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November 1976

Standard Spacecraft Economic Analysis, Volume 2 Final Report of Findings and Conclusions

E. D. Harris, J. P. Large

A report prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Rand
SANTA MONICA, CA. 90406

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PREFACE

This is the final report on a study of the comparative program costs associated with use of various standardized spacecraft for Air Force Space Test Program missions to be flown on the space shuttle during the 1980-1990 time period. The first phase of the study considered a variety of procurement mixes composed of existing or programmed NASA standard spacecraft designs and a new Air Force standard spacecraft design. The results were briefed to a joint NASA/Air Force audience on July 11, 1976. The second phase considered additional procurement options using an upgraded version of an existing NASA design. The results of both phases are included in this report. An executive summary of the study, R-2099/1-NASA, *Standard Spacecraft Economic Analysis, Vol. 1: Executive Summary*, is available from The Rand Corporation as a companion report.

The results of the study should be useful to NASA and Air Force space program offices involved in operational or experimental missions. They should also be of interest to those concerned with the determination of the shuttle tariff rate structure or with shuttle operations, because the impact of a variety of tariff rates is examined.

Although the study examines procurement options that affect both NASA and Air Force programs, the results should not be interpreted as representing official views or policies of NASA or the Air Force.

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I. INTRODUCTION

The purpose of this study was to examine the relative costs of using one or more of several possible standard spacecraft for Air Force Space Test Program missions during the initial 10-year operational period of the space shuttle. During the first phase of our study we considered the Space Test Program Standard Satellite (STPSS)--a design proposed by the Space Test Program Office of the Air Force Space and Missile Systems Organization (SAMSO), and two NASA candidates--the Applications Explorer Mission spacecraft (AEM) and the Multimission Modular Spacecraft (MMS). After the initial study phase was completed a fourth candidate was introduced--a larger and more capable AEM (L-AEM) configured by the Boeing Company under NASA sponsorship to meet specifications jointly agreed upon by NASA and the Air Force. The evaluation of that spacecraft is also included in the results of this study and procurement options derived using all four spacecraft are compared for the Space Test Program missions. The study was funded by NASA and conducted with the full cooperation of both NASA and the Air Force.

In the past the Space Test Program Office has procured spacecraft as required for specific missions. Generally, that has meant that a new spacecraft was designed and developed for each new mission. The Space Test Program Office has tried to reduce the cost of these spacecraft by requiring that: (1) the contractor use flight-proven components whenever possible; (2) a minimum amount of demonstration testing be done; (3) high technology solutions be avoided; and (4) the institutional aspects of the program, e.g., program office size, be minimized. To date the Standard Test Program Office has been very successful in developing spacecraft at a cost substantially lower than the experience of more traditional programs would lead one to expect.

Recognizing that a standard spacecraft produced in accord with these principles could generate substantial savings, the Space Test Program Office contracted for a spacecraft configuration study by TRW,[1] which was used as the baseline configuration for this study. Associated studies of other aspects of the STPSS operation and design were also available.[2-4]

Concurrent with the Air Force activity, NASA has for the past six years been working on another standard spacecraft configuration, the MMS.[5] Many of the low-cost aspects of the Space Test Program concept are a part of the MMS design and operational philosophy as well. The principal distinction is an emphasis by NASA on spacecraft retrieval and on-orbit servicing that would be possible with a space shuttle. That has resulted in a spacecraft design with capabilities exceeding those necessary for the Air Force Space Test Program missions. The MMS program is ahead of the STPSS chronologically--some of its components have been developed, the design is firm, and contractor bids have been received. Thus the MMS will be developed at no cost to the Air Force, and it is reasonable to ask whether both the MMS and STPSS are needed.

The availability of the AEM further complicates the issue. The AEM is the furthest along in the development cycle. Boeing is under contract to NASA to develop and build AEM spacecraft for the Heat Capacity Mapping Mission (HCMM) and the Stratospheric Aerosol Gaseous Experiment (SAGE) and again, NASA is emphasizing low cost in the spacecraft design. Although the AEM is designed specifically for two missions, it has a modular design that makes it suitable as a standard spacecraft.

An additional complication is that the AEM can be upgraded to perform some or all projected Space Test Program missions, depending on the kind of attitude control subsystem used. The question, then, of which spacecraft would enable the Space Test Program Office to meet its mission responsibilities at the lowest cost requires a comparative analysis of program costs. This report describes such an analysis. Section II presents study objectives and guidelines. Section III describes the spacecraft configurations along with necessary modifications for use by the Air Force for Space Test Program missions, and Sec. IV discusses the Space Test Program mission model. The results of the cost analysis are summarized in Sec. V, where estimates of spacecraft nonrecurring and recurring costs and the costs of the various launch options are presented. Section VI summarizes the program costs and the results of the sensitivity analyses conducted, and the conclusions of the study are presented in Sec. VII. Separate appendixes briefly cover the spacecraft and program cost analysis and the technical assessments of the relative state of the art of the major spacecraft subsystems in the AEM, STPSS, and MMS.

II. OBJECTIVES AND GUIDELINES

The two objectives of this study were to develop internally consistent cost estimates for the AEM, L-AEM, STPSS, and MMS spacecraft and, using these estimates, to determine the variation in program cost for a variety of spacecraft procurement options capable of performing the Space Test Program missions during 1980-1990. The emphasis is on relative, not absolute, accuracy in the estimates developed. The conclusions that are drawn concerning the various procurement options, although discussed in terms of total program costs, are dependent upon the relative costs of the various spacecraft. They are not affected if the magnitude of the total program costs is under- or overestimated.

The study guidelines are summarized below:

1. Spacecraft configurations are based on descriptions provided by Goddard Space Flight Center (GSFC) for the MMS, by TRW for the STPSS, and by Boeing for the AEM and L-AEM.
2. Space Test Program payloads described in *Current STP Payloads* (the so-called "Bluebook")(6) are considered representative of those that would be flown during the period 1980-1990.
3. All spacecraft are compatible with the use of solid rockets for orbit translation, which usually requires spin stabilization. The AEM and STPSS are designed with that in mind. The MMS normally uses a hydrazine propulsion module or the Interim Upper Stage (IUS) for orbit translation in a three-axis stabilized attitude, but according to GSFC it can also be spin stabilized for orbit translation.
4. Space Test Program missions are intended to be flown as secondary payloads, which implies that Space Test Program payloads would rely on solid rocket kick stages* for translation from the nominal shuttle parking orbit to the desired mission

*Although the IUS uses solid rockets, its use by the Space Test Program is considered a special case because of the high cost of that design.

orbit rather than on changing the shuttle orbit altitude and inclination to meet the payload requirements.

5. Nominally, two Space Test Program flights per year are scheduled; the minimum is one.
6. All payloads are launched using the space shuttle.
7. Servicing of payloads in orbit or retrieval of spacecraft for reuse is not considered.

III. SPACECRAFT CONFIGURATIONS

SPACECRAFT REQUIREMENTS

The nominal spacecraft requirements for the AEM, L-AEM, STPSS, and MMS, categorized by mission, communication, electrical power, stabilization and control, and reaction control system and propulsion, are shown in Table 1. Of the four spacecraft, the AEM is the smallest and has the least capability. It is about 3 ft in diameter, weighs about 210 lb, has a 150 lb payload capability, and is limited to operating altitudes less than 1000 n mi.

The L-AEM design is a derivative of the AEM.[7] The AEM basic structure provides the core of the L-AEM; additional structure increases the diameter to 5 ft. Three different configurations of the L-AEM are available: the baseline option (L-AEM-BL), the spin stabilized option (L-AEM-S), and the precision option (L-AEM-P). All have a minimum life of one year and a payload capability of 1000 lb. Both the L-AEM-S and L-AEM-P options can operate from low earth orbit to geosynchronous altitude; the L-AEM-BL option is restricted to altitudes less than 1000 n mi. The L-AEM-BL weighs about 670 lb.

The STPSS has a nominal payload capability of about 1000 lb, can be operated at altitudes up to geosynchronous, and weighs about 860 lb. It can be procured in three different configurations--a spinning version (STPSS-S), a low-cost three-axis stabilized version (STPSS-LC), and a three-axis stabilized precision version (STPSS-P).

The MMS is the most sophisticated of the standard spacecraft considered in this study; it is designed for on-orbit servicing and reuse. It has a payload capability of about 4000 lb and can also be operated up to geosynchronous altitude. The MMS weighs about 1400 lb without the solar array or space propulsion system.

AEM and MMS spacecraft have communication systems that are compatible with the Space Tracking and Data Acquisition Network (STDN), while the L-AEM and STPSS are compatible with the Space Ground Link System (SGLS). This difference in the communication system needs to be corrected before the AEM and MMS can be used for Air Force missions.

Table 1

NOMINAL SPACECRAFT REQUIREMENTS

Requirement Category	AEM	STPSS	MMS	L-AEM
<u>Mission</u>				
Operating mode	Expendable	Expendable	On-orbit service	Expendable
Mission duration, yr	1	?	1-2	1
Design payload weight, lb	80	1000	4000	1000
Max. payload weight, lb	150	1500	4000	>1000
Orbit range	LEO[a] (<1000 n mi)	LEO-Geosynch.	LEO-Geosynch.	LEO-Geosynch.
<u>Communications</u>				
Link	VHF and STDN[b]	SGLS[c]	STDN and TDRSS[d]	SGLS
Data bit rate, kbps	8	128 and 256	64	128 and 256
Data storage, bits	4.5×10^8	1×10^8	9×10^8	9×10^8
<u>Electrical Power</u>				
Bus voltage, V	28 ±4	28 + 5	28 ±7	28 ±4
Battery capacity, Ah	10	60	20 to 150	40
Peak array, W	238	380-1200	As required (≤3600)	318-1000
Housekeeping, W	28	100-200	350	104-132
Battery charging	--	Individual charge	Parallel charge	Individual charge
<u>Stabilization and Control</u>				
Pointing accuracy, deg	±1, pitch, roll; ±2, yaw	±1-±0.1, all axes	<±0.01, all axes	±1-±0.05, all axes
Pointing stability, deg/sec	±.01	±.01-.003	$\pm 10^{-6}$	~ ±.01
Orientation	3-axis	Spinning and 3-axis	3-axis	Spinning and 3-axis
<u>Reaction Control System and Propulsion</u>				
Impulse, klb-sec	2.3	1.7-2000	12-230	2.8-1065
Type	Hydrazine	Cold gas, solids	Hydrazine	Hydrazine, solids

[a]Low earth orbit.

[b]Space Tracking and Data Acquisition Network.

[c]Space Ground Link System.

[d]Tracking and Data Relay Satellite System.

(The modifications necessary to make this correction are discussed later.) Another difference is in the data rate capability of the communication systems. Both the AEM and MMS have data rates considerably less than that of the L-AEM and STPSS, i.e., 8 and 64 kbps, respectively, as compared with 128 to 256 kbps.

All of the spacecraft use 28 V systems. The basic differences are in the solar array designs and battery charging systems. The AEM has a fixed solar array capable of providing about 40 to 50 W for experimental use. The other designs treat the solar array as a mission-specific item. The peak array power for the L-AEM is 1000 W, almost as much as the 1200 W of the STPSS output; the MMS power system can handle arrays having a peak output of up to 3600 W. The battery-charging system of the MMS is different from those of the L-AEM and STPSS. All three provide for more than one battery, but an individual charging system is used by the L-AEM and STPSS, whereas a parallel charging system is used for the MMS.

In stabilization and control capability, the MMS is again superior to the other spacecraft with a pointing accuracy of ±0.01 deg and a pointing stability of $\pm 10^{-6}$ deg/sec. The L-AEM design provides essentially the same variety of options for stability and control of the spacecraft as the STPSS. The spin stabilized options are identical in capability, while the capability of the precision option exceeds that of the STPSS-P but is less than that of the MMS. The L-AEM-BL option is more accurate than the STPSS-LC option in the pitch and roll axes and identical in the yaw axis.

Both the AEM and MMS have hydrazine attitude control systems; the STPSS uses a cold gas system in combination with solid rockets for orbit translation. The MMS hydrazine propulsion modules (SPS-I and SPS-II)* provide a choice of module configurations that can be selected depending upon the delta velocity required. The reaction control system used in the L-AEM is a derivative of the hydrazine system of the SAGE version of the AEM. The major difference is that the L-AEM-P configuration has a reaction control system sized to provide three-axis stability during the solid rocket powered orbital translation phase.

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*Space propulsion system (SPS).

Consequently, it includes nozzles with relatively large thrust levels (65 and 155 lb) in addition to the normal thrusters. There seems to be no reason why the L-AEM-P configuration cannot be spin stabilized during orbit translation, therefore we have assumed it has this capability, especially for the geosynchronous missions where larger size solid motors are required than those discussed in Ref. 7. In Ref. 7 the overall length of the L-AEM, payload, and solid rocket kick stages was restricted to less than the diameter of the shuttle. This allowed placement of the spacecraft perpendicular to the shuttle longitudinal axis and hence minimized the length of the shuttle bay used for the flight. We have not restricted our application of the L-AEM in this manner.

The individual spacecraft configurations and the modifications considered necessary to allow their use by the Air Force in carrying out the Space Test Program missions are described below.

AEM

As mentioned earlier, there are two basic AEM configurations--HCMM and SAGE--which consist of the same base module with different mission-specific equipment. The HCMM configuration uses a hydrazine orbit-adjust module, while the SAGE configuration includes a second momentum wheel and a tape recorder..

For Air Force use we selected the SAGE configuration as being most appropriate. The only modifications that were considered relate to the conversion of the communication system to make it SGLS-compatible. These changes are itemized below and discussed in detail in Appendix C. Basically, the changes involve replacing some of the AEM communication equipment with the appropriate STPSS communication equipment.

- Replace S-band transmitter with STPSS S-band (SGLS) transmitter.
- Replace S-band transponder with STPSS S-band (SGLS) transponder.
- Replace command demodulator with STPSS dual signal conditioner.
- Modify pulse code modulation (PCM) encoder for dual baseband.
- Modify command decoder/processor.

Although the power system of the AEM is very limited (~ 50 W), no changes were made in this system for Air Force use. Also, the non-redundant design of the AEM was unaltered. In addition, the current AEM design does not allow for the use of encryption equipment--this was not changed because it is not a requirement for all Air Force missions considered in this study.

STPSS

Each of the three available STPSS configurations (summarized in Table 2) consists of a core and an orientation module (or a spin control module in the STPSS-S case). In addition, a variety of mission-specific equipment is available for each configuration. The core module is the same in all cases. The orientation or spin module determines the attitude stability and pointing accuracy of the spacecraft.

Table 2

STPSS CONFIGURATIONS

STPSS-P	STPSS-LC	STPSS-S
Core Module + Orientation Module • 3-axis • Precision (±0.1 deg) • 1 deg freedom solar dr. • Cold gas RCS + Mission-Specific Equipment • Solar panels (max. 1200 W) • Extra 10^8 tape recorder • Encryption unit (GFE) • Orbit transfer module (solids or IUS) • Antenna	Core Module + Orientation Module • 3-axis • Low cost (±1 deg) • 1 deg freedom solar dr. • Cold gas RCS + Mission-Specific Equipment Same choices as for STPSS-P.	Core Module + Spin Control Module • Spin • Low cost (±1 deg) • Cold gas RCS + Mission-Specific Equipment Same choices as for STPSS-P, *except* • Solar panels (max. 380 W)

The configurations used in this study are those identified by TRW in their study.(1) No changes were made except, by direction of the Air Force, the hydrazine reaction control system designed by TRW for the STPSS was not considered in this analysis because of its relatively high cost compared with the cold gas reaction control system/solid rocket option.

MMS

The basic MMS, summarized in Table 3, consists of three primary modules, plus a variety of mission-specific equipment, all of which are attached to a structural subsystem. For Air Force use, we have: (1) retained the attitude control module without modification; (2) added one 20 Ah battery to the power module so that it would have the

Table 3

MMS CONFIGURATIONS

MMS	MMS-AF
Attitude Control Module + Power Module • Two 20 Ah batteries + C&DH Module • TDRSS- and STDN-compatible + Mission-Specific Equipment • Antenna • Solar panels (as required) • Space propulsion (SPS-I, SPS-II, IUS) • Solar drive • Extra tape recorders (8×10^9 bits) • Extra batteries (one 20 Ah or three 50 Ah)	Attitude Control Module + Power Module • Three 20 Ah batteries + C&DH Module • SGLS-compatible • [Data rate 128-256 kbps][a] + Mission-Specific Equipment Same as above, *except* • Solid rockets for orbit translation

[a]Additional option.

same energy storage capacity as the STPSS; and (3) changed the communication system to be compatible with SGLS.

Listed below are the detail modifications to the MMS communication module needed to achieve this compatibility. Again, these modifications consist mainly of replacing MMS communication equipment with STPSS equipment that performs a similar function.* We have also identified the necessary changes to increase the data rate to 128-256 kbps but have not considered them as requirements.

SGLS Compatibility

- Replace S-band transponder with STPSS S-band SGLS transmitter and receiver.
- Replace or modify command decoder with STPSS decoder.
- Replace premod processor with STPSS dual baseband unit.

Increase Data Rate

- Replace data bus controller
- Replace clock and format generator
- Replace standard computer interface

} with STPSS bus controller (data formatter).

- Replace remote interface unit with STPSS data interface unit.

Although the parallel battery-charging design used in the MMS power module has been of some concern to the Air Force, we do not consider it necessary to change it (see Appendix B), since we assume that the power regulation unit will have adequate redundancy to meet Air Force requirements and that the MMS power system will be a flight-proven design prior to the missions considered in this study.

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*It should be noted that if the Air Force Solar Infrared Experiment (SIRE) is flown on the MMS, these changes in the communication module will have already been made prior to any of the missions considered in this study. As noted below (Sec. V) we have based our MMS cost estimates on this assumption, hence the nonrecurring cost associated with these changes is not included in the study.

IV. SPACE TEST PROGRAM MISSION MODEL

In accordance with the directions provided by the Work Statement for this study, Space Test Program missions[6] to be flown during the 1980-1990 time period are divided into three payload groups (Table 4). The principal distinguishing feature of each group is the spacecraft requirements. For example, payloads in groups I and III all require a spacecraft with *nominal* capability and either *three-axis* or *spin* stabilization. We have taken this to mean that these missions could be flown on the AEM, STPSS-S, STPSS-LC, L-AEM-S, or L-AEM-BL spacecraft. Those payloads in group II require a spacecraft with a *high* capability and *three-axis* stability. This requirement can only be met by the STPSS-P, L-AEM-P, or MMS.*

Of the estimated twenty flights to be flown during the 10 years between 1980 and 1990, the Work Statement indicated that about 75 percent (15 flights) would be in payload group I, 10 percent (2 flights) in payload group III, and 15 percent (3 flights) in payload group II. Using the estimated division between large (over 150 lb) and small payloads given in the Work Statement for each of the payload groups, we can presume a total of 114 payloads for the nominal case or about 6 payloads per spacecraft.

As mentioned in Sec. II, Ref. 5 provided a listing of only 52 Space Test Program payloads that were to be considered as representative of those that would be flown between 1980 and 1990. We analyzed these payloads in terms of their spacecraft requirements for accuracy, stabilization, and weight. The results of that analysis are shown on the right-hand side of Table 4 to allow direct comparison with the guidance given in the Work Statement for this study.

We found that the overall division of payloads between group II and groups I and III was a little different from that suggested by the Work Statement, i.e., only 11 percent, rather than 15 percent, of the payloads fell into payload group II. We also found that the

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*For this reason, group II is distinguished from groups I and III in the discussion that follows.

Table 4

SPACE TEST PROGRAM PAYLOAD CATEGORIES

Payload Groups	Percent of STP Flights	Number of STP Flights	Number of Experiments			Spacecraft Requirements		Percentage of Total Payloads		Percentage of Large Payloads[a]		Percentage of Small Payloads	
			Small	Large	Total	Stability	Capability	Gr. I & III or Gr. II	Bluebook	Gr. I & III or Gr. II	Bluebook	Gr. I & III or Gr. II	Bluebook
I	75	15	60	15	75	Spin or 3-axis	Nominal	85	89	15	10	85	90
III	10	2	24		24	Spin or 3-axis	Nominal						
II	15	3	12	3	15	3-axis	High	15	11	20	20	80	80
Total		20	96	18	114					16	11	84	89

[a] ≥150 lb.

percentage of small payloads in groups I and III was larger, i.e., 90 percent, rather than 85 percent. Appropriate adjustments for these relatively minor mismatches caused an increase in the total number of Space Test Program payloads from 114 to 151, which is equivalent to about 7.5 payloads per spacecraft. In addition to this, the preliminary status of the mission model suggested that the number of payloads in the program and the number of payloads per spacecraft should be included in the sensitivity analysis.

As indicated on Table 5, we have also divided the Space Test Program missions[5] into eight different orbits that distinguish between orbit altitude, inclination, and spacecraft orientation. The first orbit (1-S and 1-E) is a low earth orbit with an altitude of about 250-300 n mi. We have divided the missions of this orbit into those that are sun-oriented and those that are earth-oriented. As you may see, 45 percent of the Space Test Program payloads would fly in this

Table 5

SPACE TEST PROGRAM MISSION CATEGORIES

Number	Type	Orbit (n mi)	Inclination (deg)	Launch Range	Percentage of Payloads	No. of Payloads
1-S	Sun-synchronous, sun-oriented	250-300 circular	98.4	Western	17	20
1-E	Sun-synchronous, earth-oriented	250-300 circular	98.4	Western	28	32
2	Elliptical	7000 × 200	Polar	Western	28	32
3	Geosynchronous, sun-oriented	19,372 circular	Low (28.5)	Eastern	8	9
4	--	10,000 circular	Low (28.5)	Eastern	4	5
5	12 hr	21,000 × 900	63.4	Eastern	7	7
6	Geosynchronous, earth-oriented	19,372 circular	Low	Eastern	2	3
7	--	3200 × 150	30	Eastern	2	3
8	--	180 circular	Polar	Western	2	3

orbit. The second orbit is a highly elliptical one (7000 × 20 n mi) having an additional 28 percent of the Space Test Program payloads. The missions in both of these orbits are launched from the Western Test Range (WTR). The missions flown on the WTR (orbits 1, 2, and 8) represent about 75 percent of the Space Test Program payloads. The payloads flown out of the Eastern Test Range (ETR) all require large orbit translations; e.g., up to geosynchronous. The last column in Table 5 indicates the number of Space Test Program payloads in the nominal case that are flown in each of the orbits during the 1980-1990 time period. The total number of Space Test Program payloads in the nominal case is 114.

In Fig. 1 these orbits are related to the perigee and apogee altitude ranges of individual payloads. The payloads are identified by page number in the bluebook[5] at the top of the figure. Each payload generally has a wide range of acceptable operating altitudes, which has made it reasonably easy to collapse the Space Test Program payloads into eight orbits.*

In addition to ordering the Space Test Program payloads according to orbit parameters, they were also matched with each of the spacecraft being considered in this study. In making these assignments, we have considered: payload weight, maximum altitude, orientation, power availability, data rate, pointing accuracy, and stability. The resulting match between individual Space Test Program payloads and the various spacecraft is illustrated in Table 6. Space Test Program payloads are identified by bluebook page number. Of the 52 payloads in the bluebook, 6 were not included in the mission model for various reasons (see footnotes to Table 6). Of the remaining 46 payloads, the AEM with its 150 lb payload capability and 1000 n mi altitude limitation can accommodate only 10 (22 percent). The spinning versions of the L-AEM (L-AEM-S) and STPSS (STPSS-S) can both handle 26 percent of the total payloads. The baseline version of the L-AEM is limited to orbital altitudes of less than 1000 n mi and to earth-oriented missions and therefore can only accommodate

*It is recognized that when these payloads are actually flown, a larger number of orbits may be used depending upon the capabilities of the spacecraft and payload requirements; this should not affect the results of this study.

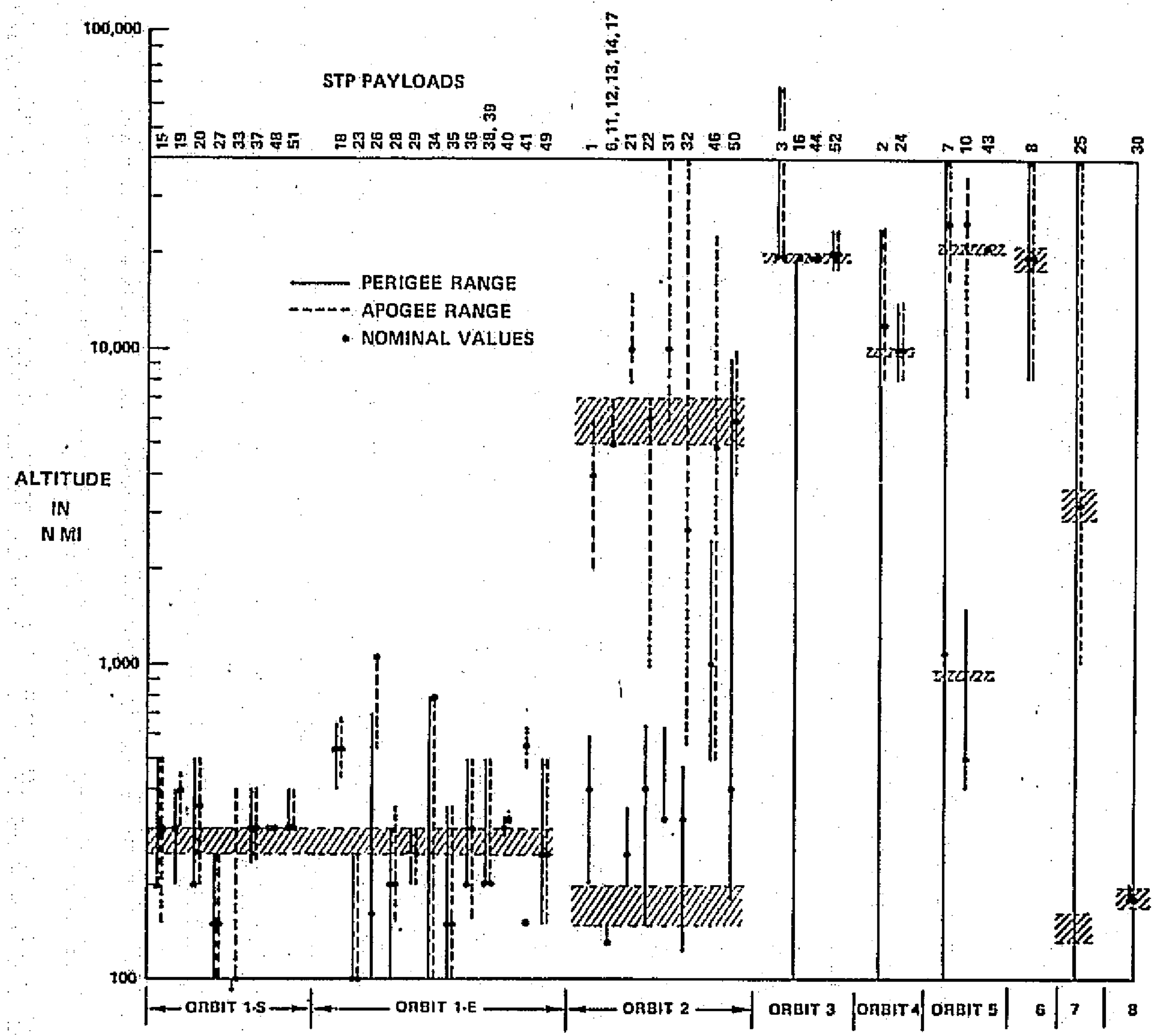

Fig. 1—Baseline Space Test Program orbit options

Table 6

SPACECRAFT MISSION CAPABILITY

Space Test Program Payloads (Bluebook Page Number)	Spacecraft							
	AEM (150 lb, <1000 n mi)	L-AEM-S	L-AEM-BL	L-AEM-P	STPSS-S	STPSS-LC	STPSS-P	MMS-AF
1				X		X	X	X
2				X		X	X	X
3[a]				X		X	X	X
6				X		X	X	X
7		X		X	X	X	X	X
8[b]				X		X	X	X
10		X		X	X	X	X	X
11				X		X	X	X
12				X		X	X	X
13				X		X	X	X
14				X		X	X	X
15				X		X	X	X
16		X		X	X	X	X	X
17				X		X	X	X
18	X		X	X		X	X	X
19				X			X	X
20				X			X	X
21				X		X	X	X
22				X		X	X	X
23	X	X	X	X	X	X	X	X
24		X		X	X	X	X	X
25				X		X	X	X
26	X	X	X	X	X	X	X	X
27		X		X	X	X	X	X
28		X	X	X	X	X	X	X
29	X		X	X		X	X	X
30				X			X	X
31[c]		X		X	X	X	X	X
32				X		X	X	X
33				X		X	X	X
34	X	X	X	X	X	X	X	X
35	X		X	X		X	X	X
36	X		X	X		X	X	X
37				X		X	X	X
38	X		X	X		X	X	X
39	X		X	X		X	X	X
40[c]		X	X	X	X	X	X	X
41[d]			X	X		X	X	X
43				X			X	X
44[e]				X		X	X	X
46[f]		X		X	X	X	X	X
48				X			X	X
49	X		X	X		X	X	X
50				X		X	X	X
51				X		X	X	X
52				X		X	X	X
Total payloads	10	12	13	46	12	41	46	46

[a] Payloads 4 and 5 eliminated--excessive altitude (69,000 n mi) and already flown.

[b] Payload 9 eliminated--excessive altitude (69,000 n mi).

[c] Assumes that only a portion of the payload is spun.

[d] Payload 42 eliminated--inconsistent data.

[e] Payload 45 eliminated--SIRE mission exceeded TRW STPSS design power level.

[f] Payload 47 eliminated--insufficient data.

28 percent of the payloads. The low-cost STPSS (STPSS-LC) spacecraft can handle 89 percent of the payloads, whereas all three precision configurations (L-AEM-P, STPSS-P, and MMS) can handle all of the payloads.

Consistent with the Work Statement guidelines, we have assumed that those payloads that require spinning can be accomplished on a three-axis stabilized spacecraft by allowing portions of the payload to spin. We have also assumed that the total payload integration costs for the mission model will not vary substantially as a function of the procurement option. A further assumption that we have made is that those payloads having accuracy requirements in excess of the capability of the L-AEM-P, STPSS-P, and MMS really have attitude determination requirements rather than pointing accuracy requirements.

In the analysis of program cost that follows (Sec. VI), we have considered only those spacecraft and combinations of spacecraft that can accommodate the entire Space Test Program mission model. We will be evaluating the various procurement options on a constant performance basis.* To expand the mission model up to 114 payloads of the nominal case, we have linearly extrapolated the characteristics of the 46 payload model given in the bluebook.

*It is clear that some procurement options, such as the pure MMS option, will have excess capability. However, we have not attempted to determine the value of this excess capacity for the Space Test Program.

V. SPACECRAFT AND LAUNCH COSTS

SPACECRAFT

Estimating the costs of the AEM, L-AEM, STPSS, and MMS presented an interesting problem, because each was at a different stage of development. The AEM was well along in the development process, and the contractor, Boeing, was confident that the ceiling price would not be exceeded. Should the L-AEM be developed, Boeing would have AEM experience to build on. The three STPSS configurations were the result of a short study by TRW, and they lacked the specificity of the AEM and MMS. Since preliminary designs generally change, and changes generally increase cost, one needs to question whether an estimate of current STPSS designs would be representative of final cost. The MMS was somewhere between the AEM and STPSS; some hardware had been developed, design was complete, and NASA had gone out to industry for bids. Thus the situation was one in which some costs were known, some were partly known, and others were unknown. We needed to develop estimates that would reflect relative differences in the size, complexity, and capability of the spacecraft as currently specified.

Recurring Costs

An examination of existing parametric cost-estimating models showed that they had been developed from data on conventional spacecraft, i.e., spacecraft for which low cost was not a dominant consideration. Thus a procedure was required that would provide comparable estimates of the various spacecraft but estimates in keeping with current experience. The method adopted was to develop a model calibrated to reflect AEM experience, in essence saying that AEM costs are known and those of the other spacecraft can be extrapolated from that base using conventional scaling techniques. Estimates of Unit 1 cost for each spacecraft are shown in Table 7. These estimates include allowances for modifications of the AEM and MMS to meet Air Force requirements.

By using the same model for all estimates it can be argued that they should be comparable. The point has been made, however, that such

Table 7

ESTIMATED UNIT 1 COST

(In millions of
1976 dollars)

AEM	2.3
L-AEM	
Spin	3.9
Baseline	4.8
Precision	5.7
STPSS	
Spin	4.6
Low-cost	5.7
Precision	6.9
MMS	
Basic	8.9
SPS-I	9.4

a procedure ignores an important element of spacecraft cost. The AEM and L-AEM are not comparable to the STPSS and MMS, because they consist of a single module produced by a single contractor. With two, three, or even four contractors involved in production, integration, and test of the different modules, additional costs could be incurred. Whether that would produce a significant cost difference is a matter of some disagreement, but the assumption made here is that it would not. While that assumption may favor the STPSS somewhat and the MMS even more, if it had any effect at all it would be to strengthen the conclusions of the study.

As a check on the spacecraft estimates, they were plotted against weight (Fig. 2) and compared with a regression line from the *SAMSO Unmanned Spacecraft Cost Model* (third edition).[8] All are within the standard error of estimate (the dashed lines) of the regression line. The AEM has a higher relative cost than the other spacecraft because of a lower percentage by weight of structure. All other spacecraft have costs lower than would be predicted by the SAMSO model, and that seems appropriate because the model was derived from data on conventional spacecraft.

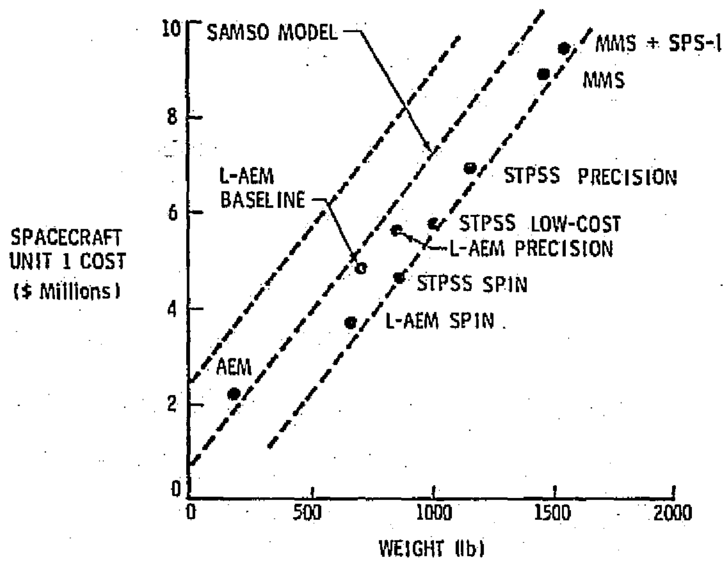

Fig. 2—Spacecraft unit cost versus weight

Cost-quantity effects in spacecraft depend more on the size of each individual procurement than on the cumulative quantity procured. A block buy of six may reduce total cost by 20 percent, but a buy of six spacecraft one at a time may produce no cost reduction. Since the manner of procurement could not be specified in this study, cost reduction was related to annual production rate according to the following empirically derived schedule:

Annual Production	Cost (%)
1	100
2	90
3	87
4	85

In estimating spacecraft costs it was further assumed that:

1. Procurement of the AEM by the Space Test Program Office begins at Unit 9. The first eight units will be procured by NASA prior to 1980.

2. Procurement of the MMS by the Space Test Program Office begins at Unit 5. The first four units will be procured by other agencies prior to 1980.
3. NASA procures two MMS per year during the decade considered. The Air Force buy is incremental to NASA procurement.
4. USAF procures MMS for SIRE, which means that an Air Force-compatible communication and data-handling subsystem would be developed for MMS and would be available to the Space Test Program Office for the missions discussed in this study at no additional cost.

Nonrecurring Costs

Nonrecurring costs were estimated for the STPSS and L-AEM only; for the other spacecraft those costs would not be borne by USAF and would be irrelevant in comparisons of USAF outlays. The *SAMSO Unmanned Spacecraft Cost Model* provided the basic estimating equations, which were derived from a sample of up to 28* space programs over the period 1959-1972. Some spacecraft had been deleted from the sample because they were developed "under tight monetary constraints and under a philosophy that required the use of proven technology." STPSS is precisely such a program, so the output of the SAMSO model was modified to fit the Space Test Program Office philosophy.

An initial assumption was that the first spacecraft manufactured and tested would be a flight model, i.e., there would be no qualification test model. It was later decided that a qualification test model would be desirable, and the estimates were modified to reflect that decision. The higher estimate is the one included in the final program costs.

For the L-AEM nonrecurring costs the basic estimate provided by Boeing was scaled up to include a test model, but as shown in Table 8 the difference between L-AEM and STPSS nonrecurring costs is striking. When L-AEM costs are estimated in the same manner as those for the STPSS, the differences are far less. It is possible to construct a

*Sample size varied for each spacecraft subsystem.

Table 8

SPACECRAFT NONRECURRING COSTS

(In millions of 1976 dollars)

Spacecraft	Estimates Based on SAMSO Model		Estimates Based on Boeing Study
	STPSS	L-AEM	L-AEM
Spin	15.9	--	--
Low-cost (baseline)	20.7	18.0	8.6
Precision	23.4	19.6	9.1
Spin + low-cost	25.3	--	--
Spin + precision	28.1	23.0	11.3
Low-cost + precision	26.1	25.3	11.9
Spin + low-cost + precision	30.9	28.7	14.5

rationale for some degree of difference, e.g., L-AEM would be a follow-on to AEM, and there would be some transfer of learning. Also, STPSS consists of modules that are developed separately, then integrated, and each module is essentially a separate spacecraft. Configuration changes in L-AEM are handled on the basis of different kits rather than different modules. Nevertheless, the discrepancy between the estimates based on the SAMSO model and those based on Boeing figures is too great to be ignored. In the discussion of program costs in Sec. VI the impact of that discrepancy on the issue of spacecraft selection will be examined.

LAUNCH COSTS

The other major category of cost in the 10-year program considered is the cost to launch spacecraft and place them in orbit at the specified altitude and inclination. The basic launch vehicle is the space shuttle, but at present neither the cost nor the guidelines for allocating cost among users has been determined. Estimates of cost range from $15 million to $30 million, of which the users may pay all or nothing. The intent of the study was not to estimate launch costs but to examine whether those costs could influence the choice of spacecraft. Consequently, launch costs were assigned to each payload based on a range of assumptions: Space shuttle launch cost was \$15.4 million or \$30 million. Costs are allocated on a basis of weight or according to either of two NASA-proposed tariff schedules, or are not allocated at all, i.e., only a service charge is incurred.

In the initial phase of this study a NASA formula suggested as a basis for prorating launch cost considered weight, length, inclination, and altitude as independent variables, i.e.:

$$\begin{aligned} \text{SRU} = {} & .00215 \text{ length} + .0238 \text{ length}^2 + .000203 \text{ weight} \\ & - .00000000169 \text{ weight}^2 - .000122 \text{ inclination} \\ & + .00442 \text{ inclination}^2 + .00109 \text{ altitude} + .000232 \text{ altitude}^2 \end{aligned}$$

where SRU = Service Rendered Units which may not exceed 100. It represents a percent of total launch cost. Length is in feet, weight in pounds, inclination in degrees, and altitude in nautical miles. If the SRU exceeds 100 ft it is assumed to be truncated at 100.

A formula proposed since the earlier phase* consists of prorating the dedicated shuttle cost on the basis of whichever of the load-factor ratios below is larger:

1. $\dfrac{\text{payload length (in feet)}^{\dagger}}{60}$

2.[‡] $\dfrac{\text{payload weight (in pounds)}}{\text{shuttle orbital capacity (in pounds) to the desired inclination and altitude}}$

In this study, we have assumed a direct relationship between load factor, as determined above, and the cost factor for prorating the

*Private conversation with Mr. Edwin G. Dupnick at the Johnson Space Center of NASA, October 1976.

†Payload length is the sum of the lengths of the Space Test Program payload, spacecraft, and solid kick stages.

‡For this study we have used a nominal shuttle capacity of 65,000 lb for ETR launches and 39,000 lb for WTR launches. A nominal altitude of 150 n mi has been used. Solid rocket kick stages are used to translate the spacecraft to higher orbits. Payload weight is the sum of the weights of the Space Test Program payload, spacecraft, and kick stages.

dedicated shuttle cost. In some formulations of this tariff rate, the load factor is multiplied by as much as a 1.4 cost factor; we have not used this in our study. Because the launch cost is very sensitive to payload length when using this NASA tariff, an attempt was made to minimize launch cost by placing payloads laterally rather than longitudinally in the shuttle bay whenever the payload length was less than 13 ft. Launch costs estimated using the above method are identified as the *modified NASA tariff*.

The other cost-allocation schemes considered were: a full allocation by weight, i.e.,

$$\frac{\text{payload weight}}{\text{shuttle orbital capacity}} \times \$15.4 \text{ million ,}$$

plus a service charge of $1 million; an allocation of only half the shuttle cost plus a service charge; and, a service charge only.

KICK STAGES

A variety of solid propellant kick stages were required, and to simplify the task of assigning a cost to each kick stage a simple cost-estimating relationship was derived from the cost of several existing stages:

$$C = 2900\ W^{.585}$$

where C = stage cost in 1976 dollars, and
W = stage weight (lb).

Where the IUS was used, a cost of $4.3 million was charged.

VI. PROGR COST

In this section, we discuss the total program costs for a variety of procurement options, each of which is capable of performing all of the Air Force Space Test Program missions. For this constant-performance comparison, program cost is used as the principal measure for distinguishing among procurement options. The analysis described in this section was accomplished in two phases. In the first phase, procurement options using the AEM, STPSS, and MMS spacecraft were compared. In the second phase, additional procurement options using the L-AEM spacecraft were derived and evaluated. The configuration of the L-AEM spacecraft was defined partly as a result of the outcome of the first phase of this analysis; for that reason the sequential nature of the analysis is preserved in the discussion that follows.

NOMINAL CASE

We defined a nominal case as a baseline for estimating the cost to carry out the Space Test Program missions during the 1980-1990 period, and a number of excursions from that baseline were made to test the sensitivity of the results to assumptions about the number of payloads, payloads per spacecraft, etc. The nominal case includes all three versions of the STPSS. The nominal program size is 114 payloads with a maximum of 6 payloads per spacecraft.* In keeping with the Air Force Space Test Program position that its payloads always have a secondary status, they are always taken to an altitude of 150 n mi by the shuttle;

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*As mentioned in Sec. IV, the Work Statement for this study indicated that the number of payloads (defined as the set of experiments combined on one page of the bluebook)[6] to be flown per spacecraft could vary from a combination of one large payload plus four small payloads to as many as twelve small payloads. In Sec. IV we found that for the nominal size program (114 payloads), the average number of payloads per spacecraft would be about 6 but that it might increase to 7 or 8. For this study, we have treated this assumption as a maximum value rather than as an average value while we allocated the Space Test Program payloads to specific spacecraft; this will be discussed later in this section when we describe the sensitivity excursions.

solid rocket kick stages (not the IUS) are then used for translation into the proper orbits. Both ETR and WTR launches of the shuttle are considered. We have assumed that the shuttle cost of $15.4 million will be prorated by weight and that a service charge of $1 million per launch will be made.

The number of spacecraft that would need to be procured for each of four different procurement options is shown in Table 9. The four options are: all-STPSS, all-MMS, AEM plus STPSS, and AEM plus MMS. An option consisting of all three types of spacecraft would not be cost-effective in view of the magnitude of the nonrecurring cost associated with providing the STPSS-P, given that the program already includes the MMS.

Table 9

NUMBER OF SPACECRAFT

(Nominal case)

Spacecraft Type	Procurement Options			
	STPSS	MMS	AEM/STPSS	AEM/MMS
AEM	0	0	3	4
STPSS-S	0	0	0	0
STPSS-LC	19	0	16	0
STPSS-P	5	0	5	0
MMS	0	24	0	20
Total	24	24	24	24

It can be seen that the STPSS-S configuration is never procured in the nominal case, because there are only a few payloads that can be spin stabilized, and they are distributed over the eight different orbits in such a way that it is always more costly to use an STPSS-S spacecraft than to load up the STPSS-LC or STPSS-P spacecraft. When we consider programs with a larger number of payloads, the spin configuration is included in the procurement mix.

The costs associated with these procurement options are shown in Table 10, broken out by the spacecraft, kick stages, and launch operations. The cost of the all-solid kick stages is nearly insignificant

Table 10

PROCUREMENT COSTS IN NOMINAL CASE

($ millions)

Cost Item	Procurement Options			
	STPSS	MMS	AEM/STPSS	AEM/MMS
Spacecraft	167	190	155	172
Kick stages (solids)	4	6	4	5
Launch (100% prorated)	51	67	51	63
Total	222	263	210	240

(about 2 percent of the total). Launch costs represent about 25 percent of the total cost.

The lowest-cost procurement option is the AEM/STPSS combination, but the all-STPSS option is within 10 percent of the AEM/STPSS cost. Given the uncertainties of the various spacecraft designs used in this study, *we consider program options having costs within 10 percent of each other as indistinguishable*. Consequently, for the nominal case, both the AEM/STPSS and all-STPSS cases are preferred alternatives. The all-MMS case is not a good option for the Space Test Program missions, because it offers more capability than is needed by most of the payloads, and that capability must be paid for.

PAYLOAD VARIATIONS

Those results can be considered valid only if they obtain for conditions other than those established somewhat arbitrarily. To test their sensitivity to the original assumptions, several other cases were examined: (1) the maximum number of payloads per spacecraft was increased from 6 to 13; (2) the number of payloads in the program was allowed to range from 92 to 228; (3) the IUS was used as a kick stage for missions with large payload weights and high altitude requirements; (4) the percentage of shuttle costs prorated to Space Test Program payloads was varied from 0 to 100 percent; (5) criteria other than weight were used for allocating shuttle cost; (6) shuttle cost was increased from \$15.4 to \$30 million; and (7) lower development cost was assumed for the STPSS to reflect the elimination of the qualification test model. Of the above cases, maximum payloads per spacecraft, payloads in the Space Test Program, allocation criteria for launch costs, and shuttle cost were found to be the most important in terms of program costs.

The variation of total program cost with maximum payloads per spacecraft is illustrated in Fig. 3. As the maximum increases, the reduction in program cost for the all-MMS case is much larger than for any of the other options. This is partly because of the large payload

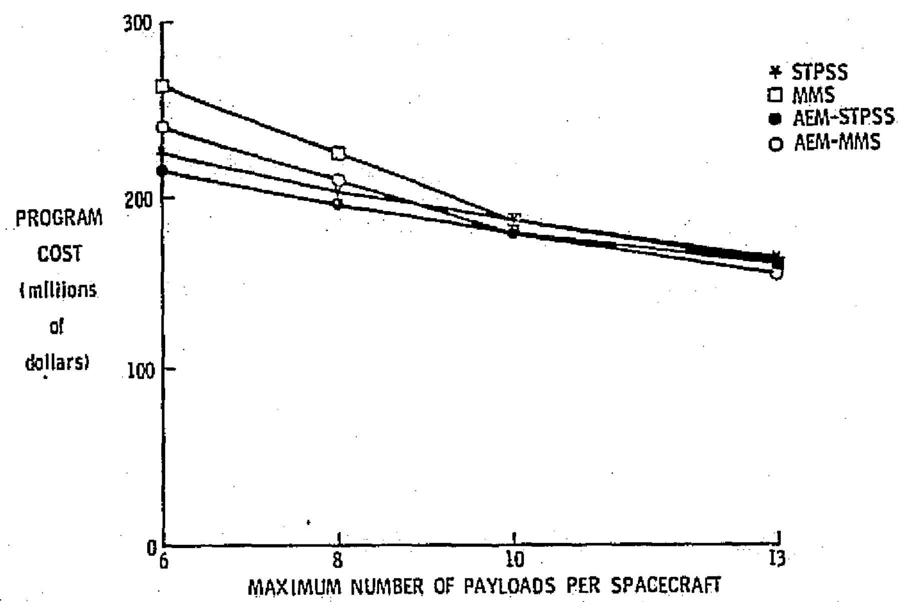

Fig. 3—Effect of the maximum number of payloads per spacecraft (nominal case)

capability of the MMS. The result is that the ability to distinguish between the procurement options on the basis of cost disappears when the maximum number of payloads increases above 10. *However, the total program cost is about 30 percent lower than in the nominal case (maximum number of payloads = 6) when the number of payloads is allowed to increase to 13.* We have found that to be true across a wide number of excursions.

It should be noted here that assuming a maximum number of payloads per spacecraft of 13 results in an average number of payloads per spacecraft of only 5 to 8, depending on the procurement option. The largest benefit is from orbits 1 and 2 where the majority of Space Test Program payloads occurs. To illustrate that, Fig. 4 presents a detailed breakdown of the distribution of the actual maximum number of payloads per spacecraft by orbit for the all-STPSS procurement option. For orbit 1-S, for example, if the assumed maximum number of payloads per spacecraft is allowed to increase from 6 to 13, the actual maximum number of payloads assigned to a spacecraft increases from 5 to 10.* The difference between the actual number of payloads assigned to a spacecraft and the upper limit occurs in all orbits because of the limited number of payloads in each orbit. In orbit 1-S, for example, the mission model includes only 20 payloads, which were distributed evenly between two spacecraft when the assumed maximum number of payloads per spacecraft was increased to 10. Consequently, the average number of payloads per spacecraft for a given procurement option does not increase substantially as a result of allowing the assumed maximum number of payloads per spacecraft to increase from 6 to 13.

The main difficulty associated with increasing the number of payloads per spacecraft lies in the payload-integration area. Although the specific performance limits of each spacecraft were imposed while allocating payloads, payload-integration problems and costs were not explicitly examined. Based on the saving in program costs identified as a result of increasing the maximum number of payloads per spacecraft, it appears that a systematic study of the payload integration problems and costs would be useful.

Figure 5 illustrates the variation in program cost as a function of Space Test Program size. Here program size was doubled to a total of 228 payloads to see if economies of scale might preferentially benefit the MMS and thereby alter the ordering of the procurement options.

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*While 13 payloads are never allocated to a spacecraft in the example shown in Fig. 4, this is not the case for other procurement options, especially those including the MMS.

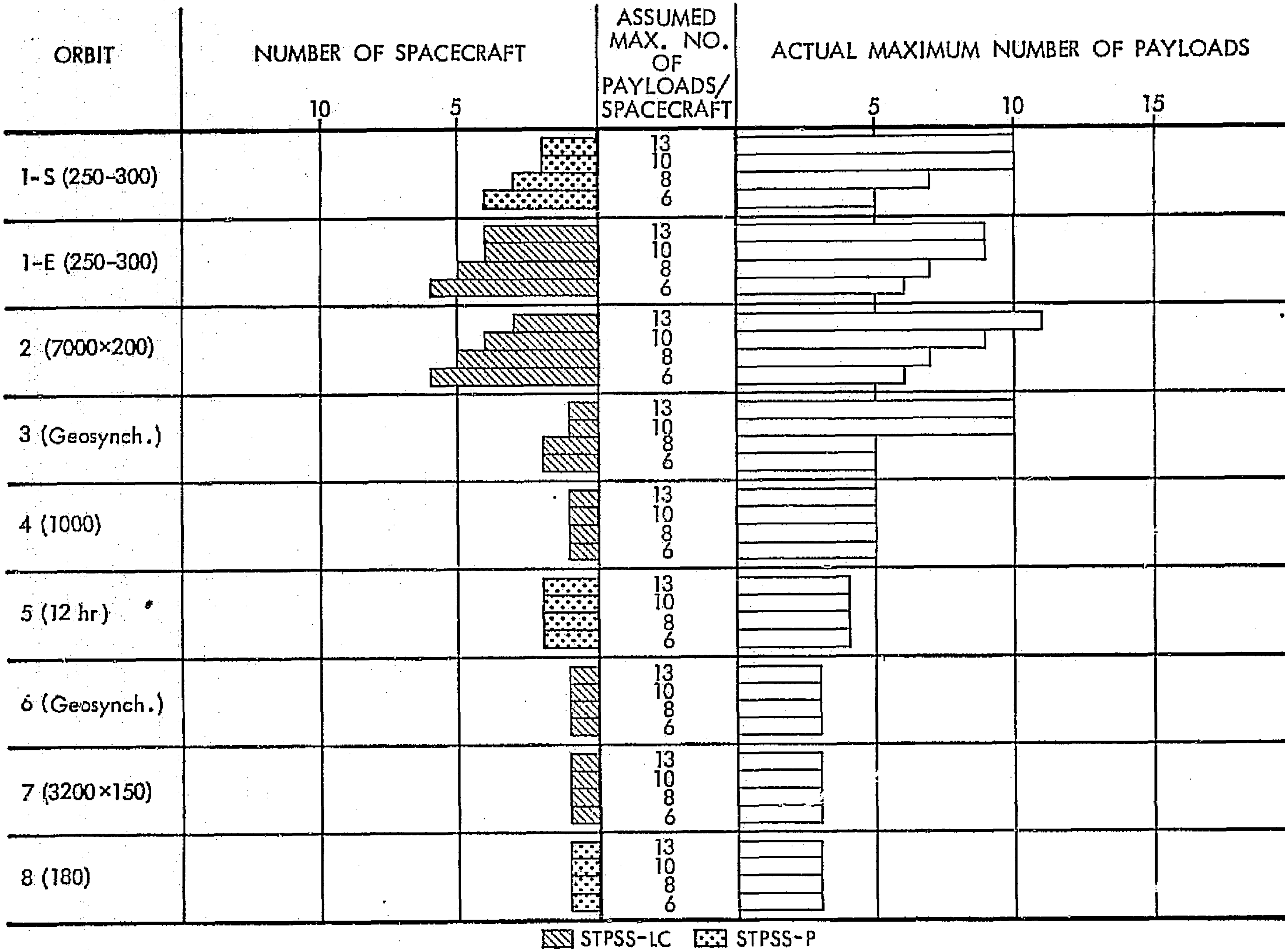

Fig. 4—All-STPSS nominal case

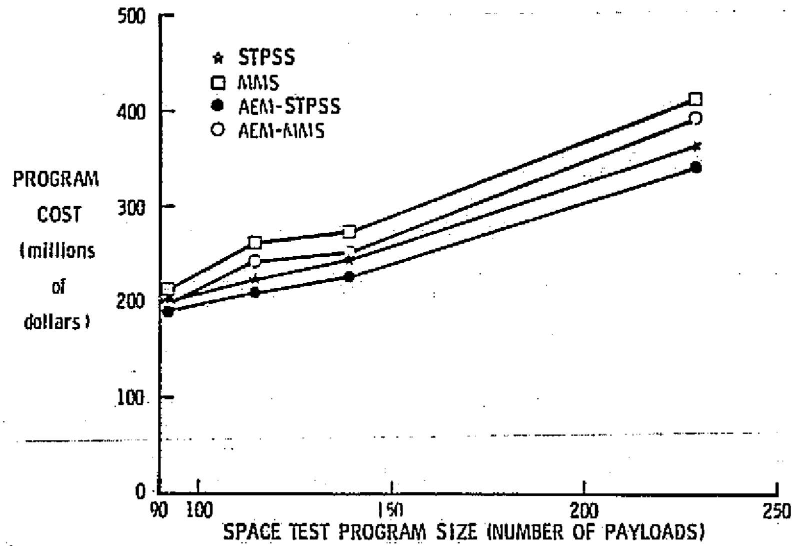

Fig. 5—Effect of Space Test Program size (nominal case)

As shown, no such effect was found. The ordering of the various procurement options remained unchanged, whereas the program cost increased nearly linearly.

LAUNCH COST VARIATIONS

Table 11 displays program costs for the nominal case where the shuttle launch cost is assumed to be $15.4 million prorated among users on the basis of payload weight. Excursions were performed to test the sensitivity of the rank ordering of program costs to shuttle launch cost and the procedure adopted for allocating shuttle costs among users. The results of the variations considered are also shown in Table 11. For ease in reading the table, all costs more than 10 percent above the lowest cost in each row are enclosed in parentheses--all other costs are considered to be essentially the same.

In looking at the other cases it is clear that increasing the shuttle cost to $30 million per launch has no effect on relative results, although the magnitude of program costs increases about 15 percent. Assuming that Space Test Program payloads get a free ride on the shuttle and pay only a service charge of $1 million per launch does not change the conclusions either. The STPSS looks slightly worse

Table 11

EFFECT OF SHUTTLE COST AND TARIFF SCHEDULES[a]

Case	No. of Payloads in Programs	Max. No. of Payloads per Spacecraft	Program Cost ($ millions)			
			STPSS	MMS	AEM/STPSS	AEM/MMS
Shuttle cost = $15.4 million	114	13	160	162	157	156
	114	6	222	(263)	210	(240)
	228	13	244	247	244	240
	228	6	373	(418)	342	(392)
Shuttle cost = $30 million	114	13	181	189	178	183
	114	6	249	(306)	237	(279)
	228	13	279	290	279	284
	228	6	424	(489)	391	(461)
Service charge of $1 million only	114	13	139	135	136	129
	114	6	195	(220)	183	201
	228	13	209	204	209	196
	228	6	322	(347)	293	(323)
NASA tariff	114	13	202	204	199	198
	114	6	297	(342)	286	(321)
	228	13	315	316	333	321
	228	6	514	(558)	490	538
Modified NASA tariff	114	13	161	(181)	156	(173)
	114	6	226	(277)	210	(258)
	228	13	244	(267)	240	(265)
	228	6	(376)	(454)	339	(432)

[a]For a given row, program costs within 10 percent of the lowest value are *not* in parentheses.

and the AEM/MMS slightly better, but the only definite conclusion is still that the MMS is not attractive when the maximum number of payloads per spacecraft is 6.

The effect of two different NASA-proposed tariff schedules is also shown. In the case called *NASA tariff*, where launch cost is allocated on a basis of payload length and weight, altitude, and orbital inclination, relative costs are unchanged from the first two cases. Adaptation of a more recent tariff schedule, *modified NASA tariff*, altered these results somewhat; both the pure MMS and the AEM/MMS options have relatively higher program costs because the average length of the spacecraft-payload combinations for these options is greater than for the options using the STPSS.

The implications of the foregoing analysis for spacecraft selection that has included the AEM, STPSS, and MMS, may be summarized as follows:

1. When the upper limit on the number of payloads that can be assigned to a spacecraft is 10 or more, program costs are essentially the same in all cases.
2. When the number of payloads per spacecraft is limited to 6, the STPSS and AEM/STPSS offer lowest program costs in virtually all cases.
3. When shuttle charges are determined largely by payload length as is the case when the modified NASA shuttle tariff is used, the AEM/STPSS combination has the lowest program cost.
4. Given the stipulated AEM, STPSS, and MMS capabilities, the uncertainties in the Air Force Space Test Program mission model, and the uncertainties in the shuttle tariff schedule, none of the alternatives considered offers a clear-cut advantage over the others, although those options that include the STPSS are generally preferred.

UPGRADED AEM

As an additional excursion, the possibility of modifying some spacecraft designs to give them greater capability was considered. Specific modifications considered include: increasing the STPSS payload capability to 1500 lb; increasing the AEM payload capability to 300 lb; and changing the AEM capability to allow sun orientation and/or geosynchronous altitude operation. Of these, only the last promised a sizable impact on program cost because of the increased number of Space Test Program payloads that could be captured (from 22 to 72 percent). To obtain a first-order approximation of the cost of an AEM having such a capability, the cost of the STPSS cold-gas reaction control system was added to the cost of the basic AEM. Such a reaction control system would be needed for the AEM to operate at geosynchronous altitude. This configuration is referred to henceforth as the upgraded AEM.*

*We have assumed that the upgraded AEM is limited to a payload of 150 lb, a data rate of .8 kbps, experimental power of 40-50 W and no encryption capability--the same as the basic AEM.

Table 12

EFFECT OF THE UPGRADED AEM[a]

Case	No. of Payloads in Program	Max. No. of Payloads per Spacecraft	Program Cost ($ millions)					
			STPSS	MMS	AEM/STPSS	AEM/MMS	Upgraded-AEM/STPSS	Upgraded-AEM/MMS
Nominal	114	13	(160)	(162)	(157)	(156)	(148)	99
	114	6	(222)	(263)	(210)	(240)	(172)	146
	228	13	(244)	(247)	(244)	(240)	(233)	175
	228	6	(373)	(418)	(342)	(392)	298	294
Increased estimates of upgraded AEM cost	114	13	(160)	(162)	(157)	(156)	(175)	121
	114	6	(222)	(263)	(210)	(240)	(215)	183
	228	13	244	247	244	240	(281)	231
	228	6	373	(418)	342	(392)	368	371

[a]For a given row, program costs within 10 percent of the lowest value are *not* in parentheses.

Table 12 compares the cost of upgraded AEM/STPSS and upgraded AEM/MMS combinations with those considered in the previous nominal case. In that excursion the upgraded AEM/MMS combination appeared to have program costs more than 20 percent below those of the other procurement options. The principal reasons for this are: (1) with the additional performance capabilities, the relatively low-cost upgraded AEM is a substitute for the more expensive STPSS on nearly all missions, and (2) when the upgraded AEM is used in combination with the MMS, the nonrecurring cost of the STPSS is not incurred.

To test the sensitivity of the above result to the estimated cost of the upgraded AEM, nonrecurring cost was increased by $10 million and unit 1 recurring cost was increased from $2.44 million to $4.88 million. The results, also shown in Table 12, indicate that the upgraded AEM/MMS combination continues to be the preferred procurement option.* Other candidates become competitive only when the program size is expanded to 228 payloads.

In this last case, an upgraded AEM spacecraft with costs of that magnitude would probably also have greater payload, power, and data

*The use of the modified NASA tariff increases the program cost of the MMS and AEM/MMS options relative to the other options shown in Table 12, and thereby would not alter this observation.

rate capabilities. Furthermore, it would probably also be a redundant design to minimize the single-point failure modes. *Because of the potential value of such a spacecraft it seemed highly desirable that an upgraded AEM having many of the above characteristics be designed and evaluated for use in the Air Force's Space Test Program.*

LARGE-DIAMETER SHUTTLE-LAUNCHED AEM (L-AEM)

Under NASA sponsorship the Boeing Company undertook a configuration and cost study for a 5 ft diameter AEM that would be designed for shuttle launch and would include the capabilities ascribed above to the upgraded AEM. Revised Boeing cost estimates (as described in Appendix A) were used to compute program costs for a variety of procurement options including the L-AEM. Table 13 shows those options compared with others for the nominal case. Where the L-AEM is used, all three configurations (baseline, spin, and precision) were considered; but for the same reasons discussed earlier for the STPSS, the spin configuration is included only when the mission model includes 228 payloads.

Two procurement options are included that use the MMS but none that uses the STPSS in combination with the L-AEM. There are two reasons for this. First, the MMS has been used primarily when its use would decrease the total number of spacecraft necessary to fly the designated payloads as a result of its large payload capability (4000 lb); the payload capabilities of the STPSS and L-AEM are identical,

Table 13

EFFECT OF THE L-AEM[a]

Case	No. of Payloads in Program	Max. No. of Payloads per Spacecraft	Program Cost ($ millions)							
			STPSS	MMS	AEM/STPSS	AEM/MMS	L-AEM	AEM/L-AEM	L-AEM/MMS	AEM/L-AEM/MMS
Nominal	114	13	(160)	(162)	(157)	(156)	135	133	139	132
	114	6	(222)	(263)	(210)	(240)	186	181	187	186
	228	13	(244)	(247)	(244)	(240)	198	208	212	199
	228	6	(373)	(418)	(342)	(392)	306	297	(373)	323
Higher L-AEM nonrecurring cost	114	13	(160)	(162)	157	156	148	146	150	143
	114	6	(222)	(263)	210	(240)	199	195	200	197
	228	13	(244)	(247)	(244)	(240)	212	222	223	211
	228	6	(373)	(418)	342	(392)	320	311	(384)	335

[a]For a given row, program costs within 10 percent of the lowest value are *not* in parentheses.

so we always chose the lower-cost L-AEM. Second, consideration of both the L-AEM and STPSS in a single procurement option would mean that the nonrecurring cost associated with developing both spacecraft would have to be included in the total program cost.

Table 13 illustrates that all of the procurement options that use the L-AEM are preferred over those made up of the three original spacecraft. In fact, the lowest-cost L-AEM option is about 15-20 percent less costly than the lowest-cost non-L-AEM option, and that assumes that the nonrecurring cost of the L-AEM would be paid for by the Air Force. If the L-AEM is developed by NASA, the L-AEM options are even more attractive.

In Sec. V, the uncertainty surrounding our estimates of the nonrecurring costs of the L-AEM spacecraft configurations was discussed. The nominal case in Table 13 includes the lower set of estimates, because we feel that they more closely reflect the nonrecurring costs of the L-AEM. However, the effect of higher nonrecurring costs for the L-AEM on the choice of a procurement option has been examined. The second set of estimates in Table 13 shows that when L-AEM development cost is increased, the AEM-STPSS combination is also attractive for some conditions. As mentioned earlier, however, it is not known whether the L-AEM would be developed (if it is developed) by NASA, the Air Force, or jointly. The L-AEM would probably be suitable for NASA missions as well as for the Air Force Space Test Program missions used in this analysis. In the case described here, we assume that the Air Force would underwrite all the nonrecurring costs of the L-AEM. If either of the other two development alternatives was followed, the attractiveness of the L-AEM would be enhanced. Consequently, *we can conclude from these excursions that development of the L-AEM would be more appropriate for the Air Force's Space Test Program than the development of the STPSS and that the use of the L-AEM in combination with the AEM and/or the MMS would constitute alternative cost-effective procurement options.*

In our analysis of the L-AEM spacecraft for Air Force Space Test Program missions, we found that the L-AEM-BL configuration was able to accommodate only 28 percent of the missions, primarily because of limitations on its maximum operating altitude and orientation. Consequently, in the L-AEM procurement options we have substituted the more expensive and more versatile L-AEM-P configuration when the L-AEM-BL configuration would have been adequate except for those limitations. To evaluate the effect of increasing the capability of the L-AEM-BL configuration to allow geosynchronous altitude and sun-oriented operations we have increased the cost of the L-AEM-BL to allow for an increase in size of the hydrazine reaction control system.* Options containing this configuration are labeled L-AEM-1.

Table 14 compares the four procurement options based on the L-AEM, with four options based on the L-AEM-1 design. As expected, the program costs for the procurement options based on the L-AEM-1 design are lower than those based on the L-AEM design; but, given the accuracy of the spacecraft designs and cost-estimating procedures, most of the options are comparable. *This means that giving the L-AEM-BL more capability is worthwhile but not essential in deciding on the procurement option for conducting the Air Force Space Test Program missions.*

Table 14

EFFECT OF UPGRADING THE L-AEM[a]

(L-AEM-1)

Case	No. of Payloads in Program	Max. No. of Payload per Spacecraft	Program Cost ($ millions)							
			L-AEM	AEM/ L-AEM	L-AEM/ MMS	AEM/ L-AEM/ MMS	L-AEM-1	AEM/1 L-AEM-1	L-AEM-1/ MMS	AEM/ L-AEM-1/ MMS
Nominal	114	13	135	133	139	132	130	127	135	129
	114	6	185	181	187	186	174	171	177	178
	228	13	198	208	(212)	199	190	200	(211)	194
	228	6	(306)	297	(373)	(323)	292	276	(365)	(315)

[a]For a given row, program costs within 10 percent of the lowest value are *not* in parentheses.

Earlier in this section, it was shown that an *upgraded AEM* in combination with the MMS provided the lowest total program cost. The

*It is assumed that the additional sun sensor required for sun orientation would be part of the payload package and therefore would not affect the cost of the L-AEM-BL.

upgraded AEM differs from the L-AEM in that it has the payload, data rate, and power limitations of the original AEM; L-AEM capability is greater in all of these areas. Table 15 displays a comparison of the program costs for the four procurement options derived from the L-AEM and the two options using the upgraded AEM. Again, the upgraded AEM/MMS procurement option is the preferred solution (as indicated by the parentheses), but by less of a cost margin than before. This result occurs for the same reasons as stated earlier (p. 36), except in this case the L-AEM spacecraft is displaced by the cheaper upgraded AEM

Table 15

COMPARISON OF THE L-AEM AND UPGRADED AEM[a]

Case	No. of Payloads in Program	Max. No. of Payloads per Spacecraft	Program Cost ($ millions)					
			L-AEM	AEM/ L-AEM	L-AEM/ MMS	AEM/ L-AEM/ MMS	Upgraded-AEM/STPSS	Upgraded-AEM/MMS
Nominal	114	13	(135)	(133)	(139)	(132)	(148)	99
	114	6	(185)	(181)	(187)	(186)	(172)	146
	228	13	(198)	(208)	(212)	(199)	(233)	175
	228	6	306	297	(373)	322	298	294
With AEM redundancy	114	13	(135)	(135)	(139)	(134)	(167)	113
	114	6	185	186	187	(196)	(209)	175
	228	13	198	217	212	217	(275)	(224)
	228	6	306	318	(373)	337	(363)	(369)

[a]For a given row, program costs within 10 percent of the lowest value are *not* in parentheses.

rather than the STPSS. However, the limited capability of the upgraded AEM, i.e., 50 W of power and a maximum payload of 150 lb, makes this conclusion somewhat tenuous in view of the uncertainty associated with Air Force Space Test Program missions for the 1980 to 1990 period. Any major growth in payload power or weight requirements would mean procurement of more MMS and fewer upgraded AEM; that would quickly decrease any total program cost advantage that the option might have. To illustrate this, three to four additional MMS in the upgraded AEM/MMS option would eliminate the difference in program cost between the pure L-AEM option and the upgraded AEM/MMS option for the nominal case.

In addition, one of the current Air Force requirements of new spacecraft is to minimize single-point failure modes in the spacecraft design. As indicated in Appendix I, that was one of the specifications for the L-AEM design and has been accounted for in its recurring cost. To illustrate the effect on program cost of increasing AEM redundancy so that the L-AEM and the upgraded AEM options will be more comparable, an excursion was made in which it was assumed that whenever an AEM or upgraded AEM is included in an option, two spacecraft would be flown in the same shuttle.* The results are shown in Table 15. It can be seen that for the case of 114 payloads and 6 payloads per spacecraft, several L-AEM options are within the lower 10 percent cost category; for a mission model with 228 payloads, the L-AEM options are clearly preferred over the upgraded AEM/MMS option.

Considering that the program cost advantage indicated for the upgraded AEM/MMS option over the L-AEM option could be lost in either of the two ways mentioned above, i.e., by growth in the power and/or weight requirements of the Air Force Space Test Program mission model, or by spacecraft design requirement for minimizing single-point failure modes, *we conclude that the L-AEM spacecraft, or some very similar design, would provide a basis for minimizing the Air Force Space Test Program costs*. The L-AEM could be used individually or in combination with the AEM and/or the MMS. This conclusion is reinforced by the analysis of a variety of procurement options that considered the uncertainties in the spacecraft costs and designs, the Air Force Space Test Program mission model, and the shuttle cost and tariff schedule.

The procurement results for the nominal case that include the L-AEM are shown in Table 16. A comparison of these options indicates that the L-AEM-P configuration comprises about 75 percent of the buy, with the balance being shared by the AEM, L-AEM-BL, and/or MMS; the L-AEM-S is never used in the nominal program.

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*This idea was suggested by Boeing as a way of achieving the desired level of redundancy without redesigning the entire spacecraft. Physically it is possible to have two AEM spacecraft side by side within the envelope of the L-AEM.

Table 16

PROCUREMENT RESULTS USING L-AEM
(Nominal case)

Spacecraft Type	Number of Spacecraft			
	L-AEM	AEM/L-AEM	L-AEM/MMS	AEM/L-AEM/MMS
AEM-AF	—	1/3	—	1/4
L-AEM-S	—	—	—	—
L-AEM-BL	4/6	3/3	—/6	—
L-AEM-P	12/18	12/18	10/18	10/18
MMS	—	—	4/—	3/2
Total	16/24	16/24	14/24	14/24

NOTE: 13/6 maximum number of payloads/spacecraft.

The distribution of the program cost of the pure L-AEM procurement option is illustrated in Fig. 6. About $134 million is spent procuring spacecraft and solid rocket kick stages. The launch costs are shown for both WTR and ETR. For the ETR launches, the launch costs are very similar for the three allocation schemes. However, the original NASA tariff rate that is a function of spacecraft payload weight and length, altitude, and orbital inclination imposes a disproportionally high cost on WTR launches. For the $15.4 million shuttle case, the WTR launch costs exceed $100 million. The most significant factor is the orbit inclination. The use of the modified NASA tariff rate redresses this drastic cost imbalance. The variation in shuttle cost considered in this study does not appear to greatly alter the launch costs, providing the earlier NASA tariff rate is not used.

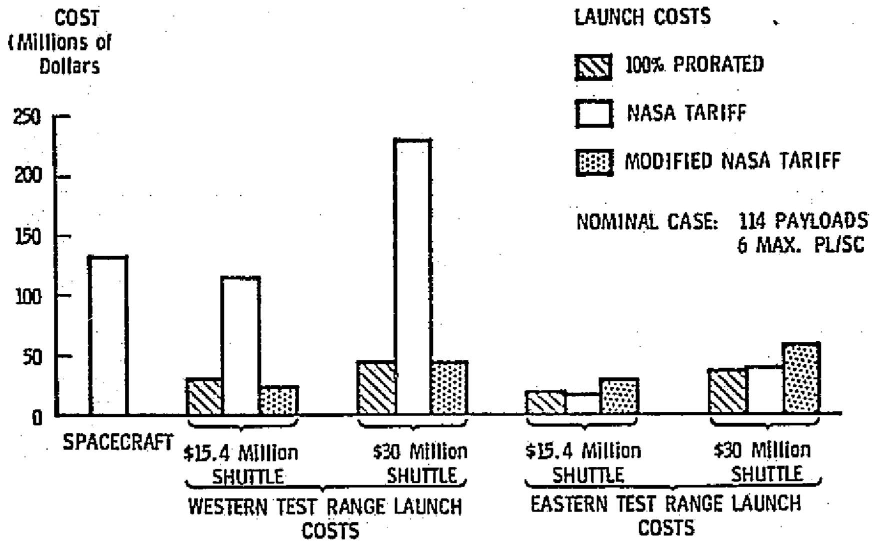

Fig. 6—Distribution of program costs
(L-AEM option)

VII. CONCLUSIONS

Four major conclusions have been drawn from this study. *First, program cost does not provide a basis for choosing among the AEM, STPSS, and MMS spacecraft given their present designs.* Only when the modified NASA tariff schedule was used for allocating the shuttle launch cost did the STPSS options become preferred; with the uncertainty in the appropriateness of this tariff schedule, this case does not provide sufficient basis for recommending the STPSS development.

Second, the availability of the L-AEM spacecraft, or some very similar design, would provide a basis for minimizing the cost of the Air Force's Space Test Program. The L-AEM could be used individually or in combination with the AEM and/or MMS as the missions require. The *upgraded AEM* options, although having program costs similar to the L-AEM options, provide less capability for handling growth in the Space Test Program mission model.

Third, the program costs are very sensitive to the maximum number of payloads flown per spacecraft. An increase from 6 to 13 in the maximum number of payloads per spacecraft would result in about a 30 percent lower program cost; the major portion of this savings occurs by increasing the maximum number of payloads to 10. An analysis of this potential should be undertaken.

Fourth, launch costs, as determined by a variety of formulas, generally did not affect the preferred procurement option, although they substantially change the total program costs. The modified NASA shuttle tariff rate structure considered during the second phase of the study corrects the drastic cost imbalance that the original NASA tariff imposed on Air Force launches from the Western Test Range. Secondary payload status, an underlying assumption for the Air Force's Space Test Program, is not yet accounted for in any of the NASA tariff rate structures for the shuttle. Incorporation of the concept of a secondary payload could reduce the total program costs presented in this report, but it probably would not affect the spacecraft procurement decision.

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Appendix A

ESTIMATES OF COST

Spacecraft traditionally have been very expensive to produce because of stringent weight and performance requirements, heavy emphasis on reliability, and small production quantities. Various parametric cost-estimating models have been developed from experience over the past 15 or so years, and those models reproduce the cost of the traditional spacecraft with acceptable accuracy. Initially, it was thought that such a model could be used to estimate the costs of the AEM, L-AEM, STPSS, and MMS. Such a model would have insured cost-comparability among them, perhaps at the sacrifice of absolute accuracy in some instances.

It developed, however, that models based on 15 years of spacecraft data estimate costs that are higher than those experienced in the Air Force Space Test Program and those in the AEM contract. The SAMSO cost model, for example, estimates the nonrecurring and recurring cost of HCMM at about $14 million, mainly for development; Boeing's ceiling estimate was approximately $5 million, and at the time of the Rand study it did not appear that the ceiling would be exceeded. At the same time, GSFC was estimating a unit cost of under $10 million for MMS compared to the SAMSO model's estimate of about $19 million. The GSFC estimate was based on some hardware development; component costs were based on vendor quotes and analogy with known costs.

At both ends of the spectrum, then, costs were known to a reasonable degree of accuracy. The problem was to ensure relative accuracy between the AEM and MMS and to estimate L-AEM and STPSS costs that would reflect their relative complexity. The decision was made to develop a cost model based on a combination of AEM costs and traditional scaling curves. That would assume implicitly that if Boeing could produce an AEM for about $2 million, all spacecraft manufacturers could be equally efficient in producing larger spacecraft based on a philosophy of low cost, use of flight-proven components, etc.

Cost-estimating equations for spacecraft subsystems are typically of the type

$$Y = aX^b \quad \text{or} \quad Y = a + bX^c$$

where Y = cost, and

X = weight or other subsystem characteristic.

In the SAMSO model, for example, the cost of the attitude control system is given by

$$\text{ACS cost in thousands of 1974 \$} = 14.72 \text{ (ACS dry weight)}^{.90}$$

In developing a model for this study the b-value, 0.90, was used with an a-value based on AEM. That procedure gave the following equations* (all these costs are in thousands of 1976 dollars):

Structure, thermal control, interstage	$= 4.8 \text{ (weight)}^{.74}$
Electrical power system	$= 5.65 \text{ (weight)}^{.84}$
Altitude control system	$= 14.7 \text{ (weight)}^{.9}$
Communications and data handling	$= 25.4 \text{ (weight)}^{.9}$

In addition, the costs of system test and integration, program management, quality assurance, reliability, etc., must be included, and they add about another 50 percent to the total. On top of that are the costs of special components, such as tape recorders, hydrazine tanks, and solar panels not included in the basic configuration.

Component costs, even those of existing, flight-proven components, vary considerably and add another measure of uncertainty to the total. Vendor quotes, for example, can vary by more than an order of magnitude. As shown below, the range of bids for a PCM encoder was from $21,400 to $611,000; in that same case the second-lowest bid was $41,200. Also,

*It may be noted that the ACS estimating equation is essentially the same as the one cited above for the SAMSO model. Apparently inflation effects have been offset by factors such as a low-cost design approach and the cost-quantity effect.

RANGE OF BIDS

Item	Range ($ thousands)	Ratio
S-band transmitter	29.1-39.8	1:1.37
Magnetometers	17.7-25.7	1:1.45
Rocket motor assembly	21.2-31.8	1:1.50
Louvers	9.6-28.1	1:2.93
Command decoder and remote command processor	62.3-1188.0	1:19.1
PCM encoder	21.4-611.0	1:28.6

component price is highly dependent on quantity procured, i.e., the quantity ordered at one time, not the total quantity over time. The table below shows what may be an extreme case, but it illustrates a point on which vendors agree--six S-band transponders bought one at a time will cost substantially more than six procured in one buy.

INFLUENCE OF SIZE OF BUY ON COST

Buy	Unit Price ($)	Cost-Reduction (%)
1	306,000	0
2	294,000	3.9
3	267,000	12.7
4	227,000	25.8

The same principle obtains at the system level, but the cost there is more a function of production rate than quantity. A manufacturer may have a fixed, sustaining cost of, say, $1 million per year whether he builds one spacecraft or four. The hypothetical example below illustrates the effect of rate in such a situation.

Annual Rate	Sustaining Cost per Spacecraft
1	1,000,000
2	500,000
3	333,333
4	250,000

The equation used to adjust recurring costs for quantity effects was:

$$f = .8 + 1.97\ n^{-1}$$

where f = adjustment factor applied to cost

n = total number of spacecraft procured

$f = 1$ if $n \leq 10$.

Cost-Estimating Equations

AEM Cumulative cost = 2.28 n (f).

STPSS

Spin $= 2.866\ f\ n + 1.743\ f_1\ n_1$

Low-cost $= 2.866\ f\ n + 2.812\ f_2\ n_2$

Precision $= 2.866\ f\ n + 3.995\ f_3\ n_3$

where n = number of core modules

n_1 = number of spin models

n_2 = number of low-cost modules

n_3 = number of precision modules.

MMS

Regular: Cumulative cost $= 8.965\ n_1\ f$

SPS-I $= 9.350\ n_2\ f$

Calculation of f includes 20 MMS procured by NASA over 10-year period.

L-AEM

Baseline: Cumulative cost $= 4.815\ n_1\ f$

Precision $= 5.678\ n_2\ f$

Spin $= 3.706\ n_3\ f$.

The remainder of Appendix A consists of tables showing estimated 10-year program costs of spacecraft and shuttle launches for various procurement options.

Table A-1

SPACECRAFT COSTS--NOMINAL CASE

($ millions)

Maximum AEM and L-AEM-BL altitude = 1000 n mi
AEM and L-AEM-BL orientation = Earth only

Payloads	114		228	
Payloads/spacecraft	13	6	13	6
Spacecraft Type	Cost			
AEM	2.3	6.8	11.4	16.0
STPSS				
Nonrecurring	22.9	22.9	26.9	26.9
Spin	---	---	26.2	40.5
Low-cost	57.1	90.7	67.8	110.9
Precision	35.0	34.3	41.8	47.5
Total	117	155	174	242
AEM	2.3	9.1	11.4	20.5
MMS	108.8	162.5	155.6	251.1
Total	111	172	167	272
STPSS				
Nonrecurring	22.9	22.9	26.9	26.9
Spin	---	---	20.0	41.6
Low-cost	62.5	109.6	88.4	150.1
Precision	34.5	34.2	43.6	52.1
Total	120	167	179	271
MMS	117	190	176	297
L-AEM				
Nonrecurring	9.8	9.8	11.3	11.3
Spin	---	---	14.3	31.1
Precision	86.9	123.9	108.9	168.3
Total	97	134	135	211
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	20.4	29.6	19.0	31.4
Spin	---	---	15.2	32.4
Precision	29.5	28.7	33.1	36.5
STPSS				
Nonrecurring	18.4	18.4	18.4	18.4
Low-cost	41.5	73	63.5	106.2
Total	122	162	164	239

Table A-1 (Cont.)

Payloads	114		228	
Payloads/spacecraft	13	6	13	6
Spacecraft Type	Cost			
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	18.7	26.8	17.9	29.8
Spin	--	--	14.3	31.0
Precision	65.1	93.4	88.2	133.7
Total	96	132	135	209
AEM	2.3	6.9	11.4	16.0
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	14.2	13.6	4.5	4.3
Spin	--	--	21.5	31.3
Precision	65.8	94.4	88.8	135.3
Total	94	127	141	201
L-AEM				
Nonrecurring	9.8	11.9	11.3	11.3
Baseline	--	26.8	--	--
Spin	--	--	3.8	10.8
Precision	58.5	93.4	76.6	104.1
MMS	33.7	--	58.5	142.5
Total	102	132	150	269
AEM	2.3	9.1	11.4	20.5
L-AEM				
Nonrecurring	9.8	9.8	11.3	11.3
Spin	--	--	3.8	10.8
Precision	58.5	96.2	76.6	104.1
MMS	25.5	16.5	34.3	80.4
Total	96	132	137	227

Table A-2

SPACECRAFT COSTS WITH ADDED CAPABILITIES:
UPGRADED AEM AND L-AEM-1

($ millions)

Maximum AEM and L-AEM-BL altitude = Geosynchronous
AEM and L-AEM-BL orientation = Earth and sun

Payloads	114		228	
Payloads/spacecraft	13	6	13	6
Spacecraft Type	Cost			
AEM	17.1	33.0	37.8	59.5
STPSS				
Nonrecurring	22.9	22.9	26.9	26.9
Spin	—	—	31.6	27.2
Low-cost	32.3	32.5	25.3	44.7
Precision	38.2	38.4	45.1	53.0
Total	111	127	167	211
AEM	12.2	26.7	46.1	66.6
MMS	56.1	73.4	65.4	127.6
Total	68	100	112	194
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	52.2	85.3	67.6	115.8
Spin	—	—	14.3	31.0
Precision	27.3	25.9	31.1	34.7
Total	91	123	128	196
AEM	2.3	6.9	11.4	16.0
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	47.5	72.7	54.6	91.2
Spin	—	—	21.6	31.3
Precision	27.4	26.2	31.3	35.1
Total	89	118	133	188
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	35.5	85.1	52.2	72.5
Spin	—	—	3.8	10.8
Precision	17.5	25.9	21.0	26.9
MMS	33.7	—	58.5	142.5
Total	99	123	150	267
AEM	2.3	9.1	11.4	20.5
L-AEM				
Nonrecurring	11.9	11.9	14.5	14.5
Baseline	35.5	60.3	52.3	72.5
Spin	—	—	3.8	10.8
Precision	17.5	27.1	16.4	20.8
MMS	25.5	16.5	34.3	80.4
Total	93	125	133	220
L-AEM				
Nonrecurring	9.8	9.8	11.3	11.3
Spin	—	—	14.3	31.1
Precision	86.9	123.9	108.9	168.3
Total	97	134	135	211

Table A-3

LAUNCH COSTS--NOMINAL CASE

($ millions)

Maximum AEM and L-AEM-BL altitude = 1000 n mi
AEM and L-AEM-BL orientation = Earth only
Space shuttle cost/launch = $15.4 million
Kick stages = Solid rockets

Payloads	114		228	
Payloads/spacecraft	13	6	13	6
100% weight attribution				
AEM/STPSS	37	51	65	93
AEM/MMS	41	63	68	111
STPSS	37	51	60	94
MMS	41	67	65	112
L-AEM	36	50	58	90
AEM/L-AEM	36	50	62	90
L-AEM/MMS	34	51	57	96
AEM/L-AEM/MMS	33	50	57	89
50% weight attribution				
AEM/STPSS	26	37	48	69
AEM/MMS	27	44	46	77
STPSS	26	38	42	68
MMS	28	46	43	77
L-AEM	26	37	42	66
AEM/L-AEM	26	37	46	67
L-AEM/MMS	24	37	39	68
AEM/L-AEM/MMS	24	37	41	66
Service charge				
AEM/STPSS	16	24	30	44
AEM/MMS	14	24	24	42
STPSS	16	24	25	43
MMS	14	24	22	41
L-AEM	16	24	25	43
AEM/L-AEM	16	24	29	44
L-AEM/MMS	14	24	22	41
AEM/L-AEM/MMS	14	24	24	42
NASA tariff				
AEM/STPSS	79	127	157	241
AEM/MMS	83	144	149	258
STPSS	79	126	131	235
MMS	83	146	134	252
L-AEM	85	134	142	247
AEM/L-AEM	85	133	167	254
L-AEM/MMS	78	134	129	242
AEM/L-AEM/MMS	77	133	141	242
Modified NASA tariff[a]				
AEM/STPSS	35	47	61	87
AEM/MMS	60	90	97	158
STPSS	34	46	55	84
MMS	61	97	92	163
L-AEM	38	51	76	92
AEM/L-AEM	44	49	76	99
L-AEM/MMS	42	51	71	92
AEM/L-AEM/MMS	38	53	66	95

[a]Assumes that whenever possible, the spacecraft and its kick stages will be oriented perpendicular to the shuttle axis.

Table A-4

LAUNCH COSTS FOR UPGRADED AEM

($ millions)

Maximum AEM altitude = Geosynchronous
AEM orientation = Earth and sun
Space shuttle cost/launch = $15.4 million
Kick stages = Solid rockets

Payloads	114		228	
Payloads/spacecraft	13	6	13	6
100% weight attribution				
AEM/STPSS	34	42	61	81
AEM/MMS	29	43	59	93
50% weight attribution				
AEM/STPSS	25	33	47	63
AEM/MMS	21	31	43	70
Service charge				
AEM/STPSS	17	24	32	46
AEM/MMS	12	20	28	44
NASA tariff				
AEM/STPSS	86	126	168	249
AEM/MMS	65	109	152	259
Modified NASA tariff[a]				
AEM/STPSS	39	50	74	95
AEM/MMS	45	64	89	139

[a]Assumes that whenever possible, the spacecraft and its kick stages will be oriented perpendicular to the shuttle axis.

Table A-5

LAUNCH COSTS FOR THE L-AEM-1[a]

($ millions)

Maximum L-AEM-BL altitude = Geosynchronous
L-AEM-BL orientation = Earth and sun
Space shuttle cost/launch = $15.4 million
Kick stages = Solid rockets

Payloads	114		228	
Payloads/spacecraft	13	6	13	6
100% weight attribution				
L-AEM	36	49	57	89
AEM/L-AEM	35	49	62	81
L-AEM/MMS	33	50	56	91
AEM/L-AEM/MMS	33	49	56	88
50% weight attribution				
L-AEM	26	37	41	66
AEM/L-AEM	26	36	45	62
L-AEM/MMS	23	37	39	66
AEM/L-AEM/MMS	24	37	40	65
Service charge				
L-AEM	16	24	25	43
AEM/L-AEM	16	24	29	42
L-AEM/MMS	14	24	22	41
AEM/L-AEM/MMS	14	24	24	42
NASA tariff				
L-AEM	85	133	141	247
AEM/L-AEM	85	133	167	246
L-AEM/MMS	76	133	128	237
AEM/L-AEM/MMS	77	133	141	241
Modified NASA tariff[b]				
L-AEM	38	51	76	92
AEM/L-AEM	44	49	76	90
L-AEM/MMS	41	51	71	92
AEM/L-AEM/MMS	39	54	65	95

[a]Only the L-AEM-BL configuration is modified to give it geosynchronous and sun orientation capability.

[b]Assumes that whenever possible, the spacecraft and its kick stages will be oriented perpendicular to the shuttle axis.

Appendix B

POWER SUBSYSTEM: A COMPARISON OF AEM, STPSS, AND MMS

by

N. E. Feldman and P. A. CoNine

BASIC DESCRIPTION OF THE AEM[1]

The AEM spacecraft comes in two versions: both have a standard 28 V power bus, a single 10 Ah rechargeable nickel cadmium (NiCd) battery, and are powered by two fixed arrays (not sun tracking) with approximately 23 sq ft of solar cells. (For further details, see Table B-1.) The solar-cell arrays can provide a peak power of 238 W end-of-life (EOL) when the sun angle is most favorable. Because the arrays do not sun track, the average power produced during illumination is about 130 W. However, to optimize power output in the orbit planned for SAGE, the two solar arrays are driven to an angle of ±50 deg with respect to the local horizontal. These motors are shown in the power subsystem diagram of Fig. B-1.

Up to 50 W can be provided to the experiment module with a voltage regulation of 28 V ±2 percent. Voltage regulation to the experiments is relaxed for peak pulse loads above 50 W, e.g., the regulation is relaxed to ±5 percent when the experiments require a peak pulse load of 120 W.[2] This peak pulse load option is used on the SAGE vehicle, where the specification states that this 120 W load must be handled for a maximum of 4 sec. Although the 4 sec time period is the specified value, the spacecraft may be able to handle this amount of experiment power for up to a few minutes.

The HCMM vehicle power budget during normal orbital operation, i.e., standby, is:

Experiment	22 W
Telemetry	4 W
Attitude control and determination	12 W
Power circuitry	12 W
Total	50 W

Table B-1

POWER SYSTEM COMPARISONS

Characteristic	AEM[(1)]	STPSS[(3)]	HMS[(4)]
Voltage level	28 ±4 V dc at bus ±2% to experiments	28 ±5 V dc at bus optional 28 V ±0.5 V to experiments (±1.8%)	28 ±7 V dc[m]
Array			No array on base module
Average power during illumination	133 W[a]	1200 W max[i]	
Average power over low altitude orbit	68 W[a]	500-600 W nominal[i]	1200 W max, bus rating[n]
Material	N/P silicon	N/P silicon	
Resistance	1 to 3 ohm-cm	2 ohm-cm	
Size of solar cells	2 × 2 × 0.03 cm	2 × 4 × 0.036 cm	
Efficiency	11%	∼ 10%	
Cover glass thickness	6 mils	6 mils	
Total dimensions of array	Each panel consists of 2 strings × 82 cells in series × 5 in parallel × 6 panels on each of two non-sun-tracking paddles	Each panel consists of 2 strings of 96 cells in series by 3 in parallel (50 W/panel-EOL max) up to 24 panels	
Total area of array	23.2 sq ft	6 sq ft/panel	
Array power/ft^2 EOL	10.3 W/ft^2[b]	8.3 W/ft^2	
Total weight of array and support structure	19.6 lb	132 lb	
Spacecraft power consumption, excluding experiments	∼ 50 to 80 W[c]	92-197 W[j]	350 W
Power available for experiments	40 to 50 W[d]	∼ 400 W nominal[i]	850 W max
Kind of battery	NiCd	NiCd	NiCd
Battery rating	10 Ah	3 × 20 Ah[k]	2 × 20 Ah baseline or up to 3 × 20 Ah or 1 to 3 × 50 Ah[o]
Battery coefficient, Ah/lb	0.49	0.38	0.40
Number of batteries	1	3	1 to 3
Depth of discharge[e]	14% (BOL); 16.6% (EOL)[f]	25%	25% low earth orbit; 50%[p] synchronous orbit
Power available during eclipse[g]	46 Whr	420 Whr	280 Whr for 2 × 20 Ah battery or 1050 Whr for 3 × 50 Ah battery[q]
Weight of battery, power conditioning and distribution	51.1 lb	253.3 lb	334 lb[q]
Battery charging method	Across both solar arrays in parallel	Separate control for each battery	One power regulating unit for all batteries[r]
Dissipation of excess power	"Shunt resistors"[h]	"Shunt modules"[ℓ]	Peak power tracker,[s] excess power is left on the array, there is a 2 to 5°C rise in array temperature

NOTES TO TABLE B-1

From Refs. 1 and 2:

[a]This is the average power produced by the stationary array during illumination. At an optimum sun angle, a maximum of 238 W can be produced. Assuming a low earth orbit illumination interval of approximately 60 min, the solar array power output is 7952 Wmin corresponding to 7952/60, or 133 W. Average power available for the orbit is 68 W, which can be derived in the following way:

$$\frac{7952 \text{ W} \cdot \text{min}}{59.1 \text{ min} + \frac{42.9 \text{ min}}{0.75}} = 68 \text{ W}$$

where 59.1 min is the period of illumination and 42.9 min is the period of occultation during low earth orbit. The factor of 0.75 is the derived overall battery efficiency.

[b]Based on maximum array output of 238 W.

[c]The HCMM vehicle, excluding experiments, uses 59 W during a data pass. The SAGE vehicle uses 47 W to 79 W for the portions of the mission discussed in the text. The remainder of the power produced during illumination is used for battery charging.

[d]Fifty watts could be available for an appreciable fraction of the orbit, but the orbital average power that could be made available for experiments and telemetry of the experimental data is no more than 40 W. This assumes 68 W orbital average available: 12 W for attitude, 12 W for power subsystems, and 4 W for housekeeping telemetry.

[e]Depth of discharge is given for the low orbit case, which is the higher stress one because of the high frequency of occultation. Depth of discharge for synchronous orbit can be as high as 62 percent.

[f]During prelaunch, launch, and completion of the acquisition phase, the depth of battery discharge reaches 61.5 percent (Ref. 1, pp. 1-26). This is a one-time condition. The AEM requires only an 8 Ah battery, but a space-qualified 10 Ah battery was readily available. It proved to be more practical to incorporate the standard battery rather than to redesign the battery and charging circuits. Thus, the lower depth of discharge values (0.14 or 0.166 rather than 0.25 as on STPSS and MMS) reflect overdesign, not high risk, on STPSS or MMS designs.

[g]Calculated using depth of discharge for low earth orbit.

[h]In shunt loads, based on battery Ah and temperature monitors.

From Ref. 3:

[i]Reference 3, p. 6-1, lists a total nominal orbital average system power of 500 W to 600 W, with 400 W for experiments. Page 3-5 of the same report discusses using up to 24 panels, which would provide 1200 W in the three-axis stabilized configuration with sun tracking arrays. In the spin stabilized configuration, however, the solar arrays are

NOTES TO TABLE B-1 (Cont.)

mounted to the six faces of the space vehicle; it should be noted that not all solar cells are exposed to the sum simultaneously on this spacecraft, therefore, about $1200/\pi$ of 382 W are available on this design.

[j]Electrical power consumption of the standard STPSS modules, excluding experiments, is determined by the stabilization system used: spinning spacecraft, 92 W; three-axis earth reference, 136 W; three-axis stellar (and wheels), 185 W; three-axis stellar with hydrazine, 197 W.

[k]TRW does not recommend using batteries smaller than 20 Ah for missions requiring less than 500 W because the nonrecurring costs associated with designing a smaller capacity battery and with interface redefinition would increase program cost by about $200K to $300K. Recurring battery cost savings due to using the smaller battery are not substantial, since, typically, cell hardware contributes only 20 percent to battery total cost, with the other 80 percent due to test and quality control requirements.

[ℓ]Excess power generated by the STPSS solar array is shunted into resistive modules on the surface of the spacecraft and radiated into space.

From Ref. 4:

[m]Page 22 says, "28 ±7 V dc negative ground."

[n]The power subsystem can support an orbital average load of 1200 W in any orbit from 500 to 1665 km and at geosynchronous altitude. This includes being able to accommodate a peak load of 3 kW for 10 min, day or night. These determine the peak and average power requirements of the power regulating unit and batteries.

[o]The choice of various numbers of batteries and two sizes allows a large variation in battery capacities to be chosen to suit the particular experiment: 20, 40, 50, 60, 100, or 150 W.

[p]The most recent specification calls for a 60 percent depth of discharge in synchronous orbit instead of 50 percent.

[q]The baseline power module weighs about 254 lb, including the case, louvers, and all module attachment hardware. The heat sink louvers, which prevent thermal runaway of the switching semiconductors, weigh 12 to 13 lb. The weight of the power subsystem frame or box, i.e., without electronics, just structure, is about 54 lb; and the attachment hardware is about 25 lb. Thus, the 254 lb power system module, excluding thermal and structural elements, weighs about 262 lb. Each 20 Ah battery weighs about 50 to 53 lb; each 50 Ah battery weighs about 100 to 110 lb. Thus, for the baseline case, the weight of the battery and power conditioning is about 354 lb; and, for 3 × 50 Ah batteries, the total weight can be as much as 585 lb. Note that these figures include some structure but do not include the vehicle harness, i.e., power distribution.

NOTES TO TABLE B-1 (Cont.)

[r]While all the batteries are connected to a single power regulator unit, the unit has been designed to compensate for loss of a single cell, or even an entire battery, without jeopardizing the total power system.

[s]NASA Goddard's MMS program office has decided to use a peak power tracker rather than the separate battery charging modules, plus shunt modules typically used in direct energy transfer systems. The tracker works by tracking the peak power point of the solar array. When peak power is not required, the power regulating unit forces the solar array operating point to a lower level. Therefore, no excess power is produced which would have to be dissipated. The peak power tracker lends itself to simpler interfaces than the direct energy transfer system with shunt module dissipators.

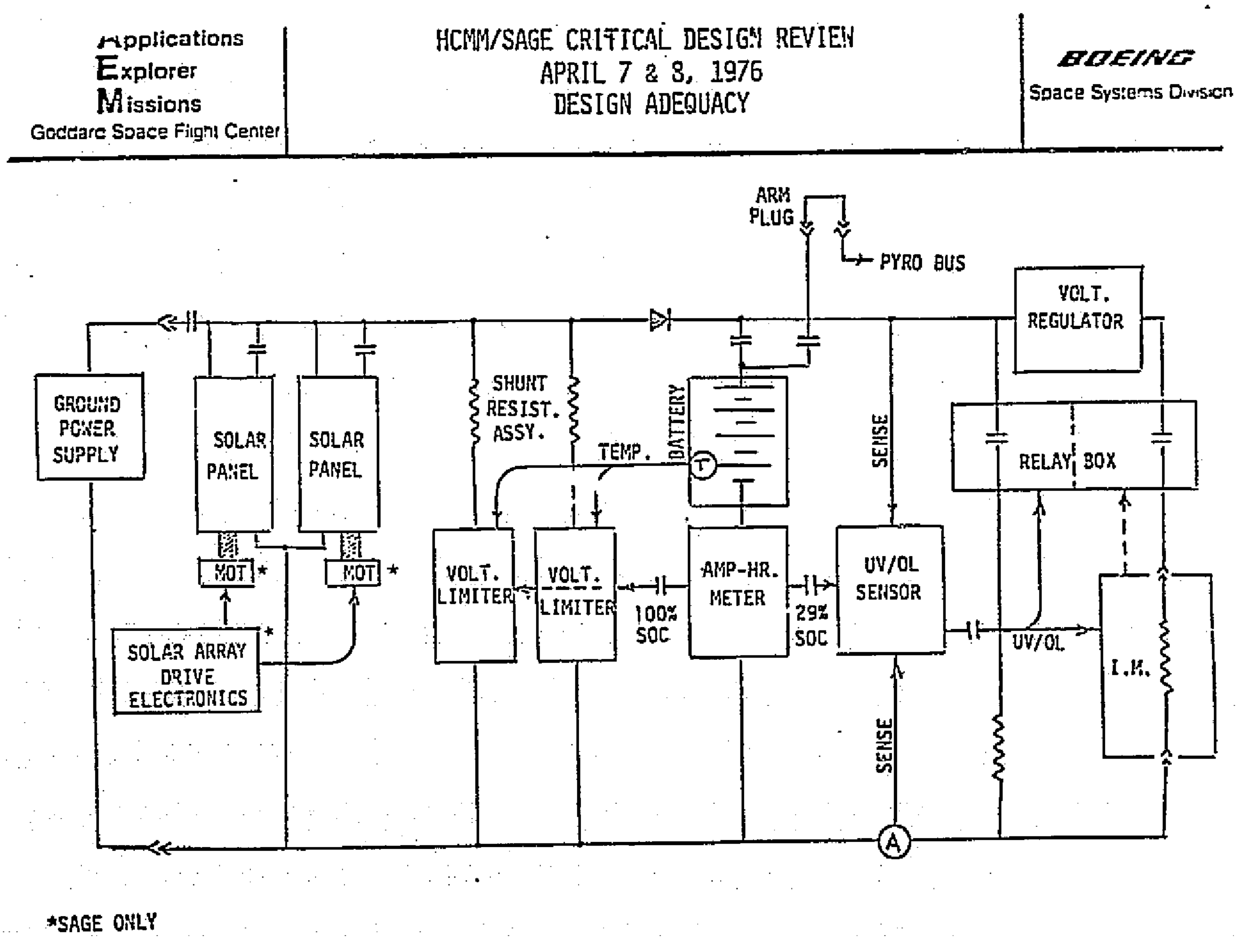

Fig. B-1—Power and distribution subsystem block diagram

The duration of the data pass is 10 min during illumination and 15 min during occultation. The HCMM vehicle power budget during data pass is roughly as follows:

Experiment	24 W*
Telemetry	35 W
Attitude control and determination	12 W
Power circuitry	12 W
Total	83 W (10 to 15 min)

The remainder of the energy produced during illumination is used for battery charging and this energy is later used by the spacecraft during eclipse. During the eclipse, 46 Whr of energy are available from the battery; this is about 75 percent of the energy used in charging the battery. Examination of the power system performance for the HCMM and SAGE missions indicates that about half the energy out of the arrays is used for battery charging.

On the SAGE vehicle, there are some high short-duration loads (less than 4 sec) from the experiment and from the tape recorder.† The timing for the experiment module is such that the tape recorder peak demands and experiment peak demands do not occur at the same time; the power system is not adequate for this. The telemetry subsystem requires 18 W to 21 W, except during tape dump (once per day), when this subsystem uses 51 W of power (500 sec duration). The total SAGE power demand during tape dump is:

Standby power to experiment	9 W
Telemetry	51 W
Attitude control and determination	16 W
Power circuitry	12 W
Total	88 W (500 sec)

*Tables of subsystem electric load demands provided by Boeing show an HCMM payload total power consumption of 34 W during a data pass. However, a total of the entries adds only to 24 W. Either there is an error in a table entry, or else there is a mistake in addition.

†One such load is the 120 W pulse option (2 to 4 sec duration) to the experiments. The experiment module, which includes the experiment and a tape recorder, requires only 9 W during standby but can draw a maximum pulse power of 117 W during acquisition (4 sec duration).

The maximum experiment power for durations of more than a few seconds is required by the SAGE experiment (during the track interval), not the HCMM experiment. The power breakdown for the SAGE vehicle for the 180 sec track interval during data taking is as follows:

Experiment	43 W
Telemetry	19 W
Attitude control and determination	16 W
Power system circuitry	12 W
Total	90 W (180 sec)

The power consumed by experiments plus telemetry can be high for short periods of time, e.g., it is 59 W for 10 to 15 min and 62 W for 3 min.

DESCRIPTION OF STPSS[3] AND COMPARISON WITH AEM

The STPSS spacecraft also has a 28 V bus, but its voltage regulation is not quite as stringent as the AEM (±5 V rather than ±4 V, as shown in Table B-1). Additional power regulation equipment (±1.8 percent regulation) can be added if the experiments require it (optional), but the associated weight and power loss are not mentioned. The STPSS spacecraft is equipped with three 20 Ah batteries and up to 24 solar panels may be used in two arrays. These arrays can provide up to 1200 W maximum (during illumination) in the three-axis stabilization configuration with sun tracking. Use of the same 24 panels around a spinning spacecraft will generate only about $1200/\pi$, or 380 W.* Spacecraft subsystems, excluding experiments, require approximately 100 to 200 W, depending on which one of four stabilization techniques is used. A block diagram of the STPSS power subsystem is shown in Fig. B-2.

The STPSS spacecraft can supply substantially more power for experiments than the AEM, i.e., 400 W compared to 40 W. Short-term peak load data comparable to those available for the AEM are not available for the STPSS. Other characteristics, shown in Table B-1, are relatively standard.

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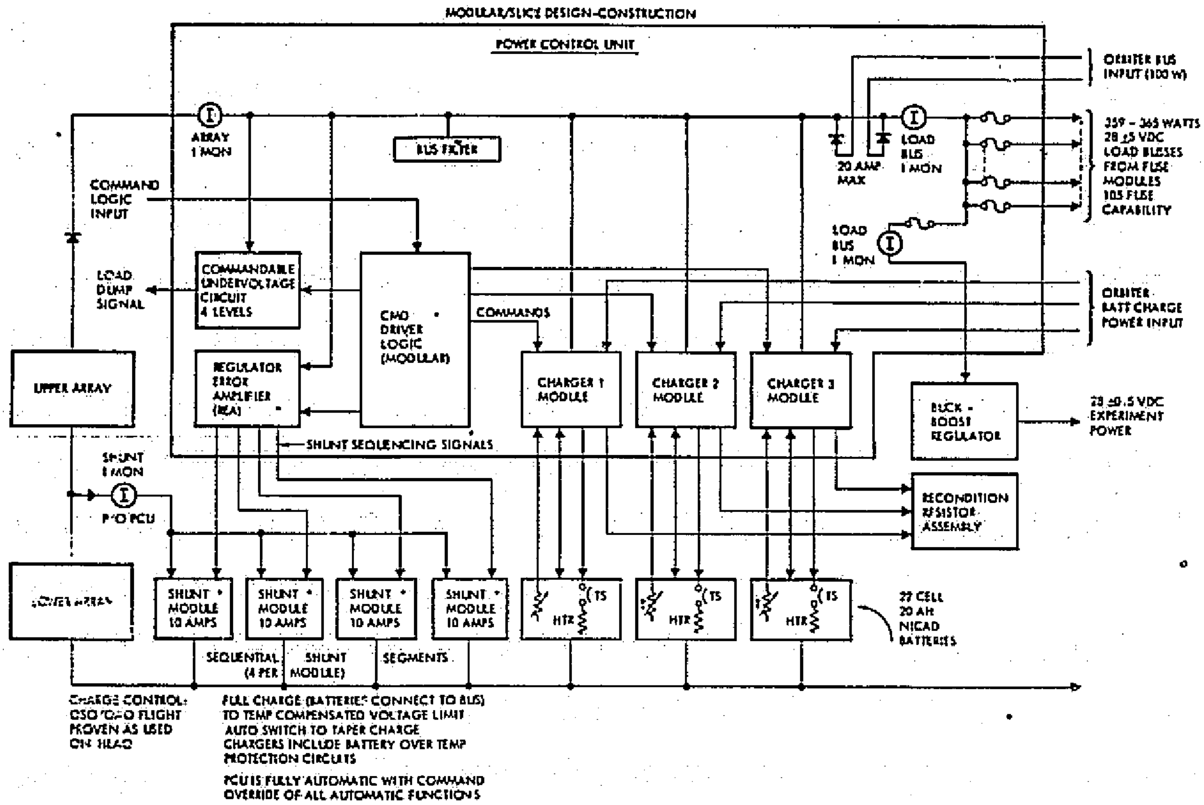
*The average power available for experiments over an orbital period also depends on the orbit.



SOURCE: Ref. 3.

Fig. B-2—STPSS standard satellite power subsystem

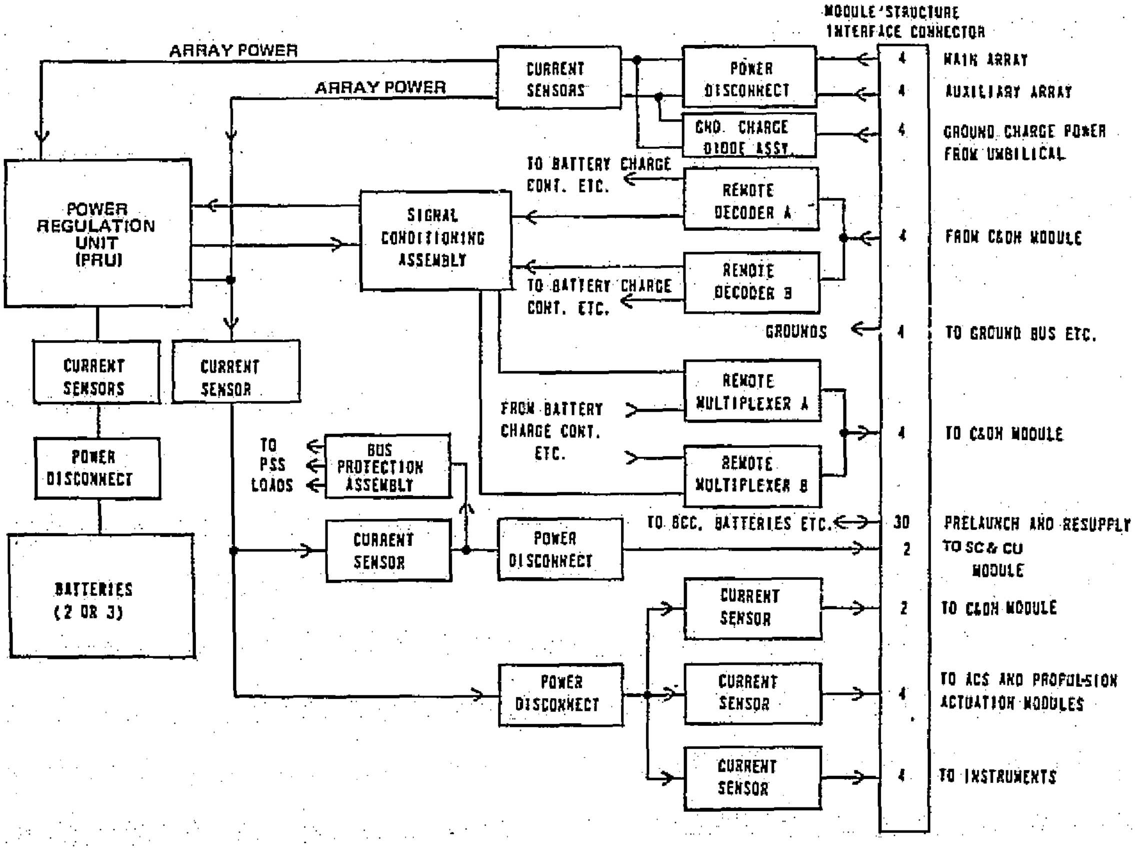
DESCRIPTION OF MMS[4] AND COMPARISON WITH STPSS

The MMS is the largest spacecraft of the three. The MMS base module does not include an array and the assumption is made that any array that is adequate for each payload can be easily incorporated.

The MMS power regulation system has been designed with an emphasis on simplified interfaces and substantial redundancy. The spacecraft is designed to be able to handle orbital average powers up to 1200 W (this would require a peak power from the array of 2400 W or more in a low altitude earth orbit). Power to the spacecraft loads and the batteries is controlled through a switching type of series regulator--the PRU, or power regulating unit. The PRU is designed to adapt to power array levels between 600 and 3600 W; its efficiency ranges from about 0.88 to 0.96. The nominal battery configuration is two of 20 Ah each. However, one to three batteries with either 20 or 50 Ah ratings can be accommodated.

When the MMS is shuttle launched, there should not be a large cost impact associated with integration and testing for every new array, since the shuttle imposes less size constraints and lower stresses (vibration, acoustic) than previous launchers.

All of the MMS batteries and spacecraft loads are controlled by a single PRU (see Fig. B-3). In the event of a single battery cell



SOURCE: Ref. 4.

Fig. B-3—MMS block diagram--power subsystem module

failure caused by a short circuit, the PRU can change its (voltage/temperature) operating point to accommodate the lower battery terminal voltage; while this will underutilize the undamaged batteries by one cell out of 22, the total energy available will still be more than if the battery with the failed cell were placed off line.* In the STPSS,

*In the three-battery case, two cells out of 66 are sacrificed because of the one cell failure, while open circuiting a single battery sacrifices 21 cells.

each battery has its own charge control unit. The latter is frequently considered a more reliable system in the event of a single point failure and has been the system considered preferable by the Air Force. Replacement of the MMS power system with one similar to that used on the STPSS would require a substantial amount of redesign.

The PRU, however, has considerable redundancy: two peak power tracking circuits, two bias supply circuits (bias converters with separate fuses), three control logic circuits, and six switching regulators (each rated for 600 W or 18 A maximum). With little additional cost or time, it is possible to arrange two regulators in parallel to supply each of three batteries, with separate logic control for each pair of regulators. The battery outputs would be diode isolated from the load bus. These modifications would result in a battery charging system more analogous to that of the STPSS.

The unregulated bus voltage (28 ±7 V) was selected to permit extraction of the full Ah rating from the battery, even after several years of aging when the discharge voltage may have decreased to as low as 21 V. On the high side of the voltage range, the batteries require a maximum of 33.4 V at the terminals under worst case charging conditions (highest current level and a battery temperature of 0°C). Because the PRU has a voltage clamp at 35 V, the tolerance was set at ±7 V for symmetry. The ±7 V tolerance requires that the experiments incorporate a preregulator with a larger dynamic range than would be required for the AEM or STPSS (±4 V and ±5 V, respectively). The PRU locates the peak power point by hunting around the equilibrium value at a 70 Hz rate. The resultant 0.5 V peak-to-peak 70 Hz ripple (at a 7 A load) that the PRU imposes on the bus also must be removed by the preregulator at the input of each experiment (it is not practical to filter out so low a frequency).

The PRU is a series regulating element and thus tends to provide lower efficiency than the conventional shunt regulators, e.g., the direct energy transfer systems used on the AEM and STPSS. At synchronous altitudes, this shows up as about a 5 to 10 percent lower efficiency for the PRU approach. In addition, the PRU approach may be as much as 10 percent heavier than the direct energy transfer systems. In low earth orbits, e.g., altitudes around 300 n mi, it has been claimed that an optimized PRU may provide up to 30 percent more power than the direct energy transfer systems for arrays with long thermal time constants (τ). This is because the array is more efficient at lower temperature when it first comes out of eclipse and the PRU takes full advantage of this. For an array like Skylab, the thermal constant is about 20 min. Thus, it takes 60 min (3τ) to get to 90 percent of the final ΔT, and this is the whole illumination period. For lightweight arrays such as the Flexible Roll Up Solar Array (FRUSA), the thermal time constant is only a few minutes and the improvement over a direct energy transfer system in low earth orbit may be no more than 5 to 10 percent.

OVERVIEW

Because many maximum or average power levels can be defined for each space vehicle, Table B-2 summarizes some of the more useful values. Shorter-term peak power levels available for experiment packages may be limited by a variety of considerations unrelated to the factors that dominate in Table B-2. The regulated 28 V ±2 percent power supply for experiments on the AEM, for example, is limited to 50 W maximum; however, the regulator can supply 120 W at 28 V ±5 percent for up to 4 sec. Short-term peak power levels may be limited by the excess output of the solar array, by the battery energy storage capacity, by the surge current limit of the battery, or by the peak power handling capability of some component in the power conditioning subsystem. Short-term power levels—that is, those lasting seconds to minutes—are generally only a factor of 2 to 10 times the average power level, but only penalties such as cost, weight, or reliability inhibit the use of larger factors. Because the complete power subsystems of the STPSS and MMS are not as well defined as for the AEM, and no power-time profiles are available for each experiment, no short-term peak power summary is shown.

There is no doubt that the peak power tracker design of the MMS can squeeze more power out of a given array in a low altitude orbit than a direct energy transfer system, but the primary justification for its use on the MMS is that the array characteristics and array integration into the space vehicle need not be optimized—any handy oversized array is acceptable and can easily be integrated. In this case,

Table B-2

POWER SUMMARY

Characteristic	AEM	STPSS	MMS
Peak array power possible, W	238[a]	1200	≥3600[b]
Average array power available to space vehicle during illumination, W	133	1200	3600
Average power available over a low altitude orbit,[c] W	68	500-600	1200[d]
Average spacecraft housekeeping power, excluding experiments and associated telemetry, W	28	100-200	350
Continuous or average power available for experiments over a low altitude orbit, W	40	400	850

[a]The 238 W is the peak of the power curve which roughly resembles a positive half sine wave, since the array is not sun tracking.

[b]The 3600 W is set by the peak power handling capability of the PRU; actually, there is no maximum since still higher power arrays would merely be used less efficiently. The excess electrical power would not be drawn from the array, which merely results in a slightly higher array temperature.

[c]This assumes that power is supplied at a constant rate to the spacecraft loads over the entire low altitude orbit and that the battery capacity is adequate to store the energy required over the period the array is occulted.

[d]The power bus is rated for 1200 W maximum, limiting the total load which can be supplied.

however, optimizing the array power output is not likely to prove necessary. Thus, there is a clear dichotomy in emphasizing peak power tracking for efficiency in a multipurpose vehicle.

Some of the ±7 V variation of the MMS bus must be due to series voltage drop in the PRU. In addition to this slow dc variation, there is a superimposed 70 Hz ripple caused by hunting of the peak power tracker about the optimum. While this has been measured to be about

0.5 V peak to peak at a 7 A load (it is limited by the low impedance of the batteries), it may be as much as 3 V peak to peak around the maximum 40 A load. Virtually all experiment packages will require their own preregulators to remove both variations, i.e., the ±7 V dc and 3 V peak-to-peak 70 Hz ripple. Series type preregulators are simple, lightweight, and reliable, but excess power must be available, since their efficiencies over so large a range is poor, i.e., 50 to 60 percent. Furthermore, the additional preregulator dissipation at each experiment package increases thermal problems. Switching regulators (dc-to-dc converters) are more complex, heavier, and require more filtering to control electromagnetic interference but offer efficiencies of 85 to 90 percent or more.

The entire problem can be eliminated by installing one large preregulator (e.g., 28 V ±2 percent) for the entire spacecraft. Where this decision has been made late in a program, it has resulted in spacecraft with unnecessary duplication--the experiments already contained preregulators and too much expense and delay was involved in removing them once they had been designed into the experiment packages. A new MMS specification, which provided for only a one year life and less extreme battery and ripple conditions, would place much less burden on the experiment packages.

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3. Taber, John E., *Space Test Program Standard Satellite Study*, TRW Systems Group, 23590-6008-TU-00, October 30, 1975.

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Appendix C

COMMUNICATIONS AND DATA HANDLING SUBSYSTEM: A COMPARISON OF AEM, STPSS, AND MMS

by

P. A. CoNine

Table C-1 summarizes the communications and data handling (C&DH) subsystems for the AEM, the MMS, and the STPSS. It can readily be seen that the three C&DH systems are substantially different and not compatible. Major differences include frequencies, modulation, formats, data rates, polarization, and security equipment. None of the C&DH equipment on the three spacecraft is beyond or even pushing the state of the art. Most of the equipment on the AEM and STPSS has been used on previous spacecraft. While some of the MMS equipment will be new, it is presently in the latter stages of development. Because the STPSS missions are not concerned with cross-linking data to another spacecraft, it is not necessary to pay any further attention to the TDRSS transponder.

DESCRIPTION OF THE AEM

The AEM spacecraft is currently being built by Boeing in two versions: the HCMM and the SAGE. The HCMM has a VHF command and housekeeping telemetry system and an S-band telemetry unit for experimental data; the SAGE vehicle has all communications at S-band frequencies. The command and telemetry formats are compatible with the NASA-STDN satellite tracking and telemetry system. The HCMM spacecraft is the only one in this study with a VHF command receiver and housekeeping transmitter; however, the communication system has been designed so that it can become S-band-compatible (as on the SAGE) merely by changing the transponder/transmitter-diplexer units. No further consideration will be given to the VHF system.

The AEM telemetry system has a low data rate of 1 or 8 kbps, although on the SAGE tape recorder playback can be as high as 1 Mbps. The command rate is a low 600 bps. The memory is small and is used

Table C-1

C&DH CHARACTERISTICS OF AEM, MMS, AND STPSS

Characteristic	AEM-HCMM[a](1)		AEM-SAGE(1)	MMS(2)	STPSS(5)
Telemetry and Command Band	VHF	S-band	S-band	S-band	S-band
Tracking System Compatibility	STDN	STDN	STDN[b]	STDN	SGLS
Uplink Frequency, MHz	148	2025-2120	2025-2120	2025-2120	1750-1850
Uplink Subcarrier Modulation	PCM/FSK/AM/AM	n.a.	PCM/FSK/AM/PM	PSK	Ternary FSK with AM
Command Format	60 bits	n.a.	60 bits	96 bits[f]	43 bits[j]
Command Bit Rate	600 bps	n.a.	60 bps	2K, 1K, 125 bps[g]	2 kbps
Downlink Frequency, MHz	136	2280	2200-2300	2200-2300	2200-2300
Telemetry Format:					
Word length	8 bits	8 bits	8 bits	8 bits(4)	8 bits
Minor frame length	256 words	128 words	128 words[d]	128 words	Variable(3)
Major frame length	64 minor frames	64 minor frames	64 minor frames	128 minor frames	Variable(3)
Maximum Bit Rate	1.024 kbps[b]	8.192 kbps	1 Mbps[e]	64 kbps	8-128 kbps[k]
Power Output	1/4 W	2 W	1 W, housekeeping 2 W, experiment	1.7, 3.5, or 7.1 W[h](4)	2 W[l]
Communications Security	Not available	Not available	Not available	Not available	Available
Antenna Polarization	RHCP	RHCP	RHCP	1 with RHCP and 1 with LHCP(4)	RHCP(3)
Memory Size	256 words × 32 bits/word and 256 words × 16 bits/word[c]		256 words × 32 bits/word and 256 words × 16 bits/word	16K bits × 18 bits/word[i]	8K words × 32 bits/word[m]
Tape Recorder Capacity	None	None	4.5×10^8 bits	Up to 9×10^8 or up to 8×10^9 bits (optional)	10^8 bits

[a]Two versions of the AEM spacecraft are currently being designed by Boeing: HCMM and SAGE. The HCMM vehicle uses the VHF band for commands and for housekeeping telemetry and S-band for downlink experimental data. The SAGE mission uses S-band for commands, telemetry, and data.

[b]Data rate during the boost phase is 8192 bps.

[c]Commands are compared with words in a 256 word, 16 bits/word PROM (Programmable Read Only Memory). Delayed commands are stored in the remote command processor, which consists of a 256 word, 32 bits/word CMOS/RAM (Complementary Metal Oxide Subtrate/Random Access Memory) semiconductor memory (pp. 1-134 to 1-136 of Ref. 1).

[d]Assumed the same as the HCMM vehicle because no change is indicated.

[e]Tape recorder playback rate. Real time data rate is limited to 1 kbps or 8 kbps. A new encoder would be required if higher bit rates are needed.

[f]Reference 3 lists the command format as fixed at 96 bits (48 bit introduction and 48 bit command word). Page 34 of Ref. 2 lists the command format as 48 bits (which can be assumed to be only the command word portion of the total format).

[g]With use of the 2000 bps command rate, a single 5 min command contact per day is required for loading of commands in the on-board computer. This command load will allow the computer to operate the spacecraft for periods of 24 to 72 hr.

[h]Mission selectable.

[i]The MMS C&DH computer includes storage for attitude control information, as well as commands. The STPSS C&DH computer is used only for storing commands, and a separate computer handles attitude control. Therefore, the apparent large difference in the capacities of the two C&DH computers is one of definition not actual capability.

[j]SGLS itself has variable command formats. Page 8-3 of Ref. 5 shows a 43 bit format as TRW's conception of what is required.

[k]By changing subcarriers, this can be increased to 256 kbps. This is SGLS's maximum capacity.

[l]If appreciably higher data rates or more services are desired, there is provision for the standard 2 W transmitter to be used to derive a higher power transmitter (e.g., 20 W) in the payload segment.

[m]Word length deduced from data bus supervisory line formats, p. 8-5 of Ref. 5.

chiefly for storing commands for later processing and for verifying received commands with those stored in memory.

DESCRIPTION OF STPSS AND COMPARISON WITH AEM

The STPSS spacecraft is designed for Air Force missions. It has an S-band communication system which can handle a maximum command rate of 2 kbps and telemetry rates of 256 kbps. It is SGLS-compatible and uses ternary frequency-shift keying (FSK) coding. An on-board computer can handle stored commands, telemetry storage, format control, and memory dumps. Data and commands can be encrypted if necessary.

The C&DH for the STPSS spacecraft is far more sophisticated and has a much greater capacity than that on the AEM (see Table C-1). It is doubtful if experiments of the size that would be carried on the AEM would require as sophisticated a system as presently envisioned for the STPSS. However, currently planned AEM telemetry and control equipment probably could not be used because of the basic incompatibility of the NASA-STDN and AF-SGLS systems.

To make the AEM compatible with the SGLS system requires replacing the S-band transmitter and the S-band transponder, the command demodulator, and modifying or replacing the PCM encoder and the command decoder/processor. Personnel at Boeing indicate that the "black boxes" can be replaced one-for-one with SGLS-compatible equipment without causing major spacecraft redesign. It appears that SGLS-compatible equipment exists that could be used on the AEM. Encryption and decryption units can be added to SGLS equipment if required, but not to STDN. There is some question whether the AEM can meet the signal isolation requirements of encrypted missions. However, Boeing personnel state that an SGLS-compatible AEM can have encryption capability. Items such as the sequencer timer and remote command processor are one-time programmable, with the programming dependent on the spacecraft and mission, and could be used with the proper programming. The STPSS's bus controller, computer, and data interface units are more sophisticated than anything currently on the AEM. The functions that these would handle on the AEM are done as part of the PCM encoder and the command decoder/processor, although those done on the AEM are simpler.

Changes required to make the AEM compatible with SGLS are summarized in Table C-2.

DESCRIPTION OF MMS AND COMPARISON WITH STPSS

The MMS is a large NASA multimission modular spacecraft. Like the STPSS, the C&DH system is capable of transmitting high data rates and has a computer on board for data processing and formatting. However, as is shown in Table C-1, the MMS and STPSS C&DH systems differ substantially because of the STDN-SGLS incompatibilities. The uplink frequency, uplink subcarrier modulation, antenna polarization, communication security protection, and command format differences necessitate the following changes:

1. Replace the STDN transponder with an SGLS transponder.
2. Replce the phase-shift keying (PSK) demodulator with an SGLS signal conditioner (includes PSK demodulator).
3. Modify the signal conditioner output, modify the command decoder input, or add a suitable piece of equipment between the two to make the signal conditioner and the command decoder compatible.
4. Redesign the MMS omni antenna.

Further details on interchanging STDN/SGLS communication components are summarized in Table C-3. While the differences between the two C&DH systems are substantial, it is possible that proper preliminary design of the spacecraft would enable communication black boxes to be interchanged with minimal impact. However, if a decision is made late in the design cycle, substantial problems will most likely occur. Available STPSS equipment could be used directly on the MMS. Capabilities are similar, so sizes, weights, and power requirements should be also.

Table C-2

C&DH CHANGES REQUIRED TO RUN STP MISSIONS ON THE AEM

AEM[a] Equipment	Changes to AEM for STP Compatibility	STPSS Equipment
Antennas	Usable	Antenna
Hybrid	Usable	Hybrid
S-band transmitter S-band transponder	Replace[b]	Receiver Transmitter
Command demodulator	Replace[c] Add (if necessary)[d]	Dual signal conditioner Decryption unit
Command decoder/processor	Modify or replace[e]	Command decoder
PCM encoder	Modify (if necessary)[f] Add (if necessary)[g]	Dual baseband unit Encryption unit
Tape recorder	Usable[h]	Tape recorder Bus controller (data formatter)[i]
Sequencer timer[i]	Modify (if necessary) Not on AEM[j]	 Computer
Remote command processor	Modify[j] Not on AEM[k] Usable[ℓ]	 Data interface unit Harness

[a]Only AEM S-band equipment as on the SAGE will be considered.

[b]The AEM spacecraft uses one antenna and transmitter for experimental data transmission and another antenna and a transponder for receiving commands and broadcasting housekeeping information. Because of differences in the uplink frequencies, at least the receiver portion of the transponder must be replaced. If the current AEM communication configuration is to be maintained, a transponder and a transmitter or two transmitters and one receiver, are required. It may be possible to use STPSS receivers and transmitters on the AEM. Otherwise, several other SGLS-compatible transmitter/receivers have flown or will fly on FLTSATCOM (Fleet Satellite Communication System), P72-1, P72-2, and the S-3.

[c]The STDN-compatible AEM command demodulator operates with binary FSK coding. SGLS uplinks are ternary FSK so this unit must be replaced. The receiver-demodulator unit on the S3 vehicle may be an appropriate replacement for the receiver and demodulator on the AEM (capacity is 1000 bps).

[d]AEM requirements do not include a secure uplink. If a secure uplink is required, then a decrypter must be added between the signal conditioner and the command decoder and these items modified accordingly.

NOTES TO TABLE C-2 (Cont.)

[e]The command decoder processor can be retained for clear uplinks. However, the Air Force Satellite Control Facility (AFSCF) command format would have to be compatible with the decoder and new software would be required. This affects the STPSS.

[f]The SGLS ground system can process PCM signals, however, some modification may be necessary because the AEM uses biphase L Manchester coding and the STP biphase M. However, the current AEM encoder has no provision for dual baseband, which may or may not be necessary for small STP missions run on the AEM. The STPSS dual baseband unit is not directly substitutable on the AEM because it does not include encoding provisions. The P72-1, P72-2, and S-3 spacecraft have had PCM encoders with bit rates of 8, 32, and 16 kbps, respectively. These could probably be used on the AEM if higher data rates are desired.

[g]Boeing personnel state that encryption is possible of the AEM; there appears to be some question about signal isolation, however.

[h]The optional AEM tape recorder has a larger capacity than STPSS.

[i]Data formatting on the AEM occurs in the PCM encoder. Timing is provided by the sequencer timer. There is no item as sophisticated as the bus controller on the AEM; and for small experiments, it is probably not required. There should be little impact in setting the sequencer timer for STP missions. The AEM is not capable of transmitting data rates as high as the STPSS. Therefore, experiments with real time data rates over 8 kbps cannot be run on the AEM.

[j]The AEM remote command processor is not the same as the STPSS computer. The AEM processor is used simply for verifying commands and storing them for future execution. Modifying the remote control processor for SGLS-type commands should not be a major undertaking because commands are unique to a given spacecraft and its mission anyway.

[k]Experimental data on the AEM go directly to the PCM encoder. Data interface units are not really necessary on the small spacecraft.

[ℓ]Boeing says that the AEM spacecraft can be modified for SGLS compatability merely by replacing black boxes.

Table C-3

C&DH CHANGES REQUIRED TO RUN STP MISSIONS ON THE MMS

MMS Equipment	Changes to MMS for STP Compatibility	STPSS Equipment
	Payload module[a]	Antenna
	Payload module[a]	Hybrid
Diplexer		Diplexer
	Replace[b]	Receiver
Transponder		Transmitter
	Payload module[a]	Switch
PSK demodulator	Replace[c] Add (if necessary)[d]	Dual signal conditioner Decryption unit
Central command decoder[e]	Modify or replace[f] (software change)	Command decoder
Premodulation processor	Replace[g] Add (if necessary)[h]	Dual baseband unit Encryption unit
Tape recorder	Usable[i]	Tape recorder
Data bus controller[e]		
Clock and format generator[e]	Usable[j]	Bus controller (data formatter)
Standard computer interface		
Computer	Usable[k]	Computer
Remote unit	Usable[ℓ]	Data interface unit
Harness and connectors	Usable with proper design[m]	Harness
Signal conditioning and control unit	Unique and necessary[n] to MMS vehicle	

[a]The antenna or antennas and their components are considered payload-unique on the MMS. The requirement for hybrids and switches would depend on the exact placement and design of the MMS antenna system. It can be assumed that for Space Test Program missions that the STPSS antenna can be used on the MMS.

[b]Because of differences in uplink frequencies, the STDN transponder cannot be used for SGLS. Reference 5 shows a receiver and transmitter rather than an integrated transponder; however, these could be combined into an SGLS transponder.

[c]The modulation differences necessitate replacing the PSK demodulator with an SGLS signal conditioner, which includes an FSK demodulator.

[d]STPSS system requirements do not include a secure uplink. However, if a secure uplink is to be considered, it is then necessary to add a decrypter

NOTES TO TABLE C-3 (Cont.)

between the signal conditioner (that replaces the MMS PSK demodulator) and the command decoder. A KIR 23 would be considered appropriate for STPSS missions. The KIR 23 output and the decoder input would have to be made compatible by modifying the decoder input or adding a suitable piece of hardware. Further, uplink communications security equipment imposes constraints on the command word format, which in turn influences the decoder. Hence, if a secure uplink is employed, it would be necessary to modify the MMS decoder so that it is compatible with the communications security unit.

[e]These items form the STACC (Standard Telemetry and Command Components) central unit as shown in Ref. 6.

[f]The MMS command decoder can be retained for clear uplinks. However, the AFSCF command format would have to be compatible with the decoder and new software is required. The decoder could also be replaced with the STPSS one.

[g]The premodulation processor (PMP) generates a 1.024 MHz subcarrier, which is modulated by the telemetry data stream. The MMS ranging signal is not combined with the subcarrier in the PMP but is combined in the transponder; SGLS transponders usually do not accomplish the combining in the transponder (unless the transponder performs the baseband assembly function). The PMP can be retained if the SGLS transponder incorporated in the MMS departs from normal practice and combines the ranging signal with the subcarrier. If the SGLS transponder selected performs the baseline assembly function, the PMP will not be required. The PMP also includes electronics for TDRS compatibility which would serve no useful purpose on satellites communicating with the satellite control facility. It is desirable that a baseband assembly unit be substituted for the PMP.

SGLS has a capability of using two subcarriers. The need for two subcarriers at most is infrequent; the penalty for the capability of having two is also small. While it cannot be demonstrated at this time that two subcarriers are necessary, the capability of having two subcarriers available as an option is desirable.

[h]Most STP missions do not require secured downlink; thus the basic MMS configuration for STP application need not have communications security equipment. However, the communications system design must be such that it can readily accept communications security equipment without costly modifications. For those missions requiring secured downlink, communications security equipment must be added to the MMS between the telemetry format generator and the premodulation processor for downlink protection. A KG-46 is considered to be appropriate for STP programs and is expected to be available in time for use on the MMS. The spacecraft must comply with Tempest requirements to protect the classified data. Proper design practice will provide a high degree of confidence that Tempest requirements can be satisfied with little or no modification. There should be 90 dB isolation between the data and the clock, the input and output signal leads should be well shielded, and the input and output signal leads should be run in separate cables and connectors. The encryption unit would be GFE.

[i]The MMS tape recorder has a larger capacity than the STPSS one and so should satisfy all Space Test Program missions.

NOTES TO TABLE C-3 (Cont.)

[j]The MMS telemetry format and data rates offer a great deal of flexibility and can be used by STP; they will probably accommodate a large percentage of the payloads. However, there may be some penalties involved in accepting the fixed minor frame length (128 words), the fixed number of subcommutated words (4), and the fixed major frame length (128 minor frames). Supercommutation of the minor frame words and/or of the subcommutated data is provided in the MMS design and will add the flexibility. A recent change to the MMS clock will permit data rates of 128 and 256 kbps.

[k]The MMS computer is larger than that of the STPSS because it handles attitude control as well as C&DH. However, there is adequate room in the MMS computer for STP data handling.

[ℓ]The MMS remote unit is usable for STP missions assuming that the data bus controller, clock and format generator, and standard computer interface used is that of the MMS. Using the STPSS bus controller rather than these units would require using an STPSS data interface unit.

[m]Assumes an initially compatible design.

[n]Involved with solar panel deployment on MMS and is required. The STPSS vehicle has nothing comparable. It can be assumed that the changes that must be made in the decoder will not jeopardize this function.

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Appendix D

ATTITUDE CONTROL AND STABILIZATION SUBSYSTEM: A COMPARISON OF AEM, STPSS, AND MMS

by

T. B. Garber

The function of the attitude control and stabilization system is to provide the means of orienting the satellite in some specific attitude and then to maintain that orientation with acceptable angle and angular rate errors. In addition, the stabilization and control system should also be able to provide the information necessary for after-the-fact attitude determination.

Table D-1 presents the performance specification and the physical characteristics of the attitude control systems that have been proposed for three spacecraft, NASA's AEM and MMS, and the Air Force's STPSS. In the case of the STPSS design, three different attitude control systems can be incorporated into the spacecraft depending upon the level of performance required.

Of the three spacecraft designs, that of the AEM is the most firm. As can be seen from Table D-1, the performance requirements of the AEM attitude control system are quite modest. The performance of the AEM control system should, under normal conditions, exceed the specifications, with pointing errors roughly one-half those shown.

Basically, the AEM spacecraft is inertially stabilized in roll and yaw by virtue of the angular momentum of a wheel spinning about the pitch axis, normal to the orbital plane. Control of the spacecraft about the pitch axis is achieved by modulating the pitch wheel's angular rate. Errors in the spacecraft's pitch and roll attitudes are detected by a horizon scanner.

To remove the small roll and yaw errors that result from both external and internal disturbances, electromagnets are used to generate the necessary torques. A three-axis magnetometer provides the required knowledge of the earth's magnetic field vector. In addition to damping precessional and nutational spacecraft motion, the electromagnets also provide the necessary torque to unload the pitch wheel (desaturation).

Table D-1

ATTITUDE CONTROL AND STABILIZATION SYSTEMS

Characteristics and Specifications	AEM	STPSS I	STPSS II	STPSS III	MMS
Type of Stabilization	Three-Axis	Spin	Three-Axis	Precision Three-Axis	Precision Three-Axis
Performance:					
Attitude control	±1° pitch, roll ±2° yaw	1°-2° spin axis	1°-2° all axes	0.1° all axes	Less than 0.01° all axes
Rate control	±0.01°/sec all axes	--	0.01°/sec	0.003°/sec	Less than 10^{-6}°/sec all axes (long term)
Attitude determination	±0.5° pitch, roll ±2° yaw	--	0.2°-0.4°	0.02°	--
Control Torques:					
RCS	None	Cold gas, N_2	Cold gas, N_2	N_2; N_2H_4 option	Hydrazine (optional)
Momentum wheels	Pitch bias wheel, roll wheel option	None	None	3, reaction	4, reaction wheels
Electromagnets	3	None	Option	Option	3, pitch, roll, yaw
Nutation damper	None	1	None	None	None
Sensors:					
Earth	Mounted on pitch wheel	1	2, conical scan	None	None
Sun	3 head sun sensor	1	2	2	Both fine and coarse (solar array)
Star	None	None	None	2 strapdown trackers	2 strapdown trackers
Magnetic	3 axis magnetometer	None	Option	Option	3 axis magnetometer
Gyros	None	None	2 rate	4 rate (1 standby)	3 axis + redundancy
Accelerometers	1	None	None	None	None
Miscellaneous:					
Computer	Minimal	None	Yes, dedicated	Yes, dedicated	Yes, shared
Control system weight	29 lb	95 lb	165 lb	289 lb	253 lb (not including N_2H_4 RCS weight)

The AEM attitude control system does not include reaction jets as a means of torque generation. Thus there are no limits on operational lifetimes due to fuel considerations. However, magnetic torques are relatively weak and as a consequence control time constants tend to be large--on the order of an orbital period. Also, magnetic torques decrease with increasing altitude and for the AEM design, they become ineffective for altitude in excess of 1000 n mi.

The simplest of the STPSS designs utilizes spin stabilization. Thus, ideally, the spin axis of the vehicle is inertially fixed. No provisions are made for a despun platform.* A mechanical nutation damper is provided to remove unwanted spin axis wobble and cold gas jets are used to reorient or stabilize the direction of the spin axis. Sun and earth sensors are used for attitude determination.

The second STPSS design is a low-cost, three-axis system with performance specifications similar to those of the AEM spacecraft (see Table D-1). The attitude control system of this version of the STPSS differs from that of the AEM in that a pitch momentum wheel is not used to provide roll-yaw stabilization and cold gas reaction jets are the primary means of generating control torques. Two conical scan earth sensors provide pitch-roll attitude information, while a rate gyro is used to detect yaw attitude errors.

Since, without a pitch momentum wheel, this version of STPSS does not have any inherent stability, disturbances from either internal or external torques must be countered by the reaction control system. For low altitude orbits where aerodynamic and gravity gradient disturbance torques can be large, control system fuel requirements for a one-year mission might be excessive. This situation could be alleviated by adding electromagnetic torques and a magnetometer to the control system so that almost continuous use of the reaction jets would not be necessary.

The third version of the STPSS is designed to attain precise pointing accuracies and rate control. To improve performance relative to the low-cost three-axis design, two star trackers, two rate gyros,

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*In essence, the body of the AEM spacecraft is a despun platform with the pitch wheel inertially stabilized.

and three reaction wheels are added to the stabilization and control system and the two earth sensors are removed. Also, with the addition of the star trackers, a star catalog and the spacecraft's ephemeris must be ground-supplied periodically and thus an on-board computer becomes mandatory. Pointing accuracies of 0.05 deg per axis can be expected from the precision STPSS design.

Unlike the AEM design, the three reaction wheels of the precision STPSS have no momentum bias and are used only to provide reaction control torques. The primary function of the cold gas reaction jet system is to unload the wheels when they approach saturation. As in the case of the low-cost STPSS design, electromagnetic torques and a magnetometer could be added as a supplement to the cold gas system if secular disturbance torques become a problem.

The final spacecraft design to be considered is MMS. The attitude control system of this spacecraft is very similar to that of the precision STPSS. The major difference is that the MMS uses electromagnetic torques to unload the reaction wheels rather than a jet reaction system. However, a hydrazine jet reaction system can be added as an option.

The pointing accuracy specification of the MMS is ±0.01 deg per axis, which is better by a factor of five than that claimed for the precision STPSS. Since the same model strap-down star tracker assembly is proposed for both the MMS and the precision STPSS, the superior performance projected for the MMS must result from either a better gyro reference unit or more frequent stellar updates.

Considering the relatively modest STPSS attitude control performance specifications, it is apparent that all five spacecraft designs of Table D-1 are well within the state of the art. In all cases the major components that have been selected, such as earth sensors, reaction wheels, or star trackers, are developed items of equipment with a history of previous spacecraft use. The AEM and the STPSS spin stabilized configuration have the least complex attitude control systems, while the precision STPSS and MMS vehicles have the most complex systems.

Appendix E

REACTION CONTROL/PROPULSION SUBSYSTEM: A COMPARISON OF AEM, STPSS, AND MMS

by

J. R. Hiland

Comparative technical evaluations were made for the reaction control/propulsion subsystems contained in the three basic spacecraft designs discussed in this study. There are two versions of the AEM spacecraft: HCMM and SAGE. The STPSS designs encompass three basic configurations: (1) spin stabilized, (2) three-axis stabilized (low-cost), and (3) three-axis stabilized (precision). The MMS spacecraft is a single three-axis stabilized design that can employ several subsystem options within this basic categorization.

The reaction control/propulsion subsystems discussed herein use either cold gas (GN_2) or hydrazine (N_2H_4) as the propellant and perform functions such as spacecraft stabilization, reaction wheel unloading, orbit adjustment, and orbit transfer. Solid propellant rocket motors, which in some cases are also used for stabilization and orbit transfer, are considered separately and not included in this discussion.* Cold gas and hydrazine RCSs consist, essentially, of the same basic components, i.e., tank(s), fill and drain valves, isolation valves, pressure regulator and/or transducer, filters, thrusters, plumbing, and, in cases where the RCS is a separate module, some mounting structure and electrical harness. In this analysis, when the RCS is a secondary subsystem to a particular spacecraft module (usually orientation or attitude control system), the structure and harness is assumed accountable to the primary subsystem. The primary difference in cold gas versus hydrazine system components is in their relative complexity and hence cost. Other potential differences in degree of technological development within a given propellant type have essentially been nulified

*In this study, the stable of solid rocket motors described in Ref. 1 were used for the kick stages to provide orbit translation and circularization.

by the commonly adopted design goal of using flight-proven components where possible for the RCSs evaluated.

Table E-1 shows component breakdowns of the RCS for the various versions of the three spacecraft and is used as a basis for the discussion that follows. The development status of a component is indicated by either a P for flight-proven, PM for flight-proven but requiring some modification for the subject applications, or N if the item represents new hardware, such as plumbing or structure. For costing purposes in this exercise, however, new plumbing or structure can probably be treated as flight-proven, since the technology involved is not new; only the tailoring of these items for each specific configuration is required.

DESCRIPTION OF THE AEM REACTION CONTROL SYSTEM

Only the HCMM version of the AEM uses a reaction control system and it is a small hydrazine system packaged as a separate module. This orbit adjust subsystem provides a nominal 262.4 ft/sec velocity correction capability with the maximum spacecraft weight of 285.5 lb to circularize the orbit and minimize nodal drift. All components are flight-qualified and currently in production. The single 0.287 lb thrust chamber is from the NASA/GSFC IUE program and the propellant flow control valve (included as part of the total thruster assembly) will consist of two single-seat Wright Components, Inc., valves welded together in a series redundant configuration, each valve seat being controlled by a separate coil. The dual version valve, while a minor modification, has been tested by Hamilton Standard and is expected to meet all requirements. The hydrazine tank with elastomeric diaphragm is from the IUE program and needs only very minor modifications to the plumbing and mounting connections. The rest of the RCS is quite straightforward.

DESCRIPTION OF THE STPSS COLD GAS REACTION CONTROL SYSTEM AND COMPARISON TO AEM

There are two cold gas RCSs contemplated for the STPSS. The three-axis version shown in Table E-1 uses twelve 0.1 lb_F thrusters in both the low-cost and precision orientation modules for on-orbit control and reaction wheel unloading. The spin control module of the spin-stabilized

Table E-1

RCS SYSTEM COMPONENT BREAKDOWN

Item	Quantity	Size	Status[a]	Unit Weight (lb)	Total Weight (lb)	Total Cost ($)
AEM-HCMM, Orbit Adjust Module, Hydrazine						
Tanks	1	9.6" dia. (400 psi)	P	2.7	2.7	
Thrusters	1	$0.287\ lb_F$	PM	0.8	0.8	
Valves						
Drain and fill	2		P	0.15	0.3	
Isolation						
Miscellaneous						
Press. regul.						
Press. transd.	1		P	0.6	0.6	
Filters						
Plumbing			N		0.7	
Structure			N		3.8	
Total dry weight, lb					8.9	
Propellant weight, lb					10.6	
Total wet weight, lb					19.5	
STPSS 3-Axis,[b] Orientation Module, Cold Gas						
Tanks	2	13" dia. (4000 psi)	P	17.0	34.0	30K
Thrusters	12	$0.1\ lb_F$	P	0.5	6.0	60K
Valves						
Drain and fill	1		P	0.1	0.1	
Isolation	4		P	0.4	1.6	
Miscellaneous						60K
Press. regul.	2		P	1.2	2.4	
Press. transd.	1		P	0.2	0.2	
Filters	1		P	0.3	0.3	
Plumbing			N	2.0	2.0	
Structure						
Total dry weight, lb					46.6	150K[c]
Propellant weight, lb					18.4	
Total wet weight, lb					65.0	
STPSS 3-Axis and Orbit Transfer, Transfer/Orientation Module, Hydrazine						
Tanks	1	36" dia.	PM[d]	56.0	56.0	80K
Thrusters	12	$0.1\ lb_F$	P[e]	0.5	6.0	240K
	4	$4\ lb_F$	P	0.6	2.4	100K
	1	$300\ lb_F$	P	50.0	50.0	125K
Valves						
Drain and fill	2		P	0.25	0.25	
Isolation	3		P	0.8	2.4	
Miscellaneous						100K
Press. regul.						
Press. transd.	1		P	0.4	0.4	
Filters	1		P	0.3	0.3	
Plumbing			N	6.0	6.0	
Structure						
Total dry weight, lb					124.0	645[f]
Propellant weight, lb					666.0	
Total wet weight, lb					790.0	

Table E-1--Continued

Item	Quantity	Size	Status[a]	Unit Weight (lb)	Total Weight (lb)	Total Cost ($)
MMS-SPS-I, Propulsion Module, Hydrazine						
Tanks	1[g]		P	10.2	10.2	20K
Thrusters	12	0.2 lb_F	P	0.6	7.2	144K
	4	5 lb_F	P	1.25	5.0	48K
Valves						
Drain and fill	2		P	0.25	0.5	4K
Isolation	4		P	0.7	2.8	20K
Miscellaneous						
Press. regul.						
Press. transd.	2		P	0.5	1.0	10K
Filters	2		P	0.25	0.5	4K
Plumbing	25 ft		N		5.0	5K
Harness			N		14.0	
Structure			N		29.0[h]	
Total dry weight, lb					75.2	
Propellant weight, lb					55.0	
Total wet weight, lb					130.2	
MMS-SPS-II, Propulsion Module, Hydrazine						
Tanks	1	36" dia. × 55.5 cylindrical	PM[i]	125.0	125.0	100K
Thrusters	12	0.2 lb_F	P	0.6	7.2	144K
	4	5 lb_F	P	1.25	5.0	48K
Valves						
Drain and fill	2		P	0.25	0.5	4K
Isolation	4		P	0.7	2.8	20K
Miscellaneous						
Press. regul.						
Press. transd.	2		P	0.5	1.0	10K
Filters	2		P	0.25	0.5	4K
Plumbing	25 ft		N		5.0	5K
Harness			N		14.0	
Structure			N		81.0[h]	
Total dry weight, lb					242.0	
Propellant weight, lb					1050.0	
Total wet weight, lb					1292.0	

[a] P = flight-proven; PM = flight-proven but requires some modification; N = new hardware.

[b] Spin module cold gas system is same as three-axis except uses 8 thrusters of 4 lb_F each, which weigh and cost the same (0.5 lb/$5K each). System dry weight is reduced by 2 lb.

[c] TRW estimates that $100-150K should be added to this value for integration and test costs.

[d] Uses 2 end forgings from Viking Orbiter tank and existing elastomeric diaphragm.

[e] Flight-qualified but have not flown.

[f] TRW estimates that $200-300K should be added to this value for integration and test costs.

[g] SPS-I can employ 1, 2, 3, or 4 tanks providing propellant weights of 55, 110, 165, or 220 lb and corresponding system dry weights of 75, 87.2, 99.4, or 111.6 lb.

[h] Includes propulsion module structure, drive electronics, remote interface unit, GN_2 and miscellaneous.

[i] Existing flight-qualified tank developed for Viking Orbiter (VO-75) but will replace surface tension expulsion device with an elastomeric (AF-E-332) bladder.

version of the STPSS uses the same cold gas system, except that the twelve 0.1 lb_B thrusters are replaced with eight 4 lb_F thrusters of the same basic configuration. The unit weights and costs of these thrusters are estimated to be the same as the three-axis units. All components in both cold gas systems are flight-proven.

While the component development status of both the AEM hydrazine system and the STPSS cold gas systems appears to be about the same, different costing bases will be required to reflect the relative degrees of component complexities between them, particularly for tanks and thrusters. Hydrazine tanks typically use diaphragms or bladders for propellant expulsion and gaseous nitrogen (GN_2) for pressurization and require two drain and fill valves per tank. Cold gas tanks simply contain GN_2 under high pressure (in this case, 4000 psia) thus eliminating the diaphragm/bladder and one drain and fill valve. Hydrazine thruster assemblies typically consist of propellant flow control valves, injector thermal standoff and capillary feed tubes, catalytic decomposition chamber, injector, thrust nozzle, heaters (for thrust, chamber, valves, and catalyst bed), temperature sensors, and in some cases, filters and cavitating venturis; whereas cold gas thruster assemblies consist essentially of solenoid valves and a thrust nozzle. Hence, a sizable component cost differential is justifiable between these two types of RCSs, as well as some anticipated difference in system integration and test costs.

DESCRIPTION OF THE STPSS ALTERNATIVE HYDRAZINE REACTION CONTROL SYSTEM AND COMPARISON TO AEM

An alternative to the STPSS three-axis version spacecraft is to use a transfer/orientation module in place of the cold gas equipped orientation module and solid rocket propulsion for orbit transfer. This transfer/orientation module contains (in addition to attitude control system equipment) a hydrazine RCS to perform all of the spacecraft functions, such as three-axis stabilization, reaction wheel unloading, and orbit transfer and adjustment. Table E-1 shows the component breakdown for this system.

The 36-in. diameter spherical tank will be fabricated using the end forgings from the Viking Orbiter tank and incorporating an existing flight-proven elastomeric diaphragm. The 0.1 lb_F thrusters are flight-proven. The 300 lb_F thruster, as purchased, has a very heavy valve and gimbal mount assembly, which will be removed for this application. The $125K cost shown in Table E-1 is the estimate after these changes.

In comparison to the AEM hydrazine system, this RCS is larger (employs more components and of larger unit size) but is basically the same technologically; the required fabrication modifications and the indicated deviations from flight-proven status appear not of significant magnitude to warrant much, if any, variation in the costing basis employed.

DESCRIPTION OF THE MMS REACTION CONTROL SYSTEM AND COMPARISON TO STPSS

Two hydrazine RCS/propulsion systems have been configured to accommodate the various missions being considered for the MMS. The first, SPS-I, meets the orbit adjust and reaction control requirements for spacecraft in the 2500 lb class that would be launched by a Delta 2910. The second, SPS-II, meets the requirements of orbit transfer, orbit adjust, and reaction control for spacecraft in the 4000 to 10,000 lb class and would be used only by missions that are shuttle-launched. Component breakdowns of each system are shown in Table E-1.

The SPS-I system can use 1, 2, 3, or 4 of the tanks shown to provide propellant capacities of 55, 110, 165, or 220 lb, depending upon specific mission requirements. Two additional fill and drain valves and a filter and pressure transducer (totaling 2.0 lb) are required with each additional tank.* As indicated, all components in the SPS-I system are flight-proven or flight-qualified except for plumbing, harness, and structure, and for costing purposes these items can probably be treated as flight-ready per earlier discussions. The total SPS-I system is estimated to have a nonrecurring cost of $900K and a recurring cost of $600K.

The SPS-II system is the same as SPS-I except that it uses a large single cylindrical tank and, hence, requires more structure. The tank

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*Efforts are under way to do without these items as tanks are added.

(36 in. in diameter by 55.5 in. in length) is an existing flight-qualified design that was developed for the Viking Orbiter (VO-75) program. It presently has a surface tension device for propellant expulsion, which will most likely be replaced with an elastomeric (AF-E-332) bladder. Such replacement would entail about a 25 percent modification to the overall tank assembly. As indicated in Table E-1, the structure weight is increased from 29 lb to 81 lb compared to SPS-I. However, it should be noted that these weights include propulsion module structure, drive electronics, remote interface unit, GN_2, and other miscellaneous items; hence, some care in cost bookkeeping appears warranted for both the SPS-I and SPS-II systems. The total SPS-I system costs are estimated to be \$500K nonrecurring and \$750K recurring on the basis that the SPS-I system will be built first.

In comparing these two MMS hydrazine systems with the STPSS cold gas systems, the same comments apply as presented earlier in the comparison of STPSS cold gas systems and the AEM hydrazine system, i.e., a different cost base is required for cold gas hydrazine components. With respect to the STPSS hydrazine system, the same cost base should apply with perhaps some minor adjustments for the required component modifications noted herein. Moreover, the 0.2 lb_F and 5 lb_F thrusters of the MMS systems are estimated at \$12K each compared to \$20K and \$25K each for the 0.1 lb_F and 4 lb_F thrusters in the STPSS hydrazine system. This difference is probably reconcilable on the basis that the MMS thrusters have single-seat/single-coil propellant flow control valves versus dual-seal/dual-coil valves in the STPSS thrusters and perhaps less contractor testing and paperwork required, since the MMS thrusters are standard NASA items.

REFERENCE TO APPENDIX E

1. Taber, John E., *Space Test Program Standard Satellite Study*, TRW Systems Group 23590-6008-TU-00, October 30, 1975.

Appendix F

STRUCTURAL SUBSYSTEM: A COMPARISON OF AEM, STPSS, AND MMS

by

M. M. Balaban

AEM STRUCTURAL SUBSYSTEM

The principal elements of the AEM(1-3) structural subsystem that are of interest for a shuttle application consist of a base module and an instrument module. The base module structure contains support subsystems for the HCMM and SAGE missions, including all appendages and mechanisms to support these subsystems. The differences between these missions have no effect on the primary structural subsystem.

The base module consists of an 18 in. long hexagonal body with six longerons tied to a 7 in. conical structure that mates with a standard Scout series 25E adapter. Open truss bulkheads rigidize each end of the hexagonal enclosure. This design provides approximately 7.3 sq ft of usable flat surface for experiment mounting.

The structural elements of the base module are primarily sheet and stringer aluminum. Side panels of the hexagon are 0.012 in. thick clad 2024-T3 aluminum sheet riveted to the six corner longerons. Panel edge members, equipment support stiffeners, and truss-type bulkheads are also formed from 2024-T3 aluminum sheet. The longerons are standard Burner IIA extrusions, specifically shaped for hexagonal structure corners.

The truss-type bulkheads at either end of the hexagonal body provide structural rigidity, with good accessibility to the interior. These bulkheads are 2024-T3 formed parts attached to the body longerons. The forward bulkhead ties to the four longerons that serve as attach fittings to the instrument module. The center diagonal is easily removed by disconnecting fasteners at each end so as to provide better access for installing or removing interior components.

The aft bulkhead supports the modular orbit-adjust system for HCMM missions. The orbit-adjust system, which is fabricated, tested, and serviced as a separate module, is bolted to the aft bulkhead at three points. Shims are bonded to the aft bulkhead to provide proper lateral and angular alignment once the spacecraft mass properties have been determined.

The instrument module contains the mission instruments and the supporting electronics. This module is connected by low-heat-conduction, bolted-in fittings at four of the six longeron forward ends so as to provide direct load transfer. Fiber glass blocks and thermal blankets reduce heat conduction to less than 0.2 W/°C. This type of attachment fitting was used in the Burner IIA and P42-1 units. The four structural attach points feed acceleration loads directly into the base module longerons.

The total weight of the AEM structural subsystem is 47.7 lb, consisting of 27.2 lb of primary structure, 17.5 lb of secondary structure, and 3 lb of mechanisms.

MMS STRUCTURAL SUBSYSTEM

The primary structural elements of the MMS[4,5] for shuttle operation are the module support structure and the transition adapter. The power, attitude control and stabilization, and C&DH module skins are secondary structural elements in that they support elements of the spacecraft subsystems.

Module Support Structure

The module support structure provides structural continuity between the transition adapter, subsystem modules, and propulsion module. Its construction is basically a three-dimensional truss, with the six corners as the primary load points. (Electrical connectors and other insignificant loads may be hung on the struts themselves.) The Rockwell technical proposal[6] for fabrication shows the structural elements to be primarily sheet, angles, and channels. The corner fittings appear to be 60 deg V-shaped channels especially designed for triangular corners.

Transition Adapter

The transition adapter is the interface between the module support structure and the mission adapter. During shuttle boost, it is also the element that connects to the flight support system. The attachment points are provided by three load pins. The drogue point is the attachment element to the remote manipulator system of the orbiter, used for initial contact in the retrieval operation. The transition adapter also supports operational or mission-unique elements such as solar arrays (and associated mechanisms), booms, and antennas.

Structurally, the transition adapter is a ring with an I-beam cross section. It contains chromated machined fittings, formed extrusion, and sheet metal components. Flanges and webs are formed from annealed material then heat treated to the T-6 (temper) condition. Standard mechanical fasteners are used for component joining. Final machining of mating surface and drilling of subsystem attach holes take place after structural assembly. Additional details are available in Ref. 6.

Spacecraft and Structural Weights

Table F-1 shows the weights budgeted for MMS subsystems in their baseline configurations. The MMS total weight including payload will be defined by GSFC for each mission based upon spacecraft and launch vehicle configuration.

Table F-1

BASELINE MMS STRUCTURE WEIGHT SUMMARY

Subsystem	Baseline Configuration Weight (lb): Total	Baseline Configuration Weight (lb): Structural and Thermal Components
Module support structure	168	150
Transition adapter	115	115
C&DH module	199	103
Power module	358	107
Attitude control and stabilization module	371	117
Thermal control	3	3
Electrical integration	98	0
Total	1312	595[a]

[a]The thermal weight breakdown is as follows: louvers = 39 lb, blankets = 6 lb, other = 3 lb. Total thermal weight = 48 lb. The net structural weight is then 595-48 = 547 lb.

STPSS STRUCTURAL SUBSYSTEM

The description of the STPSS structure presented here provides only the overall dimensions and configuration.(7) Additional details, such as individual member materials and thicknesses are not available because no actual design has yet been undertaken. The STPSS consists mainly of a core module and an orientation module.

Core Module

The core module has the shape of a thin hexagonal nut. It connects to the shuttle orbiter at two trunnions and a stabilizing fitting. Box beams spread the load from the trunnion to the central ring, which is the primary load-carrying member. Honeycomb panels define the hexagonal perimeter of the core module. They also provide mounting surfaces for equipment on the interior and thermal radiators on the exterior. The panels transfer the load to the trunnions and directly to the central ring via the webs.

Orientation Module

Each orientation module is also hex-nut shaped and mates with the core module at the central ring. The two versions of the three-axis stabilized module (i.e., the "orientation" version and the transfer/orientation version) have identified structure except for brackets that connect the appropriate propulsion unit. The spin orientation module is thinner because its equipment does not require as much volume.

Spacecraft Weights

Table F-2 summarizes the spacecraft structural component weights. The TRW estimate of structural weight was deduced from HEAO* data. The HEAO spacecraft, which carries a 7000 lb payload with a safety factor of 3, weighs about 20 lb/axial length (in.). Taking a 1500 lb payload weight for the STPSS spacecraft, and a safety factor of 2, TRW deduced a structural weight of 25 lb/in.

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*High Energy Astronomical Observatory--a spacecraft that was actually designed and analyzed by TRW.

Table F-2

STPSS STRUCTURAL COMPONENT WEIGHT SUMMARY

Item	Structural Component Weight (lb)
Core module	240
Spin control orientation module	70
Three-axis orientation module	150
Precision three-axis module	150
Solar array	
Standard 50 W subpanel (19" × 45")	3.0 ea.
"Picture frame" (boom, hinges, etc.)	2.0-2.6 ea.

SOURCE: Reference 7.

COMPARISON OF STRUCTURAL SUBSYSTEMS

The AEM is primarily aluminum sheet and stringer construction, using standard Burner IIA extrusions for longerons. The conical shell that interfaces between the spacecraft and the Scout F booster is probably the most "exotic" structural element from a structures standpoint. However, it too is formed from aluminum sheet, and fabrication appears to be well within the state of the art and, in addition, will not be used on STPSS missions.

The module support structure of the MMS is a simple 3-D truss. The subsystem modules utilize honeycomb panels that frame into aluminum stock edges. The transition adapter is of more complex construction; however, the fabrication procedures appear to be based on proven techniques.

The basic structure of the STPSS appears to use more nonstandard components, i.e., rings and diverging box beams. The structural weight is also a higher percentage of the instrument payload weight than it is in the AEM and MMS. Additionally, alignment may be a more critical aspect of STPSS construction because loads have to be transferred between the inner cylinders of the core module and orientation module with minimal edge moments. The additional complexity of the STPSS structure will be reflected primarily as a fabrication cost, rather than as one of development risk.

In summary, the AEM and MMS structural subsystems appear to use proven techniques and, for the most part, standard members. The STPSS certainly is no simpler in construction and probably more costly on a relative basis.

REFERENCES TO APPENDIX F

1. *Applications Explorer Missions (AEM)*, Mission Planner's Handbook, Goddard Space Flight Center, National Aeronautics and Space Administration, May 1974.

2. *HCMM Base Module Specifications*, Goddard Space Flight Center, National Aeronautics and Space Administration, S-733-55, November 1975.

3. *Heat Capacity Mapping Mission Base Module*, Part 2—Technical, *Type I and Type II Missions*, Boeing Aerospace Company, Seattle, Washington, March 17, 1975.

4. *Low Cost Modular Spacecraft Description*, Goddard Space Flight Center, National Aeronautics and Space Administration, May 1975.

5. *Mechanical System Specification for the Multimission Modular Spacecraft (MMS)*, Goddard Space Flight Center, National Aeronautics and Space Administration, November 1975.

6. *Technical Proposal for a Multimission Modular Spacecraft Fabrication Program*, Rockwell International, January 9, 1976.

7. Taber, John E., *Space Test Program Standard Satellite Study*, TRW Systems Group, 23590-6008-TU-00, October 30, 1975.

Appendix G

THERMAL CONTROL SUBSYSTEM: A COMPARISON OF AEM, STPSS, AND MMS

by

W. D. Gosch

COMPARATIVE EVALUATION OF THE THERMAL CONTROL SUBSYSTEM ON THE STPSS AND AEM SPACECRAFT

There are two major differences between the thermal control subsystem of the STPSS and that of the AEM (Table G-1). First, the AEM design uses louvers, while the STPSS relies on radiators and heaters for controlling spacecraft component and structure temperatures. Second, the STPSS requires high-temperature insulation around the

Table G-1

THERMAL CONTROL ELEMENTS OF THE AEM, STPSS, AND MMS SPACECRAFT

	Spacecraft					
	AEM		STPSS			MMS
	3-Axis			3-Axis		
Element	Type I	Type II	Spin	Low-Cost	Precision	3-Axis
Spacecraft weight (lb)	214	274	888	1043	1167	1312
Thermal control weight (lb)	~ 3	3+[a]	(b)	(b)	(b)	39
Thermal control elements:						
• Louvers	1	2	—	—	—	6
• Radiators	X	X	X	X	X	X
• Heaters	X	X	X	X	X	X
• Multilayer insulation	X	X	X	X	X	X
• Thermal coatings	X	X	X	X	X	X
• High-temp. insulation			X	X	X	
• Interface insulators	X	X	X	X	X	X

[a] A second louver and radiator are added for this mission.

[b] Structure and thermal control weights combined: core middle = 250 lb, spin module = 75 lb, orientation (low-cost and precision) = 160 lb. TRW did not determine actual weights of the thermal control elements but they indicate it would be on the order of 10-15 lb.

solid rocket kick stage motor. This motor is imbedded inside the hexagonal modules and must be thermally isolated during and after firing to prevent excessive heat transfer to the spacecraft modules.

The louvers specified for the AEM were flight-qualified on the Mariner '64 and '71. The Boeing STP 72-1 and the S3 programs used a total of 17 louver assemblies identical to the ones proposed for the AEM spacecraft.

Multilayer insulation blankets for shielding the spacecraft from the heat generated by the solid rocket motors during and after firing are made of materials that can withstand the higher temperatures, such as titanium.

The "low temperature" multilayer insulation blankets are used to decouple the spacecraft from the external environment. For the AEM the blankets consist of an outer layer of aluminized 1 mil Kapton, 10 layers of doubly aluminized 1/8 mil perforated mylar separated by silk net spacers, a single layer of Dacron scrim cloth to act as a filter, and an inner layer of aluminized 1 mil Teflon (Teflon side facing the base module). The STPSS uses similar insulation blankets on the entire outer surface of each module with the exception of cutouts for the radiator panels.

On the AEM, heaters are used in the thermal control system solely for maintaining the orbit adjust system component (thruster valves and catalyst bed) temperatures within the design limits during the initial velocity trim. The heaters are subsequently commanded off and remain inactive for the remainder of the mission. They could be reactivated at any time by ground command if required. The total heater power required during velocity trim is 3 W.

The STPSS uses a heater for the solid rocket motor. It is thermostatically actuated to ensure adequate temperature levels at the time of firing. The STPSS also uses thermostatically controlled component heaters with sufficient power to maintain component temperatures above the minimum allowable under the coldest conditions.

Thermal control coatings used on the AEM and STPSS provide interior and exterior radiation control. Interior coatings enhance the internal radiation heat transfer from bay to bay. Coatings are used on the external surfaces to reduce the temperature effects of direct or reflected sunlight. These surfaces include the backside of the solar array, the louver radiator surface (AEM), the thermal control trim radiator, the shunt dissipater panel, solar array and antenna appendages, and the S-band antenna.

Radiators for dissipating heat generated inside the spacecraft are used on both the AEM and the STPSS. In the case of the STPSS (which has no louvers) the control of component temperatures within the spacecraft is achieved with a combination of radiators, second surface mirrors, and thermostatically controlled heaters. On the AEM, component temperature control is achieved with louvers and thermal-control trim radiators. The baseline design radiator for the AEM spacecraft radiator is sized to satisfy the HCMM mission requirements and is painted white. The radiator's properties can be adjusted by paint stripes to attain the desired trim.

Since most of the elements of the AEM thermal control subsystem have been flight-proven on previously designed Boeing spacecraft, they should be considered at least state of the art if not off-the-shelf. The same holds true for the TRW-proposed STPSS design.

COMPARATIVE EVALUATION OF THE STPSS AND MMS SPACECRAFT

To date, contracts have not been awarded for the design, development, or production of either the MMS or the STPSS. Consequently, the information available for making a comparative evaluation of the MMS and STPSS is less detailed than for the AEM-STPSS evaluation. However, based on the information from GSFC, Aerospace Corporation, and TRW, thermal control subsystem concepts are sufficiently well defined that a reasonable comparative technical evaluation can be made.

The same two differences between the AEM and STPSS are indicated for the STPSS and MMS (Table G-1). The MMS spacecraft uses two louvers on each of three modules: power module, ACS module, and the C&DH module. As previously stated, the STPSS relies on radiators, second surface mirrors, and thermostatically controlled heaters for maintaining the spacecraft structure and components within specified temperature limits. Louvers are generally considered to be more expensive than heaters. However, personal contact with a thermal control system engineer at

GFSC revealed that their analysis of the spacecraft heat balance, using louvers rather than heaters and radiators, indicated it is more economical to use louvers. The propulsion module for the MMS spacecraft (either SPS-I or SPS-II) is mounted at the base of the spacecraft structure and is thermally isolated from the structure and modules. A small quantity of heat is transferred at the interface between the structure and propulsion module and is accounted for in the thermal control analysis of the entire spacecraft. As noted previously the STPSS spacecraft uses a solid propellant rocket motor for propulsion and must be thermally isolated from the modules with high temperature multilayer insulation to prevent excessive heat transfer into the modules during and after firing.

The design objectives for both spacecraft, from a thermal control point of view, are generally the same, namely, thermally isolate each individual module from the environment and other parts of the spacecraft. The same basic design philosophy of using low-cost, proven elements for the thermal control subsystem appears to apply to the MMS and the STPSS. Thermal control elements for the MMS can be considered as at least state of the art if not off-the-shelf.

Appendix H

PROGRAM OPTIONS FOR THE SAMSO SPACE TEST PROGRAM

by

S. H. Dole and L. N. Rowell

Alternative approaches (i.e., different mixes of spacecraft, orbits, and payloads) to carrying out a complete Space Test Program during the 1980-1990 period were generated so that different sets of total program costs could be computed and compared. This appendix includes only a representative sample of the alternative program options that were examined in this study. First, the STPSS mission model is discussed and disaggregated into eight categories of orbits, and then the various standard spacecraft configurations considered in this study are identified with the payloads in these orbit categories according to their ability to accommodate the payload requirements. After this, the procurement options are determined for a variety of conditions.

ANALYSIS OF PAYLOADS IN THE STPSS "BLUEBOOK"(1)

We adopted the premise that we could consider the payloads given in Ref. 1 to be "representative" of those that would be orbited, thus the payloads in the bluebook were analyzed, as follows. Of the 51 payloads listed therein, four were eliminated because they required special spacecraft, or because they had already been launched into space (Nos. 4, 5, 9, 45),* and one (No. 42) was eliminated because the orbit was not clearly defined. The remaining 46 payloads were categorized according to their orbital orientation and apogee altitude and perigee altitude requirements. The standard orbits that were selected to provide a means of grouping payloads (and the number of bluebook payloads captured by each) are:

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*The numbers are those of the bluebook pages where the payloads are described.

Orbit Number	Description
1-S	Sun-synchronous (98.4 deg inclination), 250 to 300 n mi circular, sun-oriented [8]*
1-E	Sun-synchronous, 250 to 300 n mi circular, earth-oriented [13]
2	Elliptical, 7000 × 200 n mi, polar [13]
3	Geosynchronous (19,372 n mi) circular, low inclination, sun-oriented [4]
4	10,000 n mi circular, low inclination [2]
5	12 hr orbit, 21,000 × 900 n mi, 63.4 deg inclination [3]
6	Geosynchronous circular, low inclination, earth-oriented [1]
7	3200 × 150 n mi, 30 deg inclination [1]
8	180 n mi circular, polar [1]

The velocity increments required to place the spacecraft into the above standard orbits are given in Table H-1. These ΔVs were used for the selection and sizing of appropriate kick stages.

The payloads were also ordered according to the spacecraft capabilities that are needed to accommodate the payload. In addition to mission altitude and orientation, we also used payload weight, power, data rate, stabilization requirements, and pointing accuracy as filters for assigning spacecraft. These assignments are given in Tables H-2 to H-5 where the letters "x" or "y" indicate a compatibility between spacecraft capability and payload requirements. The letter "y" in the AEM spacecraft row applies when that spacecraft's maximum altitude capability is assumed to be geosynchronous rather than its current limit of 1000 n mi; this was one of the spacecraft design excursions that was examined in the study.

PROGRAM OPTION DEVELOPMENT

On the basis of information provided by SAMSO, it appeared that the Space Test Program would be orbiting approximately 114 payload packages during the 1980-1990 time period. Since there were only 64 representative payloads in the sample we had available to work with, it was

*Numbers in brackets are the number of the bluebook payloads accommodated by the orbit.

Table H-1

STANDARD ORBITS, VELOCITY INCREMENTS

Orbit Number	ΔV_1 (ft/sec)[a]		ΔV_2 (ft/sec)[b] [Apogee altitude (n mi)]		Total ΔV (ft/sec)	Remarks
1-S	174	[150]	172	[250]	346	250 n mi circular orbit
1-E	258	[150]	255	[300]	513	300 n mi circular orbit
	258	[150]	171	[300]	429	250 x 300 n mi orbit
2	5,581	[150]	55	[7,000]	6,136	200 x 7,000
3	7,984	[150]	4,820	[19,372]	12,804	Geosynchronous
4	6,505	[150]	4,601	[10,000]	11,106	10,000 circular
5	8,136	[150]	419	[21,000]	8,555	12 hr 900 x 2,000 (ΔV in plane)
	8,136	[150]	3,116	[21,000]	11,252	12 hr 900 x 21,000 63.4 deg (ΔV changes orbit plane inclination 35 deg at 21,000 n mi altitude)
6	7,984	[150]	4,820	[19,372]	12,804	Same as Program Option 3
7	3,536	[150]	-0-	-0-	3,536	150 x 3,200 n mi
8	53	[150]	53	[180]	106	180 n mi circular

[a] ΔV_1 is velocity increment added at shuttle altitude of 150 n mi to carry spacecraft to apogee altitude.

[b] ΔV_2 is the velocity increment added at apogee altitude to achieve desired orbit.

Table H-2

Orbit 1-S: Sun synchronous, 250-300 n mi circular, sun-oriented								
Payload number	15	19	20	27	33	37	48	51
Weight (lb)	50	10	76	250	1	12	135	3
Candidate Spacecraft AEM					y[a]	y		y
STPSS/S				x				
LC-STPSS	x			x	x	x		x
STPSS or MMS	x	x	x	x	x	x	x	x

[a] y applies when AEM maximum altitude is geosynchronous (19,382 n mi).

Table H-3

Orbit 1-E: Sun-synchronous, 250-300 n mi circular, earth-oriented													
Payload number	18	23	26	28	29	34	35	36	38	39	40	41	49
Weight (lb)	13	9	13	525	53	13	40	60	6	5	331	135	25
Candidate Spacecraft AEM	xy	xy	xy		xy	xy	xy	xy	xy	xy			xy
STPSS/S		x	x	x		x					x		
LC-STPSS	x	x	x	x	x	x	x	x	x	x	x	x	x
STPSS/P or MMS	x	x	x	x	x	x	x	x	x	x	x	x	x

Table H-4

Orbit 2: Elliptical (7,000 x 200 n mi), polar													
Payload number	1	6	11	12	13	14	17	21	22	31	32	46	50
Weight (lb)	17	16	4	8	15	8	4	44	70	18	1	5	110
Candidate Spacecraft AEM	y	y	y	y	y	y	y	y		y	y	y	
STPSS/S										x		x	
LC-STPSS	x	x	x	x	x	x	x	x	x	x	x	x	x
STPSS/P or MMS	x	x	x	x	x	x	x	x	x	x	x	x	x

Table H-5

Orbit	3[a]				4[b]		5[c]			6[d]	7[e]	8[f]
Payload number	3	16	44	52	2	24	7	10	43	8	25	30
Weight (lb)	12	3	147	13	29	2	25	19	475	(30)	43	30
Candidate Spacecraft												
AEM	y	y		y	y	y	y	y		y	y	
STPSS/S		x				x	x	x				
LC-STPSS	x	x	x	x	x	x	x	x		x	x	
STPSS/P or MMS	x	x	x	x	x	x	x	x	x	x	x	x

[a]Geosynchronous (19,372 n mi) circular, low inclination, sun-oriented.

[b]10,000 n mi circular, low inclination.

[c]12-hour orbit, 21,000 x 900 n mi, 63.4 deg inclination.

[d]Geosynchronous circular, low inclination, earth-oriented.

[e]3,200 x 150 n mi, 30 deg inclination.

[f]180 n mi circular, polar.

necessary to scale this number up by a factor of 2.48 to yield a closer approximation of the complete program. Consequently, both the numbers of payloads and their aggregated weights taken from Tables H-2 to H-5 were multiplied by 2.48 in developing the Program Options. Other numbers of total payloads in the ten-year period, 92, 138, and 228, were assumed in some of the cases to test the effect on results. As above, appropriate multiplying factors were used.

Groups of payloads (for a given orbit) were assigned to specific spacecraft with the following limits being observed:

1. Maximum payload weights that can be loaded on a single spacecraft: AEM = 150 lb; STPSS = 1000 lb or 1500 lb; MMS = 4000 lb.
2. Maximum circular orbital altitudes reachable by the spacecraft: AEM(x) = 1000 n mi; AEM(y) = 19,372 n mi; STPSS and MMS = 19,372 n mi.
3. The maximum number of payloads that can be loaded on a single spacecraft in separate program options was assumed to be 6, 8, 10, or 13.
4. Maximum experimental power: AEM = 50 W; STPSS-S = 290 W; STPSS-LC and STPSS-P = 400 W; MMS = 850 W.*
5. Maximum data rate: AEM = 8 kbps; STPSS = 128 kbps; MMS = 64 kbps.

The number of spacecraft flights for six different cases, four different program options, and four different assumed upper limits on the number of payloads that could be placed on a single spacecraft are summarized in Table H-6. As may be seen from Table H-6, the total number of shuttle flights required to place all of the STPSS payloads into orbit ranged from a minimum of 12 to a maximum of 26. The ranges in numbers of launches, as a function of the assumed payload limitations, are shown below:

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*The power limitation affected only the payload packages for the AEM and STPSS-S spacecraft; for all others, different limitations were more critical.

Table H-6

SUMMARY OF TABLES H-7 THROUGH H-23

CASES →		I	II	III	IV	V	VI
STPSS MAXIMUM PAYLOAD WEIGHT (LB) →		1000	1500	1000	1000	1000	1500
MAX. AEM ALT. (NM) →		1000	1000	1000	1000	19372	19372
TOTAL NO. OF PAYLOADS →		114	114	92	138	114	114
		NUMBER OF SPACECRAFT FLIGHTS					
PROGRAM OPTION	SPACECRAFT	(13) (10) (8) (6)	(13) (10) (8) (6)	(13) (10) (8) (6)	(13) (10) (8) (6)	(13) (10) (8) (6)	(13) (10) (8) (6)
		A	B	C	D	E	F
1 (AEM + STPSS)	AEM	1 1 2 3	1 2 3 4	0 0 1 2	0 0 2 3	7 7 9 11	6 8 10 13
	STP/LC	10 11 14 16	10 11 13 15	10 11 12 15	12 15 16 18	5 5 7 8	5 5 6 7
	STP/P	5 5 5 5	3 3 4 5	4 4 4 4	4 4 5 5	5 5 5 5	3 3 4 4
	MMS	0 0 0 0	0 0 0 0	0 0 0 0	0 0 0 0	0 0 0 0	0 0 0 0
	TOTAL NO. FLIGHTS →	16 17 21 24	14 16 20 24	14 15 17 21	16 19 23 26	17 17 21 24	14 16 20 24
		G		H	I	J	
2 (AEM + MMS)	AEM	1 2 3 4	SAME AS G	2 2 3 4	2 3 3 5	7 8 10 13	SAME AS J
	STP/LC	0 0 0 0		0 0 0 0	0 0 0 0	0 0 0 0	
	STP/P	0 0 0 0		0 0 0 0	0 0 0 0	0 0 0 0	
	MMS	13 14 17 20		11 12 13 16	13 16 18 21	8 9 10 11	
	TOTAL NO. FLIGHTS →	14 16 20 24		13 14 16 20	15 19 21 26	15 16 20 24	
		K	L	M	N		
3 (ALL STPSS)	AEM	0 0 0 0	0 0 0 0	0 0 0 0	0 0 0 0	SAME AS K	SAME AS L
	STP/LC	11 12 16 19	11 13 16 19	11 12 14 18	12 15 17 21		
	STP/P	5 5 5 5	3 3 4 5	3 3 3 3	4 4 5 5		
	MMS	0 0 0 0	0 0 0 0	0 0 0 0	0 0 0 0		
	TOTAL NO. FLIGHTS →	16 17 21 24	14 16 20 24	14 15 17 21	16 19 22 26		
		O		P	Q		
4 (ALL MMS)	AEM	0 0 0 0	SAME AS O	0 0 0 0	0 0 0 0	SAME AS O	SAME AS O
	STP/LC	0 0 0 0		0 0 0 0	0 0 0 0		
	STP/P	0 0 0 0		0 0 0 0	0 0 0 0		
	MMS	14 16 20 24		12 14 16 20	14 18 21 26		
	TOTAL NO. FLIGHTS →	14 16 20 24		12 14 16 20	14 18 21 26		

	13	10	8	6
Maximum number of payloads per single spacecraft	13	10	8	6
Number of launches	12-17	14-19	16-23	20-26

Each of the cells of the matrix represented by Table H-6 is expanded in Tables H-7 through H-23. In these tables, the total number of spacecraft required are disaggregated by orbit so that one can determine the appropriate kick stages that would provide the velocity increment necessary to translate the spacecraft from the nominal shuttle parking orbit (150 n mi) to the mission orbit. Tables H-7 through H-23 also tabulate the maximum number of payloads actually assigned to a spacecraft in a given orbit.

INTEGRATION COSTS

The costs of integrating and testing a complete spacecraft appear to be predominantly a function of the complexity of the individual payloads themselves rather than of the characterictics of the spacecraft on which they are mounted or of the number of payloads that have to be integrated into a single spacecraft. Some information provided by Mr. W. A. Myers, of Rockwell International, indicates that mission integration costs might include the costs of about three engineering man-months per payload at the low-cost end, up to total costs of possibly $1,000,000 per payload for highly complex payloads. A typical mission integration job would require one engineer per payload over a period of six to nine months. He indicated that there should be very little difference between the STPSS and the MMS relative to mission integration. The test procedures might be slightly more complicated with the MMS so the nonrecurring costs (of developing procedures) could be a little higher.

REFERENCE

1. *Current STP Payloads*, Department of Defense Space Test Program, January 1, 1976.

Table H-7

CASE I (A)[a]

PROGRAM OPTION 1: USE LEAST EXPENSIVE SPACECRAFT AND MINIMIZE NUMBER OF FLIGHTS

ORBIT	SPACECRAFT	MAXIMUM NUMBER OF PAYLOADS PER FLIGHT: 13	10	8	6
		NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	0 - 0	0 - 0	1 - 5	2 - 5
1-S	STPSS/P	2 - 10	2 - 10	2 - 8	2 - 5
1-E	AEM	1 - 3	1 - 3	2 - 5	3 - 5
1-E	STPSS/LC	3 - 10	3 - 10	3 - 8	3 - 6
2	STPSS/LC	3 - 11	4 - 9	5 - 7	6 - 6
3	STPSS/LC	1 - 10	1 - 10	2 - 5	2 - 5
4	STPSS/LC	1 - 5	1 - 5	1 - 5	1 - 5
5	STPSS/P	2 - 4	2 - 4	2 - 4	2 - 4
6	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
7	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	1	1	2	3
	STPSS/LC =	10	11	14	16
	STPSS/P =	5	5	5	5
TOTAL NUMBER OF FLIGHTS		16	17	21	24

[a]Roman capitals correspond to those in Table H-6.

Table H-8

CASES I(G) AND II(G)

PROGRAM OPTION 2: USE AEM AND MMS AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS -- NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 10	2 - 10	3 - 8	4 - 5
1-E	MMS	2 - 10	2 - 7	2 - 5	2 - 5
1-E	AEM	1 - 13	2 - 10	3 - 8	4 - 6
2	MMS	3 - 11	4 - 9	5 - 7	6 - 6
3	MMS	1 - 10	1 - 10	2 - 5	2 - 5
4	MMS	1 - 5	1 - 5	1 - 5	1 - 5
5	MMS	1 - 8	1 - 8	1 - 8	2 - 4
6	MMS	1 - 3	1 - 3	1 - 3	1 - 3
7	MMS	1 - 3	1 - 3	1 - 3	1 - 3
8	MMS	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	1	2	3	4
	MMS =	13	14	17	20
TOTAL NUMBER OF FLIGHTS		14	16	20	24

Table H-9

CASES I(K) AND V(K)

PROGRAM OPTION 3: ALL PAYLOADS ON STPSS AND MINIMIZE NUMBER OF FLIGHTS BY COMBINING PAYLOADS ON SAME ORBIT

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS -- NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	0 - 0	0 - 0	1 - 7	2 - 5
1-S	STPSS/P	2 - 10	2 - 10	2 - 7	2 - 5
1-E	STPSS/LC	4 - 9	4 - 9	5 - 7	6 - 6
2	STPSS/LC	3 - 11	4 - 9	5 - 7	6 - 6
3	STPSS/LC	1 - 10	1 - 10	2 - 5	2 - 5
4	STPSS/LC	1 - 5	1 - 5	1 - 5	1 - 5
5	STPSS/P	2 - 4	2 - 4	2 - 4	2 - 4
6	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
7	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	STPSS/LC =	11	12	16	19
	STPSS/P =	5	5	5	5
TOTAL NUMBER OF FLIGHTS:		16	17	21	24

Table H-10

CASES I(0), II(0), V(0), AND VI(0)

PROGRAM OPTION 4: ALL PAYLOADS ON MMS AND MINIMIZE NUMBER OF FLIGHTS					
		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS -- NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 10	2 - 10	3 - 7	4 - 5
1-E	MMS	3 - 11	4 - 9	5 - 7	6 - 6
2	MMS	3 - 11	4 - 9	5 - 7	6 - 6
3	MMS	1 - 10	1 - 10	2 - 5	2 - 5
4	MMS	1 - 5	1 - 5	1 - 5	1 - 5
5	MMS	1 - 8	1 - 8	1 - 8	2 - 4
6	MMS	1 - 3	1 - 3	1 - 3	1 - 3
7	MMS	1 - 3	1 - 3	1 - 3	1 - 3
8	MMS	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	MMS =	14	16	20	24
TOTAL NUMBER OF FLIGHTS	=	14	16	20	24

Table H-11

CASE II(B)

PROGRAM OPTION 1: USE LEAST EXPENSIVE SPACECRAFT AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS — NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	1 - 10	1 - 10	1 - 5	2 - 5
1-S	STPSS/P	1 - 10	1 - 10	2 - 8	2 - 5
1-E	AEM	1 - 8	2 - 7	3 - 6	4 - 6
1-E	STPSS/LC	2 - 13	2 - 10	2 - 8	2 - 5
2	STPSS/LC	3 - 11	4 - 9	5 - 7	- 6
3	STPSS/LC	1 - 10	1 - 10	2 - 5	2 - 5
4	STPSS/LC	1 - 5	1 - 5	1 - 5	1 - 5
5	STPSS/P	1 - 8	1 - 8	1 - 8	2 - 4
6	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
7	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM	= 1	2	3	4
	STPSS/LC	= 10	11	13	15
	STPSS/P	3	3	4	5
TOTAL NUMBER OF FLIGHTS:		= 14	16	20	24

Table H-12

CASES II(L) AND VI(L)

PROGRAM OPTION 3: ALL PAYLOADS ON STPSS AND MINIMIZE NUMBER OF FLIGHTS BY COMBINING PAYLOADS ON SAME ORBIT

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS -- NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	1 - 10	1 - 10	1 - 7	2 - 5
1-S	STPSS/P	1 - 10	1 - 10	2 - 7	2 - 5
1-E	STPSS/LC	3 - 11	4 - 9	5 - 7	6 - 6
2	STPSS/LC	3 - 11	4 - 9	5 - 7	6 - 6
3	STPSS/LC	1 - 10	1 - 10	2 - 5	2 - 5
4	STPSS/LC	1 - 5	1 - 5	1 - 5	1 - 5
5	STPSS/P	1 - 8	1 - 8	1 - 8	2 - 4
6	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
7	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	STPSS/LC =	11	13	16	19
	STPSS/P =	3	3	4	5
TOTAL NUMBER OF FLIGHTS:	=	14	16	20	24

Table H-13

CASE III(C)

PROGRAM OPTION 1: USE LEAST EXPENSIVE SPACECRAFT AND MINIMIZE NUMBER OF FLIGHTS

ORBIT	SPACECRAFT	MAXIMUM NUMBER OF PAYLOADS PER FLIGHT: 13	10	8	6
		NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	1 - 8	1 - 8	1 - 8	2 - 5
1-S	STPSS/P	1 - 8	1 - 8	1 - 8	1 - 6
1-E	AEM	0 - 0	0 - 0	1 - 2	2 - 4
1-E	STPSS/LC	3 - 9	3 - 9	3 - 8	3 - 6
2	STPSS/LC	2 - 13	3 - 9	4 - 7	5 - 6
3	STPSS/LC	1 - 8	1 - 8	1 - 8	2 - 4
4	STPSS/LC	1 - 4	1 - 4	1 - 4	1 - 4
5	STPSS/P	2 - 3	2 - 3	2 - 3	2 - 3
6	STPSS/LC	1 - 2	1 - 2	1 - 2	1 - 2
7	STPSS/LC	1 - 2	1 - 2	1 - 2	1 - 2
8	STPSS/P	1 - 2	1 - 2	1 - 2	1 - 2
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	0	0	1	2
	STPSS/LC =	10	11	12	15
	STPSS/P =	4	4	4	4
TOTAL NUMBER OF FLIGHTS		14	15	17	21

Table H-14

CASE III(H)

PROGRAM OPTION 2: USE AEM AND MMS AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 8	2 - 8	2 - 8	3 - 6
1-E	MMS	1 - 12	1 - 10	1 - 8	1 - 6
1-E	AEM	2 - 7	2 - 8	3 - 6	4 - 5
2	MMS	2 - 13	3 - 9	4 - 7	5 - 6
3	MMS	1 - 8	1 - 8	1 - 8	2 - 4
4	MMS	1 - 4	1 - 4	1 - 4	1 - 4
5	MMS	1 - 6	1 - 6	1 - 6	1 - 6
6	MMS	1 - 2	1 - 2	1 - 2	1 - 2
7	MMS	1 - 2	1 - 2	1 - 2	1 - 2
8	MMS	1 - 2	1 - 2	1 - 2	1 - 2

TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	2	2	3	4
	MMS =	11	12	13	16
TOTAL NUMBER OF FLIGHTS		13	14	16	20

Table H-15

CASE III(M)

PROGRAM OPTION 3: ALL PAYLOADS ON STPSS AND MINIMIZE NUMBER OF FLIGHTS BY COMBINING PAYLOADS ON SAME ORBIT

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS – NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	1 - 8	1 - 8	1 - 8	2 - 5
1-S	STPSS/P	1 - 8	1 - 8	1 - 8	1 - 6
1-E	STPSS/LC	3 - 9	3 - 9	4 - 7	5 - 6
2	STPSS/LC	2 - 13	3 - 9	4 - 7	5 - 6
3	STPSS/LC	1 - 8	1 - 8	1 - 8	2 - 4
4	STPSS/LC	1 - 4	1 - 4	1 - 4	1 - 4
5	STPSS/LC	1 - 4	1 - 4	1 - 4	1 - 4
5	STPSS/P	1 - 2	1 - 2	1 - 2	1 - 2
6	STPSS/LC	1 - 2	1 - 2	1 - 2	1 - 2
7	STPSS/LC	1 - 2	1 - 2	1 - 2	1 - 2
8	STPSS/P	1 - 2	1 - 2	1 - 2	1 - 2
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	STPSS/LC =	11	12	14	18
	STPSS/P =	3	3	3	3
TOTAL NUMBER OF FLIGHTS		14	15	17	21

Table H-16

CASE III(P)

PROGRAM OPTION 4: ALL PAYLOADS ON MMS AND MINIMIZE NUMBER OF FLIGHTS

ORBIT	SPACECRAFT	MAXIMUM NUMBER OF PAYLOADS PER FLIGHT: 13	10	8	6
		NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 8	2 - 8	2 - 8	3 - 6
1-E	MMS	2 - 13	3 - 9	4 - 7	5 - 6
2	MMS	2 - 13	3 - 9	4 - 7	5 - 6
3	MMS	1 - 8	1 - 8	1 - 8	2 - 4
4	MMS	1 - 4	1 - 4	1 - 4	1 - 4
5	MMS	1 - 6	1 - 6	1 - 6	1 - 6
6	MMS	1 - 2	1 - 2	1 - 2	1 - 2
7	MMS	1 - 2	1 - 2	1 - 2	1 - 2
8	MMS	1 - 2	1 - 2	1 - 2	1 - 2
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	MMS =	12	14	16	20
TOTAL NUMBER OF FLIGHTS		12	14	16	20

Table H-17

CASE IV(D)

PROGRAM OPTION 1: USE LEAST EXPENSIVE SPACECRAFT AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	1 - 12	2 - 8	2 - 6	2 - 6
1-S	STPSS/P	1 - 12	1 - 9	2 - 6	2 - 6
1-E	AEM	0 - 0	0 - 0	2 - 5	3 - 5
1-E	STPSS/LC	4 - 10	4 - 10	4 - 8	4 - 6
2	STPSS/LC	3 - 13	4 - 10	5 - 8	7 - 6
3	STPSS/LC	1 - 12	2 - 6	2 - 6	2 - 6
4	STPSS/LC	1 - 6	1 - 6	1 - 6	1 - 6
5	STPSS/P	2 - 5	2 - 5	2 - 5	2 - 5
6	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
7	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3

TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	0	0	2	3
	STPSS/LC =	12	15	16	18
	STPSS/P =	4	4	5	5
TOTAL NUMBER OF FLIGHTS		16	19	23	26

Table H-18

CASE IV(I)

PROGRAM OPTION 2: USE AEM AND MMS AND MINIMIZE NUMBER OF FLIGHTS

ORBIT	SPACECRAFT	MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
		NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 12	3 - 8	3 - 8	4 - 6
1-E	MMS	2 - 12	2 - 9	2 - 8	2 - 6
1-E	AEM	2 - 8	3 - 7	3 - 8	5 - 6
2	MMS	3 - 13	4 - 10	5 - 8	7 - 6
3	MMS	1 - 12	2 - 6	2 - 6	2 - 6
4	MMS	1 - 6	1 - 6	1 - 6	1 - 6
5	MMS	1 - 9	1 - 9	2 - 5	2 - 5
6	MMS	1 - 3	1 - 3	1 - 3	1 - 3
7	MMS	1 - 3	1 - 3	1 - 3	1 - 3
8	MMS	1 - 3	1 - 3	1 - 3	1 - 3

TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	2	3	3	5
	MMS =	13	16	18	21
TOTAL NUMBER OF FLIGHTS		15	19	21	26

Table H-19

CASE IV(N)

PROGRAM OPTION 3: ALL PAYLOADS ON STPSS AND MINIMIZE NUMBER OF FLIGHTS BY COMBINING PAYLOADS ON SAME ORBIT

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/P	1 - 12	1 - 9	2 - 6	2 - 6
1-S	STPSS/LC	1 - 12	2 - 8	2 - 6	2 - 6
1-E	STPSS/LC	4 - 10	4 - 10	5 - 8	7 - 6
2	STPSS/LC	3 - 13	4 - 10	5 - 8	7 - 6
3	STPSS/LC	1 - 12	2 - 6	2 - 6	2 - 6
4	STPSS/LC	1 - 6	1 - 6	1 - 6	1 - 6
5	STPSS/P	2 - 5	2 - 5	2 - 5	2 - 5
6	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
7	STPSS/LC	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	STPSS/LC =	12	15	17	21
	STPSS/P =	4	4	5	5
TOTAL NUMBER OF FLIGHTS		16	19	22	26

Table H-20

CASE IV(Q)

PROGRAM OPTION 4: ALL PAYLOADS ON MMS AND MINIMIZE NUMBER OF FLIGHTS					
		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 12	3 - 8	3 - 8	4 - 6
1-E	MMS	3 - 13	4 - 10	5 - 8	7 - 6
2	MMS	3 - 13	4 - 10	5 - 8	7 - 6
3	MMS	1 - 12	2 - 6	2 - 6	2 - 6
4	MMS	1 - 16	1 - 6	1 - 6	1 - 6
5	MMS	1 - 9	1 - 9	2 - 5	2 - 5
6	MMS	1 - 3	1 - 3	1 - 3	1 - 3
7	MMS	1 - 3	1 - 3	1 - 3	1 - 3
8	MMS	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	MMS =	14	18	21	26
TOTAL NUMBER OF FLIGHTS		14	18	21	26

Table H-21

CASE V(E)

PROGRAM OPTION 1: USE LEAST EXPENSIVE SPACECRAFT AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	0 - 0	0 - 0	1 - 5	2 - 5
1-S	STPSS/P	2 - 10	2 - 10	2 - 8	2 - 5
1-E	AEM	1 - 3	1 - 3	2 - 5	3 - 5
1-E	STPSS/LC	3 - 10	3 - 10	3 - 8	3 - 6
2	AEM	3 - 7	3 - 8	4 - 7	5 - 6
2	STPSS/LC	1 - 13	1 - 10	1 - 8	1 - 5
3	STPSS/LC	1 - 10	1 - 10	2 - 5	2 - 5
4	AEM	1 - 5	1 - 5	1 - 5	1 - 5
5	STPSS/P	2 - 4	2 - 4	2 - 4	2 - 4
6	AEM	1 - 3	1 - 3	1 - 3	1 - 3
7	AEM	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	7	7	9	11
	STPSS/LC =	5	5	7	8
	STPSS/P =	5	5	5	5
TOTAL NUMBER OF FLIGHTS		17	17	21	24

Table H-22

CASES V(J) AND VI(J)

PROGRAM OPTION 2: USE AEM AND MMS AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS- NUMBER OF PAYLOADS/FLIGHT			
1-S	MMS	2 - 10	2 - 10	3 - 8	4 - 5
1-E	AEM	1 - 13	2 - 10	3 - 8	4 - 6
1-E	MMS	2 - 10	2 - 7	2 - 5	2 - 5
2	AEM	3 - 7	3 - 8	4 - 7	5 - 6
2	MMS	1 - 13	1 - 10	1 - 8	1 - 5
3	MMS	1 - 10	1 - 10	2 - 5	2 - 5
4	AEM	1 - 5	1 - 5	1 - 5	1 - 5
5	MMS	1 - 8	1 - 8	1 - 8	1 - 3
5	AEM	0 - 0	0 - 0	0 - 0	1 - 5
6	AEM	1 - 3	1 - 3	1 - 3	1 - 3
7	AEM	1 - 3	1 - 3	1 - 3	1 - 3
8	MMS	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	7	8	10	13
	MMS =	8	8	10	11
TOTAL NUMBER OF FLIGHTS		15	16	20	24

Table H-23

CASE VI(F)

PROGRAM OPTION 1: USE LEAST EXPENSIVE SPACECRAFT AND MINIMIZE NUMBER OF FLIGHTS

		MAXIMUM NUMBER OF PAYLOADS PER FLIGHT			
		13	10	8	6
ORBIT	SPACECRAFT	NUMBER OF FLIGHTS - NUMBER OF PAYLOADS/FLIGHT			
1-S	STPSS/LC	1 - 8	1 - 10	1 - 5	2 - 5
1-S	STPSS/P	1 - 13	1 - 10	2 - 8	2 - 5
1-E	AEM	1 - 8	2 - 7	3 - 6	4 - 6
1-E	STPSS/LC	2 - 13	2 - 10	2 - 8	2 - 5
2	AEM	2 - 10	3 - 8	4 - 7	5 - 6
2	STPSS/LC	1 - 13	1 - 10	1 - 8	1 - 5
3	STPSS/LC	1 - 10	1 - 10	2 - 5	2 - 5
4	AEM	1 - 5	1 - 5	1 - 5	1 - 5
5	STPSS/P	1 - 8	1 - 8	1 - 8	1 - 5
5	AEM	0 - 0	0 - 0	0 - 0	1 - 3
6	AEM	1 - 3	1 - 3	1 - 3	1 - 3
7	AEM	1 - 3	1 - 3	1 - 3	1 - 3
8	STPSS/P	1 - 3	1 - 3	1 - 3	1 - 3
TOTAL NUMBER OF SPACECRAFT FLIGHTS:	AEM =	6	8	10	13
	STPSS/LC =	5	5	6	7
	STPSS/P =	3	3	4	4
TOTAL NUMBER OF FLIGHTS		14	16	20	24

Appendix I

L-AEM SPECIFICATIONS

THE FOLLOWING IS THE STATEMENT OF WORK FOR A SHUTTLE LAUNCHED ADAPTATION OF THE AEM FOR LARGE DIAMETER PAYLOADS THAT RESULTED IN THE L-AEM DESIGN.

5.0 CONTRACTOR TASKS

5.1 BASELINE DEFINITION

The Contractor shall design a baseline adaptation of the AEM base module for comparison with other vehicles by the Contractor. The baseline design shall be consistent with the following requirements:

- The payload interface shall be hexagonal 60 in. in maximum diameter.
- The spacecraft shall be three-axis stabilized with control capability to 0.5 deg in pitch and roll and 1 deg in yaw, with capability to be modified to control to 6 arc minutes or spin stabilized with control capability to ±1 deg.
- Solid propulsion shall be provided to inject the spacecraft into a circular orbit at altitudes up to geosynchronous altitude (orbiter altitude 150 n mi).
- A SGLS-compatible telemetry, timing, and control shall be provided using Carrier I with capability to also incorporate Carrier II for transmitting payload data at high data rates.
- Provision shall be made for payload weights up to 1000 lb.
- The power system array shall be one-axis with setable angle with 100 sq ft of array area. Two 20 Ahr batteries will be provided.
- The thermal system shall use louvers and heaters with a maximum power input from a payload of 10 W (insulated).
- No single-string failure modes.

5.2 SHUTTLE INTERFACE

5.2.1 The shuttle interface shall be defined including an adapter to support one or more spacecraft with payloads in the shuttle over the short or long spacelab tunnel or over Orbital Maneuvering System kit.

5.2.2 IUS interface shall be defined.

5.2.3 Mixed DoD payloads shall be considered.