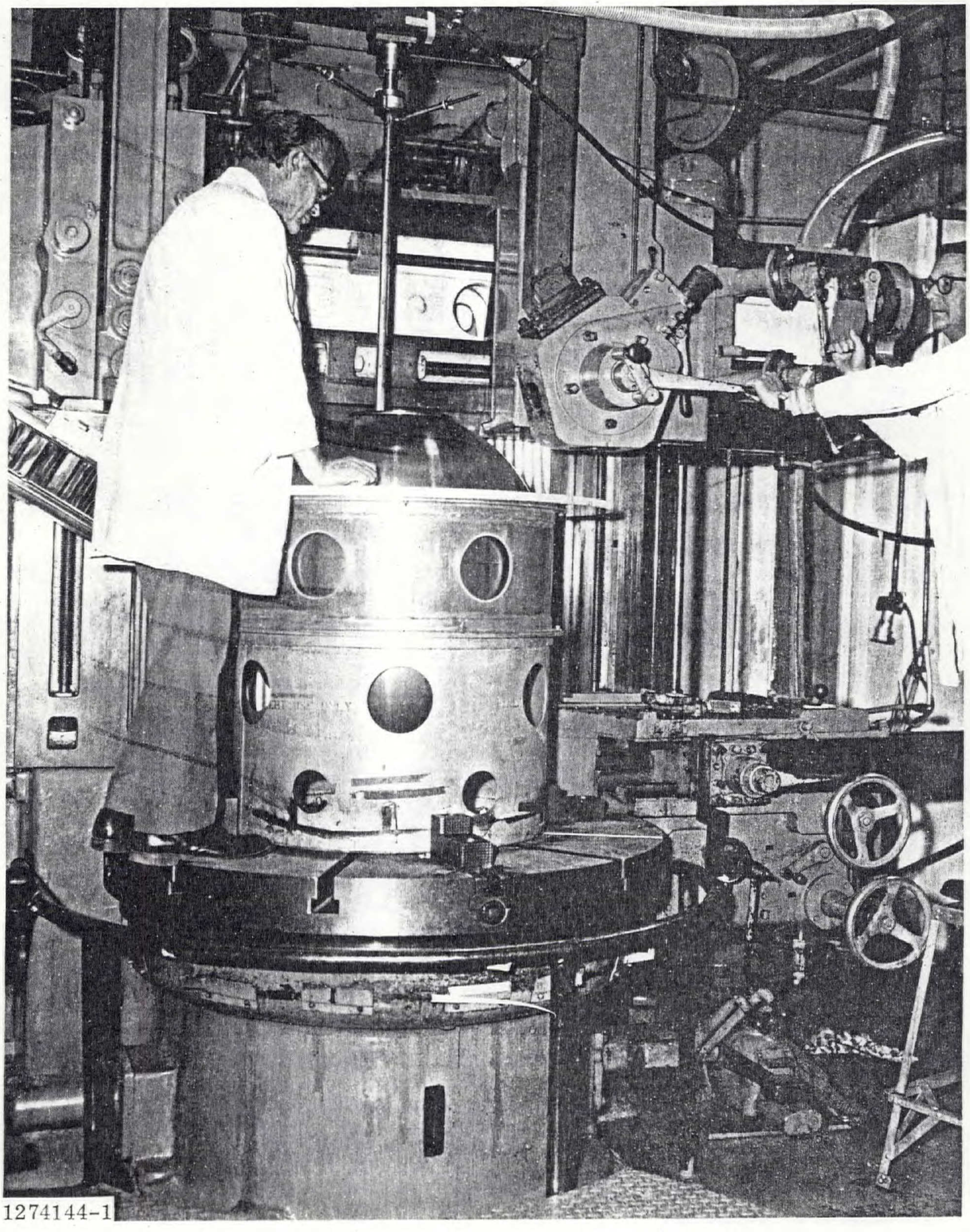


FIGURE 17. 30-INCH MOTOR IN VERTICAL TURRET LATHE WITH CUTTER BAR

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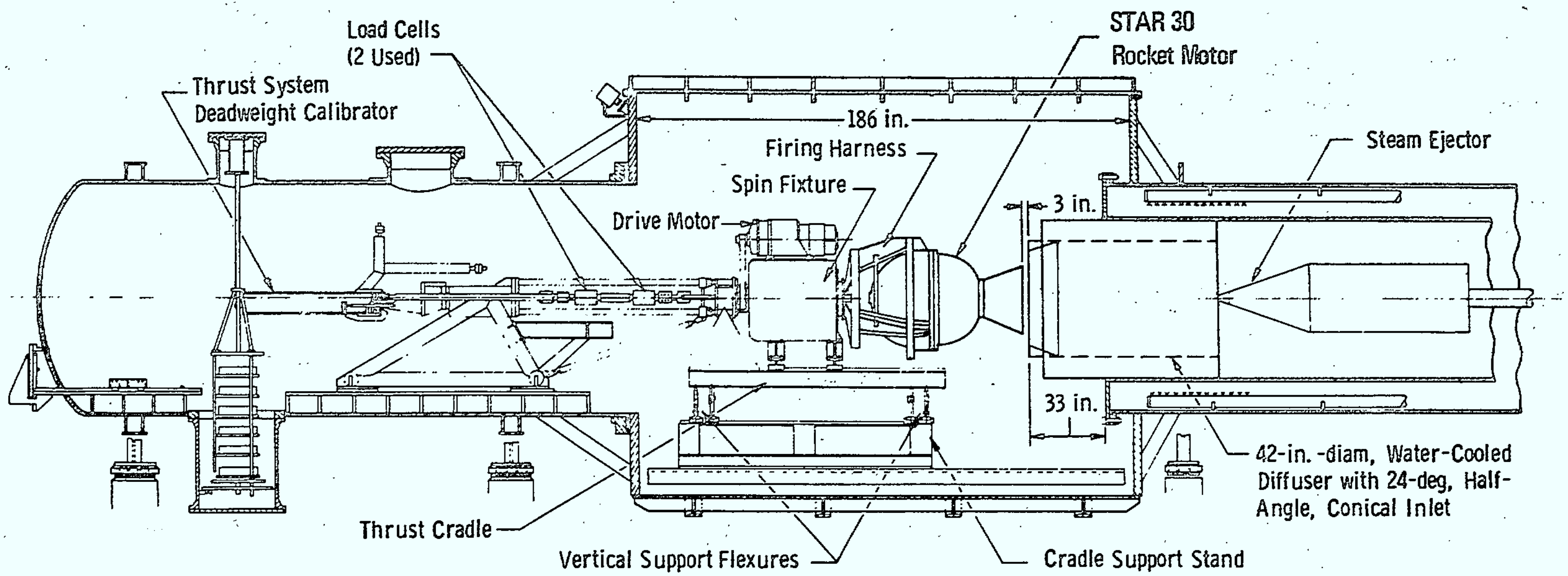

FIGURE 18. SCHEMATIC OF THIOKOL MOTOR IN AEDC TEST CELL T-3

minutes before ignition while at the specified prefiring environmental temperature. The spin rate will then be reduced to 100±10 rpm and the motor will be fired.

One of the Qualification motors will be altitude tested with a customer-supplied thermal blanket installed and with a representative thermal mass at the attachment ring for typical thermal soakback conditions. Design parameters are to be provided by SPAR at a future date to provide a basis for design of the test fixture. Since the design parameters are not currently available, we have not included any budgetary pricing for this item in Volume II at this time.

Qualification test procedures based on previous STAR motor documentation will be prepared and submitted to SPAR for concurrence. A Qualification Test Report will be prepared at the conclusion of the test program, summarizing the environmental and static test conditions, test findings, and compliance with specification requirements.

4.4 Delivery

The inert motor for the dynamic and thermal model is scheduled to be delivered to SPAR at the end of the 11th month. This model will use reclaimed fired qualification hardware.

The empty motor will be prepared for delivery by refurbishing a fired qualification motor test unit and will be available in May 1977 (11 months ARO). This unit, which uses a Thiokol case, will be provided on a loan basis for a six month period. If SPAR needs it beyond that time period, an equitable arrangement can be negotiated.

The flight motors will be fabricated using the same procedures as described for manufacture of the Qualification motors. Finished motors will be available for delivery at the end of November 1977 (17 months ARO), well in advance of the SPAR requirements for delivery of the unit in early 1978, one unit during mid 1978, and one unit during early 1979. Approximately 3-4 months are available after the completion of qualification tests to finalize on motor weight for the flight units.

Three reusable shipping containers are planned for delivery of the flight quality motors. These containers will have been used for the Qualification and inert shipments and then returned to Thiokol for subsequent flight motor shipments.

FEASIBILITY STUDY OF A GENERAL
PURPOSE SPACECRAFT BUS.