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FEASIBILITY STUDY OF A
GENERAL PURPOSE SPACECRAFT BUS

VOLUME IV
BUS PERFORMANCE COMPARISON

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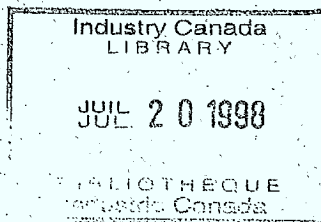
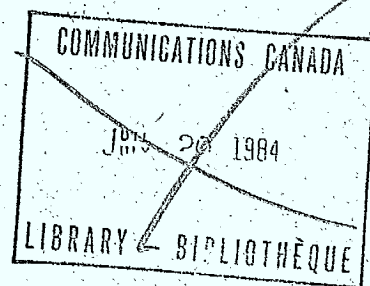
FEASIBILITY STUDY OF A
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VOLUME IV
BUS PERFORMANCE COMPARISON

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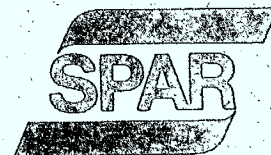
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FEASIBILITY STUDY OF A GENERAL

PURPOSE SPACECRAFT BUS

BUS PERFORMANCE COMPARISON

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June, 1975

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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

INTRODUCTION

Summarized in this volume of the General Purpose Bus (GPB) Feasibility Study Report are:

- Requirements
- Design/Performance
- Further subsystem improvements, together with associated weight savings and problem evaluation
- Comparison of the performance characteristics of the GPB with other similar capability spacecraft

In addition, Appendix A discusses the Canadian capability to carry out the GPB program (identified in the Implementation Plan Volume III) in terms of expertise, facilities and available manpower. In particular, the capabilities at Spar are discussed relative to its mechanical subsystem experience on all previous Canadian, and various manned and unmanned U.S., space programs.

2.0

REQUIREMENTS SUMMARY

The basic requirement for a General Purpose Bus (GPB) is to provide a spacecraft bus capable of meeting the requirements for all Canadian geo-stationary missions in the next decade, and as such the bus will meet the requirements for all missions between the Anik and Shuttle eras (see Figure 2-1).

This spacecraft bus will provide a cost effective vehicle for operating satellites that have the same general mission parameters such as orbit and launch vehicle and have similar payload weight and power requirements. The bus, which is a basic "service module" will consist of the spacecraft "housekeeping" subsystems, namely structure, thermal control, attitude control, propulsion, power and telemetry tracking and command subsystems.

Payload characteristics for the five payload options identified by the Department of Communications (DOC) as being the most probable payloads for possible Canadian geo-stationary missions in the next decade, are given in Table 2-1.

These payloads are:

- (a) UHF/4-6 GHz (12 channels, 5 watt TWT's) transponders.
- (b) UHF/12-14 GHz (4 channels, 20 watt TWT's) transponders.
- (c) UHF/7-8 GHz (single channel) transponders, with experimental payload.
- (d) UHF/7-8 GHz/L Band transponders.
- (e) 4-6 GHz (24 channel, 5 watt TWT's) transponder.

Not listed in this requirements summary are those payloads which were identified by Telesat late in the study, but which can likely be incorporated within the present baseline design.

PROJECTED CANADIAN SPACECRAFT BUS UTILIZATION

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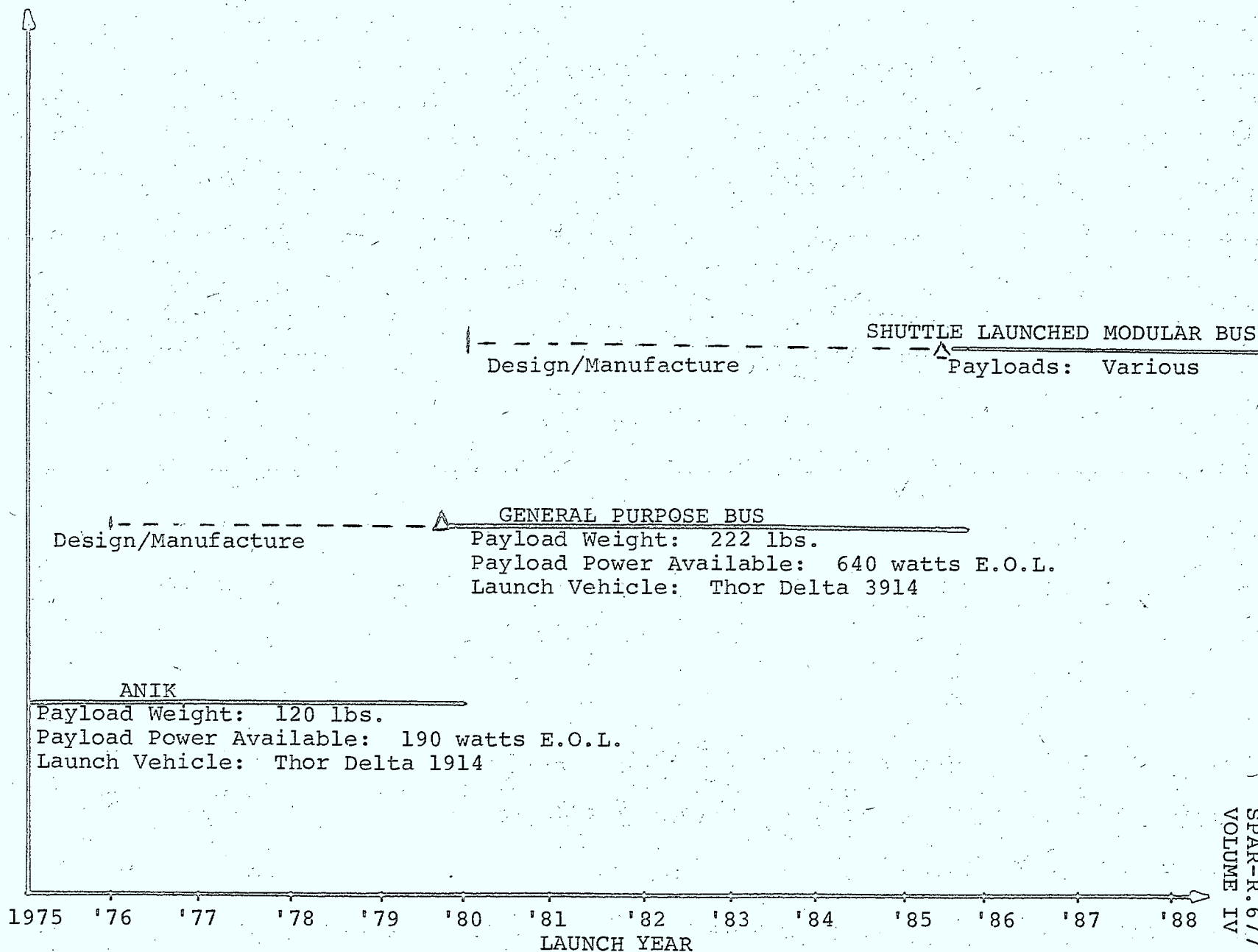


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SPACECRAFT BUS

2-2

FIGURE 2-1



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TABLE 2-1

G.P. BUS PAYLOAD CHARACTERISTICS

PAYLOAD OPTIONS	ANTENNAE			TRANSPONDER			
	TYPE	NO. & SIZE	WEIGHT	SIZE	WEIGHT lbs.	DISSIPATION watts	DC POWER
a) UHF/4-6 GHz Transponder (12 channel, 5 watt TWT)	Deployable parabola or parabola + quad helix	13 ft. dia. or 82" dia. x 100" long	55 lbs.	TBD	218.4	S*314.9 E 264.3	S 414.9 E 339.3
b) UHF/12-14 GHz Transponder (4 channel, 20 watt TWT)	Deployable parabola or parabola + quad helix	13 ft. dia. or 82" dia. x 100" long	55 lbs.	TBD	180.5	S 456.6 E 351.5	S 624.6 E 463.5
c) UHF/7-8 GHz single channel Transponder with auxiliary experimental payload	Deployable parabola	13 ft. dia.	55 lbs.	TBD	221.9	S 404.2 E 252.3	S 505.2 E 305.3
d) UHF/SHF/L Band Transponder	Deployable parabola	13 ft. dia.	60 lbs.	TBD	190.4	S 397.2 E 305.3	S 513.2 E 373.3
e) 4-6 GHz (24 channels, 5 watt TWT)	parabolic shaped beam	5 ft. parabolic or equivalent	TBD	TBD	TBD	S 416.5	S 416.5 E 347.2

*S SUNLIGHT

*E ECLIPSE

Essential requirements of the GPB are:

- The spacecraft will be designed for launch on a Thor Delta 3914 launch vehicle fitted with an 84 inch diameter fairing.
- 3-Axis stabilized spacecraft in geo-synchronous orbit, with sun orientated solar arrays.
- Operation life of 6 to 8 years.
- Station keeping accuracy $\pm 0.05^\circ$.
- Lift off weight 1,925 lb.

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3.0 DESIGN/PERFORMANCE SUMMARY3.1 Design Philosophy

- Use of proven components where feasible.
- Use of new technology, in the baseline design, where:
 - (a) shows significant advantage,
 - (b) is in process of development and is anticipated to be fully developed and available within the expected time frame of the G.P. Bus.

Major design features of the baseline General Purpose Bus configuration (see Figures 3-1, 3-2 and 3-3) are identified below.

3.2 Design Features3.2.1 General Spacecraft

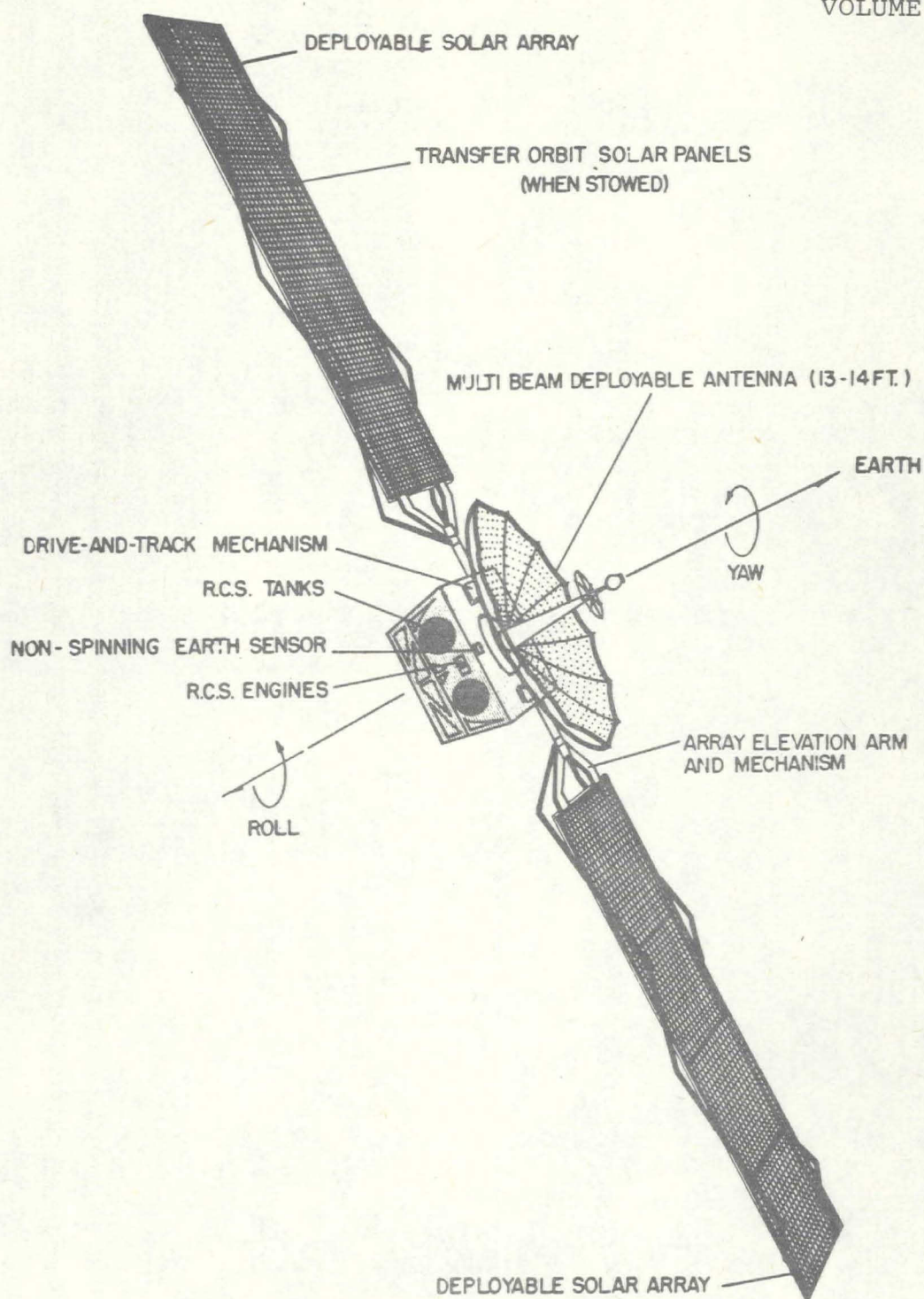
3-Axis stabilized spacecraft with the multi-beam communications antenna and TT&C bi-cone antenna mounted on the earth facing forward platform (perpendicular to the yaw axis).

Communications transponders mounted on the north and south facing panels (perpendicular to the pitch axis) to minimize solar thermal effects.

High dissipation and narrow temperature range housekeeping components (e.g. batteries) also mounted on these panels.

The deployable solar arrays, capable of supplying 835 watts at end of life, are stowed on the north and south panels during the spin phase, with parts of the arrays providing power for this phase of the mission. The arrays are attached to the drive and tracking units mounted on the forward platform.

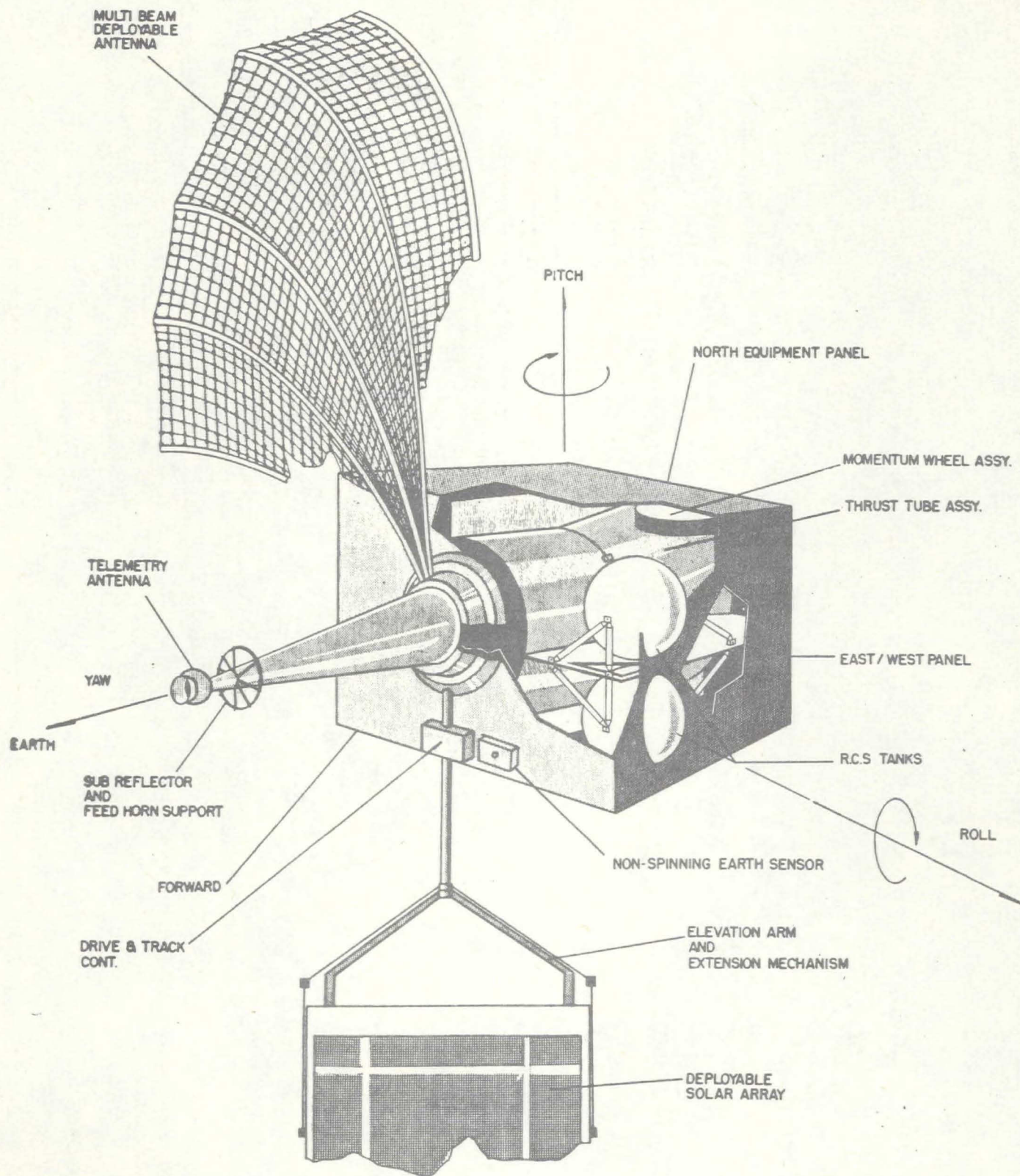
Attitude control subsystem sensors are mounted on the forward platform.



SPACECRAFT ARRANGEMENT

IN

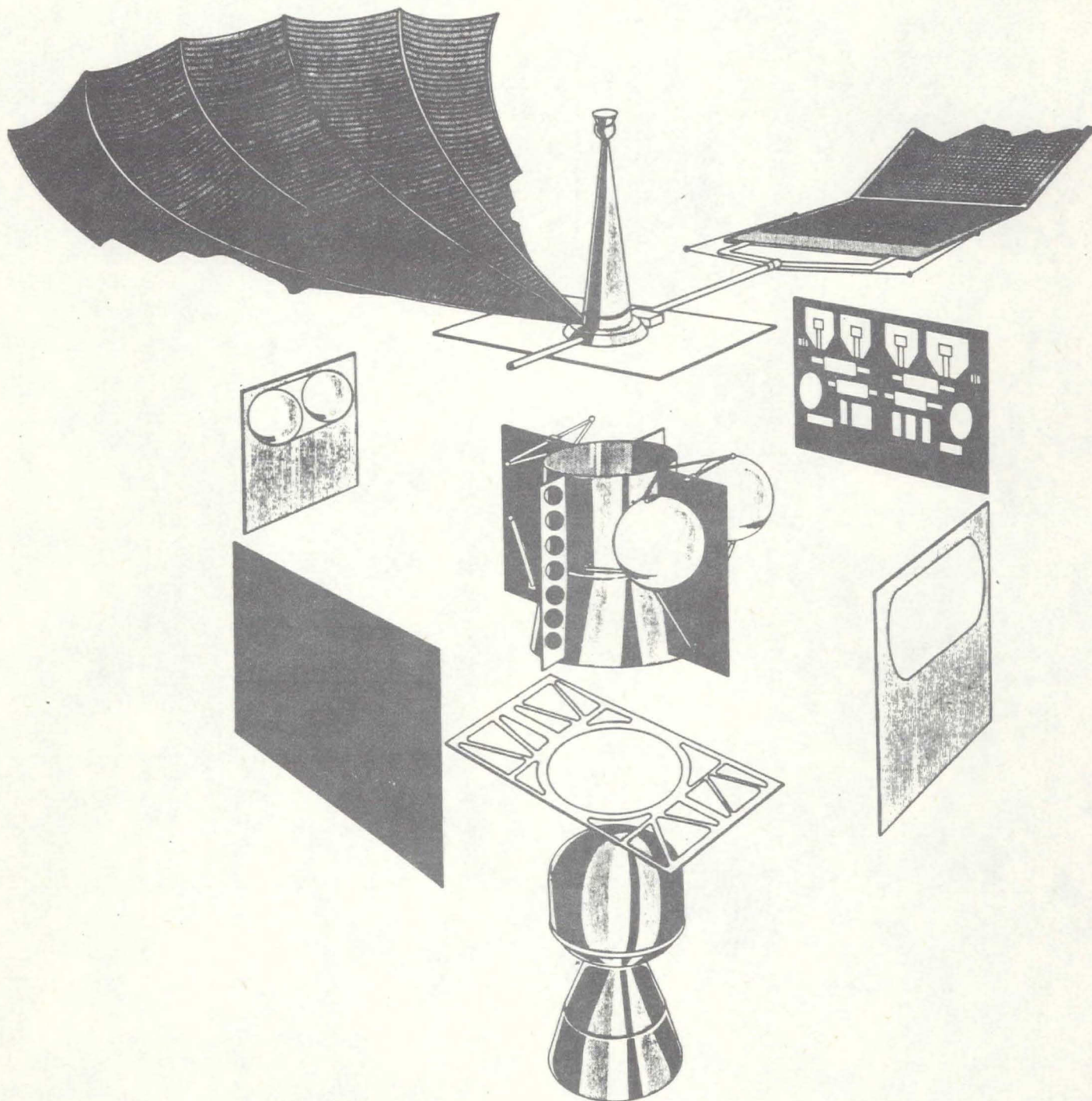
SYNCHRONOUS ORBIT



GENERAL PURPOSE BUS

FIGURE 3-2

GENERAL PURPOSE SATELLITE BUS (EXPLODED VIEW)



3.2.2 Attitude Control Subsystem

A hybrid attitude control system consisting of

- Pitch reaction wheel
- Thruster control about roll and yaw
- Single Magnetic bearing reaction wheel
- Microwave attitude sensing system

Spacecraft pointing accuracy during 3-Axis stabilized phase within $\pm 0.15^\circ$ in roll and pitch and $\pm 1.0^\circ$ in yaw.

Autonomous attitude acquisition.

Stationkeeping accuracy of $\pm 0.05^\circ$ north/south and east/west with orbit adjust intervals scheduled on a 21 day operational cycle.

Capable of a pitch slew manoeuvre of up to $\pm 0.5^\circ$ to accommodate a change of station longitude.

Eight year life.

3.2.3 Reaction Control Subsystem

Use of flight proven, high reliability subsystem components including catalytic hydrazine high thrust engines, with high performance and reliability low thrust engines (development soon to be complete), to provide very low weight propulsion subsystem.

High reliability, blow-down mass expulsion hydrazine monopropellant management employing flight qualified low weight surface tension tankage and latching valves for isolation.

Complement of two high thrust engines (HTE) and 16 low thrust engines (LTE) grouped downstream of four latching valves.

Thrusters positioned to efficiently perform north/south stationkeeping without requiring operational constraints on position of arrays during this manoeuvre.

Use of advanced performance, electrothermal hydrazine thrusters (EHT) for all low thrust functions except north/south and east/west station acquisition and keeping. Thrusters deliver Isp of 235 lbf.sec/lbm steady-state and repeatable linear impulse bits as low as 5×10^{-4} lbf.sec for optimum on-orbit attitude control.

Use of superheated electrothermal hydrazine thrusters for performing all on-orbit delta velocity functions of north/south and east/west station acquisition and keeping. Thrusters deliver Isp of at least 300 lbf.sec/lbm., operated in the steady-state mode, and their significant power requirements can be handled by existing power subsystem without impacting battery or solar array sizing.

Overall RCS weight for worst case (1985 launch) six year mission with tankage sized for worst case (1984 launch) eight year mission will be maximum of 218 lbs. (41 lbs. lighter than most weight optimum all catalytic RCS design).

RCS designed to be integrated to GPB primary structure of thrust tube and bulkheads to minimize cost and schedule and maximize reliability. Components are orientated to maximize ease of ground handling of fluids within the subsystem.

RCS accepts both regulated 27.5 V.DC and partially regulated main rail power and commands. Performs all valve and heater driving functions, power conditioning for secondary voltages and RCS telemetry signal conditioning.

3.2.4

DSA Subsystem

Use of high efficiency solar cells giving approximately 10% improved performance at end of life over typical CTS solar cells (however, only a 5%

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weight improvement assumed, to allow for cell layout/operating voltage constraints).

Array sized for a total radiation fluence of 1×10^{15} electrons per cm^2 of equivalent 1Mev electrons giving potential target life of up to 10 years in synchronous orbit.

Array capability 835 watts at 40 volts DC at end of above life (summer solstice) and 1160 watts at 40 volts at beginning of life (equinox).

In deployed state, the first natural frequency of the array will exceed 0.15 Hz.

Stowage and deployment system is a reliable conventional design using an aluminum honeycomb core/aluminum face sheet substrate and pantograph linkage deployment control system.

In stowed condition an average of 94 watts at 40 volts available with spacecraft spinning at 60 rpm.

Modular design allowing for removal of one panel per wing, reducing end of life power from 835 watts to 626 watts.

An available array orientation and power transfer system assumed in baseline design.

3.2.5 Structure Subsystem

Light weight structure consisting of

- thin wall central thrust tube
- four bulkheads in cruciform arrangement
- north, south and forward, thin face sheet, aluminum honeycomb equipment platforms
- east, west and aft closure panels

Above structural elements assembled in a closed box of high stiffness with bulkhead/panel/platform attachment by integral, contoured, thinly machined edge members using screws and replaceable locking inserts.

Apogee motor carried within the thrust tube.

RCS components integrated directly to primary structure.

Modular design to accommodate variety of payloads on the north, south and forward equipment platforms.

3.2.6 Thermal Control Subsystem

Passive thermal design supplemented with commandable thermostatically controlled electric resistance heaters, and simple construction, space proven heat pipes integrated into the North and South equipment platforms.

Light weight, high efficiency multi-layer insulation blankets used to minimize heat loss/input to spacecraft.

Caters for high power dissipation transponders and high power density TWTs (up to 50 watts per TWTs).

Batteries maintained within 0 to 10°C.

3.2.7 Apogee Kick Motor

Capable of providing velocity increments sufficient for placing spacecraft weighing 1925 lbs into synchronous orbit.

Motor can be overloaded by 10% with additional propellant enabling a 2125 lb satellite to be injected into synchronous orbit without any major redesign of motor.

Motor size has been optimized for a transfer orbit inclination plane of 28.3°.

3.2.8 System Parameters

General Purpose Bus System parameters are identified in Table 3-1. Weight breakdown for the bus and complete spacecraft are contained in Tables 3-2 and 3-3 respectively. Power requirements are identified in Table 3-4, and payload to spacecraft, weight and power ratios are presented in Table 3-5.

TABLE 3-1 GENERAL PURPOSE BUS - SYSTEM PARAMETERS

<u>LAUNCH VEHICLE</u>	:	Thor Delta 3914
<u>DESIGN LIFT-OFF WEIGHT</u>	:	
<u>CAPABILITY</u>	:	1925 lbs (excluding adapter)
<u>LIFETIME</u>	:	6 years (RCS Fuel)
		8 years RCS Tanks and Solar Array
<u>MAX. ARRAY POWER</u>	:	1160 watts BOL
<u>MIN. ARRAY POWER</u>	:	835 watts EOL
		94 watts Spin Phase
<u>BATTERY POWER</u>		560 watts (Longest eclipse)
<u>PAYLOAD CARRYING CAPACITY</u>	:	222 lbs
<u>PAYLOAD POWER</u>	:	625 watts EOL
		465 watts Eclipse
<u>SPACECRAFT POINTING ACCURACY</u>	:	+0.15° Roll and Pitch
		+1.0° Yaw
<u>STATIONKEEPING ACCURACY</u>	:	+0.05° North-South and East-West

TABLE 3-2 - G.P. BUS WEIGHT BREAKDOWN

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

*See Table 3-3 for security box and battery weights for each Payload Option

TABLE 3-3 - WEIGHT SUMMARY: GENERAL PURPOSE BUS AND PAYLOAD OPTIONS

PAYLOAD	UHF + 4-6 GHz	UHF + 12-14 GHz	UHF + 7-8 GHz + EXPERIMENTS	UHF + SHF & L BAND	24/4-6 GHz
DEFINITION	a	b	c	d	e
POWER	414.9S	624.6S	505.0	513.0S	417S
REQUIREMENTS	339.3E	463.5E	305.0	373.0E	347E
A) PAYLOAD - TRANSPONDER & ANT.	218.4	180.5	221.9	190.4	231
Antenna included in A	(55)	(55)	(55)	(60)	(55)
B) BATTERY TO OPER. A	74.8	102.08	67.1	82.06	77
C) SECURITY BOX MIL. VER.	10	10	10	10	0
X TOTAL (A + B + C)	303.2	292.58	299.0	282.46	308
Y TOTAL (BUS)	1587	1607	1607	1607	1587
TOTAL (X + Y)	1890.2	1899.58	1906	1889.46	1895
LIFT OFF WEIGHT ALLOWED	1925.0	1925.0	1925.0	1925.0	1925.0
(CONTINGENCY)	+34.8	+25.42	+19.0	+35.54	+30.0
<u>Note:</u>					
TOTAL ARRAY (INCL. HOUSEKEEPING)	100	120	120	120	100
TOTAL BATTERY & PC (56 + B)	130.8	158.08	123.1	138.06	133

TABLE 3-4 - POWER REQUIREMENTS - G.P. BUS AND PAYLOAD OPTIONS

Subsystem	Payload (a) Synch. Power-W	Payload (b) Synch. (w)	Payload (c) Synch. (w)	Payload (d) Synch. (w)	Payload (e) Synch. (w)
Communications	415	625	505	513	417
Telemetry and Command	30	30	30	30	30
Power	20	20	20	20	20
Battery Charging	30	30	30	30	30
Harness	5	5	5	5	5
DSA (Tracking, including Electronics)	10	10	10	10	10
ACS	25	25	25	25	25
RCS	15	15	15	15	15
Thermal Heaters	20	20	20	20	20
Total	570	780	660	668	573
Contingency	20	20	20	20	20
Design Total	590	800	680	688	593
Array Power Capability	626	835	835	835	626

3-13

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TABLE 3-5 - PAYLOAD/SPACECRAFT, WEIGHT AND POWER RATIOS

	Payload (a)	Payload (b)	Payload (c)	Payload (d)	Payload (e)
Payload weight	218	181	222	190	231
Spacecraft Total Weight	1890	1900	1906	1889	1895
Ratio Payload/Spacecraft Weight	.12	.10	.12	.10	.12
Payload Power Requirements	415	625	505	513	417
Spacecraft Total Power Requirements	590	800	680	688	593
Ratio Payload/Spacecraft Power	.70	.78	.74	.75	.70

4.0 FURTHER SUBSYSTEM IMPROVEMENTS

The selected configuration of the baseline GPB has recognized the current state of development of technology and components. It has also recognized, with regard to improvement in baseline performance, developments that are being currently undertaken, contemplated or which appear feasible. Where such developments could show some significant improvement in the payload carrying efficiency of the GPB at a reasonable risk or cost they have been noted, and where practical to do so, without undue compromise to the baseline configuration; the design of the baseline has allowed for adjustment and growth.

Some of these improvements together with design optimization improvements, are discussed below with applicable weight savings and problems identified.

4.1 Reaction Control Subsystem

The catalytic/electrothermal (with superheat) hydrazine RCS design presented in Volume II, Section 2.1.2.7 has been based on conservative values of major parameters. Many trade-offs remain to be performed to optimize the design, all of which will result in further subsystem weight reductions. Among these are:

- a) The superheated engine Isp assumed was 300 lbf. sec/lbm at 7-8 watts/lbf. It is expected that when development has been completed, the engine will provide at least 320 lbf. sec/lbm at power levels acceptable to the present battery and array design. This would save approximately 7.5 lb. This would also reduce engine life requirements.
- b) Further plume impingement studies may reveal a more optimum north/south stationkeeping thruster placement or alignment which could save up to several pounds of hydrazine propellant and reduce engine life requirements.
- c) The present design conservatively assumes that, because of expected Centre of Mass uncertainties and engine thrust mismatch, 10% of the north/south and 25% of the east/west station acquisition and

keeping fuel would be provided by non-superheated electrothermal engines at an Isp of 235 lbf.sec/lbm. Reduction in these uncertainties and mismatches may result in further weight savings of up to 5 lb and reduction in engine life requirements.

- d) The present design considers total north/south, 2 engine, thrust at beginning of life of 30 milli-lbf. which results in 0.7 to 1.5% inefficiency due to non-impulsive stationkeeping. Increased thrust level may be possible within the present battery solar array design which would reduce or limit this inefficiency thereby saving weight and reducing engine life requirements.
- e) The RCS has been configured about a WHECON ACS design. Additional RCS weight savings may be realized with the hybrid design through removal of offset thrusters.

4.2 Deployable Solar Array Subsystem

4.2.1 Advanced Stowage and Support System

The baseline General Purpose Bus DSA design utilizes an aluminum honeycomb substrate to support the solar cell array. Such a design has the advantage of being simple and proven. However, it may not be weight optimized for this size of array. As mentioned in Section 5.3 of Volume I of this report, this design was chosen because of the requirement for a high stiffness ($f_n > 0.3$ Hz) and minimum cost. In view of the reduced stiffness that is now acceptable ($f_n > 0.15$ Hz) a rigid frame/flexible substrate system can be designed using magnesium alloy tubes for the frame and thus be cost effective. It is estimated that such a design will give a 19 lb weight saving compared to the baseline design weight. Initial studies and build of a full size breadboard of such a system has already been undertaken by Spar under contract to D.O.C.

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4.2.2 Weight Optimized Orientation and Power Transfer System

The baseline assumes the use of a system that has been developed for another similar spacecraft application. A review of Reference 15 (Section 5.3.7, Volume I) shows that there are seven other systems that have a rating factor greater than six (the rating factor being an assessment of an applicability of these systems to the General Purpose Bus) and hence would yield a weight saving over the present baseline design. There would, however, be modifications required for some of the systems in order to be utilized on the General Purpose Bus. Allowing a weight for such modifications and taking an average of the resulting system weights shows that it should be possible to save 3.5 lb from the 22 lb presently allowed.

4.2.3 Less Conservative Radiation Degradation Allowance

Initially, as agreed with DOC, the rigid substrate design has been sized for a total fluence of 1×10^{15} e/cm² of equivalent 1 MeV electrons and the rigid frame/flexible substrate for a total fluence of 2×10^{15} e/cm².

However, for 6 mil cover slides and using cycle 19 data for solar flare protons, a maximum dosage of 7.7×10^{14} e/cm² for six years and 8.5×10^{14} e/cm² for eight years would not be unrealistic for an array with back shielding equivalent to the front (rigid substrate). Doubling these values for the flexible solar substrate design gives 1.54×10^{15} and 1.7×10^{15} e/cm² for six and eight years, respectively.

Note: It has been assumed that the entire cycle 19 solar flare proton fluence will be experienced in the first six years.

Using the above values instead of 2×10^{15} e/cm² for a flexible substrate design shows that for a six year life there is a 9.6% improvement and for an eight year life there is 6.6% improvement in cell

performance. Assuming that for reasons of cell layout and operating voltage, only half these improvements can be utilized, a further weight saving of 3.8 lb for a six year optimized design and 2.6 lb for an eight year optimized design can be effected for the rigid frame flexible substrate array.

4.3 Structure Subsystem

Significant structure weight savings are possible by using more advanced design and manufacturing techniques than assumed for the baseline design. Two such techniques are:

4.3.1 Use of Prebonded Honeycomb Sandwich Core

Prebonded honeycomb sandwich core is available from at least one manufacturer (Hexcel). Prebonded core eliminates the need to use, and hence the weight of, a separate layer of bond (FM-123) between the core and each facesheet.

When this technique is used on all panels, the weight saving per spacecraft is estimated to be 4 lb, approximately 4% of the total structure weight. A certain amount of development testing will be required to establish the necessary confidence level in areas of concentrated and high intensity loading.

4.3.2 Honeycomb Sandwich Construction Thrust Tube

A weight saving of 9 lb, approximately 9% of the total structure weight is possible by using a honeycomb sandwich construction thrust tube. Both the forward and aft portions of the thrust tube can be designed using this method of construction. It appears, however, that the weight saving is more effective for the most highly loaded aft thrust tube portion, being approximately 7 lb of the total 9 lb saving. There is no degradation of strength levels relative to the more conventional monoplly magnesium design, however, the first spacecraft lateral frequency for the hard-mounted

spacecraft is reduced by approximately 9 Hz to approximately 16 Hz. This is still satisfactory relative to the 15 Hz specification requirement; also, the total effect on the natural frequency of the spacecraft-adaptor combination will be significantly less than the 9 Hz difference mentioned above for the hard-mounted spacecraft. A development program will be required to establish manufacturing techniques, as well as strength and stability margins.

4.4 Apogee Kick Motor

As indicated in Section 3.7, the apogee motor design has a 10% propellant overload capability (without major redesign of the motor) enabling it to inject into geo-synchronous orbit a 2125 lb. spacecraft - a 110 lb. increase in usable spacecraft weight should an equivalent increase in launch vehicle capability occur.

4.5 Battery

Present design assumes a battery efficiency of 5 watts per lb. giving a total battery and power conditioning subsystem weight of 158 lbs. Any future improvements in battery efficiency will have a considerable impact on available bus payload weight.

4.6 Thor Delta Third Stage

Improvement in Delta third stage payload capability of 60 lbs, is being considered by Thiokol, through increased capability of the 364-4 third stage motor. Such an increase could result in an additional 30 lbs. (approximately) of usable payload weight.

4.7 Ion Engines

An Ion engine review is presented in Volume II, however, Ion engines for stationkeeping have not been considered practical for the baseline configuration, despite a promised weight saving of up to 40 lb. Should sufficient flight experience produce confidence in these engines, their use could be reconsidered for the latter part of the GPB utilization period.

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TABLE 4-1

G.P. BUS - POTENTIAL IMPROVEMENTS

SUBSYSTEM	POTENTIAL IMPROVEMENT	WEIGHT SAVING (LBS.)
REACTION CONTROL	o Superheated engine Isp increased from 300 to 320 lbf.sec/lbm.	7.5
	o More optimum North/South stationkeeping thruster placement	unknown
	o Reduction in centre-of-mass uncertainties and engine thrust mismatch	up to 5 lb.
	o Reduction in inefficiency due to non-impulsive stationkeeping	unknown
	o Optimum thruster complement for hybrid Attitude Control Subsystem	unknown
DEPLOYABLE SOLAR ARRAY	o Advanced stowage and support system	19
	o Weight optimized Orientation and Power Transfer System	3.5
	o Less conservative radiation degradation allowance	3.8
STRUCTURE SUBSYSTEM	o Use of Prebonded honeycomb sandwich core	4.0
	o Honeycomb sandwich construction thrust tube	9.0
APOGEE MOTOR	o Can inject 2125 lb. spacecraft by 10% propellant overloading	none, unless launch vehicle capability increased
THOR DELTA 3rd STAGE	o 3rd Stage motor performance	30 lb.
Ion Engine	o Use for N/S stationkeeping	30-40 lb.

TABLE 4-2 - G.P. BUS WEIGHT BREAKDOWN (BASELINE)
COMPARED WITH FURTHER SUBSYSTEM IMPROVEMENTS INCLUDED

SUBSYSTEM	G-P B BASELINE (WEIGHT (LBS))	G-P B WITH IMPROVE- MENTS
TT&C & Ant., no security box	35	35
ACS Hybrid	54	54
RCS (Tanks, Pressurant and struts (Fuel - 6 years	218	210
Structure (including Payload Inserts, excluding RCS Struts and Brackets)	103	90
Thermal	21	21
Apogee Motor (Motor Fired 54) (Propellant & Inserts 903)	957	957
Array (D&TM Plus Solar Panel) 800 watts	120	94
Battery & PC (Housekeeping Portion)	56	56
Harness	35	35
Safe and Arm and Balance Weight	8	8
TOTAL BUS	1,607	1,560

5.0 PERFORMANCE EVALUATION

The performance characteristics of the General Purpose Bus are evaluated by comparison with other similar capability spacecraft, such as those having the same launch vehicle and similar payload requirements.

Comparisons on the basis of available payload weight and power, and total spacecraft power, are made in Figures 5-1 and 5-2 respectively.

A qualitative evaluation of the relative advantages of the General Purpose Bus in terms of mission requirements, potential for performance improvements, reliability, design simplicity and adaptability to payload modification, etc., are discussed below by subsystem:

5.1 Attitude Control Subsystem

The major technological advantage of the General Purpose Bus attitude control subsystem lies in its use of a Microwave Attitude Sensing system, conceived by the Communications Research Centre. A two axis version has been developed to date for use on the Japanese Broadcast Satellite experiment. The roll and pitch accuracy of this sensor significantly exceeds that of any infrared horizon sensor presently in use.

The Attitude Control system configuration is unique in its use of this three-axis microwave attitude sensor together with a single magnetic bearing reaction wheel. Through these two components coupled with the relatively low impulse electro-thermal thrusters, a comparatively light weight, highly reliable and simple as well as a high pointing accuracy control system results. Subsystem weight and pointing accuracy compare very favourably with competing momentum wheel or three reaction wheel systems.

SPACECRAFT PAYLOAD CAPABILITY COMPARISONS

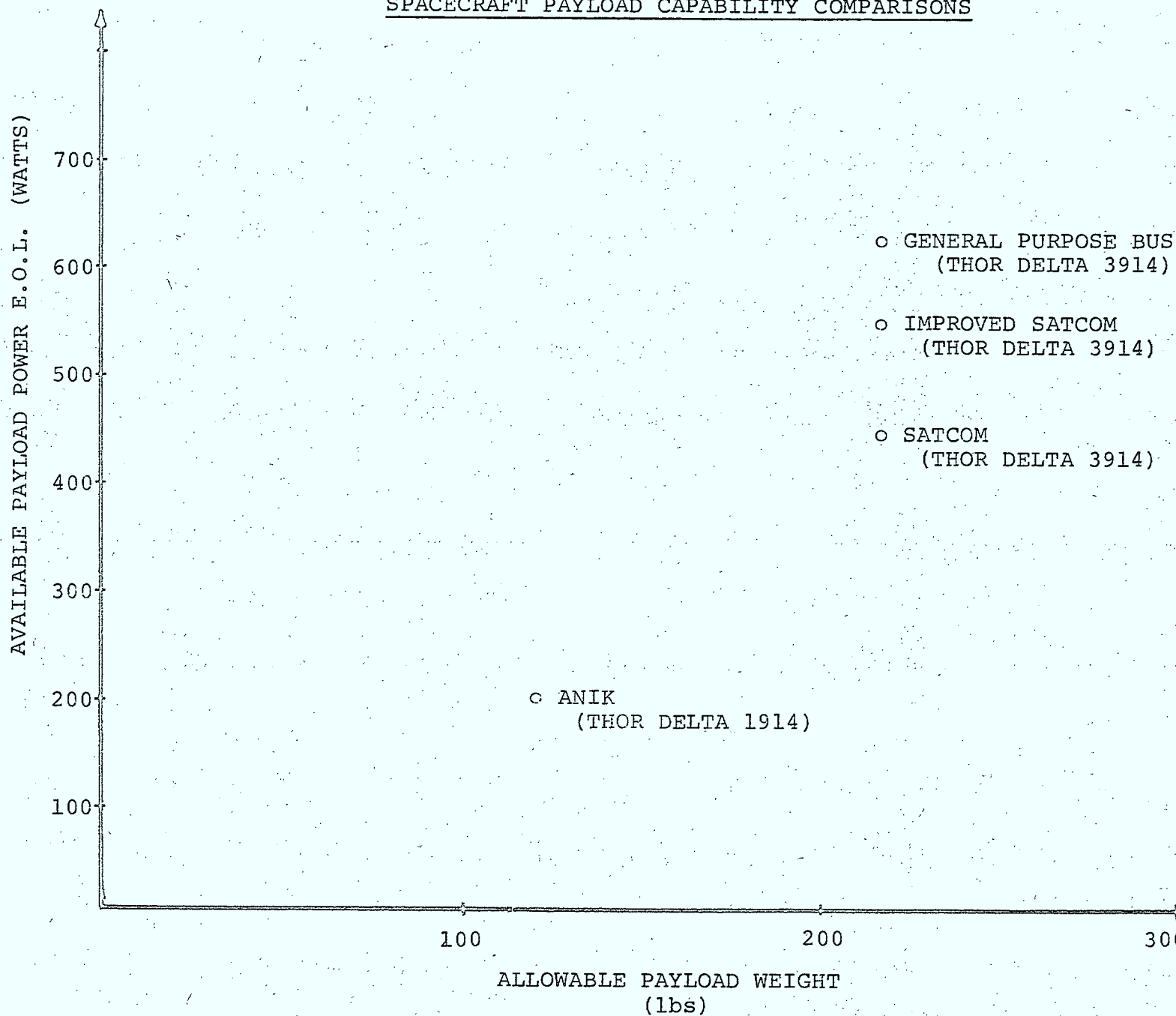
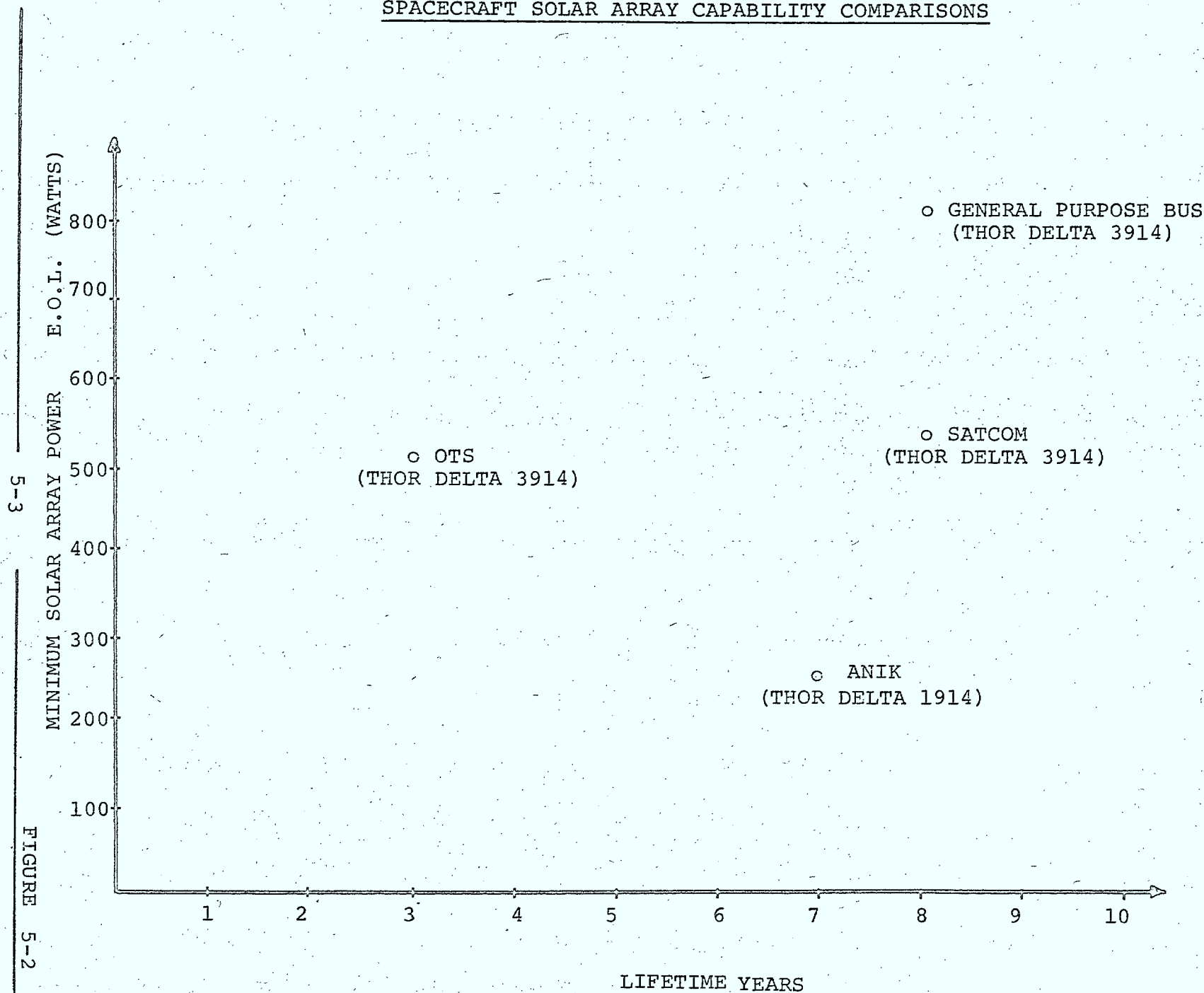


FIGURE 5-1

SPACECRAFT SOLAR ARRAY CAPABILITY COMPARISONS



5.2 Reaction Control Subsystem

Location of thrusters for north/south stationkeeping allows for use of the north and south payload platforms as primary thermal radiators. Spacecraft not presently designed for such high power dissipation transponders may not be presently incorporating this design feature and hence may require redesign for higher power payloads.

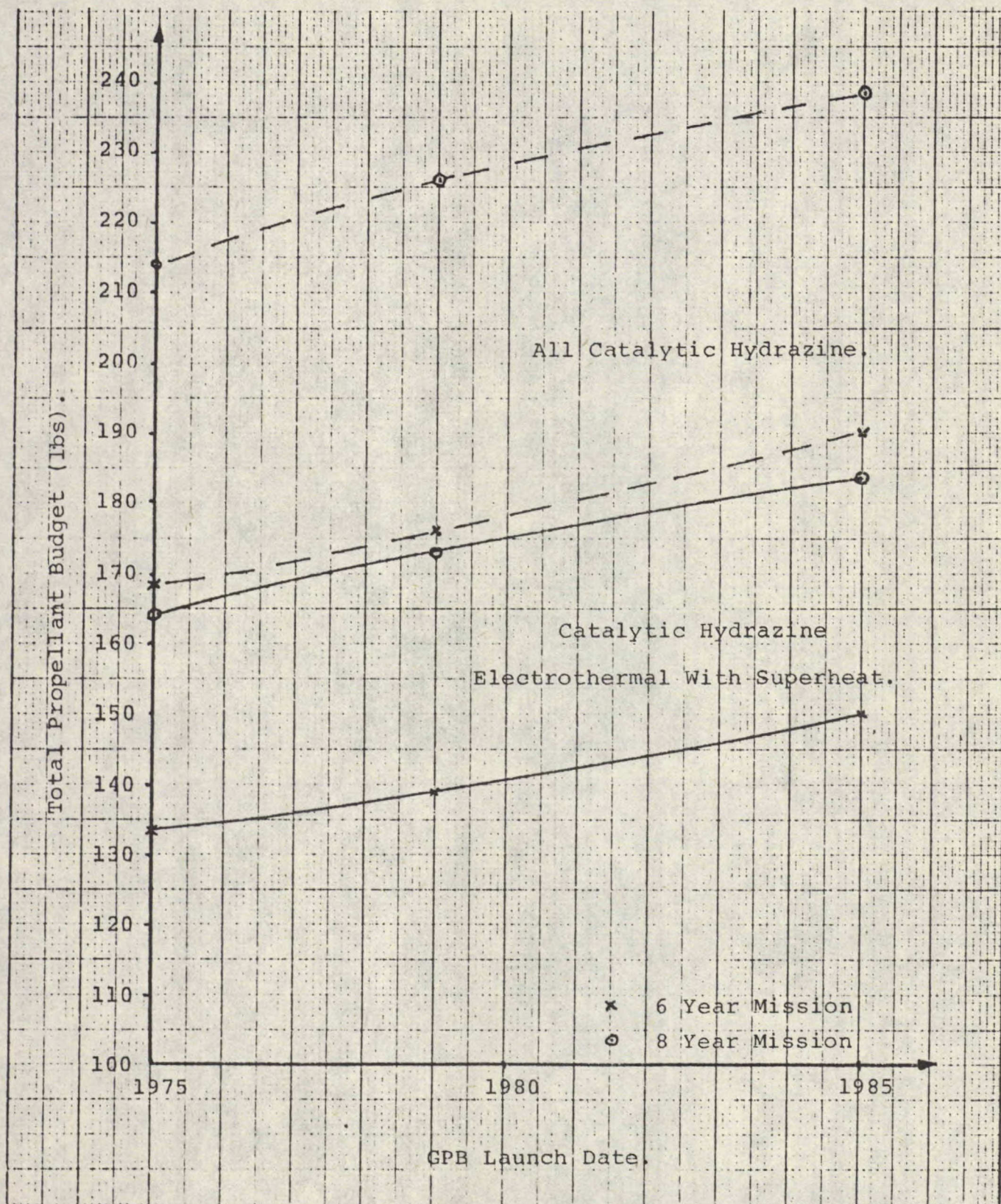
The G.P.B. is designed to provide fuel for a 6 year operational life (with tankage sized for 8 year missions) for launches between 1979 and 1985. As can be seen from Figure 5-3 the fuel requirements increase as a function of launch date in the time frame 1975 to 1985. This is due to the increasing delta velocity requirements for north south stationkeeping. Hence spacecraft designed for launch prior to 1985 will require a significant increase in RCS fuel to meet north/south stationkeeping requirements of missions scheduled for later launches.

The GPB RCS is taking advantage of rapidly developing advanced high performance hydrazine thrusters which reduce the fuel required to the quantities shown in Figure 5-3. The superheated engines used for performing stationkeeping, require significant electrical energy, approximately 425 watt hours per manoeuvre (2 burns) for time periods in excess of 2 hours. This requirement can be met by the GPB power subsystem without impacting payload requirements.

Spacecraft not presently using such superheated engines will probably require modification to their power subsystems should they wish to take advantage of RCS fuel savings by converting to the above type of RCS design.

5.3 DSA Subsystem

The deployable solar array subsystem as designed for the General Purpose Bus has certain unique features that make it a versatile system. These can be generalized as follows:



- a) The design is modular, allowing a stepwise change of power producing capability to cater for different spacecraft payload requirements.
- b) As designed, there is more than an adequate margin for an eight year life with a potential for ten years.
- c) The cell layout provides a 0 magnetic moment.
- d) The array is elevated sufficiently away from the spacecraft to avoid shadowing by the 30 ft diameter UHF antenna, except at certain times around midnight during solstice conditions. This elevation provides a thermal view factor for the spacecraft radiating platforms better than 0.9.
- e) Transfer of power is provided by part of the same array that provides power in synchronous orbit.
- f) The stowage and support system is based on well proven techniques. The system is readily adaptable to more advanced designs such as the rigid frame/flexible substrate design.
- g) A through shaft, forward platform mounted orientation and power transfer system is used. This is a reliable and modular design and permits ease of integration and testing.
- h) In the deployed condition, the array features high stiffness at a minimized distortion compatible with the spacecraft attitude control subsystem.
- i) The design allows a growth capability of at least one further panel per side, or additional 200 watts end of life.
- j) Further optimization of cell layout, cell performance, radiation degradation allowance and orientation and power transfer system will reduce the weight and size of the system further.

5.4 Structure Subsystem

The main advantage of the structure subsystem lies in its ability to accommodate, (and efficiently mass balance by adjustment of designated housekeeping components) a variety of payloads on the three main payload platforms, without any modification to the spacecraft primary structure.

5.5 Thermal Subsystem

One of the major advantages of the General Purpose Bus over its competitors is the fact that the thermal control subsystem has been designed for high power dissipation transponders with very high power density components. The latter require either the use of heat pipes or relatively large mass thermal doublers to laterally distribute dissipated power, hence maintaining acceptable component temperatures. Spacecraft which do not presently have the above types of transponders/components will require considerable modification to their thermal subsystems to accommodate these, and hence impact overall spacecraft weight, mass balance and payload location.

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APPENDIX A

GENERAL PURPOSE BUS PROGRAM

CANADIAN CAPABILITY - GROSS ASSESSMENT

INTRODUCTION

The implementation plan of the General Purpose Bus program (presented in Volume III) details program tasks and schedule and identifies facilities required for testing. This Appendix presents a gross assessment of the capability in Canada to carry out this program, particularly with regard to available manpower, expertise, and facilities available to complete the tasks.

MANPOWER

The mechanical space engineering capability in Canada is mainly centred at Spar. This is historically derived from the role played by Spar on all Canadian designed and built satellites to date, plus 15 years experience in the design and fabrication of structures and mechanisms for both manned and unmanned space missions throughout the world (see Figures A-1 through A-7). Continuity of this role is currently being sought, both in satellite and remote manipulator systems (RMS) applications.

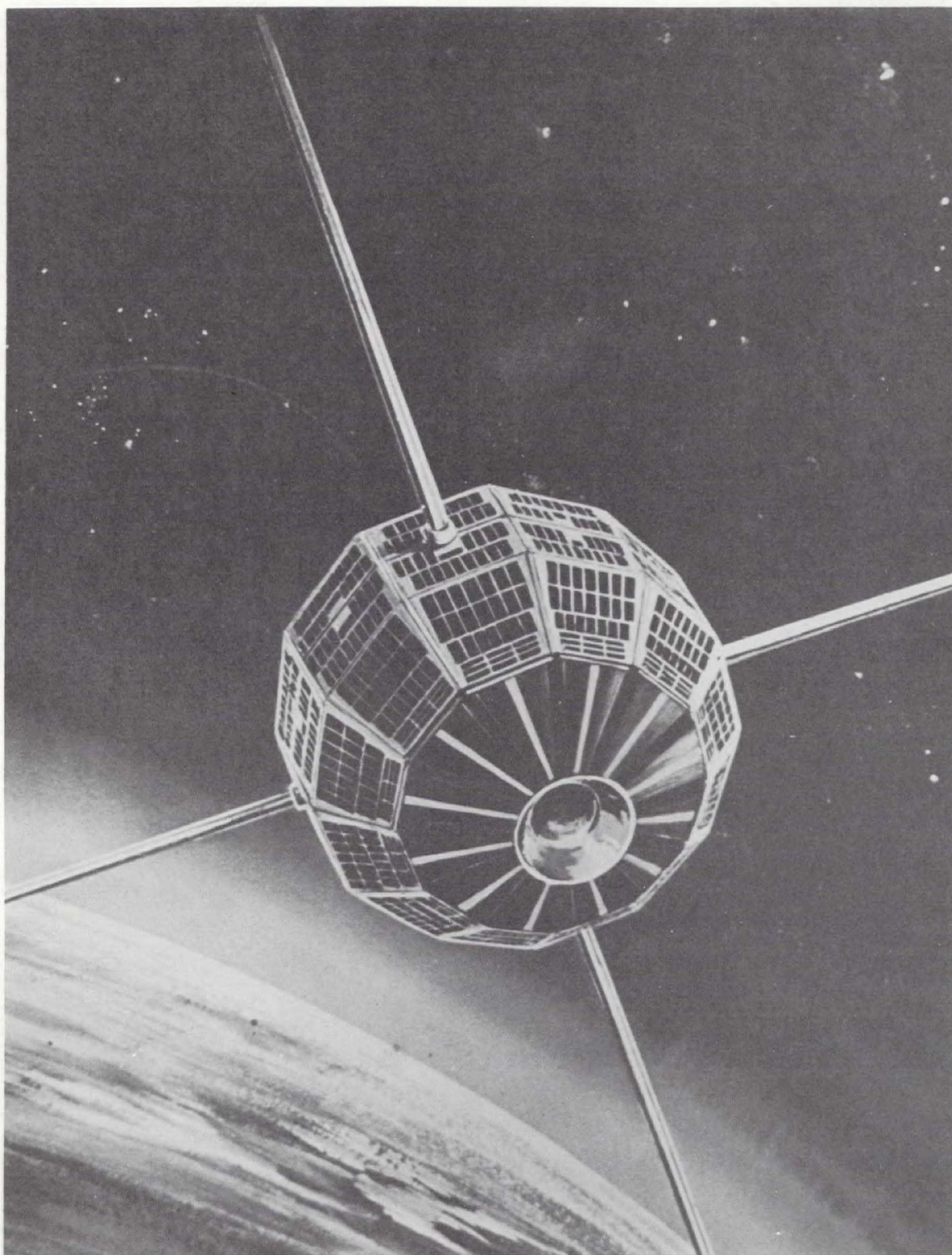
The RMS and satellite technology resource requirements are similar and each mutually supports the other in maintaining a centre of excellence for structural and mechanical space engineering at Spar.

These skills, when augmented by other Canadian expertise and resources at Communications Research Centre (CRC), SED Systems, Bristol Aerospace, Canadian Astronautics, University of Toronto Institute for Aerospace Studies (UTIAS) etc., which have been developed by Canadian investment in domestic satellite programs over the years, represent a solid base for carrying out the program outlined in this report. In those areas where additional staff are required to satisfy peak demands, it is considered to be quite feasible to rent consulting staff and hire additional staff, provided that sufficient advance notice is given of the program start-up, and funds are approved to make such commitments.

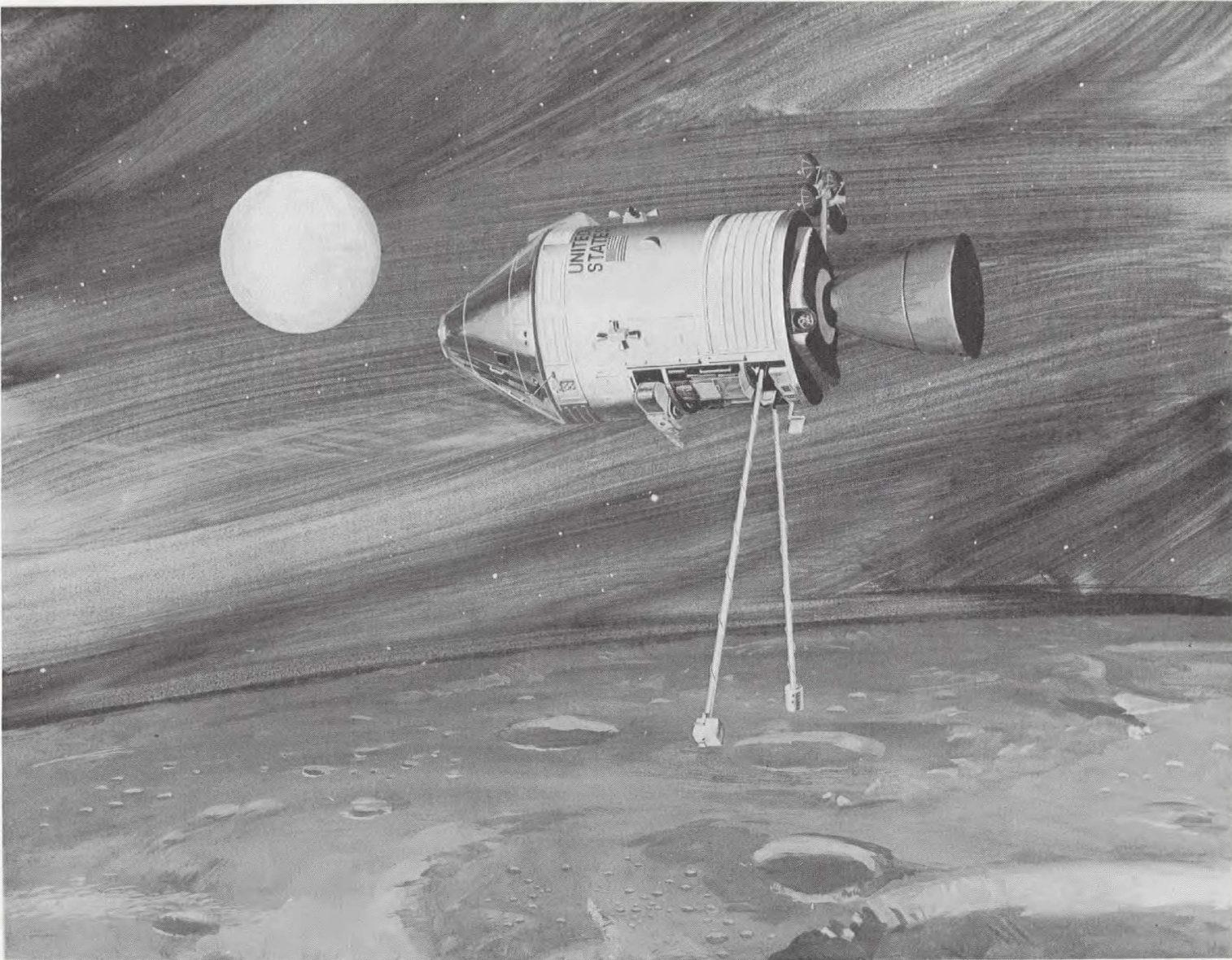
A gross indication of manpower needs for the GPB and RMS against manpower currently available is shown in Figure A-8, and illustrates this point. However, it should be noted that the availability of manpower will change as a function of the availability of applicable work as time progresses and should therefore be viewed in that context.

FACILITIES

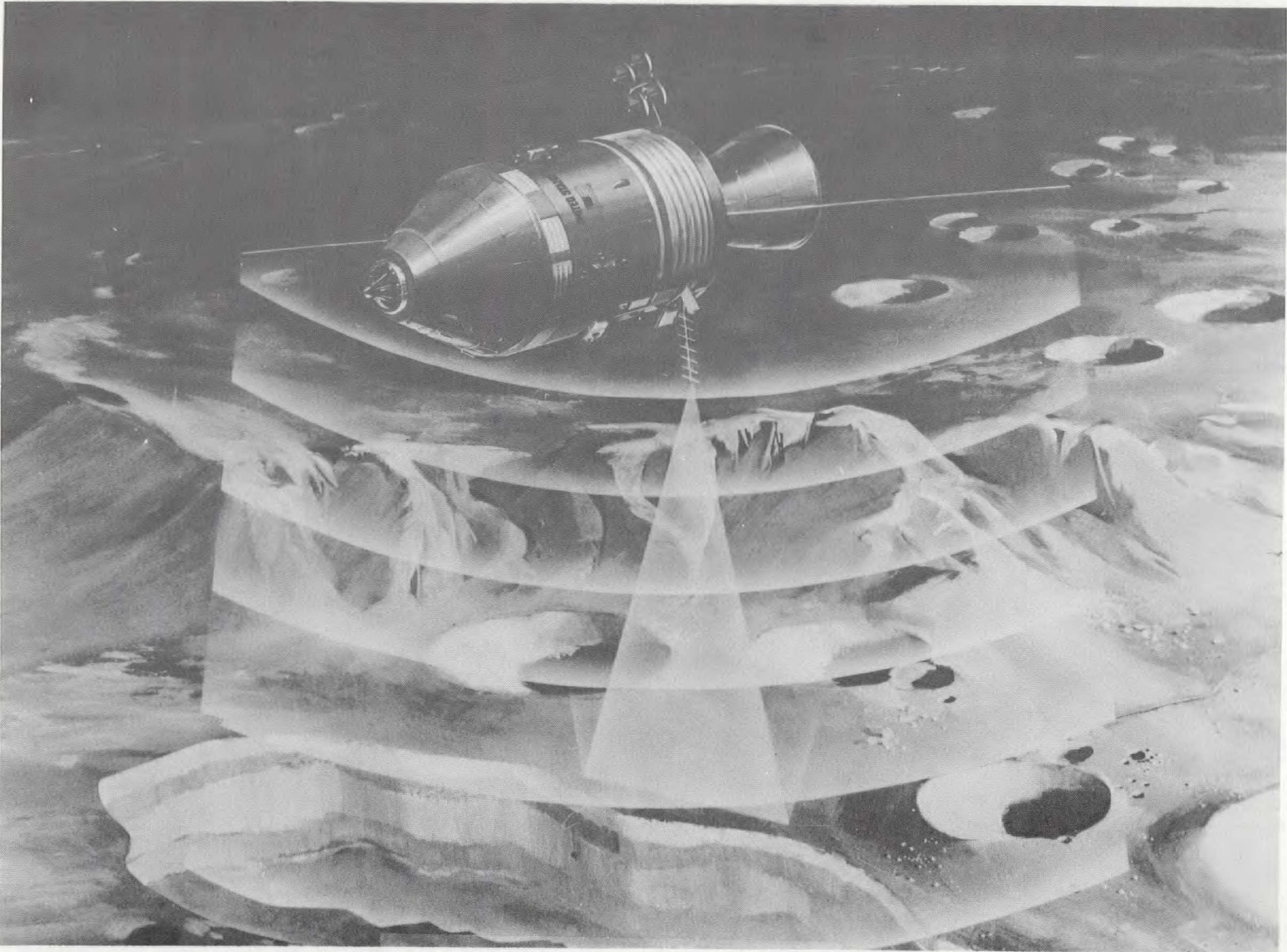
Facilities are available at the companies' plants mentioned for manufacturing, integrating and testing the GPB and subsystems; however, certain test requirements for the GPB program will of necessity require a certain amount of spacecraft level environmental testing to be conducted at outside facilities. The spacecraft assembly and environmental test facilities in the David Florida Laboratory at CRC should enable most component (if required), subsystem and system level tests for both the GPB and RMS programs to be performed in Canada, and extensive use will be made of this facility whenever possible. However, in the case of complex spacecraft level thermal balance vacuum tests, acceleration and acoustic tests, these will need to be conducted at U.S. facilities.



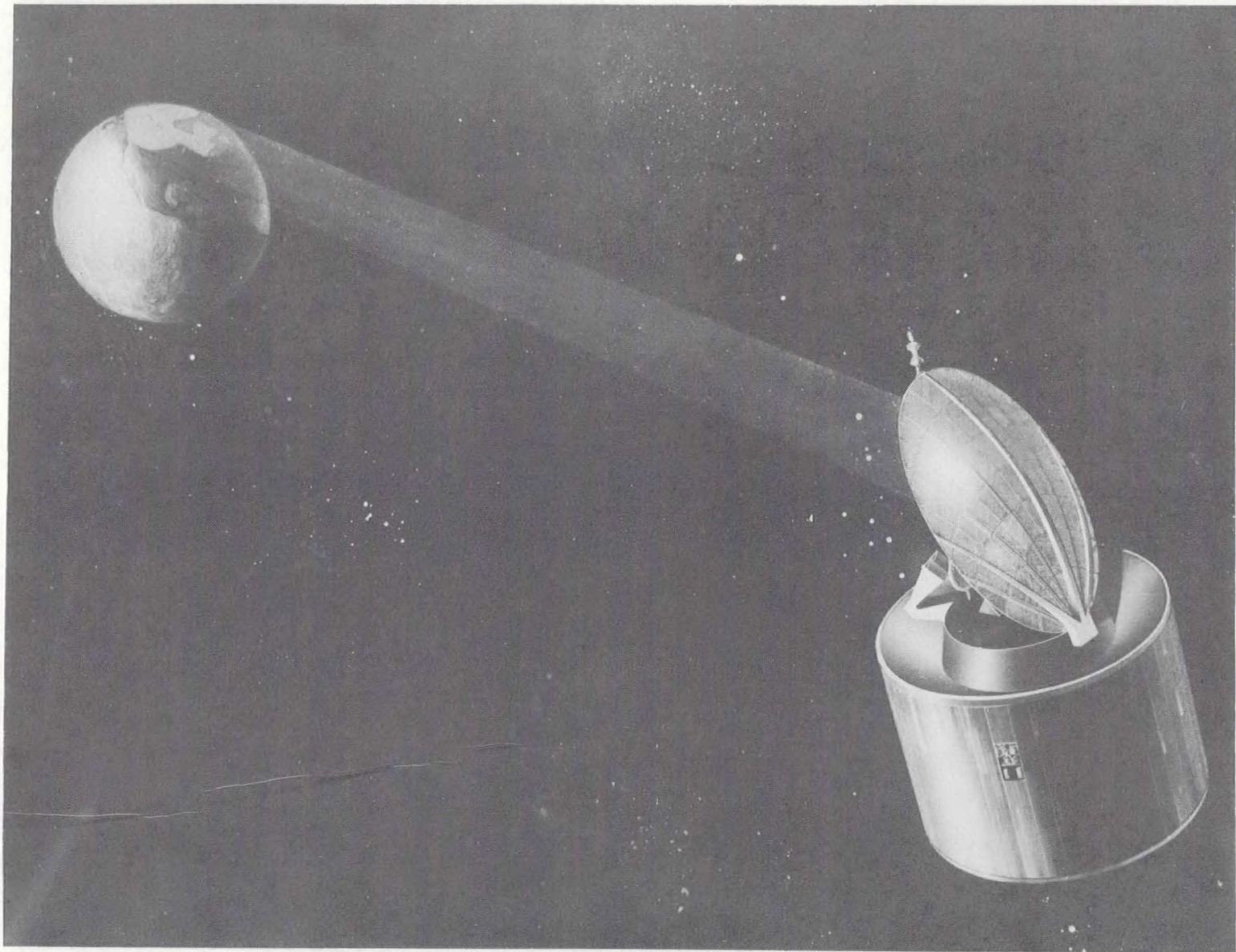
DESIGN AND FABRICATION OF ALOUETTE I AND II STRUCTURES AND EXTENDIBLE
SOUNDING ANTENNAS



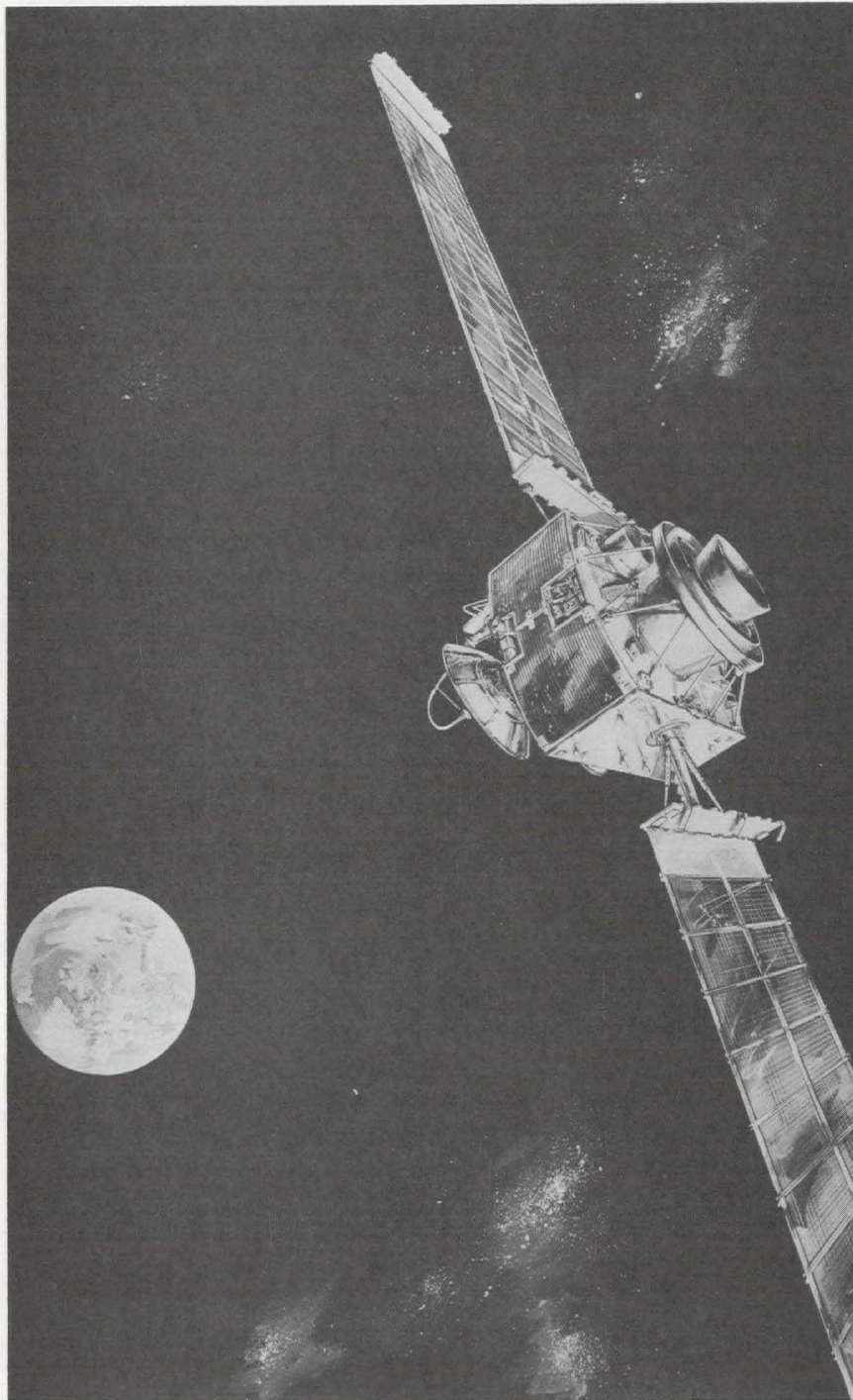
SPECTROMETER DEPLOYMENT SYSTEM ON APOLLO 15 and 16



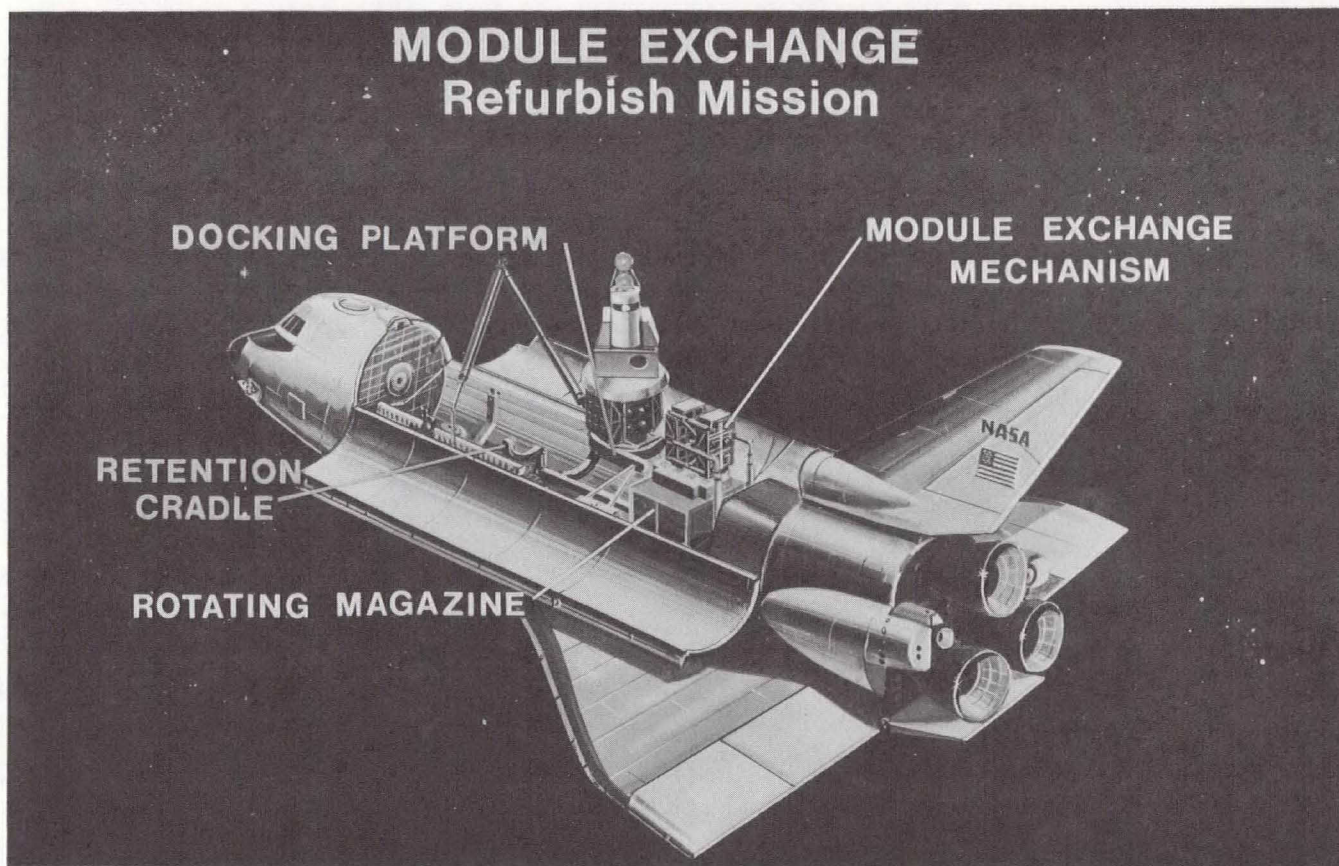
LUNAR SOUNDER ANTENNA SYSTEM FOR APOLLO 17.



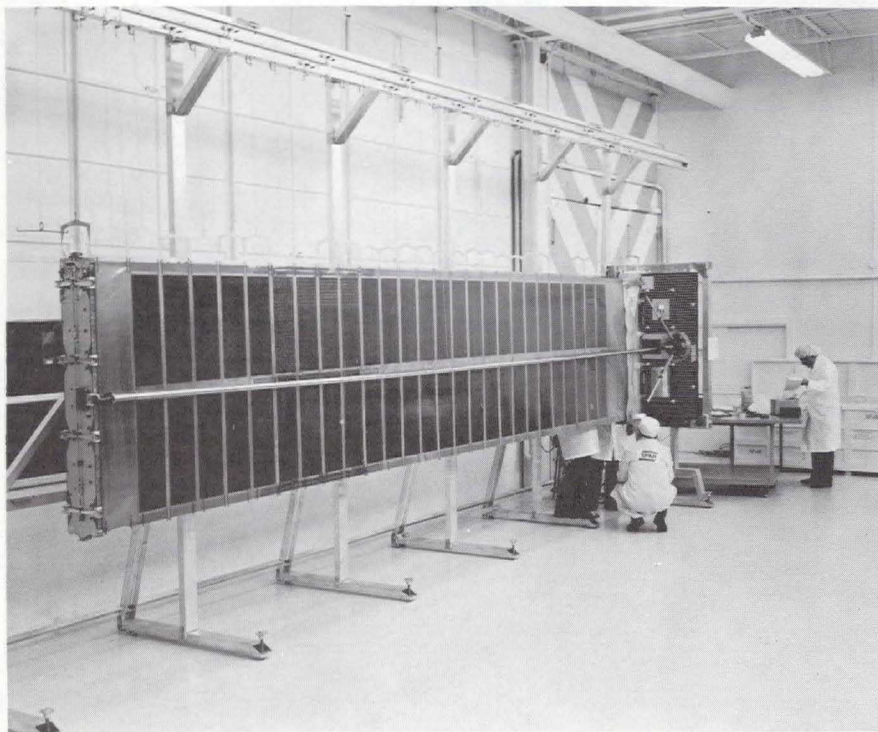
FEASIBILITY STUDIES, DESIGN SUPPORT AND STRUCTURE
FABRICATION FOR ANIK AND WESTAR COMMUNICATIONS SATELLITES



STRUCTURE, THERMAL, POWER GENERATION & ATTITUDE CONTROL
SUBSYSTEMS OF THE COMMUNICATIONS TECHNOLOGY SATELLITE.

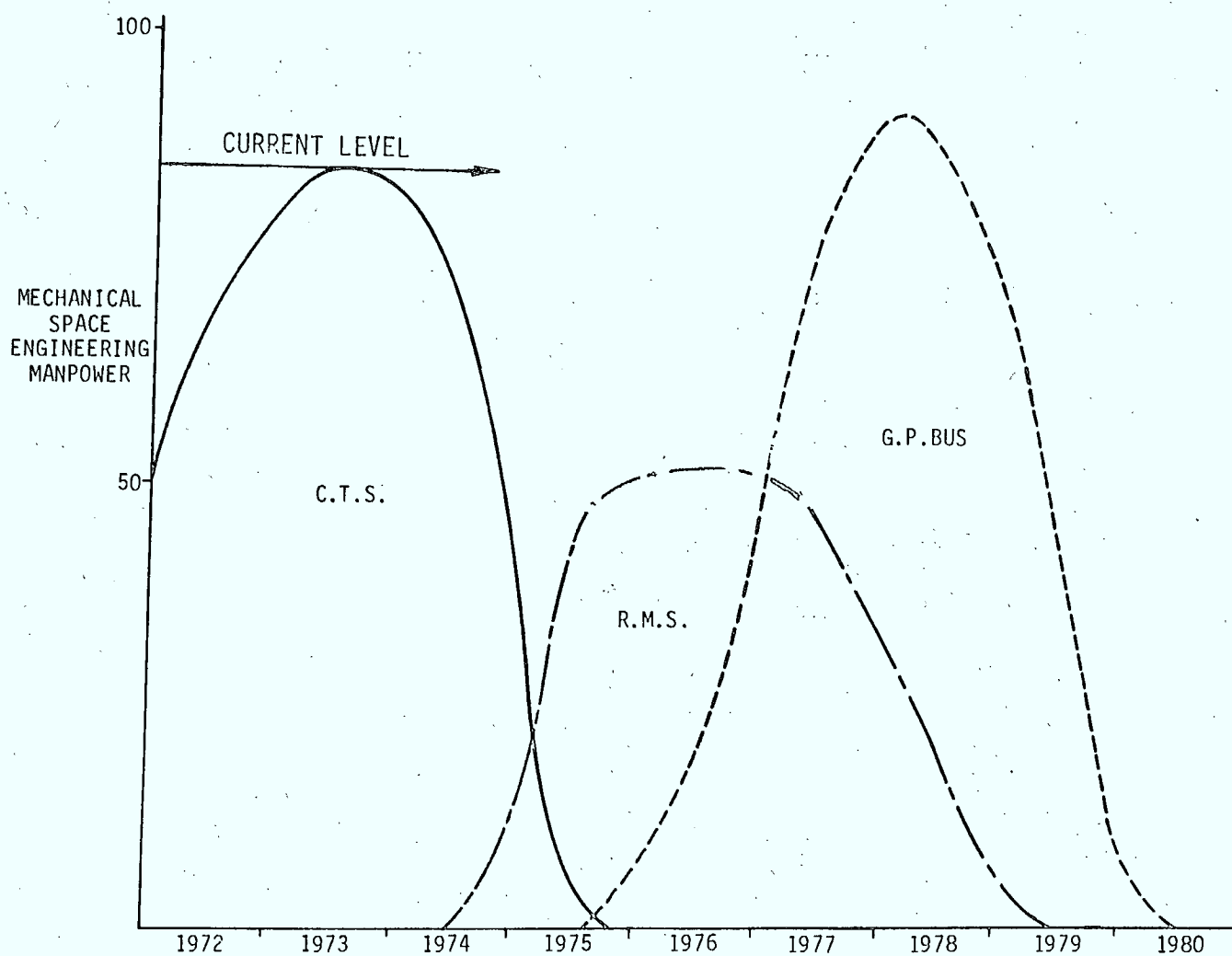


REMOTE MANIPULATOR SYSTEMS FOR SHUTTLE



DESIGN, DEVELOPMENT AND FABRICATION OF SPACECRAFT
SOLAR ARRAY POWER GENERATION SYSTEMS.

SPACE RELATED MECHANICAL ENGINEERING SKILLS AT SPAR





LOWE-MARTIN No. 1137.

