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SMALL STANDARD SATELLITE (S3) FEASIBILITY STUDY

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SUMMARY

1. This study has revealed that a family of small standard satellites weighing from 70 to 130 pounds can be designed to accommodate a large majority of the missions to be proposed in the continuing NASA magnetospheric studies program.

2. The S^3 program will give experimenters the opportunity to design and fly a well integrated set of experiments aimed at specific overall investigations.

3. Considerable savings in money and manpower can be realized with the S^3 concept. Critical spacecraft subsystems can be utilized from mission to mission without the requirement for redesign.

4. No new techniques are necessary to develop the hardware for a series of satellites to satisfy the majority of the magnetospheric missions. During the development of the first few spacecraft in the series, a set of standard off-the-shelf modules will become available for future missions.

5. It is apparent that more than one family of satellites is needed to satisfy the requirements of the other experiment groups considered, namely, atmospheric physics, solar physics, astronomy and astrophysics.

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SMALL STANDARD SATELLITE (S^3) FEASIBILITY STUDY

1. INTRODUCTION

1.1 PURPOSE AND GOALS

This report is the result of a study to determine the desirability and feasibility of a small standard satellite system designed to accommodate a variety of missions within the magnetosphere. Originally conceived as a follow-on series to the EPE satellite series, the Small Standard Satellite, S^3, will weigh from 70 to 130 pounds and be launched by the Scout vehicle. These requirements in general limit accessible regions of space to $\lesssim 10$ earth radii. Departures from the above weight estimate are possible but are strongly orbit dependent. Planned lifetimes are approximately 1 year.

The standardization envisioned is achieved by (1) selecting a basic satellite structure to be used for all missions, and (2) employing a modular concept for the spacecraft interfaces and final assembly. It is thus planned that many different missions may be accommodated by using the basic structure in combination with a stock of "off the shelf" modules. For example, it is planned that the S^3 data system be available in modular form so that the system can easily be adjusted to the requirements of a particular mission.

The over-riding goal of the S^3 program is to provide to a group of experimenters the opportunity of designing and flying a well integrated set of experiments aimed at specific investigations. While the results from each experiment will provide valuable information, the planned integration of these results will allow a more complete understanding of the phenomena being investigated by the spacecraft as a whole. These ends can best be met by allowing a particular group of experimenters the opportunity to design the entire experimental complement of the payload.

Such a program also offers ample opportunity to plan for and to carry out correlative studies between simultaneous observations taken at various positions within the magnetosphere by experiments aboard several satellites.

A similar small satellite approach has been in effect in some of the various existing individual programs, notably the INJUN series of the University of Iowa. It is desirable to extend this opportunity to the overall scientific community.

1.2 CONDENSED HISTORY

The first thoughts concerning the S^3 program and defining the feasibility study were contained in a GSFC memo by L. R. Davis dated 10 October 1964. As this document is basic to the concept of an S^3 program, it is included as Appendix A to this report.

Basically, the recognition of the need to continue the scientific mission of the EPE series of satellites and the lack of a planned research program for the region $\sim$2 - 10 earth radii led to the S^3 concept. A working group composed of GSFC employees sketched out the preliminary design objectives and constraints listed in Appendix A. In general, S^3 was then envisioned to be in the 50 to 100 lb weight class, capable of being launched by a Scout vehicle out to apogees of 10 earth radii, have 20% - 40% of the payload weight available for experiments, be easily adaptable to a variety of missions, and have a minimum one year lifetime.

The task at that early date was to effectively block out the problem so that the eventual S^3 configuration would best serve the variety of research missions which would surely be proposed.

To the above end, i.e., to best generalize the eventual S^3 configuration, a subgroup of the original working group was formed to contact and interview representative groups of experienced experimenters throughout the country. This subgroup consisted of

Dr. J. H. Trainor	Project Scientist
Dr. R. A. Hoffman	Associate Project Scientist
Mr. G. W. Longanecker	Project Manager
Mr. R. O. Wales	Spacecraft Systems Engineer
Mr. A. B. Malinowski	Flight Data Systems Engineer

After many arduous sessions, replies were gathered and tabulated from the groups of experimenters listed in Table 1. Consistent with the original S^3 concept, all but two of the groups interviewed were primarily interested in magnetospheric phenomena. The remaining two groups, representing atmospheric and solar physics, were interviewed in order to indicate the feasibility of satisfying other than magnetospheric research interests with the S^3 program.

Extended interviews were held with groups whose response was particularly detailed and informative. The types of missions and the requirements of these missions obtained from the above groups of experimenters were quite varied. However, most groups were consistent in proposing a specific mission consisting of a complete set of experiments designed to investigate a particular phenomenon of interest.

Table 1
List of Experimenters Interviewed

Experimenter	Organization	Fields of Interest
W. L. Brown G. L. Miller	Bell Telephone Laboratories	Particles; D. C. Fields, B&E
M. G. Morgan T. Laaspere	Dartmouth University	A. C. Fields, VLF
N. W. Spencer L. H. Brace G. P. Newton	GSFC, Laboratory for Atmospheric and Biological Sciences	Atmospheric Physics
G. P. Serbu	GSFC, Laboratory for Space Sciences	Plasmas
J. P. Heppner T. L. Aggson	GSFC, Laboratory for Space Sciences	Plasmas; Particles; D. C. Fields, B&E; A. C. Fields, VLF
E. J. Smith	Jet Propulsion Laboratory	D. C. Fields, B&E; A. C. Fields, VLF
C. O. Bostrom D. J. Williams	Johns Hopkins University Applied Physics Laboratory	Particles; D. C. Fields, B&E
H. S. Bridge	Massachusetts Institute of Technology	Plasmas
T. A. Chubb P. Mange C. Y. Johnson J. H. Hoffman R. W. Kreplin	Naval Research Laboratories	Solar Physics, Auroral Photometry, Plasmas, Atmospheric Physics
R. A. Helliwell R. L. Rorden J. Katsufrakis R. Smith	Stanford University	A. C. Fields, VLF
K. A. Anderson	University of California, Berkeley	Particles
C. E. McIlwain R. W. Fillius	University of California, La Jolla	Particles; D. C. Fields, B&E
C. Y. Fan	University of Chicago	Particles
W. A. Whelpley D. Enemark	University of Iowa	Particles; D. C. Fields, B&E; A. C. Fields, VLF
J. R. Winckler P. J. Kellog R. L. Arnoldy D. Cartwright	University of Minnesota	Particles; A. C. Fields, VLF
L. J. Cahill	University of New Hampshire	D. C. Fields, B&E; A. C. Fields, VLF

During the interviews the experimenters were requested to project their needs into the period 1969 through early 1970's, although it was realized that some of the missions suggested might not be necessary or desired by that time. Partially because of this, many of the replies were rather qualitative in nature. Thus it was necessary that these replies be used as guidelines in this study rather than as a definitive set of objectives. It is felt that when the missions proposed by the experimenters eventually become formalized they can be satisfied with the relatively wide range of design parameters envisioned for the spacecraft.

The results of these contacts are discussed and tabulated in a later section. While at times the responses were somewhat less than definitive, they were always enthusiastic. The scientific community definitely favors and, in fact, eagerly awaits the opportunities that can be afforded by the small satellite program envisioned here.

1.3 PARTICIPANTS

The S^3 feasibility study was conducted by the subgroup of the original working group listed in section 1.2. Valuable assistance was provided by the following persons:

Dr. D. J. Williams	Project Scientist as of November 1965
J. M. Madey and X. W. Moyer	Structural Considerations
Dr. J. V. Fedor	Appendix B, Orientation Control for Small Standard Satellite
E. I. Powers	Appendix C, General Comments on Thermal Design of Small Standardized Satellites
C. M. MacKenzie	Appendix D, Power System for Small Standard Satellite
Dr. R. A. Hoffman	Associate Project Scientist and author of Appendix E, Aspect Determination
T. V. Saliga	Appendix F, Feasible Channel Encoding, Decoding, and Synchronization Methods

P. T. Cole	Appendix G, Magnetic Recording Capabilities
A. B. Malinowski	Flight Data Systems Engineer and author of Appendix H, Data Processing System
M. Mahoney	Appendix I, S.S.S. Preliminary Data Processing Plan

2. EXPERIMENTERS' REQUIREMENTS

An outline was used to solicit experimenters' requirements for small scientific satellites in the next decade. The completed outlines were used as guidelines in order to determine the feasibility and desirability of the small standard satellite approach. Information from the outlines, together with pertinent comments is presented below.

2.1 DATA REQUIREMENTS

The difficulty of attempting to specify quantitatively data handling requirements for a possible experiment at some uncertain time in the future is inherently great. This difficulty is magnified by the realization that the very nature of the experiment is uncertain since its final configuration is dependent on results to be obtained in the interim. Nevertheless, we were very fortunate that the experimental groups were willing to give their best estimates of their onboard data handling requirements for several years into the future. The following paragraphs present the ranges of values so obtained and present various data handling features considered desirable.

2.1.1 Particle and Plasma Experiments

Individual experiment requirements were found to lie within the ranges of data parameter values shown below.

Data Parameters	Range
Number of experimental outputs	10 – 40 words
Samples per second per output	1 – 100
Word length	8 – 20 bits
Accumulation time	.01 – 10 sec
Total Bit rate	150 – 60,000 bps

The following items were considered desirable features of the on-board data handling system: data compression, ability to vary bit rate on command, data sampling synchronized to spacecraft generated signal, and a data Format capable of being re-programmed in flight.

In general it was also considered acceptable and/or desirable to duty cycle the high bit rate data.

2.1.2 Fields

The following data pertains to electric and magnetic field measurements which employ an A/D converter or which sample the output of a sequence of band pass filters:

Data Parameters	Range
Number of outputs	3 – 12 words
Samples per sec per output	.1 – 2000
Word length	5 – 10 bits
Total Bit rate	10 – 42,000 bps

The higher bit rate data were duty cycled at a $\lesssim$ 10% rate. Programmable data rates were also requested.

In addition, direct modulation of the carrier was desired by various groups. In general, 10 – 30 kc movable bandwidths and 40 – 200 kc fixed bandwidths were mentioned.

2.2 ORBITS

A summary of the types of orbits requested is shown in Table 2. In general there were several requests for each type of orbit. The capability of the Scout vehicle to meet these requirements will be discussed in Section 5 of this report.

Table 2
Orbits

Type	Apogee
a. Circular polar	low, up to approximately 1 earth radii
b. Circular polar	high, few to 30 earth radii
c. Eccentric polar	high, few to 30 earth radii
d. Circular equatorial	high, few to 30 earth radii
e. Circular equatorial	low, under the belts (500 Km)
f. Eccentric equatorial	high, few to 30 earth radii
g. Synchronous	6.6 R_e

2.3 CONFIGURATION AND WEIGHT

Nearly all the experimenters indicated that their instrumentation was compatible with the IMP packaging technique and those with experience in the IMP program were very much in favor of the design. The only exceptions are some mass spectrometers and detectors that require large surface areas.

Typical experiment complements for the majority of missions discussed would weigh from 10 to 30 pounds. These figures are consistent with total spacecraft weights* from 70 to 130 pounds, and consequently, a large number of missions can be performed with the Scout vehicle. Some experiment complements for atmospheric physics and solar physics studies may weigh up to 50 pounds. With the increased requirements of these experiments the total spacecraft weight may reach 150 to 175 pounds. Several low orbital missions with satellites in this weight range are compatible with the Scout vehicle.

2.4 STABILIZATION

Table 3 is a summary of stabilization requirements. Over half of the experimenters prefer a spin stabilized spacecraft with, in some cases, the capability of reorienting the spin axis periodically over the satellite lifetime. Desired spin rate ranges as well as other stabilization requirements are listed in the table.

2.5 POWER

Typical power requirements for the proposed experiment complements range from 4 to 8 watts. While certain missions involve experiments with high peak demands, i.e., mass spectrometers and VLF loop antennae, reasonable average power levels can be achieved by duty cycling.

Several of the experimenters suggested the avoidance of solar paddles. Paddles place restrictions on experiment look angles. Also, paddle motion in the ambient medium may produce perturbations on thermal particle measurements and consequently disrupt the science. There are engineering reasons to avoid solar paddles such as: (1) easier interface with launch vehicle and shroud, (2) more reliable mechanical system by eliminating paddle deployment function, and (3) less attitude perturbations in low circular orbits or at perigee in eccentric orbits.

*Total spacecraft weight for a Scout launch includes from 15 to 20 pounds for the fourth stage to spacecraft adapter section.

Table 3
Summary of Stabilization Requirements

Type	Measurement Accuracy	Remarks	% of Experimenters Requesting
Magnetic	Few degrees	Generally confined to use on orbit a.* One case for orbit b and/or c.	25
Gravity Gradient	~2°	Orbit g.	5
Sun Oriented	1 min, 1 deg	Orbits b, c, d.	10
Spin	~1°	All orbits. Two cases of reorientation. Orbit a. Few rps; 1 rpd - commandable. b. Reorient; 0.5-2 rpm; 6-600 rpm.** c. 3-5 rpm; 6-600 rpm**; 0.5 rpm. d. 6-600 rpm.** f. 10-30 rpm; 3-5 rpm; 6 rpm. g. Reorient; 100 rpm.	55
Low spin rate controlled precession	few tenths of degree	e. 0.5 rpm.	5

*Letters a thru g refer to orbits defined in Table 2

**Any spin rate in this range.

2.6 THERMAL

The operating temperature limits of the majority of the experiments are from -10°C to +30°C; the optimum being +10°C. The lower limit is generally set by the experiment electronics for convenience in circuit design while the upper limit is set by particle sensors. By shifting this 40°C range either 10° warmer or 10° colder, all but two of the initial experiment groups could be satisfied.

3. SPACECRAFT AND SUBSYSTEMS

Based on the responses discussed in the previous section, the study group investigated the requirements with the various in-house design areas. This section contains a possible solution showing the feasibility of a magnetospherically oriented S^3 program. Detailed discussions of the problems in most areas are contained in Appendixes B through I.

It is not clear at this time to what extent the needs of other research areas (atmospheric physics, solar physics, astronomy, astrophysics) will be satisfied by such a magnetospherically oriented program. Major problems are evident in the requirements for large area detectors, large experiment weight, high pointing accuracy, very low spin rates, and for avoiding detector contamination. It is felt that one or more separate designs may be needed to satisfy these needs.

Nevertheless, the design presented here will accommodate the large majority of the missions to be proposed in the continuing NASA magnetospheric studies program.

3.1 STRUCTURE AND MECHANICAL SYSTEMS INTEGRATION

Figures 1, 2 and 3 show the proposed configuration of the S^3 spacecraft. The structure will be formed with pop-riveted aluminum sheet metal. Most of the instrumentation will be housed in the octagonal mid-section of the spacecraft. Experiments and electronics can be packaged in trapezoidal shaped frames that are plugged in around the periphery of the octagon. The spacecraft wiring harness is affixed in the center portion of the octagon. Insertion and removal of the frames can be accomplished without handling the harness, thus enhancing reliability. Solar cell panels are mounted to the truncated sections above and below the experiment section. For higher power requirements an optional system module is under design to provide solar panels that wrap around the periphery of the spacecraft in the launch configuration (see Figure 4).

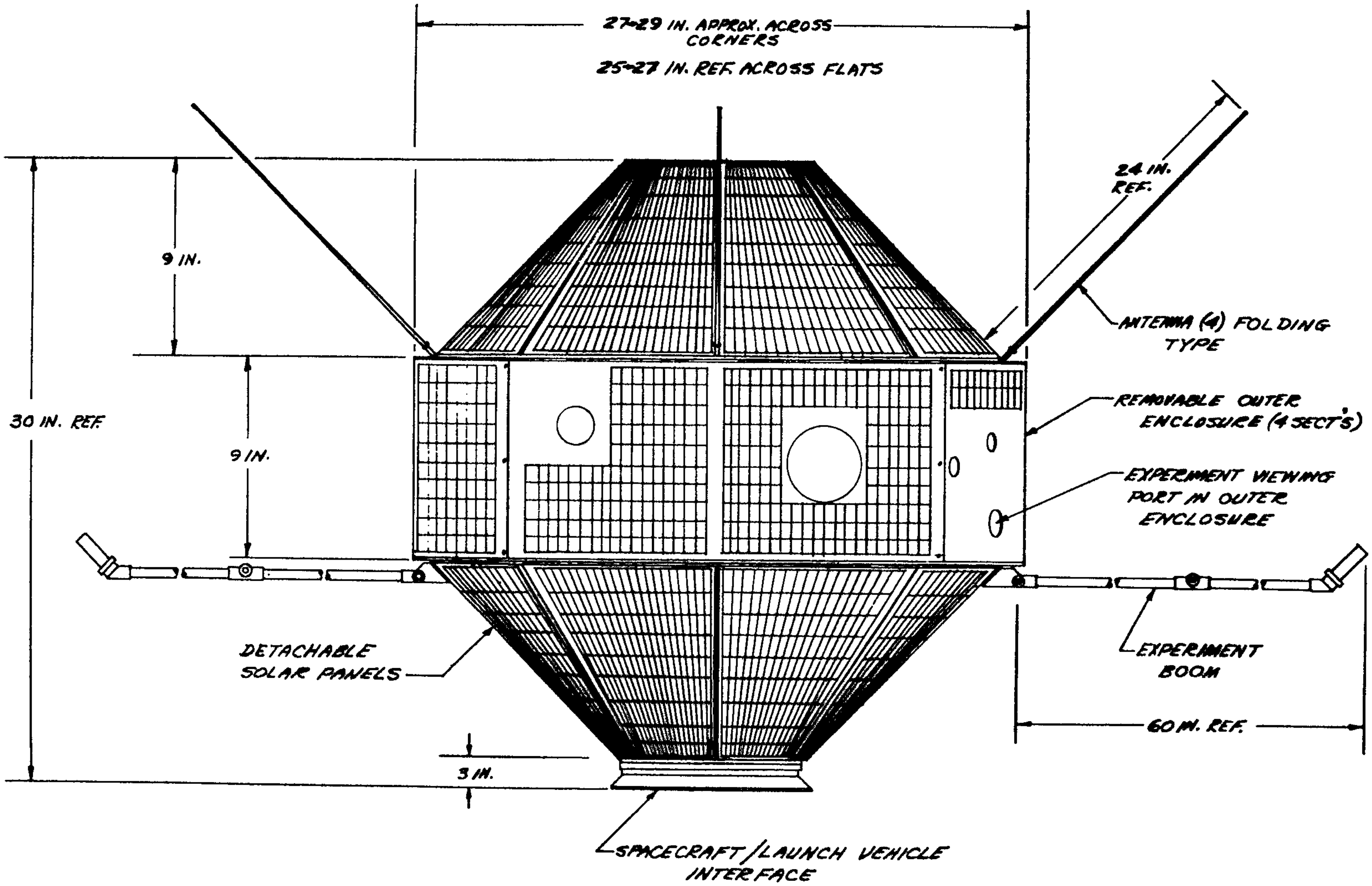


Figure 1–Preliminary Concept Typical S^3 Assembly

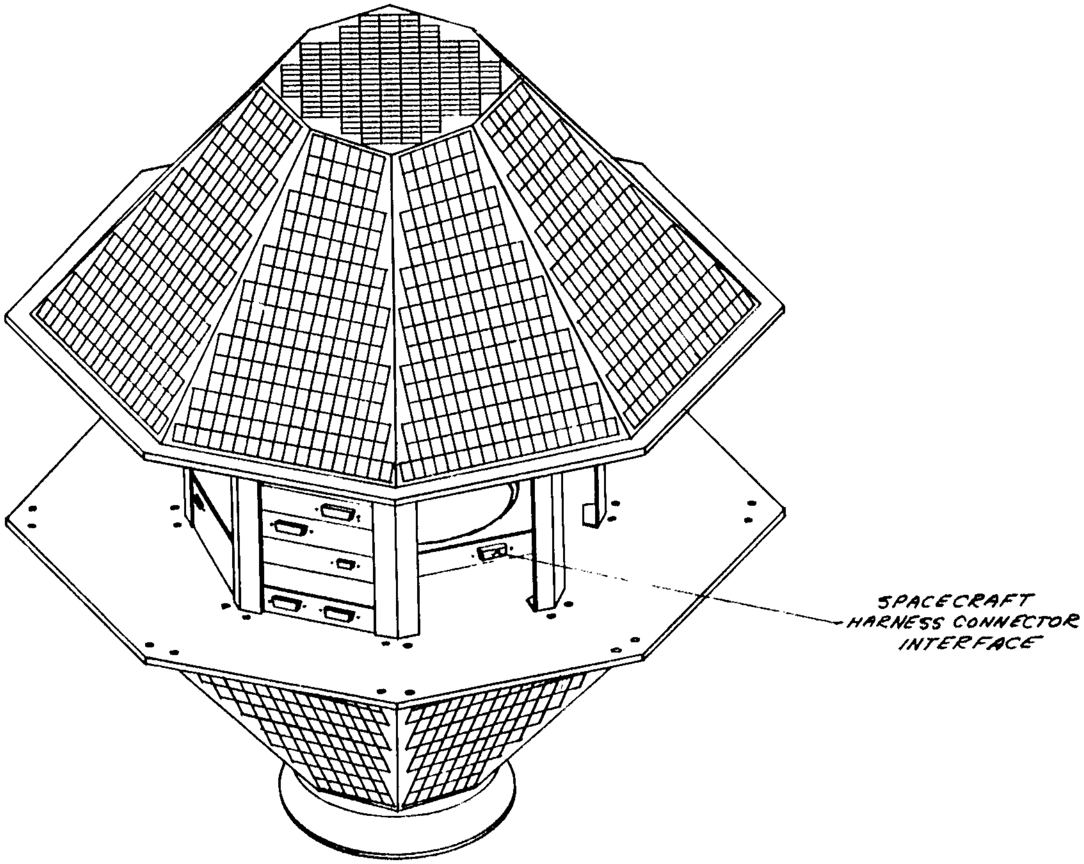


Figure 2–Preliminary Concept S^3 Assembly Less Covers

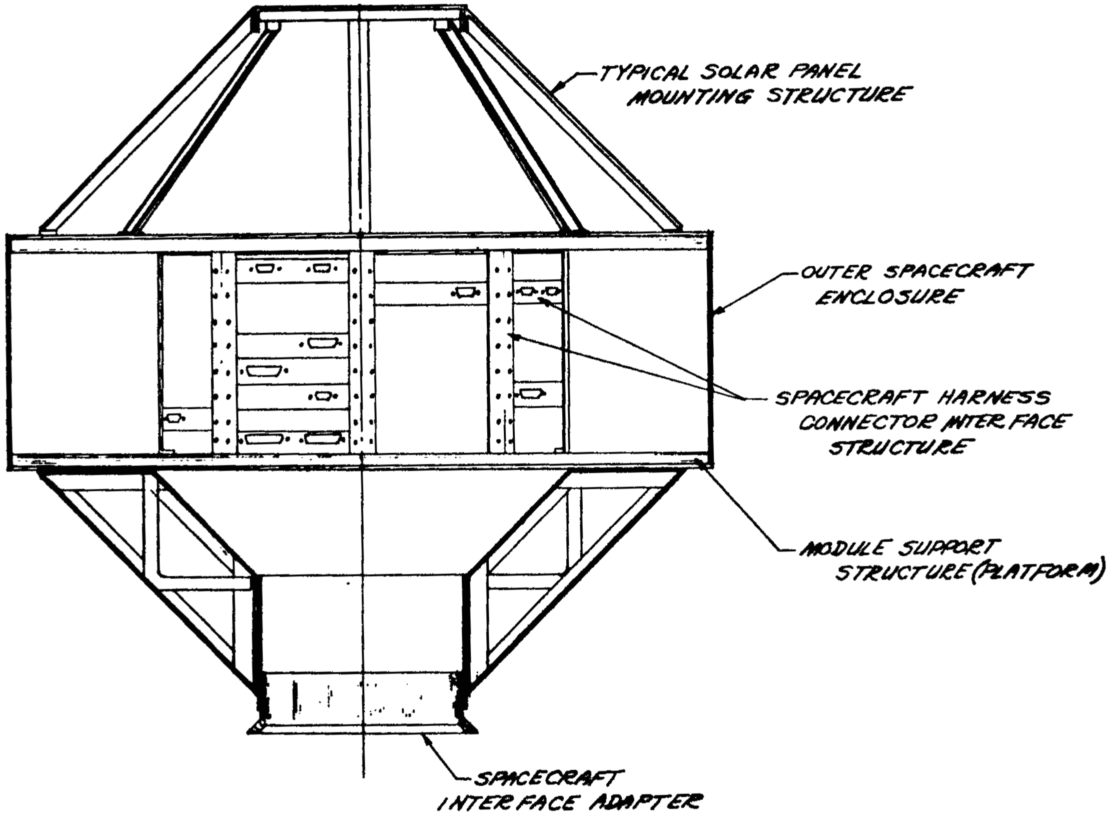


Figure 3–Preliminary Concept S^3 Structure

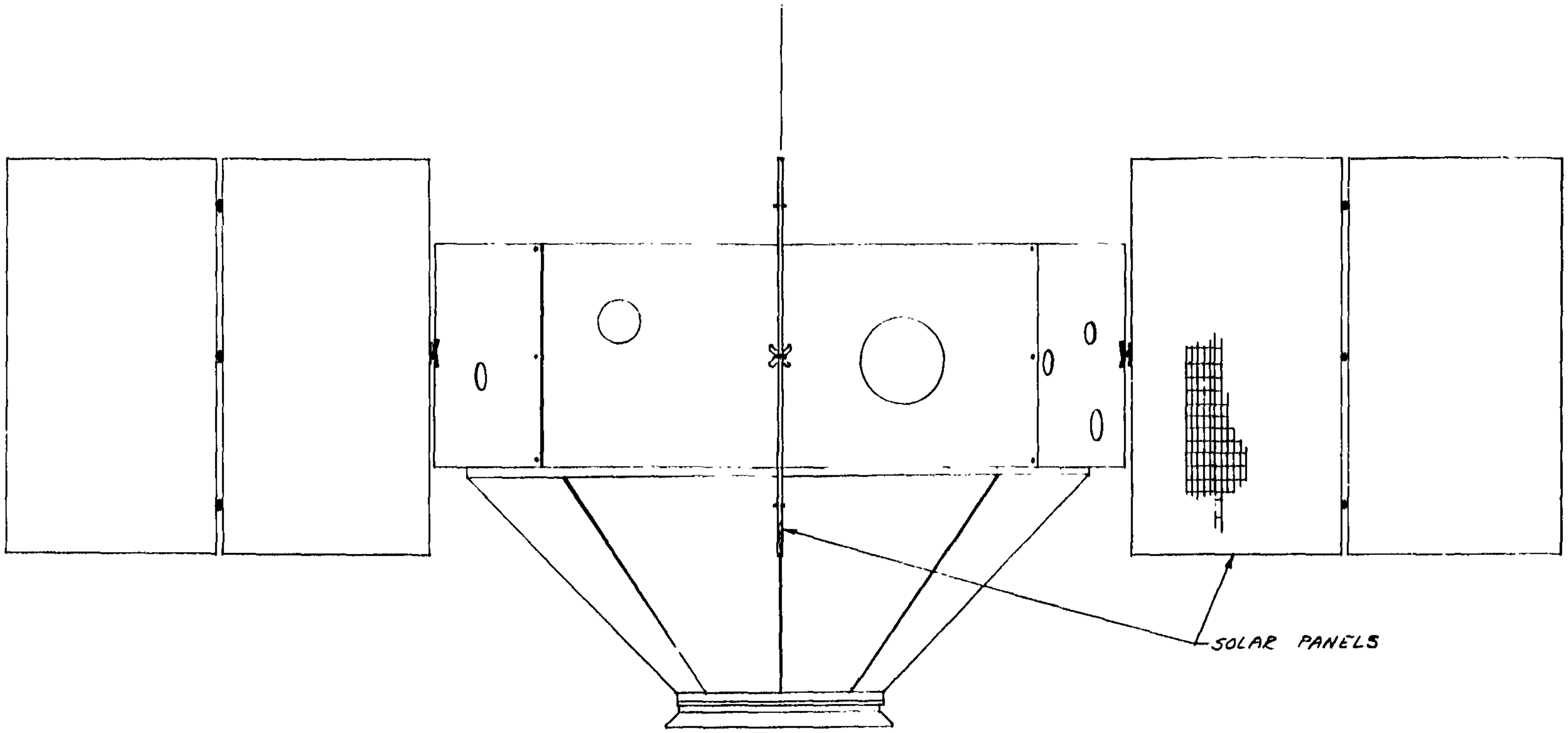


Figure 4–Preliminary Concept S^3 with Adjustable Angle Solar Panels

It is felt that this structural concept combines the best features of low cost, simplicity in design and fabrication, ease of assembly, standardization of hardware, simplicity in mechanical and electrical integration, interchangeability of subsystems, vibration isolation, and ease of thermal control. Payload power will also be approximately independent of spacecraft orientation. Table 4 lists typical weights for 70 and 130 pound versions of the S^3 for magnetosphere studies. Additional design features such as provisions for "piggyback" and kick-motor missions are feasible.

Table 4
Small Standard Satellite Weight Summary

Item	Weight (lbs.)	
	Minimum Version	Maximum Version
Solar Cell Array	7.0	13.0
Solar Cell Regulator	0.3	0.3
Undervoltage Relay and Recycletimer	0.5	0.5
Battery Pack	3.5	6.0
Power Converter	1.0	2.5
Tape Recorder	6.0	6.0
Data Encoder	1.0	1.0
Telemetry Transmitter	1.5	1.5
Tracking Transmitter	0.7	0.7
Antenna and Hybrid	1.5	1.5
Command Receiver and Decoder	2.0	2.0
Structure	9.0	17.0
Thermal Control	1.0	2.0
Harness and Connectors	1.5	2.5
Experiments	10.0	40.0
Optical Aspect System	1.5	1.5
Data Conditioning Equipment & Clock	4.0	9.0
Orientation System	3.0	3.0
Scout Separation System	15.0	20.0
	70.0	130.0

3.2 ORIENTATION REQUIREMENTS (Appendix B)

As previously noted in Table 3 (Summary of Stabilization Requirements) all types of stabilization systems were requested. The design proposes the spin stabilized spacecraft as the basic system with provision for add-on modules when magnetic or gravity gradient stabilization is used.

To orient spinning (4 - 40 rpm) spacecraft, a subliming solid jet system is proposed. With this system the spin axis can be held in any fixed orientation to within 2° for one year or programmed precessions can be provided. The gas jet components of ammonia and/or H_2S nature are directed away from the spacecraft. The possibility of contaminating some types of experiments must be considered.

3.3 THERMAL CONTROL (Appendix C)

A temperature of 10°C ± 20°C may be easily obtained for most of the missions proposed with a single passive thermal design. Thermal protection for solar cells on nonspinning or low spin axis-sun angle missions cannot be established without a more complete design of the spacecraft permitting detailed analysis of the gradients.

A spherical spinning spacecraft with solar cells distributed uniformly over most of the surface will have a mean internal temperature on the order of 5°C for all of the circular orbits considered above except the polar orbit. As the plane of the polar orbit varies from the earth sun line and the shadow time decreases from 40% to 0%, the mean temperature will rise to 25°C.

For the eccentric orbits with apogee in the earth's shadow, several hours without sunlight is possible and the thermal mass of the spacecraft may not be sufficient to hold the lower limits desired by the experimenters. Either a launch window constraint or thermal isolation of the spacecraft skin may be considered. The former may not be acceptable for some missions and the latter produces extreme variations in solar cell temperatures approaching ±100°C. To reduce the upper solar cell limit to +70°C on an isolated skin the solar cell area must permit at least 25% white painted surface and/or selective filters which reflect part of the normally absorbed thermal energy may be used. Either method will also reduce the mean temperature of the spacecraft which may not be desirable.

Similar solar cell temperature problems will be encountered if the spin axis is not nearly perpendicular to the sun line or if the spin rate is near zero (as for the gravity gradient stabilized spacecraft).

3.4 POWER SYSTEMS (Appendix D)

The 26 sided polyhedral shape shown in Figure 1 presents a cross section which is a minimum 90% of that presented by a 29" diameter sphere (Scout payload envelope is 30"). The lengthening of the shape when viewed from the side compensates for the loss of available solar cell area due to sensor openings and provides for a nearly aspect-independent power output. Assuming 80% of each surface is available for cell mounting and that the packing factor is .8 (i.e., 64% of total surface coverage) an initial power output of 31 watts will be obtained. Allowances for shadow time and decrease due to radiation damage must be made on an individual mission basis. Provisions for attached paddles are shown in Figure 4.

The power system design would follow the general concepts established for IMP and RAE. The use of silver cadmium cells for storage of energy would minimize magnetic effects. With a standard number of cells in series, the operating range of the unregulated bus voltage input to the converters or regulators is established. If the regulated output voltage requirements can be reduced to some minimal number of voltage levels (two or three), then it appears feasible that a small number of converter designs can cover all missions. It is proposed that these converters provide regulation suitable for the digital portions of the spacecraft (typically 5%) and that the experiment packages provide additional regulation where required (as for analog reference voltages) and provide converters for high voltages, etc. By reducing the amount of highly regulated power supplied, system losses are reduced.

Table 5 lists typical spacecraft system power requirements and shows amounts available for the experiment detectors from surface mounted solar cells. Figure 4 demonstrates a method of orientable solar panel attachment being considered that provides up to 100 watts when perpendicular to the solar flux.

While the number of cells in series can be standardized for all missions resulting in standard buss voltages. it is not desirable to standardize on the battery capacity. The cell capacity is dependent on such mission parameters as peak power demand, night power demand and degrees of solar cell shadowing by booms. Standardization of battery capacity would entail a weight penalty. Present state of the art batteries will provide about 14 watt hours per pound at safe discharge levels.

Table 5

Typical Experiment Power Budget (watts)

	ORBIT TYPE		
	Full Sun	10% Eclipse	40% Eclipse
Solar cell output – 31 watts (body mounted – launch)	19.5	15	4.5
Solar cell output – 22 watts (body mounted – 1 year maximum radiation damage)	10.5	7.5	Must be duty cycled

Conditions: 1 - Experiments not duty cycled.
2 - Record to transmit ratio 10:1
3 - Battery charge – discharge efficiency 66%.
4 - Subsystem requirements

	Watts	
	Record	Playback
a. Tape recorder	1.0	1.7
b. Data handling system	4.5	2.0
c. Encoder, R&RR, command receiver	.8	1.6
d. Aspect system	.7	.7
e. Transmitter	–	15.0
f. System losses for above	2.0	7.0
	9.0	28.0

3.5 ASPECT SYSTEM (Appendix E)

Most experimenters requested knowledge of the pointing position of their detector to the order of 1°. Earth and solar aspect detectors are available which meet this requirement for stable spinning spacecraft. For very low spin rates or for more accuracy, a star tracking system will be required.

Fields and particles experimenters usually requested knowledge of the magnitude and direction of the magnetic fields relative to their detectors. The onboard magnetometer would also provide some additional aspect information.

3.6 DATA SYSTEM (Appendixes F, G, & H)

3.6.1 Data Handling System

To satisfy the inherent need for data handling on the small satellite and provide the flexibility required for different missions, the data system should be developed along a modular concept whereby a series of standard modules may be easily integrated to suit the needs of a particular data system.

The primary components of this system should include:

1 - Data conditioners – A/D converters, accumulators, compressors and arithmetic modules.
2 - Reprogrammable format memory.
3 - Data buffer memory.
4 - Tape recorder.
5 - Bit parity encoders.
6 - Transmitter.

3.6.2 Reprogrammable Format

A basic telemetry format will be established for all missions such that synchronization words, spacecraft clock and other basic identifiers retain the same location within a fixed frame length. The reprogrammable format memory will contain programs to route each experiment output to the appropriate data conditioner and assign sufficient data bits (in multiples of 4 bit "bytes") for the accuracy desired. The assignments may be obtained from either prewired, ground read-in, or telemetry command link read in programs permitting adjustment of the data format during the mission. The reprogrammable format memory words will be transmitted and subsequently will be used by the ground processing computer so that a single computer program may automatically handle in-flight format changes plus all the various missions contemplated.

3.6.3 Data Buffer Memory

The reprogrammable format operates in synchronism with the telemetry bit rate. To permit collection of data synchronized with any other spacecraft generated signal a buffer core memory will be used. Portions of this memory may also be used by the experimenter for integration or for storage of data received at high bit rates (simultaneous sampling, for example).

3.6.4 Tape Recorder

For data storage between ground readout periods a tape recorder having a capacity of 12×10^6 bits is available. The ratio of record to playback speeds may be selected for each mission in the range 50:1 to 1:50 (binary ratios preferred for ground processing convenience) and with the limitation that the maximum input and output bit rate not exceed 45 kilobits per second. Start-stop capability may also be provided with some loss in total storage capacity. The data will be recorded with a frame parity check requiring 1.2% of the bits.

3.6.5 Transmitter

A 5 watt output transmitter at 136.5 megahertz is proposed as the basic module. Figure 5 shows the output bit rate capability of this unit with an omnidirectional antenna (turnstile) on the spacecraft and for the existing (18 db yagi) ground station system at various word error probabilities. Also shown are curves for the SATAN systems now being installed at most STADAN stations. For missions where the transmitter output power (5 watts) provides marginal communications system gain a 3 for 2 parity bit encoding system is proposed. This, together with a tape playback output buffer and tape drive servo system, removes data uncertainties to the level where data bit rates may be increased by a factor of 3. In addition a higher power transmitter module will be developed for missions where the power budget permits (DC power = 3 times radiated power).

Presently flying versions of command receivers and tracking beacon transmitters meet the experiment survey demands and only a minimum repackaging effort is required.

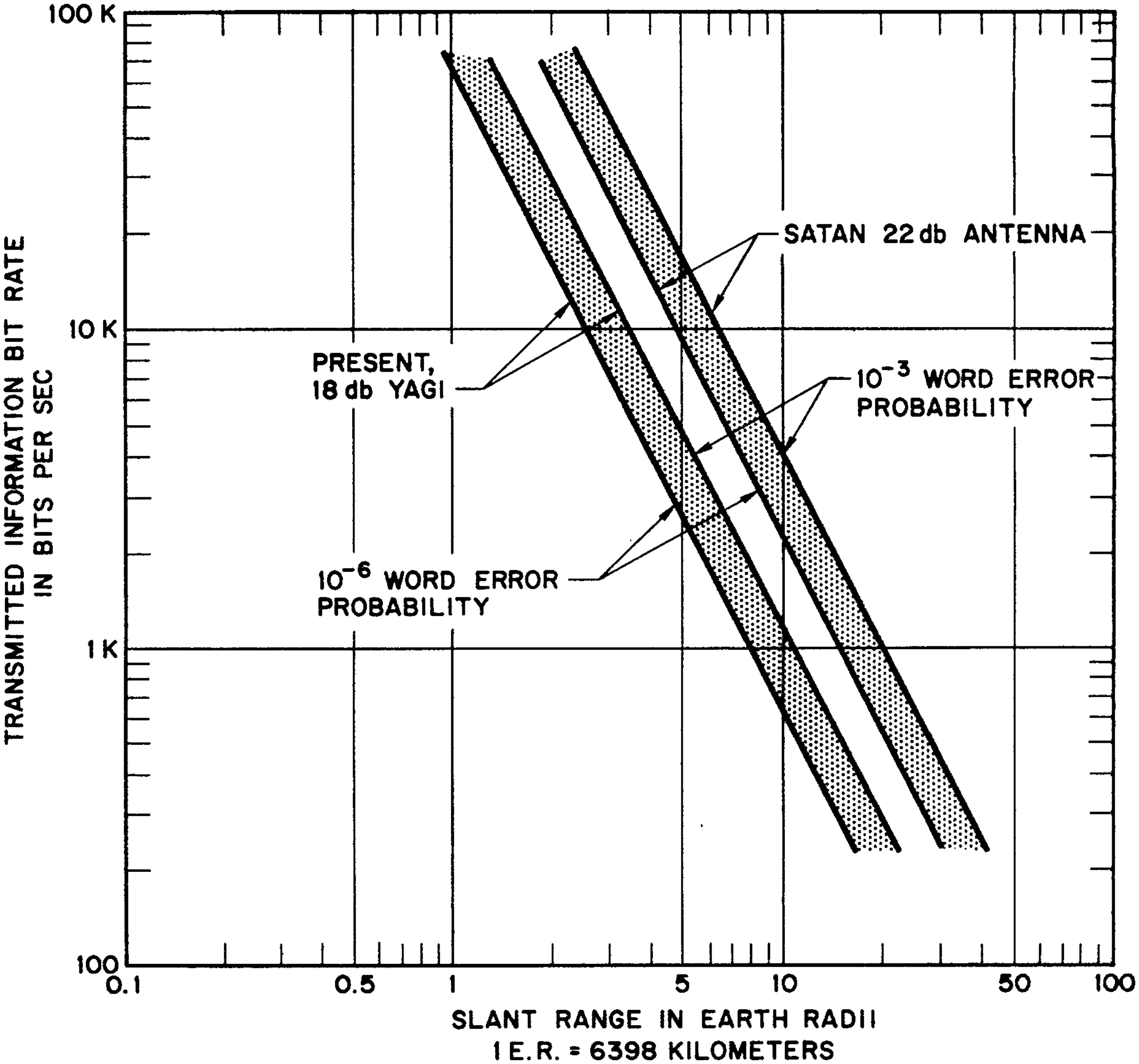


Figure 5–Output Bit Rate Capability for 5 Watt, 136.5 Megahertz Transmitter

3.6.6 Suitability of S^3 Data System

The data system described in the preceding pages, and in more detail in Appendixes F, G and H, should satisfy the broad spectrum of requirements set forth by the experimenters interested in particle and plasma measurements. These requirements have been discussed in section 2. Furthermore, the varying requirements of the many different missions considered can be accommodated with a minimum effort.

In addition, most of the requirements in the area of electric and magnetic field measurements can be satisfied by the proposed data system. The one major exception is the requirement by a number of VLF experimenters for direct modulation of the carrier. This requirement is not satisfied by the proposed data system. However, the development of a standard satellite structure along with various interface modules provides a vehicle which could be used for a number of such specialized missions.

4. GROUND SYSTEMS

4.1 SPACECRAFT GROUND SUPPORT EQUIPMENT (GSE)

Two GSE systems will be used to checkout the spacecraft during integration, environmental testing, and launch support. The first system, the Test Stand, contains the experiment excitation equipment and the spacecraft control and instrumentation monitoring equipment. Excitation and control functions are hardwired to test connectors on the spacecraft. Test points in the spacecraft are brought out through test connectors to the test stand where critical currents and voltages are monitored.

The Mobile Performance Analysis System (MPAS) is the second main piece of GSE equipment. The MPAS receives telemetry from the spacecraft via an RF link and, with suitable decoding equipment and a small computer and line printer, decommutates, sorts, compiles, performs conversions, and displays the information in a final, usable form. The MPAS equipment is van-mounted and can easily be transported to the various sites used for integration, environmental test, and launch operations.

4.2 TELEMETRY AND TRACKING OPERATIONS

For most of the proposed missions, stations in the Standard Tracking and Data Acquisition Network (STADAN) would be used to support the S^3 program.

Telemetry coverage requirements will be minimized by the use of on-board tape recorders. A few experiment groups may propose to use their own telemetry acquisition facilities.

Tracking operations for most missions can be handled with the Minitrack system. A rough approximation of the orbital position accuracy that can be obtained by Minitrack is (apogee in Km) $\times 10^{-3}$. Missions requiring higher accuracies for orbital position can use the range and range-rate system. In general, position accuracies to 1,000 meters or better can be obtained with this system with an additional 3 pounds and 0.5 watts standby power.

4.3 GROUND DATA PROCESSING (Appendix I)

Ground data reduction and processing is an integral part of the S^3 concept. As such, it is planned that the ground reduction gear and initial processing programs be capable of automatically handling in-flight format changes and a number of payloads without overloading. The changeover to various satellite configurations should take place routinely and with a minimum of effort.

Figure 6 is a general block diagram of the satellite ground reduction and handling subsystem and serves to illustrate the philosophy behind such a subsystem. The functions of the ground gear have been divided into two parts, A and B in Figure 6. In A, the primary function is to transform the data tapes recorded directly from the satellite, the station tape, into a binary tape suitable for computer use. Timing is also added and/or checked at this stage and is part of the primary function of this portion of the ground data reduction. It is further desirable for the ground processor to judge the quality of the data and to incorporate appropriate noise flags in the binary data tape.

While not necessary, the ground processor may recognize barker words and "n" bit bytes for synchronization purposes. Note that it is not at all necessary for the ground processor to know anything about the data format of the received signals.

Thus in A, the station tape has been converted to "0" and "1" bits and been written on a binary computer tape containing all the data from the satellite (experimental and housekeeping) along with appropriate timing and noise information.

In B, the binary tape generated in A enters a series of programs which perform any or all of the following functions: (1) data-frame and word recognition, (2) identification and rejection of errors, (3) insertion of orbit and generation of satellite position and parameters required by the experimenters for further analysis (e.g., B, L values), (4) generation of output tapes and/or plots and/or listings. The experimenters then receive these outputs for further analysis.

The above functions can be generalized as subroutines and as such will handle a variety of satellites. For example, the data frame and word recognition program can be parameterized so that any frame length and format can be processed. It is essentially a computer version of the programmable format memory utilized aboard the spacecraft that allows the above flexibility. Changeover to different satellites in the program is done by means of a control card.

It is important to note that the experimenters will receive a tape containing all the data from the satellite (experimental and housekeeping) along with the various parameters indicated above. Data from all experiments are kept well integrated (on the same tape) at all times, in keeping with the S^3 concept.

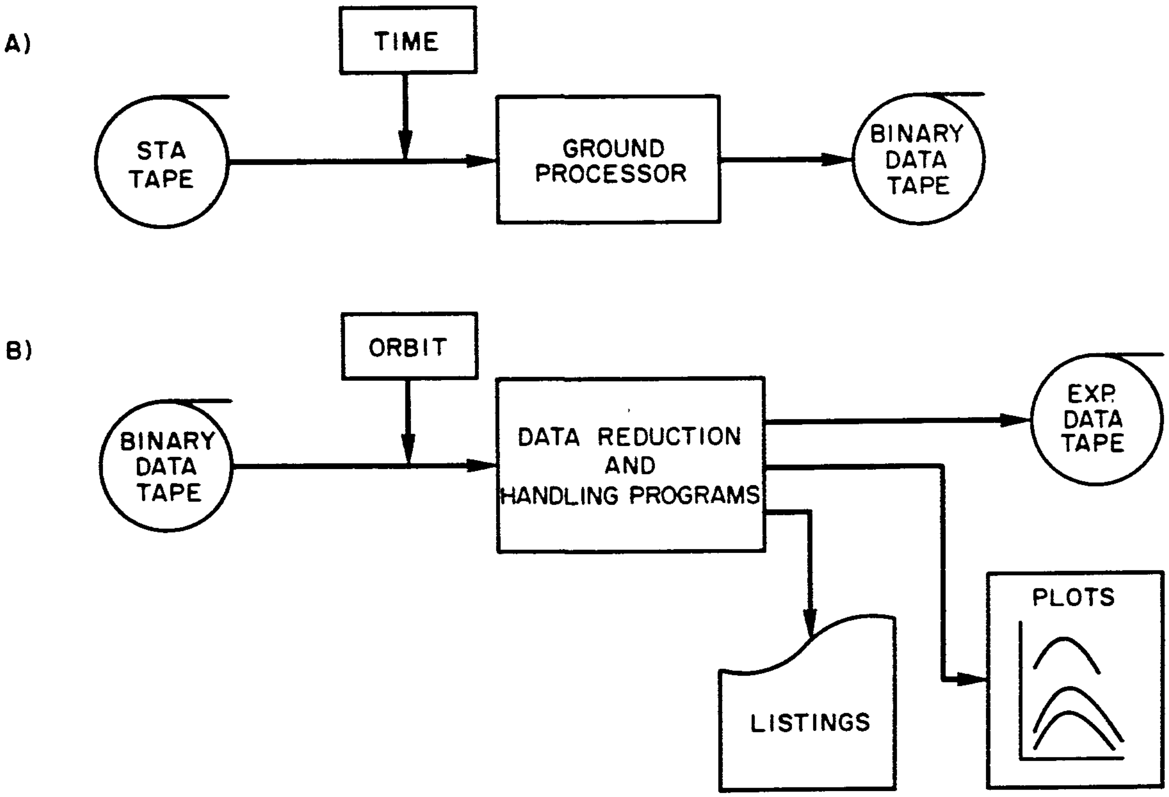


Figure 6–S^3 Ground Reduction and Data Handling Sub-System

5. LAUNCH CONSIDERATIONS

Many of the missions proposed by the experimenters during our interviews can be accomplished with configuration "B" of the Scout launch vehicle. The Scout vehicle has four solid propellant stages and is guided through the first three stages. The unguided fourth stage is spin stabilized prior to separation from the third stage. Configuration "B" consists of the Algol IIB first stage, the Castor II second stage, X259 third stage, and FW-4S fourth stage.

Circular orbit capabilities for due east launches from Wallops Station and polar launches from the Western Test Range (WTR) are shown in Figure 7. Elliptical orbit performances for due east Wallops Station and polar WTR launches are shown in Figures 8 and 9, respectively. The Wallops Station curves represent the maximum capability of the Scout since the due east launch takes maximum advantage of the earth rotational component of velocity.

Several proposed missions require orbits with near zero degree inclination. A due east launch from Wallops results in an orbit inclination of 38 degrees. Some control of the inclination angle is possible with a yaw torquing maneuver during third stage coast. However, the vehicle is not far enough down range during the maneuver to significantly lower the inclination angle. Also, the yaw torquing reduces the orbital capability of the Scout.

The obvious solution to the low inclination orbit problem is to launch from a near-equatorial site. There are two such sites in preparation. The first is the Texas tower platform being constructed for the San Marco project off the coast of Africa. The second is being developed by the French in French Guiana. Logistically this site is preferable and the French have invited United States participation.

The Vehicles Performance Branch at Langley is proposing a modification to the Scout system that has interesting possibilities. The modification consists of a velocity package (fifth stage) that significantly increases the Scout capability. The following performance figures were obtained during telephone conversations with John Canady and Bob Keynton from Langley. Injection altitude is assumed to be 100 nautical miles:

Wallops Station			WTR		
Payload Wgt. (pounds)	Apogee (nautical miles)	(Km)	Payload Wgt. (pounds)	Apogee (nautical miles)	(Km)
81	escape	–	62	escape	–
85	185,000	340,000	62.5	500,000	930,000
90	105,000	195,000	65	230,000	430,000
95	68,000	125,000	75	70,000	130,000
100	47,000	86,000	80	50,000	93,000
			90	29,000	54,000
			100	20,000	37,000

Several of the proposed missions that would not be possible with the present Scout system could be accomplished with the five stage version.

The spacecraft design will also be compatible with the Delta launch vehicle. Two or more satellites could be launched "piggyback" or an apogee kick motor could be included with the spacecraft to achieve synchronous orbit.

WALLOPS STATION LAUNCH
DUE EAST (38° Inclination)
WESTERN TEST RANGE LAUNCH
POLAR ORBIT (90° Inclination)

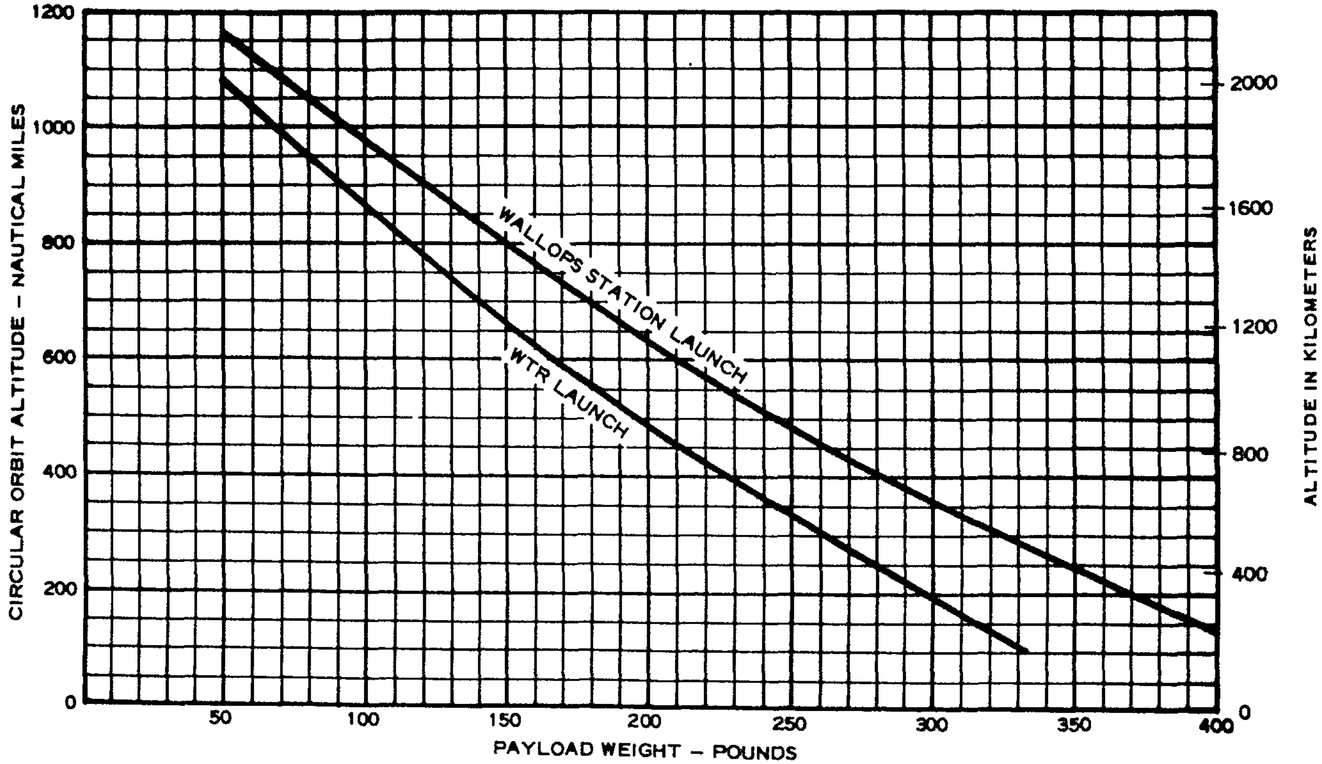


Figure 7–Circular Orbit Performance

WALLOPS STATION LAUNCH
DUE EAST (38° Inclination)

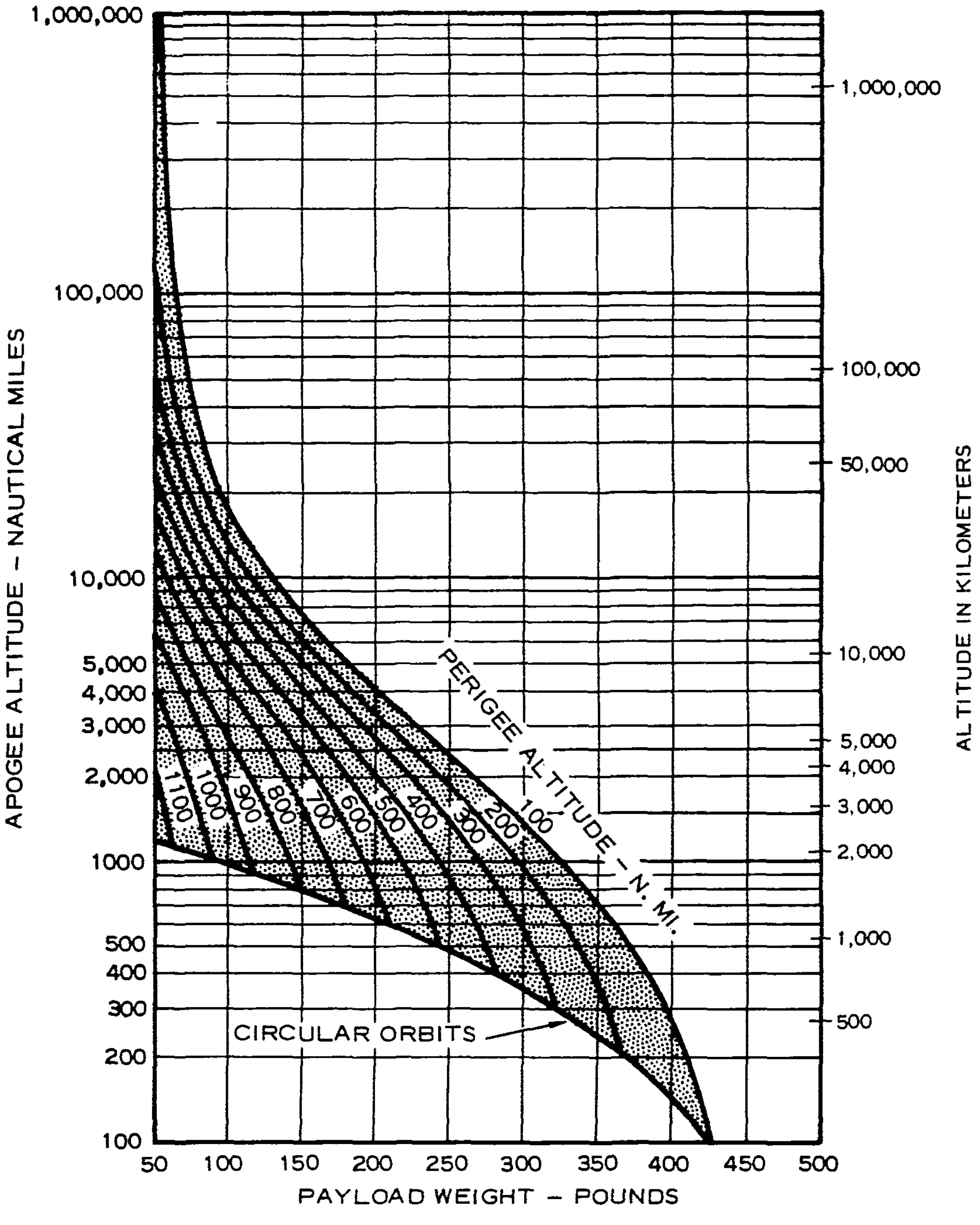


Figure 8–Elliptic Orbit Performance – Wallops Station

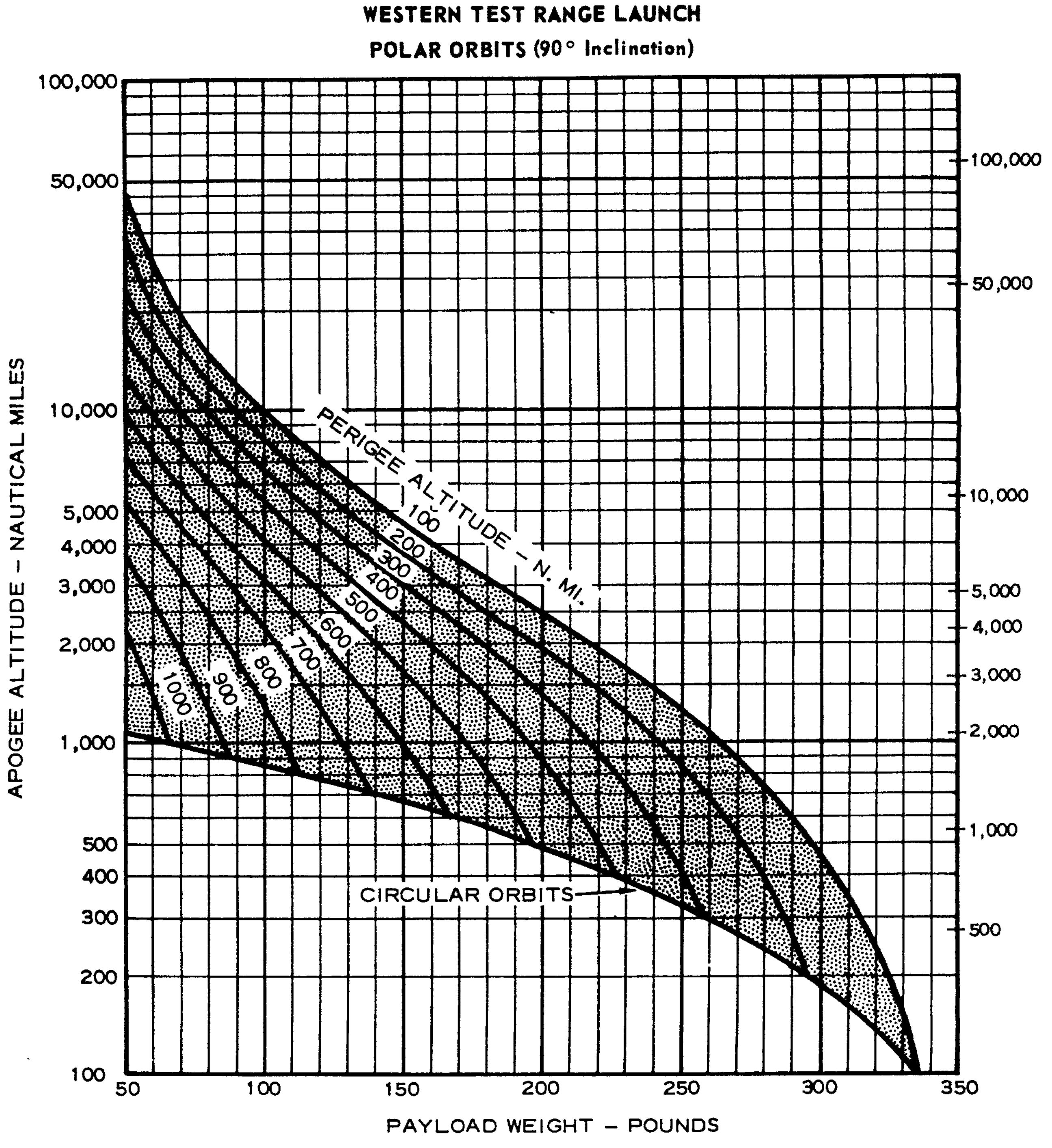


Figure 9–Elliptic Orbit Performance – WTR

APPENDIX A

MEMORANDUM

DATE: October 10, 1964

TO : Dr. John W. Townsend, Jr.
Assistant Director, GSFC

FROM : Leo R. Davis, Head
Auroral and Trapped Radiation Section, GSFC

SUBJECT : Design Study of Small Standard Satellite System for EPE Series

The purpose of this memo is to obtain approval to proceed with a design study of a small standard satellite system to continue the EPE series of studies within the magnetosphere. The study would be conducted by a working group composed of members of the Spacecraft Integration and Sounding Rocket Division, Spacecraft Technology Division, and Space Sciences Division. Appendix A is a list of the proposed working group.

The satellite envisioned would weigh 50 to 100 pounds, small enough to be launched into optimum orbits on the Scout or Delta vehicles or piggybacked with other payloads on larger vehicles. The results of the S-3 and IMP satellite series suggest that a basic system having rather general applicability to magnetospheric studies can be developed. If successful, the proposed system could be used with various sets of instruments on a number of missions over a period of years with only minor modification and improvements.

INTRODUCTION

Our concept of the magnetosphere and our knowledge of its physical structure and processes have evolved swiftly in the six years which have elapsed since Explorer 1 was launched. This is largely due to the direct measurements made on earth satellites and space probes in the magnetosphere.

The magnetosphere extends from 100 km. altitude out to about 10 earth radii at the subsolar point and flares out further in other directions, presumably forming an elongated cavity in the solar plasma with the maximum radial dimension in the anti-solar direction. Geomagnetically trapped particles populate much of the magnetosphere above 600 km. altitude. Trapped protons having energies from 100 kev to over 100 mev and electrons having energies from 10 kev to over 5 mev have been detected. In general the spectra of both the protons and electrons soften with increasing radial distance. The particles trapped below 2 earth radii are relatively stable with time. Above this region the electrons, and less frequently the protons, show large temporal fluctuations, the more rapid of which are coincident with fluctuations in the geomagnetic field. The currents resulting from the motions of the observed protons should produce a measurable distortion of the geomagnetic field. The effect of still lower energy protons may be larger and their enhancement during geomagnetic storms may produce the main phase. Low energy electrons and some protons are found to be precipitated more or less continuously into the upper atmosphere on field lines which traverse the trapped particle region. The more intense precipitations produce aurora and auroral absorption events. There are reasons to believe that many of the transient phenomena occurring in the magnetosphere, such as particle fluctuations, field fluctuations, aurora, and VLF emissions, are closely related and perhaps are all driven by energy derived from the solar plasma impinging on the magnetosphere. At present no adequate physical theory exists to explain and relate the transient phenomena.

Studies which will contribute to a better understanding of the magnetosphere require measurements of the particle intensity and spectra, measurements of the magnetic field intensity and direction, measurements of the VLF emission, and probably measurements of electric fields. These should include studies of the short term fluctuations and long term fluctuations having periods up to eleven years, the solar cycle. Table 1 lists the characteristic times of some magnetospheric phenomena. The time resolution of measurements is determined by the design of the instrumentation, the telemetry bit rate and the satellite orbit. In an elliptical orbit the satellite motion is slow enough over most of the orbit that fluctuations having periods of a few seconds or less may be measured directly. Because of the strong spatial dependence in the particle intensities and magnetic field intensities the time resolution for longer period fluctuations is the order of the orbital period. The orbit period as a function of apogee is shown in Figure 1, for both elliptical and circular orbits. By comparing the orbit periods with the characteristic times listed in Table 1, it may be seen that measurements having the required time resolution for various magnetospheric phenomena will require a number of different orbits. As an example, suitable detectors and magnetometers are now available to investigate the mechanism by which the ring current of trapped particles is produced or modulated and how this current distorts the magnetosphere. The desired measurements could probably be made in an elliptical orbit having a low inclination and an apogee of about 4 earth radii. The period of this orbit is 5.5 hours, thus providing adequate time resolution in the measurements during the main phase of a geomagnetic storm. It is quite unlikely that a successful study of the main phase can be performed with instrumentation on an IMP or EGO satellite where the orbital period of three days or longer is long compared to the duration of the main phase and where the high inclination of the orbit makes it highly probable that the satellite will pass through the region of interest at such a high geomagnetic latitude that only a few of the protons would be sampled. The high apogee orbits of course are required for continued studies of other phenomena. Since their orbital periods are much too long elliptical orbits are not well suited for studies of the changes in electron intensity which have been observed to occur above 3 earth radii and with periods of the order of an hour or less. Circular orbits in this region would provide the required time resolution for such studies. These are but a few examples of the diverse orbits required for studies of different magnetospheric phenomena.

Table 2 is a partial listing of satellites which have been instrumented for scientific studies of the magnetosphere. It includes most of the satellites which have made major contributions in this area and which have been launched with United States vehicles. At the bottom of Table 1 are listed the spacecrafts scheduled for future studies of the magnetosphere. The future IMP and EGO launches provide excellent opportunities for continued studies of the higher reaches and boundary of the magnetosphere. The future POGO launches provide excellent opportunities for continued studies of the lower edge of the magnetosphere and auroral phenomena. The planned ATS satellites which will be launched into circular orbits at 2.7 earth radii and 6.6 earth radii are of limited scientific value because of magnetic contamination and severe restrictions on apertures and appendages for scientific instruments. The EPE-D satellite, which is to be launched into a 5 earth radii apogee elliptical orbit later this year, is the only NASA satellite with instruments and orbit well suited for studies in the region from two to ten earth radii. EPE-E and EPE-F satellites are budgeted for and presumably will be used for studies in this region.

During the last two years the members of the Auroral and Trapped Radiation Section (GSFC) have been developing plans for future studies and a family of particle detectors to make the required measurements. Early in this period we became aware of the lack of opportunities for studies in the 2 to 10 earth radii region provided by the present program. In considering means to correct this deficiency, we soon concluded that a small satellite program would be most effective and economical (the reasons for these conclusions are contained in the following paragraphs). Our initial plans envisioned a small, specially designed, tightly integrated spacecraft analagous to the Injun satellites built by the State University of Iowa. However, further consideration of the design requirements and the design of previous small satellite systems have lead us to conclude that a small standardized spacecraft system can be developed which is suitable for a much wider range of studies in the magnetosphere. We are therefore proposing that such a system be developed.

The design of this satellite should be carried out in three phases. The initial phase, lasting for some six to eight months, would be spent in developing and breadboarding any special spacecraft subsystems required and in blocking out the overall spacecraft system. At the beginning of this phase, a representative group of experienced experimenters from various laboratories would be contacted in an effort to develop a general set of design objectives (See Appendix B for list of experimenters). At the end of Phase 1, a report of the results would be prepared. If the results show that a standardized system is feasible, a proposal for Phase 2 would be submitted for Headquarters' approval and funding.

In the second phase, a final spacecraft and instrumentation design would be completed. Hardware would be fabricated and tested and launched, possibly on a Scout vehicle, to test the spacecraft-experiment system. In the third phase, which would extend over several years, additional units would be built and instrumented by different groups for a number of scientific studies in the magnetosphere as a continuation of the EPE program.

It should be noted that the Auroral and Trapped Radiation Section (GSFC) in cooperation with Dr. L. Cahill and his group in the Department of Physics at the University of New Hampshire are preparing a proposal for a joint magnetic field-charged particle study to be conducted on such a small satellite as proposed here. The instrumentation envisioned would requir all or at least a substantial portion of the small satellite payload. It is our intention to submit this proposal for the first launching in the small satellite development program. We would recommend GSFC participation in the second phase contingent on approval of this proposal and its assignment to the test mission. Appendix C is a preliminary outline of this proposal.

ADVANTAGES OF SMALL STANDARD OBSERVATORIES

The relative advantages of small specially designed spacecraft and the large standard observatory have been discussed in detail by Ludwig[(1)]. The advantages of the small spacecraft are (1) greater choice in tailoring the orbit to the experiment when a small number of related experiments are involved; (2) greater choice in selecting the spacecraft orientation to suit the experiment; (3) greater ease in tailoring the environment to the experiment, i.e., eliminating magnetic or electronic interference; (4) greater ease in scheduling, integrating the payload, and environmental testing when a smaller number of experiments and spacecraft subsystems are involved and when spacecraft subsystems are less complex; (5) the experimenters and spacecraft system engineers can more readily and completely understand the overall system and thereby reduce the probability of a failure or malfunction occurring due to an unforeseen combination of events; and (6) it is possible to launch a smaller spacecraft with a shorter lead time.

Of the advantages listed by Ludwig for the large standard observatory only two are still valid when compared with the small standard observatory. These are (1) the capability of performing complex or large numbers of experiments requiring a large amount of instrumentation and power, and (2) the cost per pound of instrumentation, per watt of power, and per bit of information transmitted is about a factor of two less for the large observatory.

In magnetospheric studies where a number of different orbits are required for different phenomena and where in many cases large amounts of

[(1)]Ludwig, G. H., The Orbiting Geophysical Observatories, <u>Space Science Reviews</u>, 2 (1963), pp. 175-218.

instrumentation are not required the small satellite can be used to a distinct advantage.

Other advantages of the small satellites result from the fact that they, in general, contain sets of directly related experiments, provided by a small number of laboratories or even a single laboratory. This permits joint planning and development, by such groups, of proposals for complete missions. This in turn should provide better integrated sets of instruments and conditions for more effective comparisions of results.

A great effort has gone into the development of large standard observatories which are highly flexible in order to accommodate many types of experiments. There has been no systematic development to evolve a small standard satellite. Nevertheless, there has been a trend toward standardization in the small spacecraft systems. This standardization has come about in the building blocks or subsystems more than in the overall system. This is perhaps best shown in the genetically related series consisting of Explorer VI, Pioneer V, Explorers XII, XIV, XV, and XVIII. The octogon structure fabricated of plastic - aluminum honeycomb and the solar cell paddles developed for Explorer VI, and Pioneer V spacecraft were used with minor modifications for the Explorer XII, XIV, and XV spacecraft. The Explorer XII, XIV, and XV spacecraft used the PFM telemetry and antennae previously developed in the Vanguard program. The PFM system has been used on a number of other small satellites and common ground decoding and digitization equipment has been used to process the data.

The IMP satellite (Explorer XVIII) utilized, with some modification, the previously developed solar cell paddles, octogon configuration, and PFM telemetry. It departed from the previous satellites in this series by adopting a standard plug-in module for housing detectors and electronics. The flexibility of the IMP system has been demonstrated by the relative ease with which it has been adopted to the IMP-D and E lunar orbiter mission.

SPACECRAFT AND SUBSYSTEMS

A preliminary design study to evolve a small standard satellite system has been made by an informal working group composed of the people listed in Appendix A. A set of general design objectives was developed, a set of basic constraints assumed, and then a system design worked out, using existing subsystem designs wherever possible.

The proposed study would be carried out in a similar fashion. A more comprehensive set of objectives would be developed, which would include the ideas and requirements submitted by the experimenters listed in Appendix B. The results of the preliminary study are given in the following paragraphs to illustrate the planned approach, the flexibility that can be achieved in such a standard system, and to note subsystem areas where new developments may be required.

The preliminary design objectives are: (1) operation in orbits having apogees of 2 to 10 earth radii geocentric, either eliptical or circular, (2) capability of launch on Scout, Delta, or piggyback, (3) capability of easily accommodating various weights, and types of instruments, (4) at least 10 pounds of experiments, (5) at least 4 watts average power for experiments, (6) continuous data rate of at least 100 bits per second throughout orbit, (7) temperature inside the payload within the range -10 degrees to 50 degrees C, (8) spin stabilized, (9) no constraint on spin axis-sun angle for thermal control, (10) spacecraft environment suitable for magnetic field studies, (11) spacecraft design to have provisions for mounting rings on both ends for optional attachment of kick motor to achieve circular orbits or optional stacking for multiple launches, (13) no weight penalty for unused optional features, (14) solar cell paddles and all other appendages and booms should fold in such a fashion as to form a single compact unit during the launch phase, (15) optional reorientation system, (16) one or two year useful life.

The basic design constraints are: (1) Payload diameter less than 29 inches, the maximum permitted on the Delta. For the Scout the value is 30 inches. The constraints for piggyback payloads were not determined. (2) Payload weight design range of 50 to 100 pounds. Table 3 lists the payload weights of the Scout and Delta vehicles for several orbits of interest. As can be seen in Table 3, the Scout vehicle is capable of placing up to 80 pounds of payload into a four earth radii elliptical orbit. The Delta 3C vehicle is capable of placing a 120 pound payload into a ten earth radii apogee elliptical orbit and up to 460 pounds of equipment into a two earth radii apogee elliptical orbit. The selection of a 50 to 100 pound payload design range would permit these two vehicles to be used for achieving a number of useful orbits. For example, the Scout vehicle could be used for orbits up to five earth radii; the Delta vehicle would then be capable of placing the satellite in elliptical orbits up to ten earth radii apogee or high inclination orbits. The Delta vehicle would also be capable of launching two or more of the satellites into lower orbits. High circular orbits could be achieved using the Delta vehicle with a kick motor attached to the payload. (3) In order to permit nearly continuous recording of data, at a reasonable cost of telemeter station time, onboard recording must be used. For efficient utilization of the onboard tape recorder Pulse Code Modulation will be required.

SPACECRAFT SYSTEM

Figure 2 is a block diagram of the spacecraft system. The basic spacecraft would consist of the solar cell power system, the command receiver, the tape recorder, and data transmission system. The equipment unique to a particular set of experiments might consist of a set of detectors, an optical aspect unit, an onboard clock, and data conditioning equipment which would convert the measurements to a serial train of bits which would be fed to the tape recorder. The stored data would be read out on command and encoded in a form suitable for transmission. A low power transmitter is operated continuously for tracking purposes and could also be used for real time transmission of the data as a backup for the recorder.

A GSFC design tape recorder is available which weights 3 pounds, consumes 1.3 watts in the record mode, 2.3 watts during the readout mode, and provides three million bits storage capacity. The record speed may be changed between missions to provide a bit rate from 10 to 500 bits per second. Any record period to readout period ratio of 50 to 1 or less may be used. The first of these units was flown on UK-2 and is still in operation eight months after launch. Additional units have been delivered and are being tested for use on the OSO and AEB satellites. A data transmission rate of 6000 bits per second would result in about an 8 minute readout period. At 120 bits per second recording rate, the storage capacity would be about 7 hours.

An analysis of the communications system has shown that transmission of 6000 bits/sec from ten earth radii, using standard PCM with one error per thousand bits and including a 7.5 db margin for system deterioration, would require 7 watts radiated power. This could be reduced 3 to 5 db by using coded PCM with special synchronization. Thus it is desirable to investigate data encoding techniques to reduce the power requirements. It is proposed to study three types of systems: PCM (standard, uncoded), PCM (special synchronization, uncoded), and PCM (special synchronization and coded). The coded system to be investigated will probably use feedback shift-register generators, generating the so called psuedo-noise sequences.

Initial study of the time flutter introduced by the tape playback machine indicates that ground synchronization would be marginal near threshold conditions using uncoded PCM. Thus, it is felt that some means of eliminating the tape playback "time noise" will be required. Past methods have done this at the expense of complicated aperiodic data transmissions which also complicated ground data processing. It is hoped that a simple reliable technique can be found which allows a continuous, unambiguous data stream to be transmitted.

Existing designs could be used for the transmitters and command receiver. Power would be supplied by N on P solar cells with appropriate cover glass for radiation shielding. Conventional converter circuitry could be used.

The data conditioning equipment is considered a part of the experiment instrumentation. It is expected that a family of circuit modules will be developed which would perform the majority of the experimenter's requirements for shift registers, A-D convertors etc. Although these would be assembled into different configurations for each mission the re-use of identical modules will provide greater confidence through accumulation of test histories.

SPACECRAFT STRUCTURE

The spacecraft design presently being considered utilizes IMP type cards for electronics and detectors. The dimensions of the cards used on IMP are shown in Figure 3 and as they are assembled in a spacecraft in Figure 4. The advantages of this type structure are:

1. The plug-in feature facilitates assembly and substitution of sub-assemblies.

2. The wiring harness, which fits around the wall of the center tube (Figure 4), is relatively simple and requires no long leads.
3. This configuration provides the high ratio of moments of inertia required for spin stability.
4. Sensors may be conveniently arranged to look out the sides or through the ends.
5. The use of flat surfaces simplifies the fabrication of structural components.
6. The card frames or detector housings bolt together in stacks which are then connected in a ring to form the main structural member of the spacecraft. This results in a lighter spacecraft than when the subassemblies are simply bolted to a separate main structural member.

The external form of the IMP spacecraft retained the pillbox form of the internal components. Because of the variation in illuminated area with changing spin axis-sun angle, the pillbox configuration is poorly suited to passive thermal control. This problem could be solved by utilizing active thermal control on the pillbox shaped spacecraft or by attaching thin domes to approximate the desired spherical shape.

Figures 5 and 6 illustrate a type of solar cell panel design being considered which wraps around the spacecraft in a compact fashion. An additional advantage of the wrap-around solar panels is that their deployment is accomplished more slowly and with lower stresses. Thus they would be lighter than the old solar cell paddles. On deployment the panels could be rotated to a fixed orientation with a simple spring actuated mechanical device or they could be servoed to a sun sensor which could rotate the panels through a 90° range providing a partially oriented solar array. With this solar panel design the number of panels could be varied, with the upper limit set by fairing clearance and weight limitations.

Estimates of the structural weight for the IMP pillbox configuration are as follows:

1.	Platform	3	pounds
2.	Top cover	3	pounds
3.	Center tube	2	pounds
4.	Balance weight	1	pound
*5.	Struts	.5	pound
**6.	Solar panels less cells	5	pounds
7.	Booms	1.5	pounds
8.	S-3 type magnetometer tube	.5	pound
9.	Antenna plus fasteners	2.5	pounds

*Struts would only be included if booms were to be flown.

**Weight includes 8 - 9x10" solar panels plus four orienting devices.

SPACECRAFT CHARACTERISTICS

Power and weight estimates for two versions of the spacecraft outlined in the previous two sections have been made. The minimum version weighs a total of 50 lbs and has an eight-panel, 16-watt solar array. The maximum version weighs a total of 100 lbs and has a twelve-panel, 24-watt solar array. The maximum version differs from the minimum version mainly in that it requires a larger battery pack and power converter, and in having a deeper center tube and top cover. The weight summaries are shown in Table 4 and the power summaries in Table 5. The minimum version contains 14.8 lbs of experiments and data conditioning equipment and the maximum version 52.1 lbs. The power for these two items are 7.5 watts and 11.8 watts, respectively.

SPACECRAFT ATTITUDE CONTROL

Small scientific satellites have employed spin stabilization, gravity gradient stabilization, and magnetic field stabilization. Spin stabilization is particularly suitable when a scan over a range of directions is required. Gravity gradient and magnetic orientation are most appropriate and easily achieved in low circular orbits. In the past the spin axis direction has been determined by the pointing direction of the last stage of the launch vehicle, and only a limited adjustment in the pointing direction has been possible through a choice of the launch time and the vehicle trajectory. Indeed in many cases it has not been possible to achieve the desired spin axis orientation because of the limited range of adjustment available in the third stage pointing direction or because other critical constraints such as thermal control requirements eliminated any possible adjustment of the pointing direction. In the future as more sophisticated experiments are

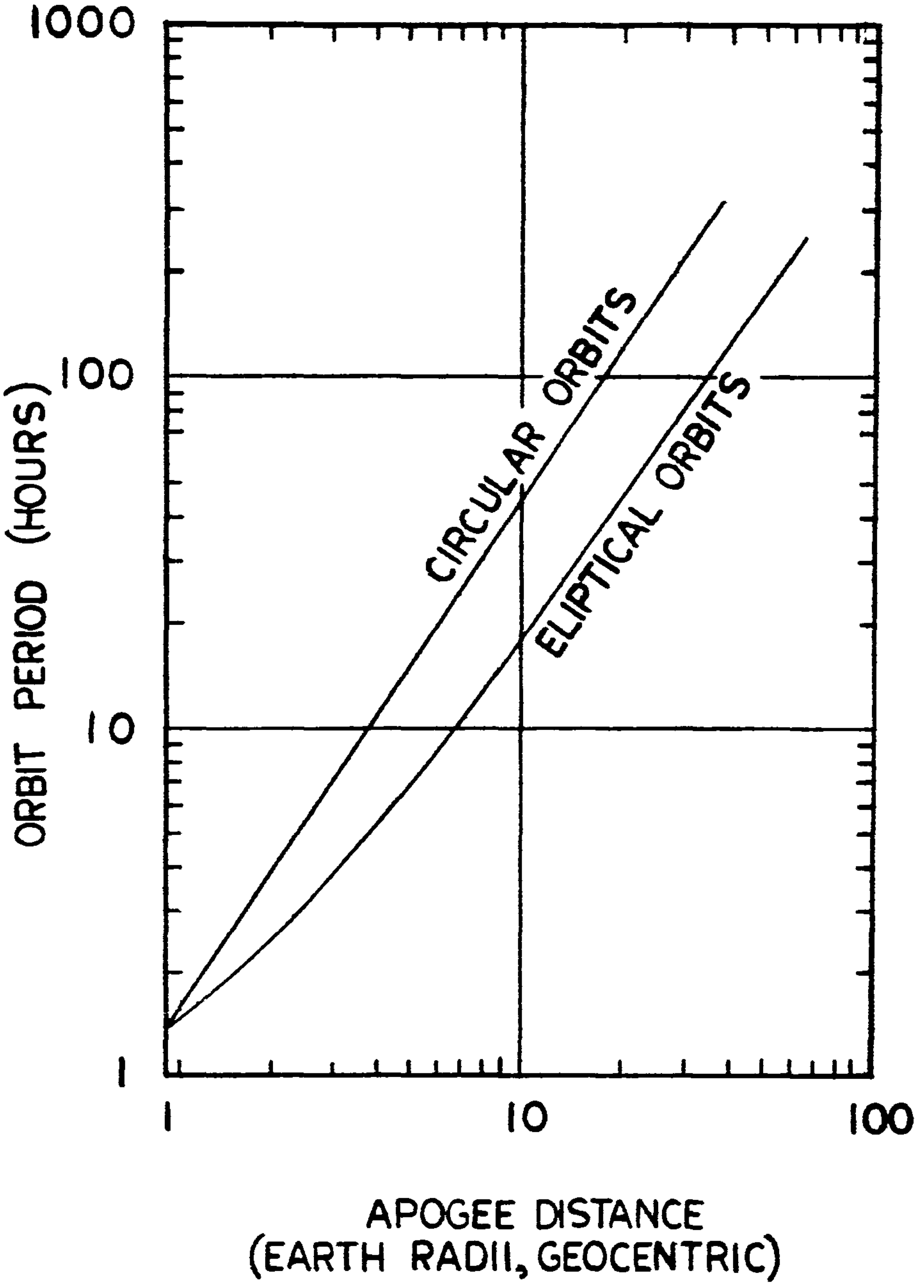


Figure 1–Orbit Period vs. Apogee Distance

performed for definitive studies of select phenomena, preferred orientations will become necessary. This problem will be compounded if the instruments are launched piggyback on a larger payload which has priority in setting constraints. Thus it would be highly desirable to develop a system for post-injection spin-up or spin-down and reorientation of the spacecraft. The initial requirements on such a system would be reorientation of the spin axis to any selected direction with an accuracy of about five degrees and the spin-up and spin-down to a selected spin rate within perhaps $\pm$ 25%. To be useful the system need not provide continuous reorientation capability throughout the life of the satellite but would satisfy many missions if it could perform only an initial reorientation. The trend in the future would probably be toward more accurate reorientation and reorientation throughout the life of the satellite.

The subliming-solid-powered system presently being developed at GSFC shows great promise in meeting the above requirements. For example, at present, a system weighing a total of 2.5 lbs. could be built which would point the spin axis of an S3 type satellite at the sun to within one degree for one year. It would be desirable if the development were continued to provide the capability of pointing in any selected direction. It would also be desirable to investigate gravity gradient and magnetic orientation systems for this satellite.

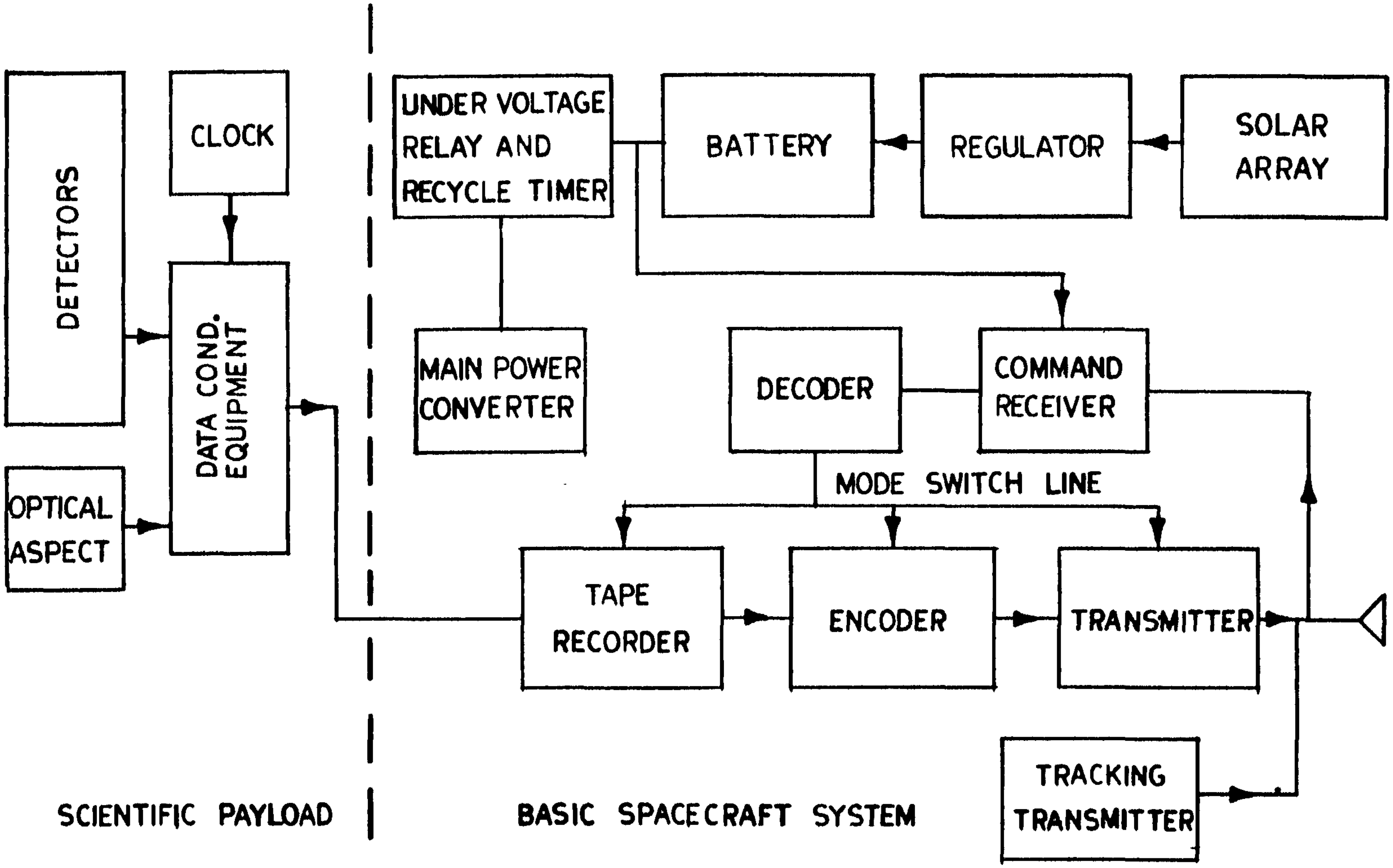
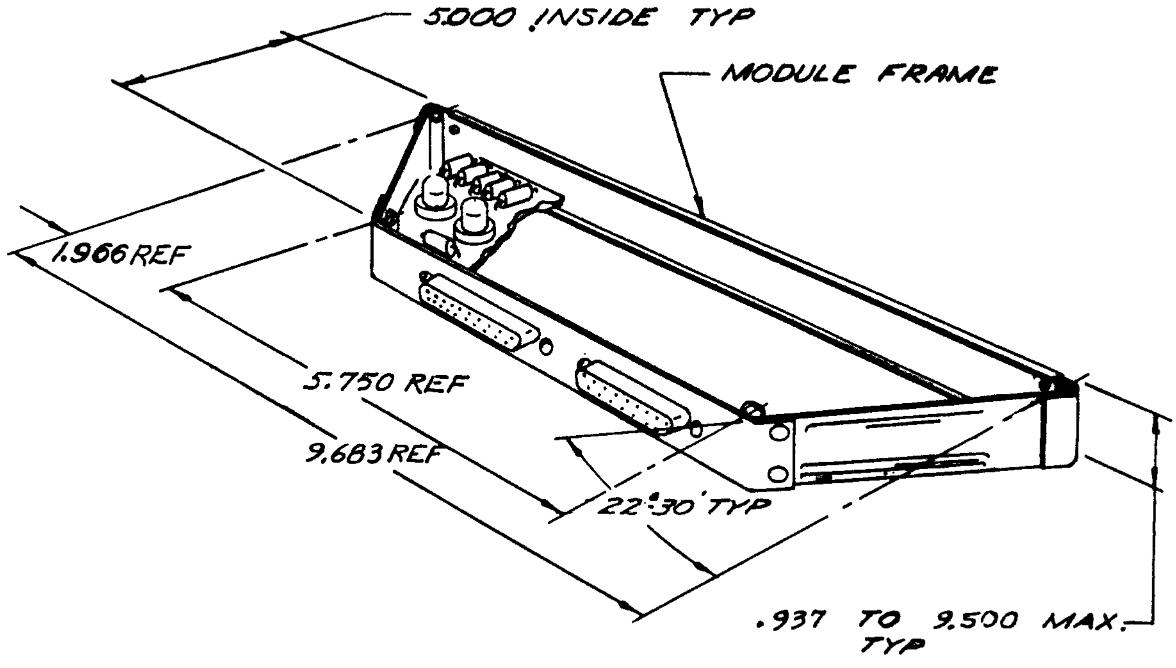
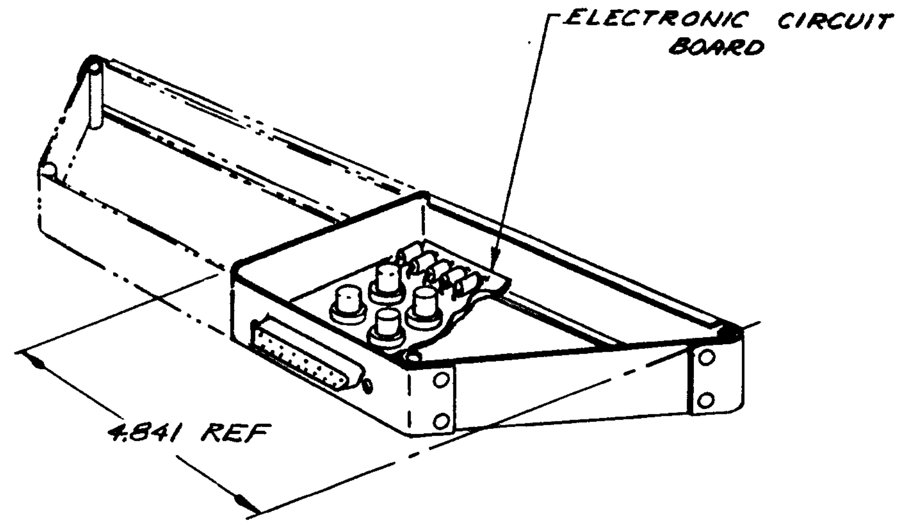


Figure A-2–Block Diagram of Satellite System



S-74 INTERPLANETARY MONITORING PLATFORM, FULL CARD



S-74 INTERPLANETARY MONITORING PLATFORM, HALF CARD

Figure A-3

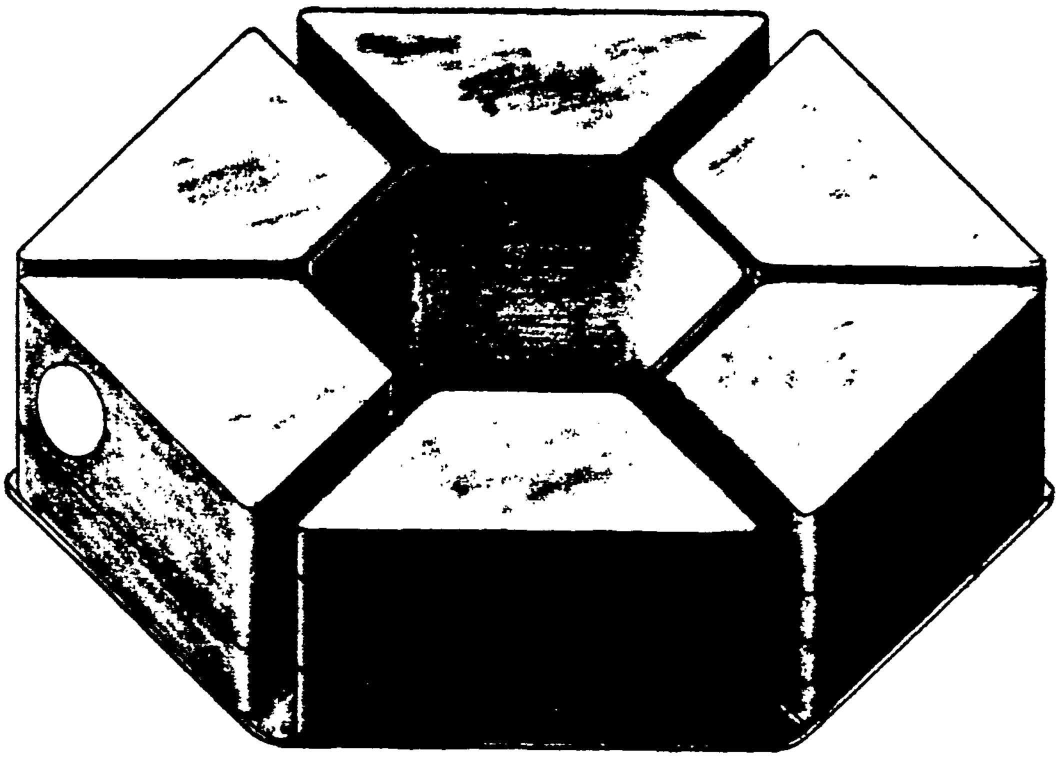
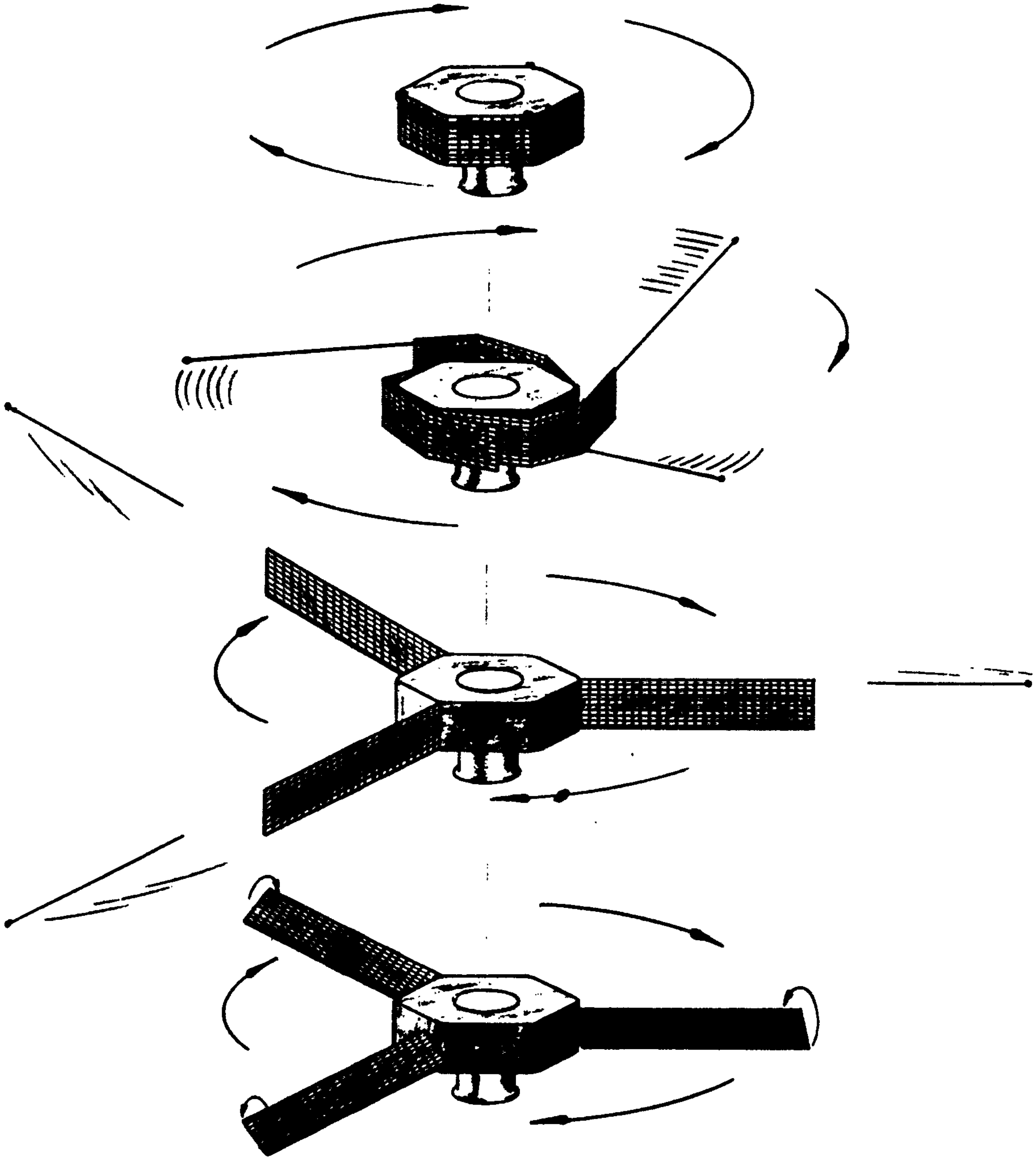
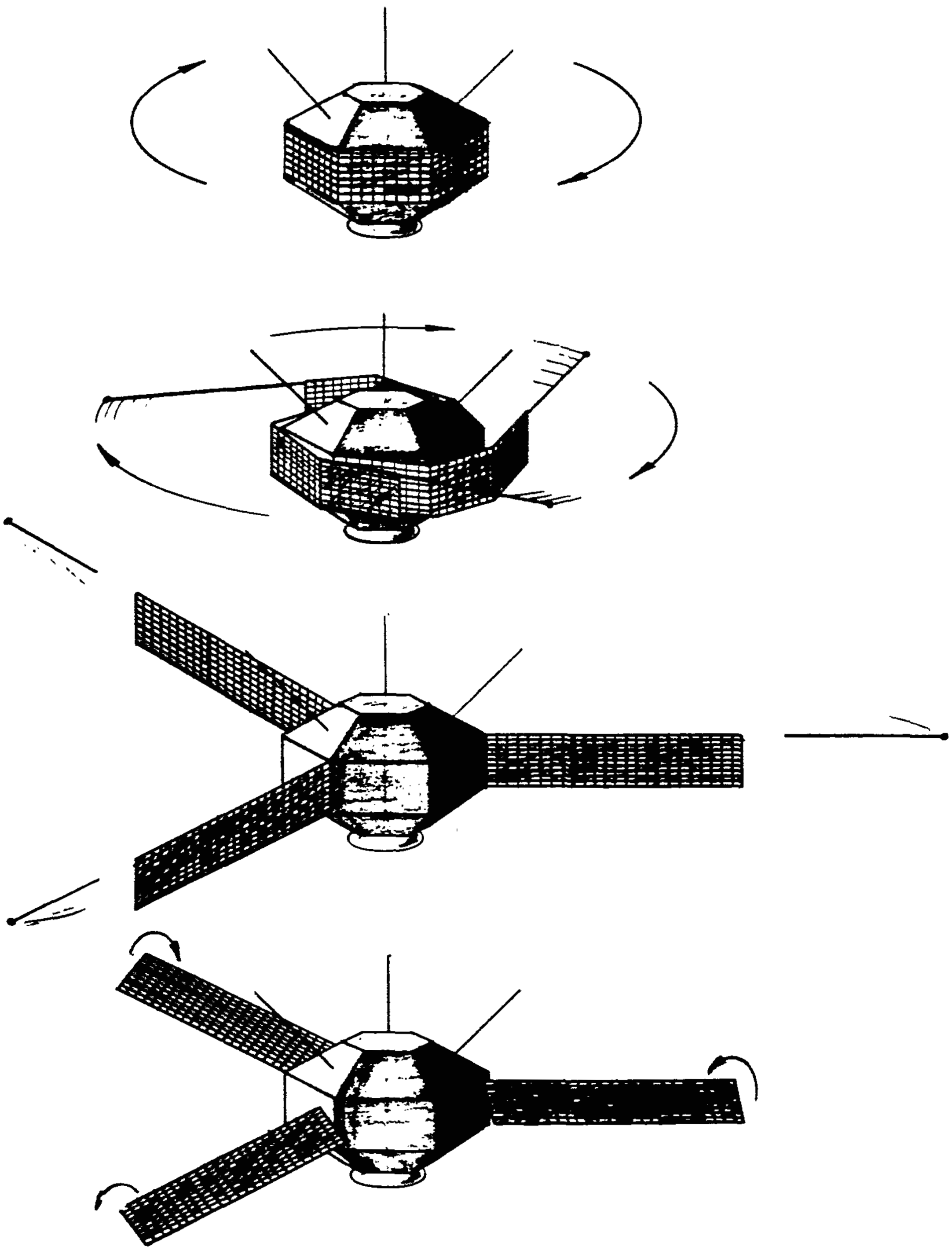


Figure A-4—Cards Assembled in Spacecraft



WRAP AROUND SOLAR CELL PANELS ON PILL BOX CONFIGURATION

Figure A-5



WRAP AROUND SOLAR CELL PANELS ON SPHERICAL CONFIGURATION

Figure A-6

APPENDIX A

(Revised 1/6/65)

PROPOSED WORKING GROUP FOR SMALL STANDARD SATELLITE STUDY

G. W. Longanecker	Project Manager
R. O. Wales	Systems Engineer
L. R. Davis	Project Scientist
Dr. G. H. Ludwig	Systems Consultant
W. P. Jones	Power System
P. T. Cole	Tape Recorder
T. Saliga	Data Encoder
H. J. Peak	R.F. System
R. E. Kidwell	Thermal Design
J. Madey	Mechanical Design
J. S. Albus	Optical Aspect
Dr. J. V. Fedor	Attitude Control
A. B. Malinowski	Data Conditioning Equipment

APPENDIX B

(Revised 1/6/65)

List of Proposed Experimenter-Consultants

K. A. Anderson	University of California
H. J. Bridge	Massachusetts Institute of Technology
W. L. Brown	Bell Telephone Laboratories
L. Cahill	University of New Hampshire
R. A. Helliwell	Stanford University
J. P. Heppner	NASA/GSFC
C. E. McIlwain	University of California, San Diego
M. G. Morgan	Dartmouth College
N. F. Ness	NASA/GSFC
B. J. O'Brien	Rice University
J. A. Simpson	University of Chicago
E. J. Smith	Jet Propulsion Laboratory
C. P. Sonett	NASA/AMES
J. R. Winckler	University of Minnesota
J. A. Van Allen	State University of Iowa
C. O. Bostrom	John Hopkins - APL
W. A. White	NASA/GSFC

APPENDIX C

Joint GSFC - U. of N.H. Proposal For Magnetic Field-Charged Particle Study

Mr. L. R. Davis, Dr. D. Evans, Dr. R. A. Hoffman, Dr. A. Konradi, Dr. J. Trainor, and Mr. J. M. Williamson of the Auroral and Trapped Radiation Section, GSFC, and Dr. L. Cahill and Dr. R. Kaufmann of the Dept. of Physics, University of New Hampshire are preparing a proposal for a joint magnetic field-charged particle payload which could be launched on a small satellite. The scientific objectives are: studies of the ring current, studies of the acceleration and loss of trapped particles, and a determination of the part played by hydromagnetic storm or disturbances in these processes.

The purpose of the joint proposal is to provide a comprehensive set of particle and field measurements which would be available to both groups of experimenters for use in studies requiring direct correlation of the field and particle measurements. The problems involved in comparison of data obtained by different experimenters on spacecraft are well known. We plan a joint processing of the data to combine the data with orbit parameters and determine the direction of the detectors for each measurement. This will include the use of the magnetometer data to determine the pitch angles of the particles being detected and the B value, leaving only the L parameter unmeasured. We feel that this method is inherently more likely to be effective than separate particle and field experiments conducted by different groups with comparisons of results only after separate analysis of the data.

The magnetometers are currently being developed at the University of New Hampshire and the particle detectors at GSFC. The magnetometer is a three axis, flux gate instrument capable of vector measurements of high precision

(4_γ accuracy with components of $4{,}000_\gamma$). The instrument will be capable of measuring field fluctuations of the order of several gammas in the frequency range of 1 to 10 CPS. A search coil would probably be included for higher frequencies.

Two types of particle detectors are being developed. Dr. David Evans is developing detectors utilizing the channel electron multiplier for measuring the directional intensity and spectra of 100 ev to 50 kev electrons and 1 kev to 200 kev protons. Dr. J. Trainor is developing a system using silicon diffused junction detectors to measure the directional intensity and spectra of 80 kev to 500 kev electrons and 400 kev to 20 mev protons. These detectors, possibly combined with the zinc-sulfide-scintillator proton detector developed by Davis and Williamson will sample the particle spectra over the complete range of 100 ev to 500 kev for electrons and 1 kev to 20 mev for protons with about three samples per decade.

Rough estimates of the power and weight requirements are 4.5 watts and 16.3 lbs. This would include data conditioning equipment. Thus the instrumentation could be launched on approximately the minimum version of the proposed small satellite.

TABLE 1 - Characteristic Times of Magnetospheric Phenomena

Solar Cycle	11 Years
Period of Solar Rotation	27 Days
Build-up Time > 5 Mev Electrons in outer-belt	Few Days
Decay Times 100 Kev Electrons in outer-belt	Few Days
Main Phase of Geomagnetic Storm	One Day
Trapped Particle Drift Period	0.1 to 10 Hours
Redistribution of 34 Mev Protons at $L = 2.5$	6 Hours
Sudden Commencement of Geomagnetic Storm	One Hour
Build-up Time 100 Kev Electrons in Outer-Belt	$\lesssim$ One Hour
Auroral Pulsations	$<10^{-2}$ to >100 sec.
Trapped Particle Period of Latitudinal Osc.	.1 to 100 Sec.
Larmor Periods of Trapped Particles	10^{-6} to 10^{-3} Sec.

TABLE 2-Partial Listing of Satellites for Scientific Studies of Trapped Radiation, Geomagnetic Field, Plasma, & VLF

Satellite	Launch Date	Launch Vehicle	Apogee (earth radii)	Satellite Weight (lb.)	Satellite Power (watts)	Telemeter Bit Rate (bits/sec)	Weight of Scient. Inst. (lb.)	Experiments
Explorer 1	31 Jan 58	Jupiter C	1.28	31				CR (First detection of trapped radiation)
Explorer 2	26 Mar 58	Jupiter C	1.44	31				CR, TR
Explorer 4	26 July 58	Jupiter C	1.35	26				TR, CR
Explorer 6	7 Aug 59	Thor-Able	7.6	142				TR, MF
Vanguard 3	18 Sept 59	Vanguard	1.6	50				MF
Explorer 7	13 Oct 59	Jupiter C	1.17	70				TR
Explorer 10	25 Mar 61	Delta	37.	78				MF, PL
Injun 1	29 June 61	Piggy-back Thor-Able	1.16	40	2.8	256	10	TR
Explorer 12	15 Aug 61	Delta	13.1	83	13	220	26	TR, CR, MF, PL
Traac	15 Nov 61	Piggy-back Thor-Able	1.16	230	14	24.3	20	TR, Neutrons
Telestar 1	10 July 62	Delta	1.9	170			1.7	TR & Communications
Alouette	29 Sept 62	Thor-Agena	1.16	319				TR & Ionosphere
Explorer 14	2 Oct 62	Delta	16.5	89	13	220	26	TR, CR, MF, PL
Explorer 15	27 Oct 62	Delta	3.7	100	16	220	23	TR, MF
Relay 1	13 Dec 62	Delta	2.07	175			12	TR & Communications
Injun 3	13 Dec 62	Piggy-back Thor-Agena	1.43	115	4	256/1000	25	TR, VLF
Telestar 2	7 May 63	Delta					1.7	TR & Communications
1963-38C	28 Sept 63	Piggy-back Thor-Able	1.16	160	18	195	10	TR, CR
Explorer 18	26 Nov 63	Delta	31.	138	33	15	35	CR, MF, PL, TR
Relay 2	21 Jan 64	Delta	2.	175			12	
EOGO-A	4 Sept 64	Atlas-Agena	25.	1000	350	1000	150	TR, MF, PL, VLF, etc.
IMP-B	64	Delta	31.	138	38	15	35	CR, MF, PL, TR
EPE-D	64	Delta	5.	105	16	220	24	TR & MF
POGO	65	Thor-Agena	1.15	1000	350	4000	150	TR, MF, VLF, etc.
EOGO-B	65	Atlas-Agena	24.	1000	350	1000	150	TR, MF, PL, VLF, etc.
ATS (6000 N.M.)	66	Atlas-Agena	2.7	782	142	200	<30	not selected-not suitable for MF
ATS (24 hours)	67	Atlas-Agena	6.6	150	98	200	<25	not selected-not suitable for MF

CR-Cosmic Radiation, TR-Trapped Radiation, MF-Magnetic Field, PL-Plasma, VLF-Very Low Frequency emission

TABLE 3 - Table of Spacecraft Weights for Scout and DELTA 3C

Orbit Characteristics		Scout Vehicle		DELTA 3C Vehicle	
		38° Inclination	Polar	38° Inclination	Polar
Circular Orbits	400 N.M. Alt.	215 lbs	167 lbs	>800 lbs	650 lbs
	800 N.M. Alt.	100	70	460	420
ELLIPTICAL ORBITS* (APOGEE IN EARTH RADII GEOCENTRIC)	2	150	110	460	400
	4	80	50	220	200
	6	45		180	160
	8			140	140
	10			120	120

*For perigee altitudes of 200 N.M.

TABLE 4 - Small Standard Satellite Weight Summary

Item	Weight (lbs.) Minimum Version	Weight (lbs.) Maximum Version
Solar Cell Array	7.0	10.0
Solar Cell Regulator	0.3	0.3
Undervoltage Relay and Recycletimer	0.5	0.5
Battery Pack	4.0	6.0
Power Converter	1.0	2.0
Tape Recorder	3.0	3.0
Data Encoder	1.0	1.0
Telemetry Transmitter	1.5	1.5
Tracking Transmitter	0.7	0.7
Antenna and Hybrid	1.5	1.5
Command Receiver and Decoder	2.0	2.0
Structure	10.0	14.0
Thermal Control	1.0	2.0
Harness and Connectors	1.3	2.0
Experiments	10.6	44.1
Optical Aspect System	0.6	0.6
Data Conditioning Equipment and Clock	4.0	8.0
	50.	100.

TABLE 5 - Small Standard Satellite Power Summary

Item	Average Power (watts) Minimum Version	Maximum Version
Command Receiver	0.1	0.1
Tape Recorder	1.3	1.3
Data Encoder	*	*
Telemeter Transmitter	*	*
Tracking Transmitter	0.7	0.7
Experiments	4.5	7.8
Data Conditioning Equipment	3.0	4.0
Optical Aspect System	0.1	0.1
Power Loss in 75% Eff. Converters	3.3	5.0
Power to Charge Batteries**	3.0	5.0
Total Power from Solar Cell Array	16.0	24.0

*Included in battery charging power

**Assuming maximum values of 10% shadow time and 3% readout duty cycle.

APPENDIX B

OPTIONAL FORM NO. 10
MAY 1962 EDITION
GSA GEN. REG. NO. 27

UNITED STATES GOVERNMENT

Memorandum

TO : Mr. Gerald W. Longanecker
Systems Integration Branch
Spacecraft Integration and Sounding Rocket Division

DATE: January 26, 1966

FROM : Dr. Joseph V. Fedor
Mechanical Systems Branch
Spacecraft Integration and Sounding Rocket Division

SUBJECT: **ORIENTATION CONTROL FOR SMALL STANDARD SATELLITE**

It is the purpose of this memo to report the Mechanical Systems Branch's orientation control capability for the small standard satellite and thereby answer your memos dated December 16, 1965 and January 10, 1966. This memo is not intended to be all inclusive; we are reporting what we have examined in the course of our work. Most of our attention has been centered around orientation control for spinning satellites, gravity gradient orientation, and to a minor extent magnetic orientation.

For active systems that require thrusters, we have pioneered work with subliming solid fuels (ART development). Subliming fuels offer several advantages over the conventional cold gas (nitrogen) system. There is a 50% in weight, increased reliability (fewer components), increased specific impulse (I_{sp} = 85 sec compared to 75 sec), and long term storage capability. Because subliming fuels require heat (usually from the spacecraft environment or directly from the sun), temperatures much below 35°F when thrust is needed are undesirable. When thrust is not needed, there is no lower temperature limit.

Figures 1 through 6 show some spinning satellite orientation control systems with pointing accuracies, weight and power. Note that these systems are sun actuated (use sun sensor to obtain an error signal and to pulse thrusters). Average spin is about 20 rpm and it is felt that this can vary from about 5 rpm to 40 rpm for a specific mission. Roll moment of inertia is assumed to be the order of 5 slug ft.^{2} and thrust levels would range from a millipound to 10 millipounds. For systems I through

III it is felt that an orientation system would cost about 50 - 60 K (thrusters, sensors and electronics) per flight unit. It should be noted that it is not economically practical to buy subliming thrusters on a per unit basis. The thruster contractor would probably not give a satisfactory price for anything less than 100 K (we are getting two flight units and two engineering test models for this price for IMP-E). For system IV (Star Tracker) the cost would be around 75 K per flight unit. No cost is mentioned for system V (hybrid system) since it would really depend on pointing accuracies and reliability desired. It is included to show what is available.

Figure 7 shows two simple gravity gradient systems that have been considered for the small standard satellite. An add-on boom the order of 100 feet would be bolted to the basic structure. This boom would provide gravity gradient stabilization and libration damping (for the BEAM configuration). There are two choices for boom deployment -- self deployed and motor driven. The self deployed system is the simpler, lighter, and least expensive of the two, but the motor driven unit has the capability of permitting retraction as well as controlled deployment rates. It is estimated that the self deployed boom system would weigh about 3 lbs. and cost 4 - 6 K (per unit) with silver plating for thermal bending control. Based on taking advantage of RAE boom and mechanism development, a motor driven boom would weigh about 6 lbs. and cost 15 - 18 K per unit.

The MAGS system (Magnetically Anchored Gravity Satellite) has a General Electric type magnetic damper for libration damping. This damper weighs about 5 lbs. and would cost 30 - 40 K per unit. Weights in the BEAM configuration would be about 2 lbs. each. It should be noted that these configurations do not give orientation control about the nominal spinning satellite roll axis. For complete three axis control, more elaborate boom systems would be needed.

It should be noted further that some sort of magnetic system (passive or active) may be needed with the gravity gradient rods for a specific mission either to orient the spacecraft prior to boom deployment or as a back-up to the yo-yo system (spinning launch vehicle assumed). Also, analysis and computer work would have to be carried out for a specific configuration to insure stability and adequate damping of libration angles. Pointing accuracy toward the earth would be about 6 to 8 degrees.

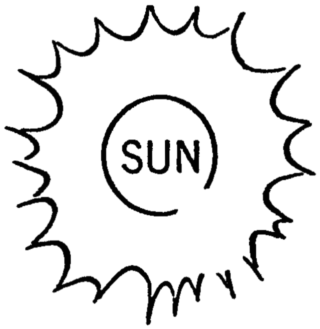
In regard to magnetic orientation systems, we have an interest in this area but manpower limitations have prevented us from developing this area. It is felt that in conjunction with, say, John Hopkins Applied Physics Laboratory, a magnetic orientation system could be specified and fabricated to meet a specific mission requirement. Our experience in this area has been that for a nominal year mission, an active magnetic orientation system is heavier (permeable material), requires more power (magnetometer and electronics), and requires ground base computer operations. It should be noted though, that for certain orientation requirements and if long life (3 to 5 years) is desired, magnetic systems are usually superior.

In summary, small, simple, light weight orientation systems are available for the small standard satellite whether it is spinning or gravity gradient stabilized. In-house capability encompasses thruster and gravity gradient control with limited magnetic capability. Remotely controlled gas bearings and ground support equipment are also available to thoroughly test out most orientation systems.

Joseph V. Fedor

Joseph V. Fedor

SYSTEM I : SUN POINTING



COARSE POINTING ~ 2°
WT ~ 3 LBS
POWER ~ 2 WATTS
FINE POINTING ~ 1/4°
WT ~ 4 LBS
POWER ~ 2 1/2 WATTS

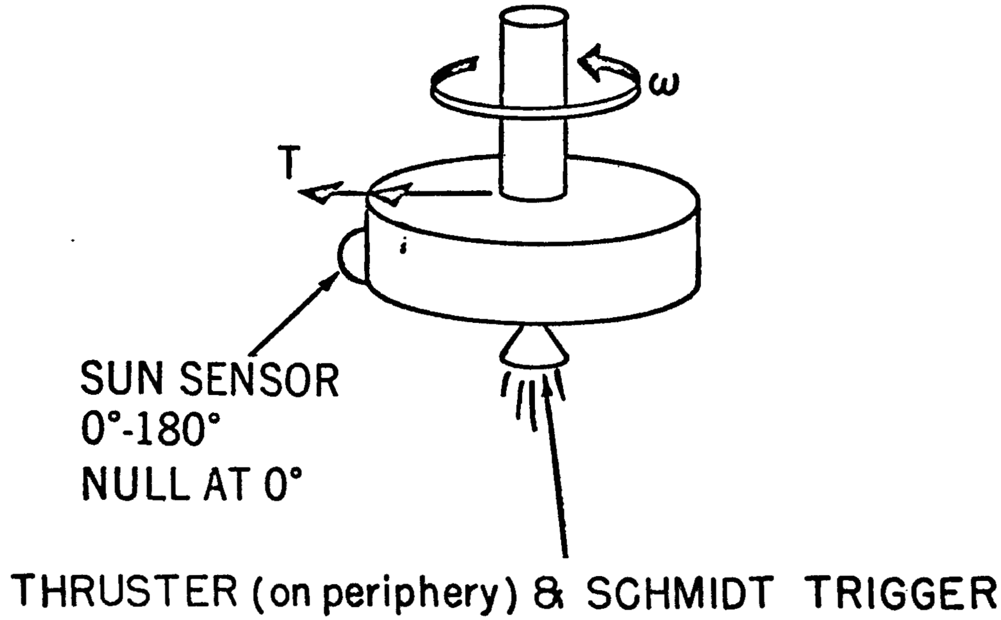


Figure B-1

SYSTEM II :
PERPENDICULAR TO THE SUN-SPACECRAFT LINE

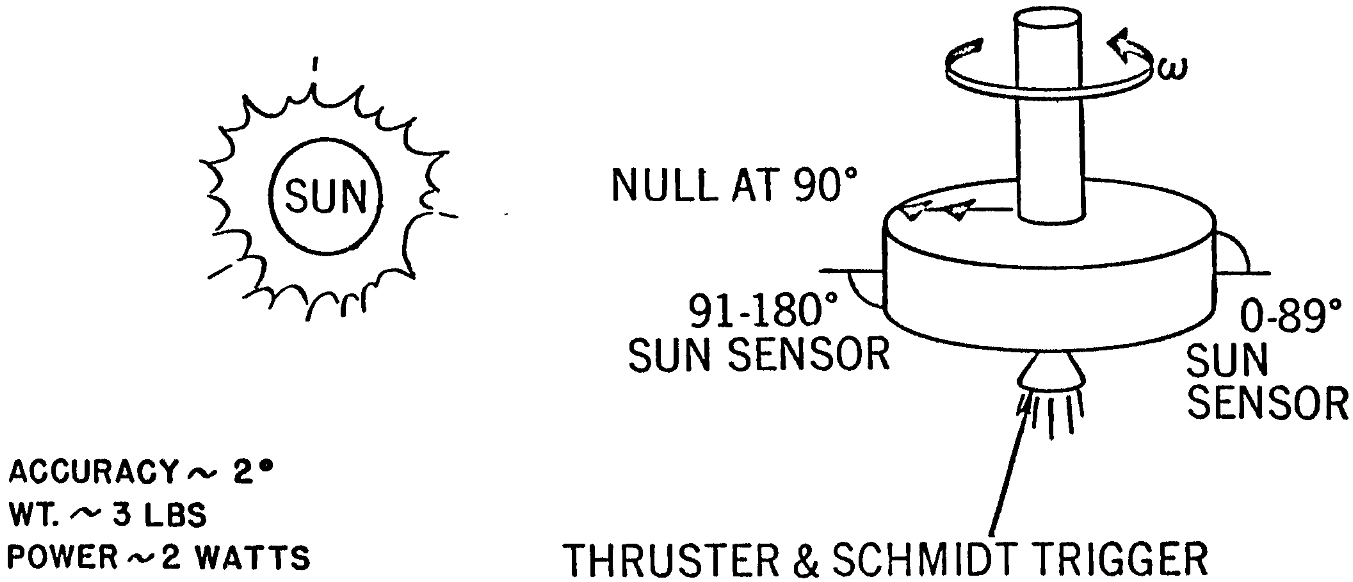
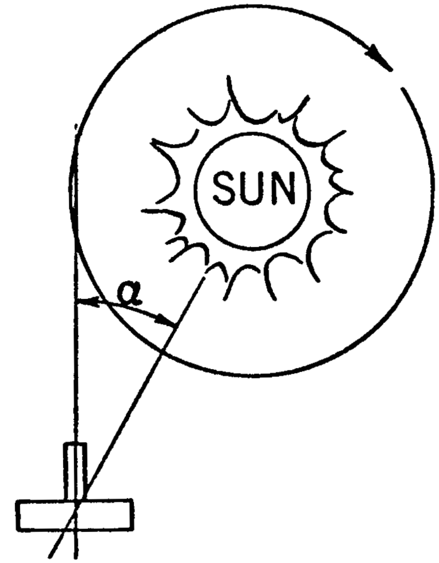
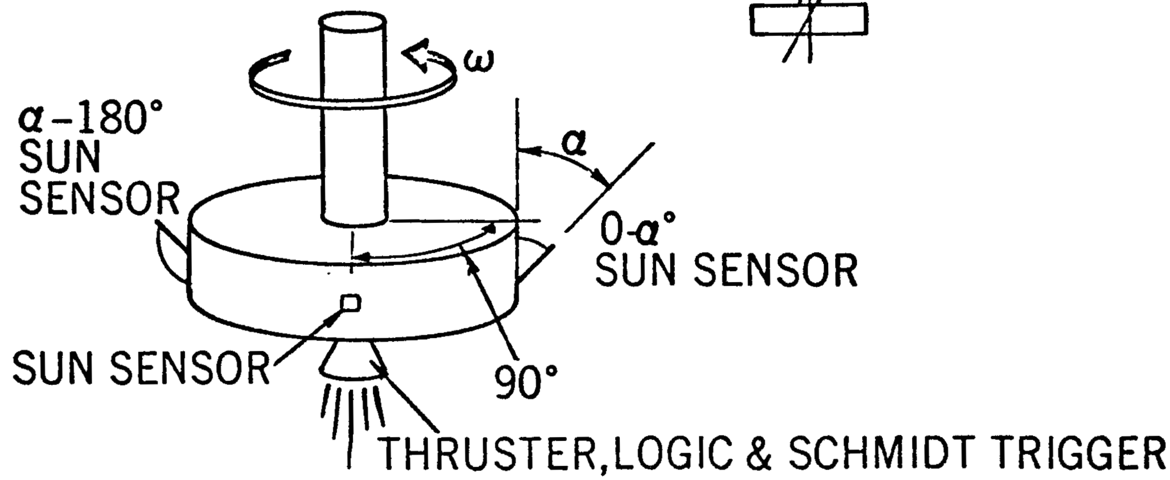


Figure B-2

SYSTEM III :
FIXED ANGLE PRECESSION ABOUT THE SUN





ACCURACY ~ 2°

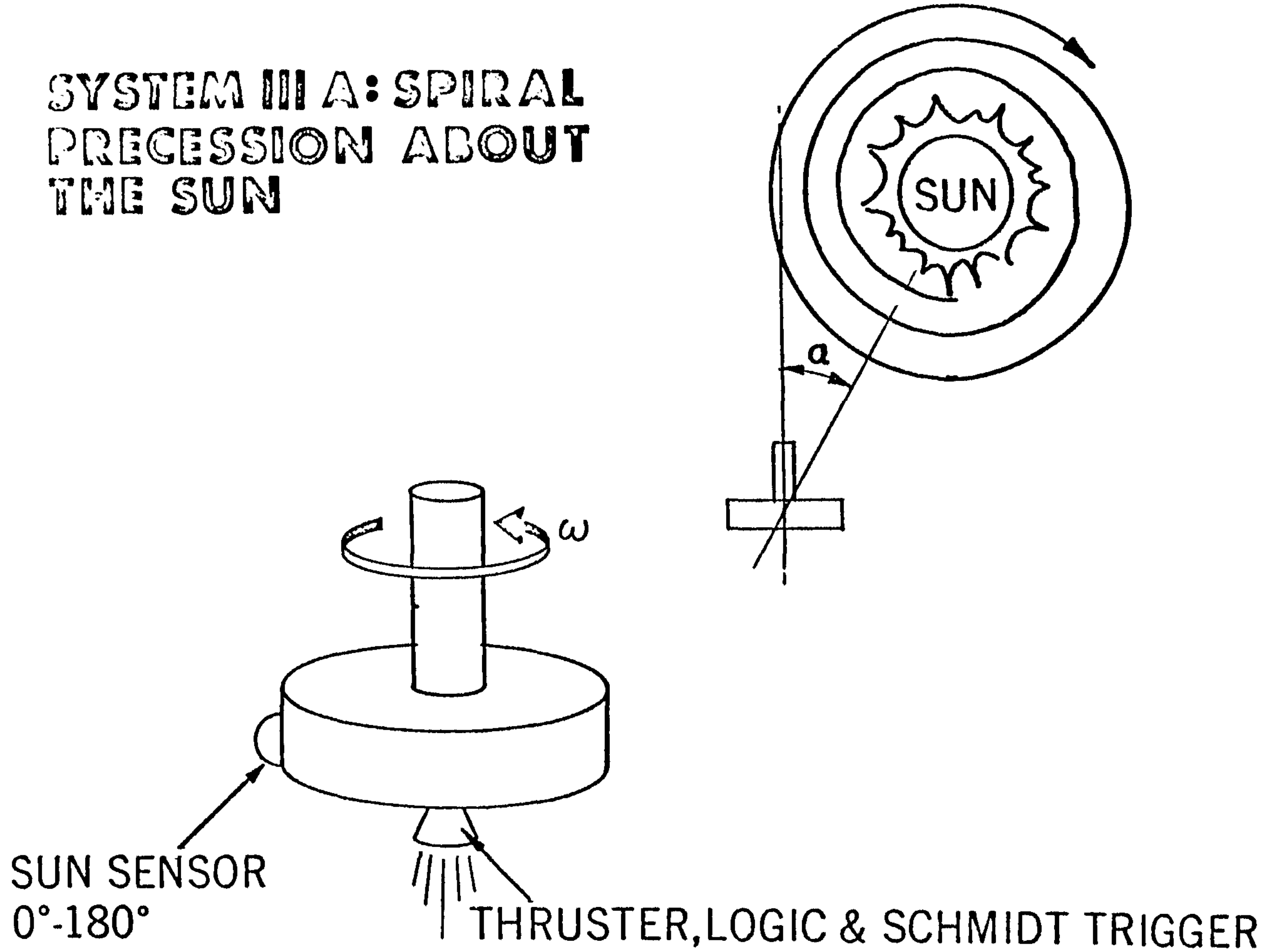
WT ~ 3-4 LBS DEPENDING ON THE NUMBER OF SWEEPS AROUND THE SUN

POWER ~ 2 WATTS

$\alpha \leq 45°$

Figure B-3

SYSTEM III A: SPIRAL PRECESSION ABOUT THE SUN



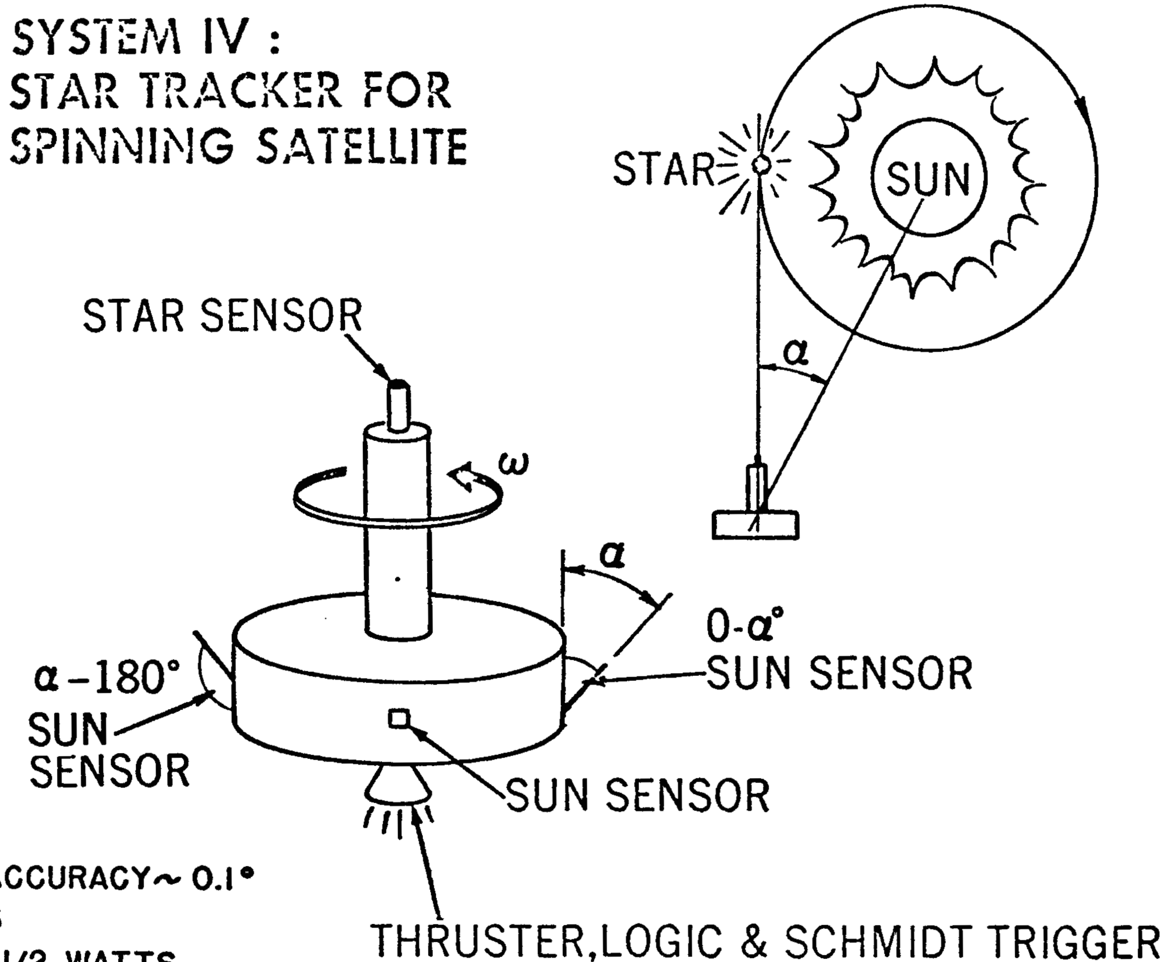
WT ~ 3-4 LBS DEPENDING ON THE
NUMBER OF SWEEPS AROUND THE SUN

POWER ~ 2 WATTS

MAX $\alpha \leq 45°$

Figure B-4

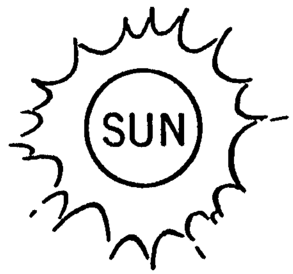
SYSTEM IV : STAR TRACKER FOR SPINNING SATELLITE

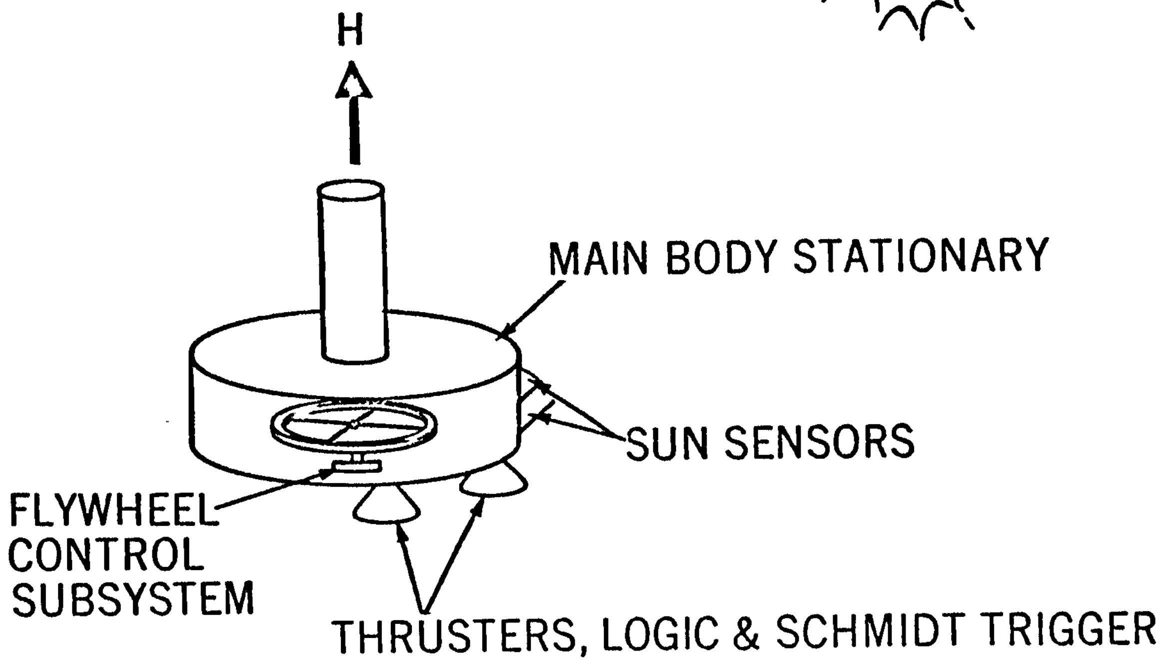


POINTING ACCURACY ~ 0.1°
WT ~ 4 LBS
POWER ~ 2 1/2 WATTS

Figure B-5

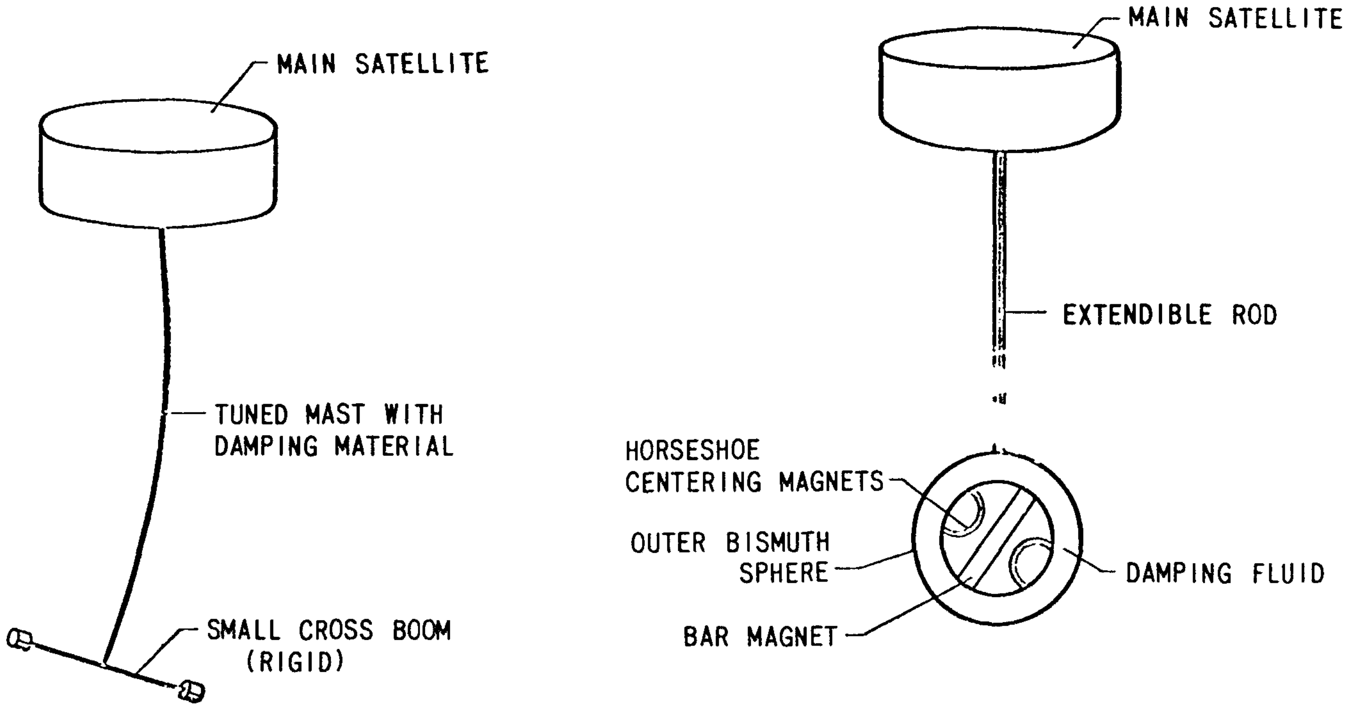
SYSTEM V : HYBRID
MAIN BODY STATIONARY; FLYWHEEL CONTAINS ANGULAR MOMENTUM





POINTING ACCURACY ~ 1°
WT ~ 5 LBS
POWER ~ 5 WATTS

Figure B-6



BEAM

MAGS

Figure B-7

APPENDIX C

OPTIONAL FORM NO. 10
MAY 1962 EDITION
GSA GEN. REG. NO. 27

UNITED STATES GOVERNMENT

Memorandum

TO : Gerald W. Longanecker
Code 722

DATE: January 26, 1966

FROM : Edward I. Powers
Code 713.1

SUBJECT: General Comments on Thermal Design of Small Standardized Satellites.

Summary

During the study phase of the "small standardized satellite" various types of orbits as well as stabilization systems are being considered. The comments contained in this memorandum briefly summarize the advantages and disadvantages associated with each of the above. The most concise approach has been to discuss each of the possible orbits and stabilization systems independently (except where considerable overlap applies). Thus, to be aware of the problems that may be encountered for a particular mission the appropriate sections pertaining to both orbit and stabilization must be reviewed.

Satellite Configuration and Tolerances

For the purpose of the study phase, the satellite is assumed to be nearly spherical in shape such that the projected area to sunlight is almost independent of solar aspect angle (angle between the solar vector and the axis of the spacecraft). A variation of 10% between minimum and maximum values can cause a 7°C change in temperature, for a uniform surface coating.

The surface of the spacecraft is covered almost entirely with solar cells. A portion of the remaining area may be used for thermal control. Although the thermal properties of the cells vary depending upon the manufacturer and the filter employed and must be checked for each application, nominal values of

$$\frac{\alpha \text{ (solar absorptance)}}{\epsilon \text{ (infrared emittance)}} = \frac{.80}{.84} = 0.95$$

were used for this study. Past experience has indicated that a tolerance in α/ϵ of ±5% should be considered in the analysis. This results in a mean temperature uncertainty of ±3.5°C.

Since the mission lifetime for most applications is one year, seasonal variations in the solar constant of 7% result in spacecraft temperature changes of almost 5°C.

If the above tolerances are assumed to be cumulative, the thermal design must be capable of absorbing ±10°C uncertainty in mean temperature. Since an overall temperature range of -10°C to +30°C has been suggested, roughly half of the working range is absorbed by the tolerances for lack of spherical symmetry, uncertainties in coating properties, and the seasonal variation in the solar constant.

General Comments on Orbits

Circular Polar-Near Earth

The thermal design for a polar near earth orbit must consider the extremes in orbital heat flux. The variation in total energy incident on the satellite between full sunlight and maximum eclipse orbits represents a mean temperature change of approximately 20°C. Since eclipse periods of less than 40 minutes are anticipated, problems associated with long cooldown periods do not apply. Internal equipment temperatures may fluctuate several degrees around a steady-state temperature. Boom or skin mounted experiments will undergo a more severe transient depending upon masses and physical locations.

Taking as an example the spherical spacecraft mentioned earlier, the mean temperature for the full sunlight orbit is approximately 20°C. The minimum sunlight condition would then produce a mean orbital temperature of 3°C. Imposing the ±10°C tolerance results in mean spacecraft temperatures from -7°C to 30°C. Internal temperature gradients due to power dissipation or orientation relative to the sun, and transient temperatures in eclipse orbits would increase the total temperature range for most components.

Circular High Altitude Orbits

The following comments apply to all of the high altitude circular orbits (i.e., polar, equatorial, synchronous).

The satellite is far enough from the earth such that heat inputs from the earth can normally be neglected. The design consists of analyzing the spacecraft for a continuous sunlight orientation. For the spherical geometry outlined above a mean temperature within the specified tolerances is generally not difficult to achieve.

Under this condition with no internal power the spacecraft temperature would be nearly 5°C. Imposing the possible temperature error of ±10°C, a range of -5°C to 15°C is reasonable.

An eclipse period of roughly one hour is encountered each orbit for the equatorial and synchronous orbits. The time in the earth's shadow when the orbit is polar depends upon the angle between the solar vector and the orbital plane, the maximum being approximately one hour. In both cases, however, the spacecraft must withstand the cooldown, which is estimated to be about 10°C for internal components.

As the altitude of the orbit is reduced for different missions, the time in eclipse decreases and becomes a significant portion of the orbital period, and earth emitted and reflected radiation become significant sources of energy. The limiting case is similar to that of the maximum eclipse orbit described in the previous section.

Eccentric Orbits

The comments on "Circular High Altitude Orbits" apply to eccentric orbits as well in that the nominal case consists of steady state sunlight.

A more critical condition, however, that must be considered is that of the long periods of eclipse when apogee is in the earth's shadow. Unlike those for circular orbits, several hours without sunlight is possible. One approach to the problem is to limit the launch window such that long eclipse periods are avoided for the prescribed duration of the mission. The maximum period of shadow acceptable from a thermal viewpoint must be established by thorough analysis.

If the launch window constraint is not feasible, it is possible to inhibit the cooldown rate of the internal components by insulating them from the surface panels with the aid of "superinsulation," properly mounted. Complications, however, may arise when the normal steady state condition is examined. Internal temperatures may be significantly higher due to component power generation. Large variations in temperature are also possible if the power level is not constant. In general, the feasibility of the design is no longer obvious and a certain loss in reliability may result.

During the long periods of eclipse the solar cells can experience very cold tempratures. Insulating the panels tends to accelerate the cooldown by isolating the panels from the warmer internal equipment. A 2-hour shade period could result in cell temperatures as low as - 100°C.

Circular Equatorial Near Earth Orbits

This type of orbit exposes the satellite to a near constant thermal environment. Eclipses range from 30 - 40% of the approximately 100 minute orbital period. A mean orbital temperature of approximately 5°C for the simplified geometry can be achieved. Imposing the variation due to coatings properties, solar constant, and projected area, the mean temperature may be -5°C to 15°C plus an additional allowance for temperature transients and temperature gradients.

General Comments on Stabilization Systems

The following text refers to magnetic, gravity gradient, and spin stabilized spacecraft.

Two severe thermal problems, high solar cell temperatures and large internal temperature gradients, can be introduced with any of these stabilization systems, when one side of the spacecraft faces the sun for relatively long periods of time. The gradient problem can be minimized by partially or fully insulating the surface panels from the experiment and equipment groups. The comments on insulated spacecraft in the section on "Eccentric Orbits" are directly applicable here although in that case insulation was mentioned as a possible method to inhibit cooldown rather than minimize a temperature gradient. The same disadvantages would also apply.

The solar cell panels when oriented toward the sun for extended periods may reach temperatures which seriously reduce the cell output voltage. The peak cell temperatures can be reduced somewhat by employing a low α/ϵ coating on "free" areas adjacent to the cells. The extent of this effect is illustrated by a simple example.

Solar cells with α/ϵ = .80/.84 = 0.95, normal to sunlight and viewing internal equipment at 30°C operate at 81°C. In order to reduce the average temperature to 70°C, 25% of the panel facing the sun must be reserved for a coating such as white paint. If the inner surface of the panel were insulated, the painted area would rise to 59% of the total panel area to reduce the temperature to 70°C.

The other method to alleviate the problem is to provide selective filters for the cells which reflect part of the normally absorbed thermal energy. If the α is lowered to 0.65, the required painted portion of the insulated panel becomes 44% for a 70°C maximum average temperature. The use of white paint to reduce maximum solar cell temperatures is effective only if there is good heat conduction between painted and solar cell areas. The use of either paint or filters to reduce solar cell temperature would also reduce the average temperature of the spacecraft, which may not be desirable.

Although the comments above are applicable to all of the stabilization systems listed, the practical extent varies considerably from system to system and orbit to orbit. The sun oriented spin-stabilized spacecraft would be effected for all possible orbits.

The problems are applicable to gravity gradient and magnetically stabilized satellites for all eccentric orbits and near earth ones when the angle between the solar vector and the orbital plane is near ninety degrees. When the angle is close to zero, both problems are probably not critical but the peak cell temperatures would have to be thoroughly analyzed. In terms of both of these systems it has been assumed that the satellite does not spin around an axis. In reality, a residual spin may exist which could minimize or eliminate the high internal gradients and possibly the excessive cell temperatures as well. The thermal design, however, must consider the non-spinning case until proven otherwise.

Spin Stabilized

The spin stabilized satellite with the axis nearly perpendicular to the sunline is the most desired attitude from the standpoint of thermal design. There is no danger in overheating the solar cells because the panels rotate in and out of sunlight. Similarly, the internal temperature gradients are expected to be small.

As the angle between the sun line and the axis is reduced, the solar panel in sunlight rises in temperature. The increase continues until a peak is reached when the angle is zero degrees. This situation and the associated problems have been described in the previous section.

The angle at which the high cell temperature and internal gradients commence cannot be established without a detailed analysis.

Edward I. Powers

Edward I. Powers
Temperature Control Section
Thermal Systems Branch

cc: Dr. J. Trainor
Dr. R. Hoffman
Dr. D. Williams
R. Kidwell
M. Schach
N. Hyman
R. Hoffman
A. Fitzkee
R. Wales

OPTIONAL FORM NO. 10
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GSA GEN. REG. NO. 27

APPENDIX D

UNITED STATES GOVERNMENT

Memorandum

TO : G. W. Longanecker - 724.8 DATE: 3/1/66

FROM : C. M. MacKenzie - 716.5

SUBJECT: Small Standard Satellite

A quick look has been taken at the problem of a power supply for a small standard satellite based on the data contained in your memo of 16 Dec 65 and our subsequent meeting.

The proposed type of satellite does not differ substantially from our work on the IMP and RAE types of satellites. Some thinking on the standardization of components on this type of satellite is being done, but work has not yet been started. A study program in this area would be a required effort in any serious consideration of a standard satellite.

The major factor preventing a single subsystem from accomplishing the multi-mission requirement is the solar array. The constraints on available surface area, multi-orientation possibilities, and the effect of the radiation belts combine to produce a horrendous analysis problem. A computer program is the only means for realistically determining the number of array designs necessary to meet all requirements. Such a program is not yet in existence.

Neglecting the solar array problem for the moment, the remainder of the power system design would follow the general concepts established for IMP and RAE. The use of silver cadmium cells for storage of energy would minimize magnetic effects from the battery. While the number of cells in series could probably be standardized for all missions, the size of the cell (capacity) is directly dependent upon such mission parameters as peak power demand, the duration of solar array eclipses, and the night power demand. Standardization in this area would entail a weight penalty for missions with short night power demands.

With a standard number of cells in series, the operating range of the unregulated bus voltage input to the converters or regulators becomes fixed. If the regulated output voltage requirements can be reduced to some minimal number of voltage levels (two or three), then it appears feasible that a small number of converter designs can cover all missions. With

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respect to converter frequency of operation vs. efficiency, there is only 1% or 2" variation in efficiency as the operating frequency is shifted from 1 kc to 10 kc. This variation is considered negligible when compared to that expected from the variations of the individual circuit components. As the frequency range is increased to 20 kc, about a 5% loss in efficiency can be expected. Even in this case, the overall efficiency will be equal to the 75% quoted in Davis' memo to Dr. Townsend (10 Oct 65).

Returning to the array problem, it has been stated that a computer study is needed to accurately study the problem. However, a first cut has been taken to show that it may be feasible to have a single array design for a multi-purpose satellite. Starting with the assumption that paddles will not be used, the amount of surface area for several typical geometric shapes has been calculated:

Sphere	2640 sq. in.
Right Cylinder Prism	3963 sq. in.
Right Octagonal Prism	3764 sq. in.

Each of the above figures fits into the envelope of a 29" diameter cylinder 29" high.

The sphere tends to be the ideal figure when considering a multi-attitude orientation; however, the loss of available area produces severe power limitations when compared to the other types of geometric designs. There are also many practical problems such as mounting provisions, low voltage output of the array, poor packing density, and a general handling problem that make this approach very unattractive. The difference between the remaining configurations is about 6%. The octagonal configuration is preferred because of the flat surfaces available for mounting solar cells.

With a solar constant of 130 watts per sq. ft., it will require about 400 sq. in. of sun-normal area to produce 24 watts of power at the end of one year (allowing for 30% radiation damage). Assuming a space utilization factor of .8 and a packing density of .8, the following equivalent normal areas are computed for the prisms:

	Cylinder	Octagon
Sun normal to top	422.7 sq. in.	380.8 sq. in.
Sun normal to side	538.4 sq. in.	495.6 sq. in.

Most of the above areas are greater than the minimum requirement of 400 sq. in. However, since the effects of the spin, the angle between the sun line and the plane of the solar cells, and the shadowing effects of booms, etc., have not been considered, the design may be marginal for many missions.

Due to the desire for a quasi-spherical shape, a computation on a truncated octagonal shape was made. The mid-portion of the proposed satellite is in the shape of an octagonal prism 12" high and 26.8" across flats. On each end is a hat approximately 8.5" high. The top of the hat is an octagonal area; the sides are planes joining the prism and the top of the hat. The slope of the planes is 45°. Using the stated assumptions of space utilization and packing density, the equivalent normal area of the top is about 380.8 sq. in., and for the side about 403.1 sq. in. The minimum area would provide about 21.8 watts at the end of a year.

In summation, the major problems in the design of the power supply for a small standardized satellite are:

1. The design of the solar array.

2. The ability to standardize the number and regulation requirements of the power supply output voltages.

It is recommended that a basic shape configuration of satellite be established, and then work with the computer programs be started so as to get more information on the solar array problem.

Charles M MacKenzie
Charles M. MacKenzie
Power Systems Design Section
Space Power Technology Branch

cc: W. R. Cherry
T. J. Hennigan
L. W. Slifer
F. C. Yagerhofer

3.

APPENDIX E

OPTIONAL FORM NO. 10
MAY 1962 EDITION
GSA GEN. REG. NO. 27

UNITED STATES GOVERNMENT

Memorandum

TO : G. W. Longanecker, Code 722 DATE: February 25, 1966

FROM : R. A. Hoffman, Code 611

SUBJECT: S^3 Feasibility Study: Aspect Determination

Attached is a copy of a document pertaining to aspect determination feasibility for the Small Standard Satellite (S^3) program. The contents have been read and approved by all those individuals listed under "Sources of Information".

R. A. Hoffman

Robert A. Hoffman

cc: J. H. Trainor
D. J. Williams
R. O. Wales

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Aspect Determination

The ground rule established for the operation of an aspect determination system is the ability to acquire uniquely on the celestial sphere the position of a specified axis on the satellite in a time short compared to the dwell time of the axis at a single position on the sphere. This rule immediately specifies that either of two sets of measurements be made:

(a) two body determination: two arcs be measured from two bodies, whose celestial coordinates are known, to the axis specified, and the phase angle from one known body to the other also be measured to eliminate the ambiguity between the two solutions attained. This determination is shown in Figure 1, where α_1 and α_2 are the two arcs, and ϕ_1 or ϕ_2 specifies which solution, S_1 or S_2, is real.

(b) three body determination: the phase angles be measured between three bodies whose celestial coordinates are known. This is shown in Figure 2.

The methods of stabilization are divided into two general categories for this discussion: those utilizing inertia (spin stabilized) and those having active attitude control (magnetic, gravity gradient, earth stabilized, sun pointing but not spinning).

In the former case the spin axis direction is usually quite stable with time, and the spin rate only very slowly changes, so the motions of all bodies relative to the spin follow simple equations, permitting accurate interpolations between all aspect sightings.

In the latter case, the specified axis is usually oscillating within a small angle about some locus which is slowly changing on the celestial sphere, so more absolute aspect determinations need be made per unit time than for the inertial system.

The effect of these two categories on the method of measuring attitude is the fact that the former provides a natural means of scanning by a detector a portion of the celestial sphere, while the latter does not.

1. Inertially stabilized spacecraft.

Case A. Two body determination.

Usually the two bodies involved are the sun and the earth. In general one arc can be acquired by measuring the angle between the satellite-sun line and the spin axis of the satellite, i.e., the latitude of the sun with respect to the satellite. Such a measurement also provides an index at the time of the sun sighting for measurements of the phase angle away from the plane defined by this arc, i.e., the longitude. Besides the original detectors of this type developed at GSFC and flown on the S-3 series, a standard detector is available for this measurement, a digital solar aspect detector, manufactured by Adcole Corporation. It provides a resolution of 1°/cos (latitude) and an ultimate accuracy of ±(1/4°)/cos (latitude) in latitude and ±1/2° in longitude, with a field of view of ±90° in latitude.

The other arc could be determined by narrow angled earth sensors which would discriminate the upper edge of the atmosphere

(~20 miles altitude). One such sensor would see the earth only over a portion of the orbit, so it would be desirable to fly several looking out at various latitudes, depending upon the particular orbit.

The resulting absolute aspect determination would be then obtained at least once each orbit to an accuracy to less than ± 1-1/2° for 90% of the time.

The specifications for the Adcole type detector and the earth telescopes depend largely upon the packaging scheme. Approximately:

weight, Adcole detectors with 180° field of view	300 gms
weight, each earth sensor	200 gms
power, Adcole	120 mw.
power, earth sensor	60 mw.

size, Adcole and earth sensors: 1" high IMP-type card with some overlay in latitude of outside shell of satellite.

bit rate per complete aspect determination:

sun-spin angle:	9 bits
roll period:	12 bits
time after clock mark of see sun:	12 bits
for the earth sensor sighting the earth:	
sun-earth phase angle:	12 bits
earth-spin axis angle:	12 bits
index of earth sensor:	2 bits
	59 bits

Spin rates:	about 4 to 40 RPM
Cost:	$20K per flight system.

The total bits per complete aspect determination could be read each roll of the satellite, which for commonly used roll periods, would result in some ten bits per second. It could also be programmed to be read only once per number of rolls or even per orbit, since spin axes are usually quite stable with time. Such resulting bit rates could be handled via the sub com.

This system utilizing Adcole detectors for 180° field of view and the earth sensors would probably satisfy the requirements of all the experimenters requesting spin stabilized vehicles, except where the desired spin rates are outside the specifications and for periods in the lifetime of the satellite or in special cases where the sun comes within about 10 degrees of the spin axis. In the last case the accuracy of the sun longitude measurement become rapidly worse. However, even when this sun-spin angle becomes small, the measurement of this angle can be made highly accurate for a stably spinning spacecraft by utilizing a sequence of observations over a period of several months. Angles as small as 3° have been measured.

Case B. Three body determination.

Another method for aspect determination obtains two or more arcs to the spin axis by measuring the angular separation about the spin axis between three or more known stars. The detector would telemeter both the relative magnitude of the stars and their angular separation. With a very rough knowledge of the spin direction from some other source (±30°), together with a predetermined star map, the spin direction can be uniquely determined.

The detector consists of a telescope, reticle, photodetector and signal processing electronics. The telescope is mounted in the satellite with the optical axis parallel or canted to the body spin axis. The reticle is opaque except for a single transparent fine radial slit centered in the focal plane of the telescope with the photodetector located behind the reticle. The signals generated will depend on the scanning disc rotational rate, slit opening angle, optical field of view, and the observed star pattern for the particular orbit of the satellite.

Stars of fourth and perhaps fifth magnitude or brighter can be utilized, depending upon the parameters of the detector. An output signal discriminator, with a level preferably commandable from the ground, would discriminate the lower limit of brightness necessary for stars observed during the particular orbit. The angle of canting, field of view, optics characteristics and photodetector would be determined by the particular mission.

In general the absolute aspect position would be determined to within ±0.5 degrees by simple manual means, to within ±0.1 degree by using a computer, and to ultimately within ±10 seconds with a highly developed version of the system.

The aspect problem of most spin stabilized missions could be met by this detector using 1" optics and a photomultiplier as the sensor. For a slit width of 0.1 degrees, an optical system of 75% efficiency, phototube dark current of 2×10^{-10} amps, and a quantum efficiency of 25% the signal would be at least 5σ over noise if one utilized stars of magnitude as small as 5. The current pulses would be larger than 10^{-9} amps, so a pulse height discriminator could provide star magnitude information. Such a detector would have the following approximate specifications:

size: (for a 4" long end window phototube) 8" long x 1 1/2" diameter plus a few cubic inches for electronics.

weight: 900 gms.

power: 1/2 watt

bit rate: if one would save data for the first five stars detected, with angular resolution to 0.1° and magnitude to the nearest 1/4:

16 bits/star x 5 stars = 80 bits.

This aspect system has not been constructed, although a breadboard model is under development by Control Data Corporation and a flight model is being considered for a spinning spacecraft. After prototype construction a flight model would cost about $25K.

Depending upon the particular orbit for a mission, two problems may be encountered in using the system over an entire orbit:

(1) Any photodetector would be sensitive to background particle and x-ray radiation.

(2) The detector would have to be protected from seeing the sun, earth and moon.

Based on past experience it appears advisable to select an aspect system capable of determining the complete aspect problem even for precessing satellites. The detector described in Case B does provide sufficient information to solve this problem. Otherwise it could be accomplished by having the detection capabilities on the spacecraft, but having programmable readout ability via ground command, which could be placed in effect as a failure mode of operation. The information from a magnetometer and even a single solar cell read twice per roll becomes most valuable for this case. The output of the latter detector follows the cosine law for the angle between the satellite sun line and the normal to the cell.

For the purposes of experiment operation and data analysis, two other pieces of information are highly desirable: phase measurement from some known direction, and a roll counter. The phase angle from the sun has been utilized in the past, but for some missions a phase angle from the spin-magnetic field plane could be used for synchronizing various detector operations on board the satellite. Either the magnetometer output or a solar sensor (when in sunlight) could be used to actuate a roll counter. Such auxiliary information requires the order of 30 bits/roll period, a large number when compared with the aspect bit rate from a stably spinning vehicle, which may be read only once every few minutes or less. However, this auxiliary information need not necessarily be read every roll period either.

2. Spacecraft With Active Attitude Control.

The complete solar aspect determination can be procured using five sensors of a modified version of the Adcole Digital Solar Aspect System.

The star aspect system would require a small motor to rotate the reticle.

Satellites whose orbit stays below about three earth radii can make use of the magnetic field angles measured by magnetometers.

Summary:

There appear to be systems either available or worthy of being developed which can provide aspect information to accuracies better than $\pm 1^\circ$ and frequently to about $\pm 0.1^\circ$ or $\pm 6'$. Even those currently available could possibly provide more accurate measurements if some further development work were done. Such systems will satisfy all missions except perhaps a few solar oriented, sun scanning spacecraft for which stabilization control and aspect constitute a primary problem in establishing the mission.

For tabulating the weight and power for a complete small standardized satellite, the following are some very approximate numbers which may be used pertaining to the aspect determination system which will probably suffice for many missions:

Weight:	1500 gms
Power:	650 mw
Size:	1 - 1" IMP-type card

Obviously the exact system for each mission will have to be selected from the subsystems available.

Sources of Information

Mr. James S. Albus, Code 711

Mr. E. J. Pyle, Code 711

Mr. Irving B. Lowen, Code 731

Mr. Leo R. Davis, Code 611

Lowen, Irving B., and Marvin S. Maxwell, "Scanning Celestial Attitude Determination System (SCADS): paper presented at the Twelfth East Coast Conference on Aerospace and Navigational Electronics; October 27-29, 1965, Baltimore, Maryland.

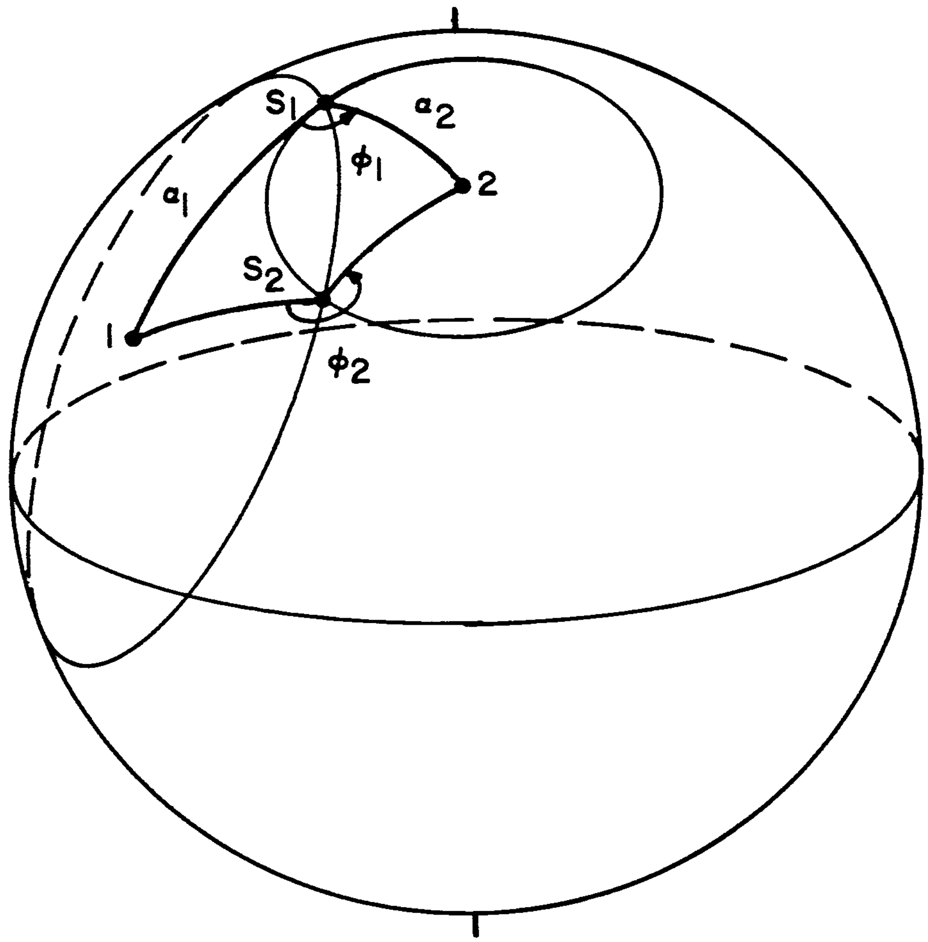

Figure E-1

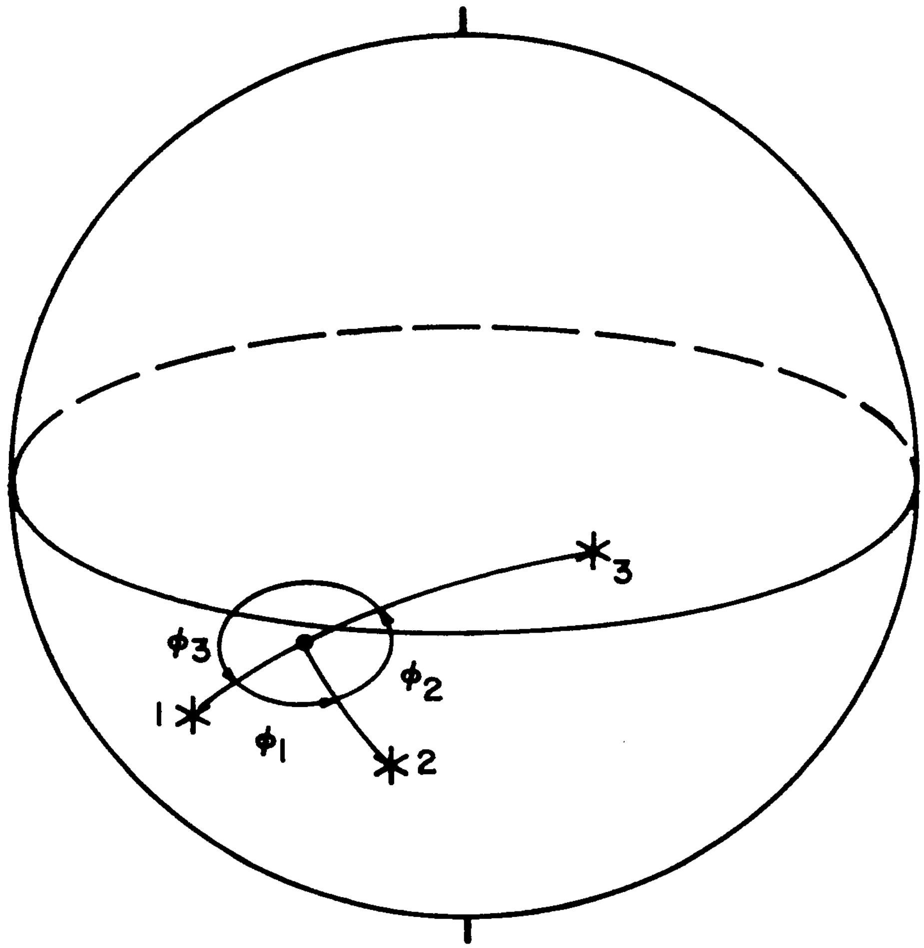

Figure E-2

APPENDIX F

OPTIONAL FORM NO. 10
MAY 1962 EDITION
GSA GEN. REG. NO. 27

UNITED STATES GOVERNMENT

Memorandum

TO : G. W. Longanecker DATE: Feb. 14, 1966

FROM : T. V. Saliga

SUBJECT: Small Standard Satellite Study Phase; Feasible Channel Encoding, Decoding, and Synchronization Methods

Ref: 1. "Comparison of Phase-Coherent and Non-Phase Coherent Code Communications," J. P. Strong, T. V. Saliga, X-711-65-425 October 1965 or TN 671

Ref: 2. W. Wesley Peterson, "Error-Correcting Codes", the M.I.T. Press, 1961.

Any coding and synchronization methods chosen for use in a flexible data system within a "standard" spacecraft series must be capable of operation

(a) with a large range of bit rates

(b) with data formats that are different from mission to mission

(c) within the constraints of the 136 mc TM band bandwidth and minitrack baseband width restrictions, and

(d) must be compatible with the RF modulation.

Since it is desirable to playback data at a high rate and power resources are very limited, coding schemes which can give a significant power gain are desirable. Because a flight tape recorder will introduce time jitter when data is recalled for transmission, consideration must be given to its deleterious effects, particularly when bandwidth-spreading channel coding is employed.

The results of this study indicate that it will be feasible to do error detection coding, to employ channel encoding which gives more than 2db. power gain, and to completely eliminate data time-jitter from the playback of the flight tape machine. The encoding functions may be accomplished with a single set of subsystems and the time-jitter reduction may be realized using a modular approach to accommodate a wide range of bit rates.

<u>The Coding Study</u>

There are a large class of codes available to the telemetry engineer, some of which are only efficient for checking the existence of errors;

others which can offer 1 to 5 db. power gain over an uncoded PCM system. All coding methods tend to increase the base bandwidth. The more the gain, the more the bandwidth required for a given type of code and a given information bit-rate. Because of the flexibility requirements and limited bandwidths available, it is not feasible to achieve a high coding gain for the SSS telemetry.

The codes specifically considered in this study were:

(a) Bi-orthogonal
(b) Orthogonal (Coherent and PFM)
(c) Maximum-Distance Binary
(d) Error-Correcting and Detecting

All of the above codes were compared with uncoded PCM and found to require excessive bandwidth for a reasonable coding gain with the exception of (c), the maximum distance codes. In addition, an error-detecting code appeared to be quite useful for data quality evaluation by the processor. These two types of coding can significantly improve the SSS data communications with surprisingly little cost in spacecraft hardware and power.

The (3,2) Maximum Distance Code

The nomenclature (3,2) used in the title implies that 2 information bits are coded into a binary code word consisting of 3 symbols. This code contains $2^2=4$ codewords in its "dictionary". The code structure is exceedingly simple. The first 2 symbols are the 2 information bits, the third is the modulo 2 sum of the information bits. When properly decoded, this system realizes more than a 2 db. power gain over uncoded PCM for a specified probability of error. Figure 1 plots probability of word error for this system and for uncoded PCM versus signal to noise ratio. (See Ref. 1 for definition of ST/(N/B)). At the lower bit error rates, the coding gain is 2.4 db. or a 1.73 power advantage. The base bandwidth is only increased by 50% over uncoded PCM for a given bit rate and the (3,2) code possesses self-synchronizing properties. This self-sync property is particularly valuable for the SSS since the encoder may be placed after the tape machine causing no reduction in data storage capacity.

Special decoding and synchronization procedures will be required in the ground data processing line. With the exception of a 4 correlator data decoder, the entire sync procedure can easily be accomplished with an ordinary PCM bit synchronizer and the on-line computer. The decoded data bits can be handled as in an ordinary uncoded PCM system. Frame sync search and lock procedures are unaltered.

A Frame Length Error Detecting Code

A problem facing the ground data processor is the determination of data quality. One simple parity check scheme, which gives a reasonable quality check, is to make an even or odd parity check and to include this check within each data word. An odd number of errors can be detected. However, with say a 9 bit per word PCM system, the inclusion of such a simple parity check extracts the rather exorbitant cost of using 11% of the available energy just for error detection. If this simple parity check were extended over a whole frame of data, then the probability of an even number of errors would become high enough to make the code of doubtful integrity. If the only source of data errors was the Gaussian channel noise, then such a simple code would still do an admirable job considering the small cost (1 bit per frame). Because SSS will use a flight tape recorder, another type of error will occur--tape data dropouts. Dropouts occur with no known statistical regularity and easily extend over 5 to 20 bits of data. A simple even or odd parity check would completely miss ½ the dropout errors that occur.

An efficient error detecting code which can detect up to 3 random errors and a relatively large number of dropout errors is the cylic Hamming Code (See Ref.1). This code also offers single-error correction capability if ever desired and can perform parity checks over any length frame. An encoder typical of what is being proposed is shown in Figure 1.

Parity checks are to be performed over a whole data frame (say 1024 bits) and the last 12 bits of the frame are reserved for parity. Thus, using only 1.2% of the available bits for parity, this code can detect any 3 random errors, any odd number of errors, any single dropout error up to 11 bits long and 99.95% of the dropout errors longer than 11 bits. When the data processor detects no errors as having occurred, he can be quite confident that none really did occur. The error detection can readily be done by the on-line computer. The only ground requirement, therefore, is the proper software.

Tape Machine Data Storage and Sync Considerations

Because of the degrading effects of data timing jitter, an experimental study was undertaken to determine its effects with a simulated flight tape recorder and a "Monitor" (Model 305) PCM bit synchronizer. The results of the study are summarized in Figure 3. The optimum loop bandwidth setting was found experimentally. By comparison of the "loss of lock" ST/(N/B) with the Figure 2 it is apparent that a larger system gain margin must be allowed for with "jittery" data and a stable data source is mandatory for use with the (3,2) coded PCM system.

Working in conjunction with P. T. Cole's section, a method has been devised for a combination servo'ed playback flight tape machine and a digital buffer which will be capable of clocking out PCM data with crystal clock time stability. An electronic simulation of the whole system has been made and its feasibility demonstrated. The development of a flight worthy system appears to be a most worthwhile goal for either uncoded or coded PCM systems.

System's Considerations and Reliability

A block diagram of the subsystems treated above is diagramed in Figure 4. The low bit-rate data from the encoder passes directly to the tape recorder with only a single digital gate in series. The parity-check contents of the encoder are shifted into the data stream through this gate at the end of each data frame. Hence, there is a very minimum amount of "in-line" hardware. If a failure should occur in the error-detecting encoder proper, only the parity checks would be affected.

Playback command of the tape recorder would be accomplished in normal fashion. The servo electronics should be an integral part of the recorder. Stable clock countdowns generated in the main TM encoder can serve as appropriate frequency references for the tape machine playback servo system and subsequent data handling circuits.

The digital re-timing buffer and channel encoder are being considered as one package since the (3,2) encoding system is not useable without "stable" data rates. With nearly all the internal components of this package being "in-line", serious thought must be given to reliability. The happy thing about the buffer and encoder package is that they can always be by-passed and jittery, uncoded PCM transmitted. However, to compensate for the coding and synchronization losses, the playback bit rate must be reduced by at least a factor of 2 to maintain the original gain margin. The block diagrams indicate a possible commandable by-pass method.

For SSS orbits which are always close-in, there may be such a large communications gain-margin available that the Buffer-Channel Encoder package would not significantly reduce transmitter power requirements or otherwise reduce the error rate. This package should therefore be considered optional depending on the available communications gain margin. The servo'ed tape machine and Error Detecting Encoder would remain the same in either case.

If the first SSS mission is a close-in mission, I would still like to include the Buffer-Encoder package as an "experiment" to prove its integrity for subsequent missions.

Memo to G. W. Longanecker
Subj: Small Standard Satellite Study Phase

Areas for Additional Study

The development of the airborne encoders should not present any unusual problems. However, additional study needs to be devoted to the synchronization procedures to be used with (3,2) code and to the development of a good servo system for the flight tape machine. A special servo design could reduce the required size of the buffer from, say, 256 memory elements to 32 or less.

Size, Weight, and Power Estimates

It is assumed that the lowest power integrated circuits available will be used for the majority of the electronics. A table below gives my estimates of power and size for the proposed subsystems. I have included an estimate for the servo electronics within the tape machine. This does not include the normal tape machine power requirements (motor, read-write electronics, etc.).

Item	Duty Cycle	Power min.	Power max.	Size-Weight
Error Detecting Encoder	Continuous	-	70mw.	Typical of 20 Digital I.C.'s
Servo Electronics	Playback only	-	80mw.	Typical of 25 I.C.'s and a 15 transistor discrete component circuit
Tape Buffer	Playback only	150mw.	600mw.	One IMP size package, 1-3/4 inch max. height.
(3,2) Channel Encoder	Playback only	-	65mw.	

Conclusions

This study indicates that for SSS missions where the communications gain-margin is low, a combination servo'ed tape machine, buffer, and channel encoder system could buy the system a 4 to 6 db. improvement. This includes a 2.4 db. coding gain and a 2 to 4 db. reduction in a normally built-in safety margin (which allowed for tape machine sync losses). The coding increases the base bandwidth 50% over uncoded PCM. Missions with an excess gain-margin could delete the Buffer-Encoder package entirely, and transmit uncoded PCM.

An error detecting encoder is proposed to enhance the estimation of data quality in the processing line. The costs to spacecraft reliability, bit efficiency, power, etc. are small.

Ground processing will require special proced~~ures but~~ it is nearly all implementable with software.

Thomas V. Saliga

Thomas V. Saliga
Flight Data Systems Branch

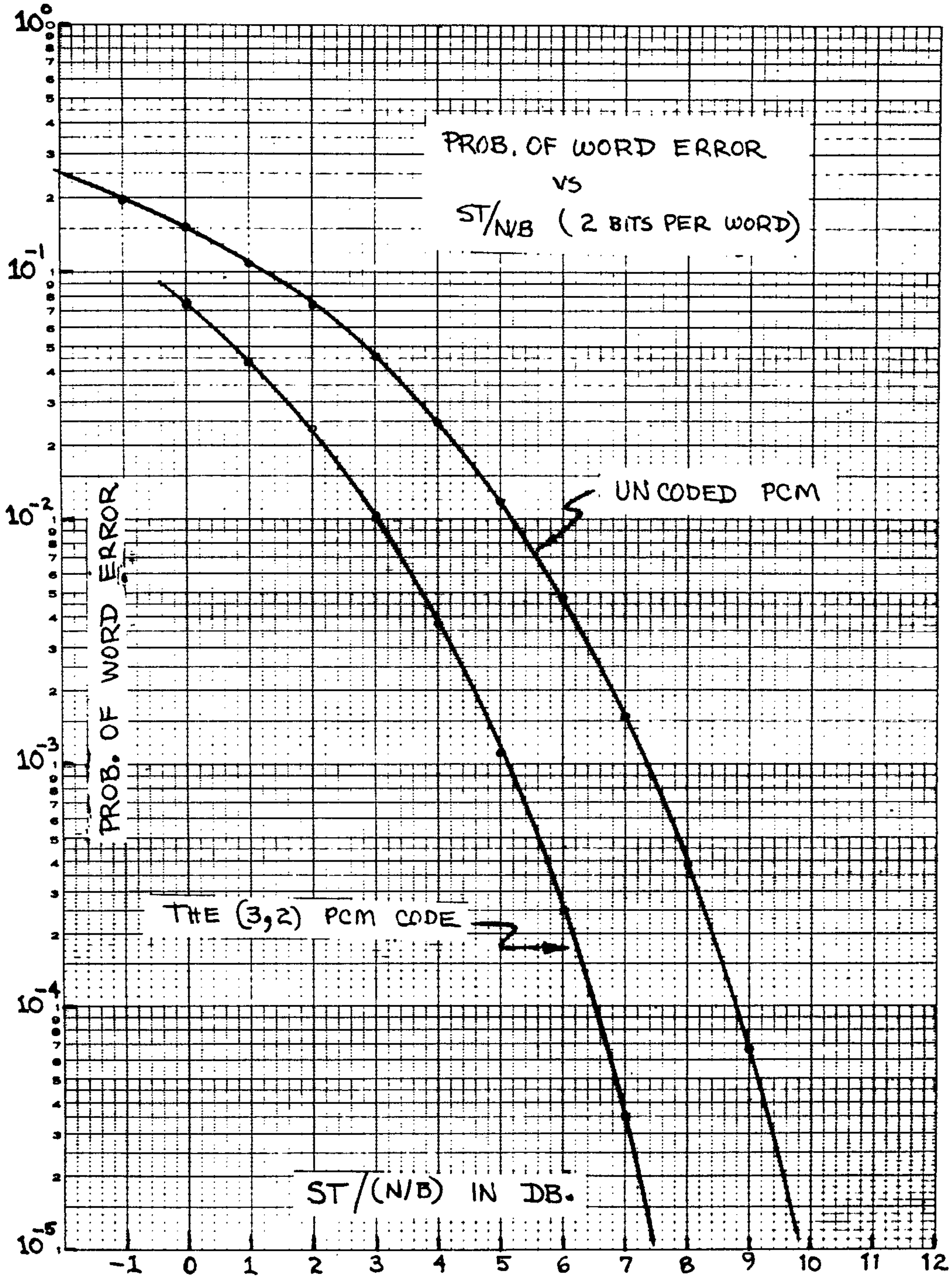
PROB. OF WORD ERROR
VS
ST/N/B (2 BITS PER WORD)
UNCODED PCM
THE (3,2) PCM CODE
PROB. OF WORD ERROR
ST/(N/B) IN DB.
10^0
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
-1 0 1 2 3 4 5 6 7 8 9 10 11 12

Figure F-1

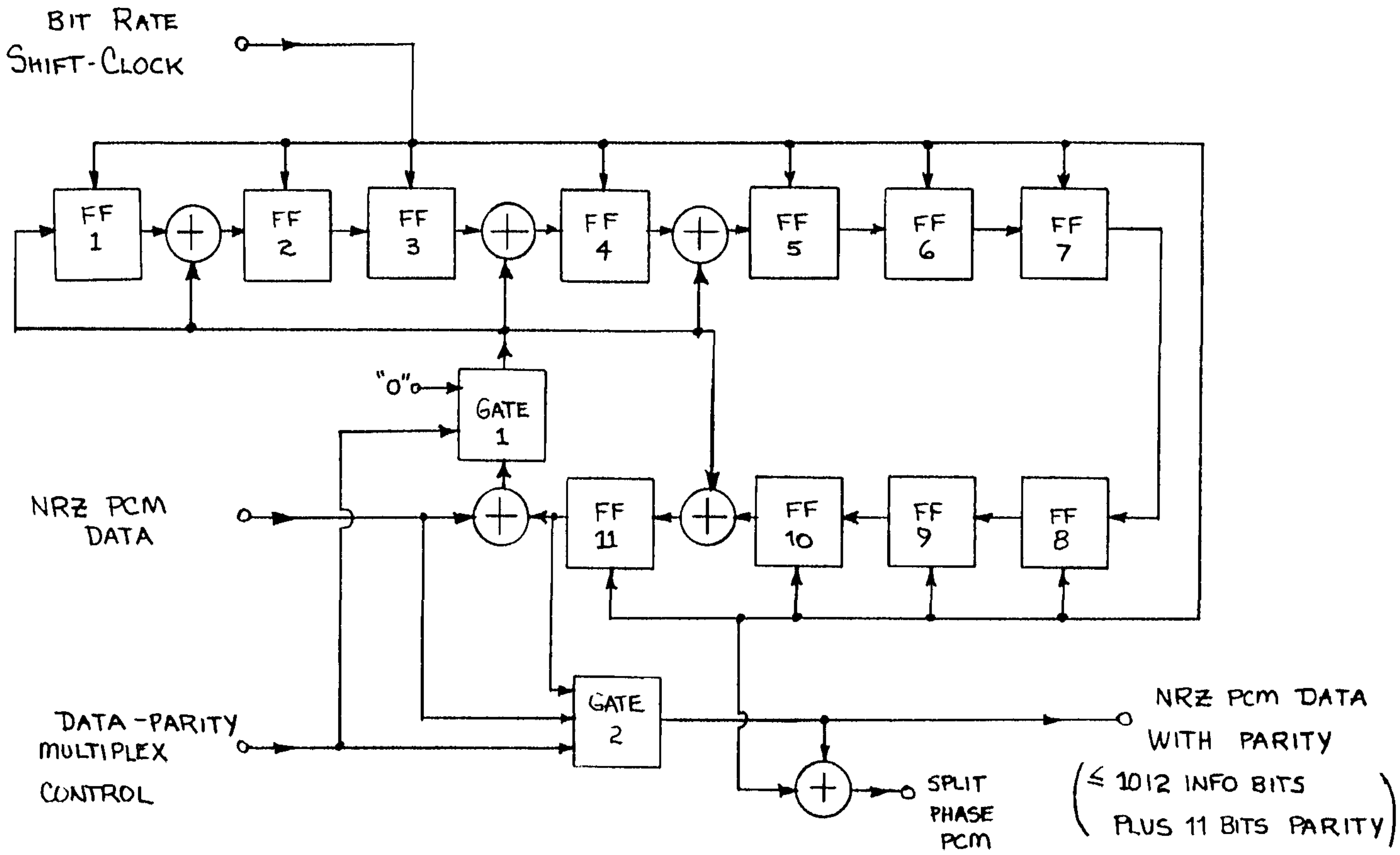

Figure F-2–Encoder for a Cyclic (1023, 1012) Hamming Code (Dist. = 4)

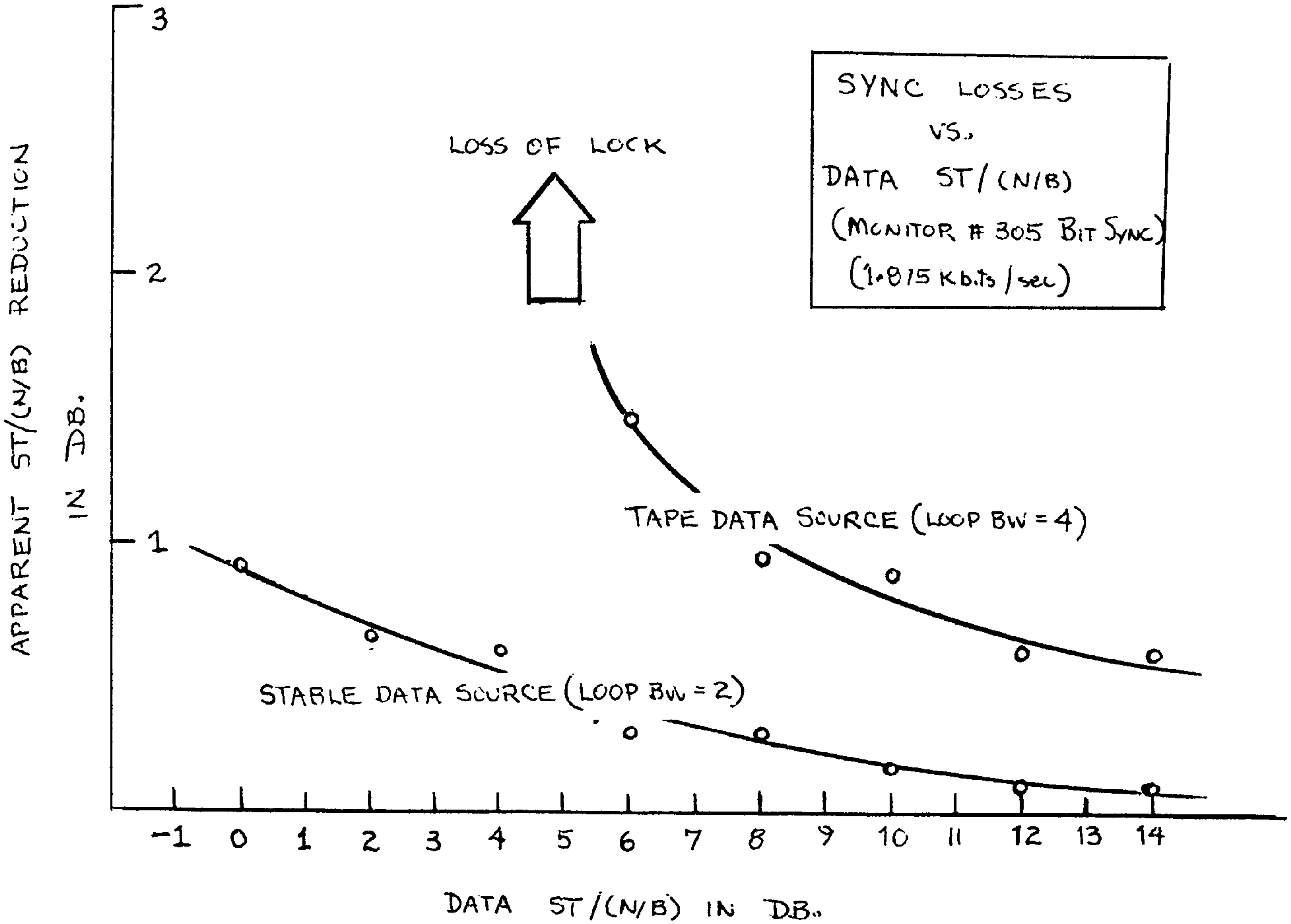
SYNC LOSSES
VS.
DATA ST/(N/B)
(MONITOR # 305 BIT SYNC)
(1.875 Kbits/sec)
LOSS OF LOCK
TAPE DATA SOURCE (LOOP BW = 4)
STABLE DATA SOURCE (LOOP BW = 2)
APPARENT ST/(N/B) REDUCTION IN DB.
DATA ST/(N/B) IN DB.
3
2
1
-1
0
1
2
3
4
5
6
7
8
9
10
11
12
13
14

Figure F-3

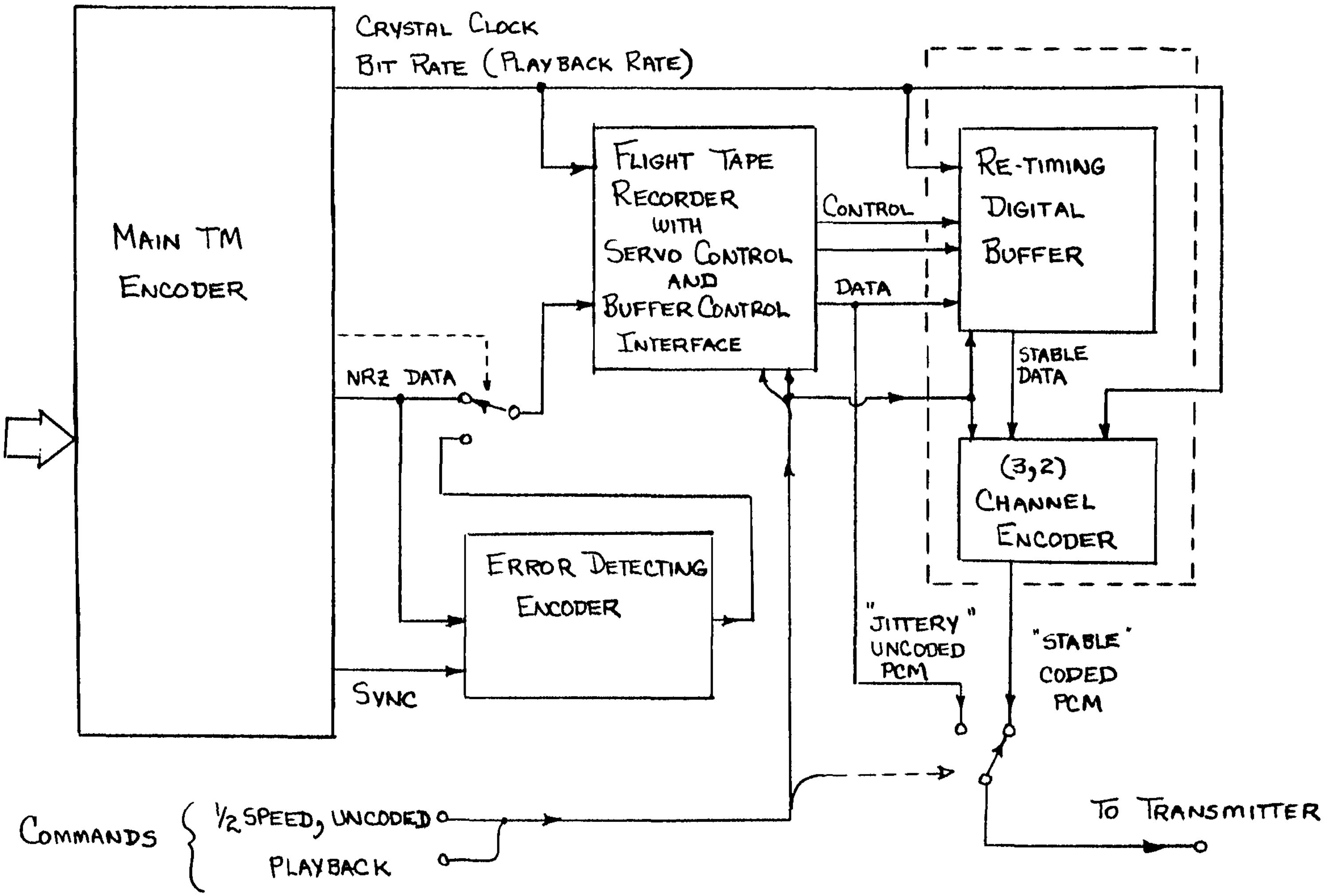

Figure F-4–Block Diagram of Telemetry Coding System

APPENDIX G

UNITED STATES GOVERNMENT

Memorandum

TO : G. W. Longanecker DATE: February 21, 1966

FROM : P. T. Cole

SUBJECT: SMALL STANDARD SATELLITE STUDY PHASE; MAGNETIC RECORDING CAPABILITIES.

The state-of-the-art for satellite magnetic tape recorders may be summarized in table form. However, since the capabilities of a recorder system are variable with maximum limits, more explanation is needed.

In-house studies for small standard satellites during calendar year 1965 have resulted in a new high packing density of 3000 bits per inch of tape using one track. Certainly a lesser packing density may be used; and in fact for conservative design considerations a less than maximum packing density that is commensurate with other mechanical parameters is desirable. For example, it would be advantageous to employ 100 feet of tape at 1500 bpi rather than 50 feet of tape at 3000 bpi, provided that the associated tape speeds are reasonable.

To fully meet the projected requirements of the "Small Standard Satellite" program, the following maximum parameters have been established:

Packing Density = 3000 bits per inch (single track)
Tape length = 300 feet
Reproduce to Record Speed Ratio = 50 to 1
Tape Speed = 15 ips

The enclosed curves show the record and playback bit rates related to time. Orbital bit rate storage capability can be extracted from Figures 1 and 2. From figure 3, the playback time and bit rate can be determined with the limitations that the playback time must never be less than 1/50 of the record time and/or less than four minutes.

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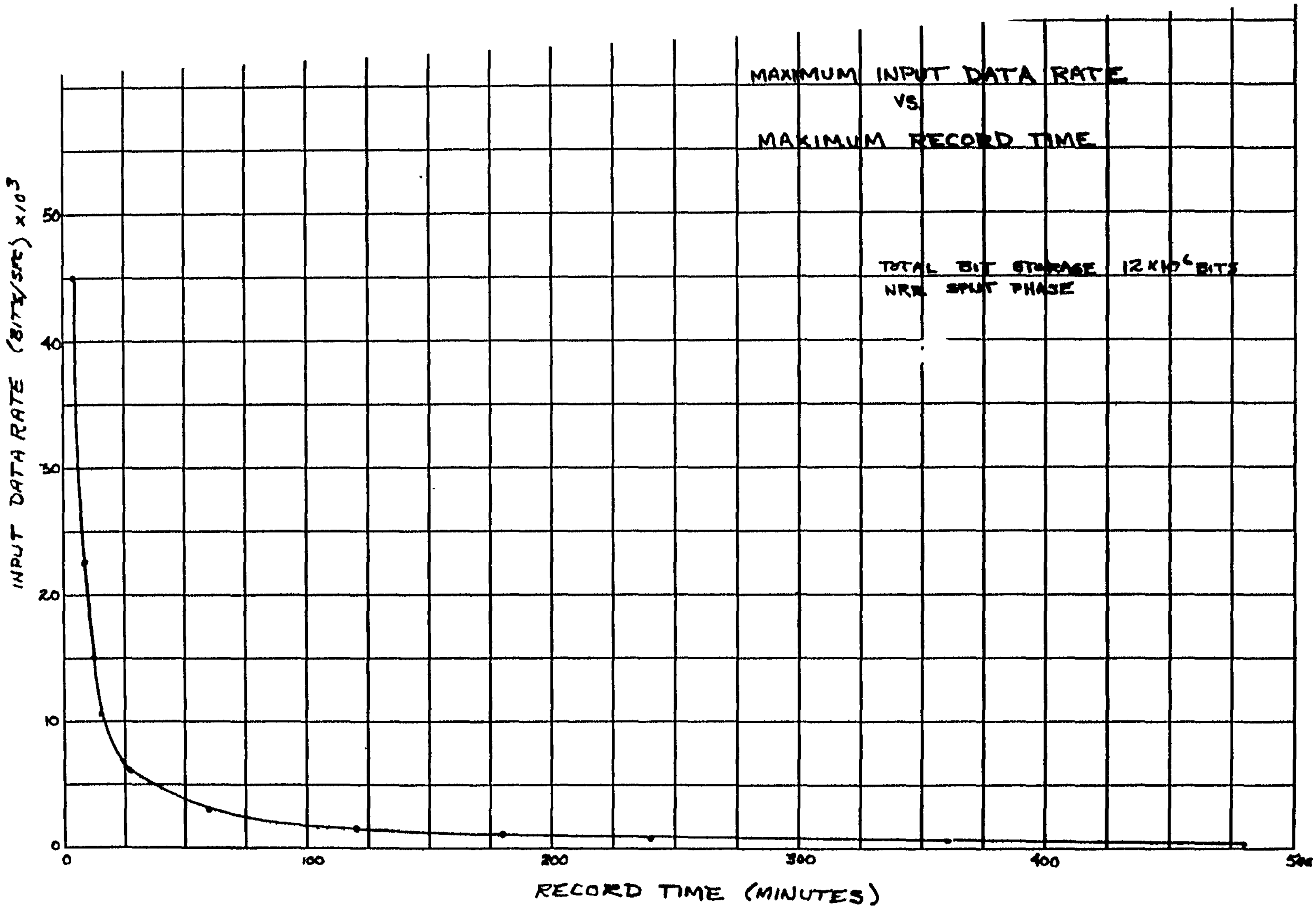
The mechanical configuration of the tape handling system will be the modularized form of endless-loop. The maximum weight and power requirements for the Small Standard Satellite recorder design will be approximately six (6) lbs. and consume 1. watt during record mode and 1.75 watts during playback mode.

For at least the first SSS system, a servo-controlled synchronous motor will be used to control the playback bit rate. A degree of bit rate control is required for compatibility with the post buffer storage concept proposed by Mr.T.V. Saliga.

In-house and out-house efforts will be continued to complete the adaptation of a servo-brushless D.C. motor to the recorder transport. This effort was late getting started in FY'66 due to funding delays. Design goals for the new motor approach are better efficiency, lower inherent tape recorder jitter and increased bandwidth of the servo control.

P. T. Cole

P. T. Cole, Head
Recording Techniques Section
Spacecraft Electronics Branch
Spacecraft Technology Division


MAXIMUM INPUT DATA RATE
VS.
MAXIMUM RECORD TIME
TOTAL BIT STORAGE 12 x 10^6 BITS
NRZ SPLIT PHASE
INPUT DATA RATE (BITS/SEC) x10^3
RECORD TIME (MINUTES)
0
10
20
30
40
50
100
200
300
400
500

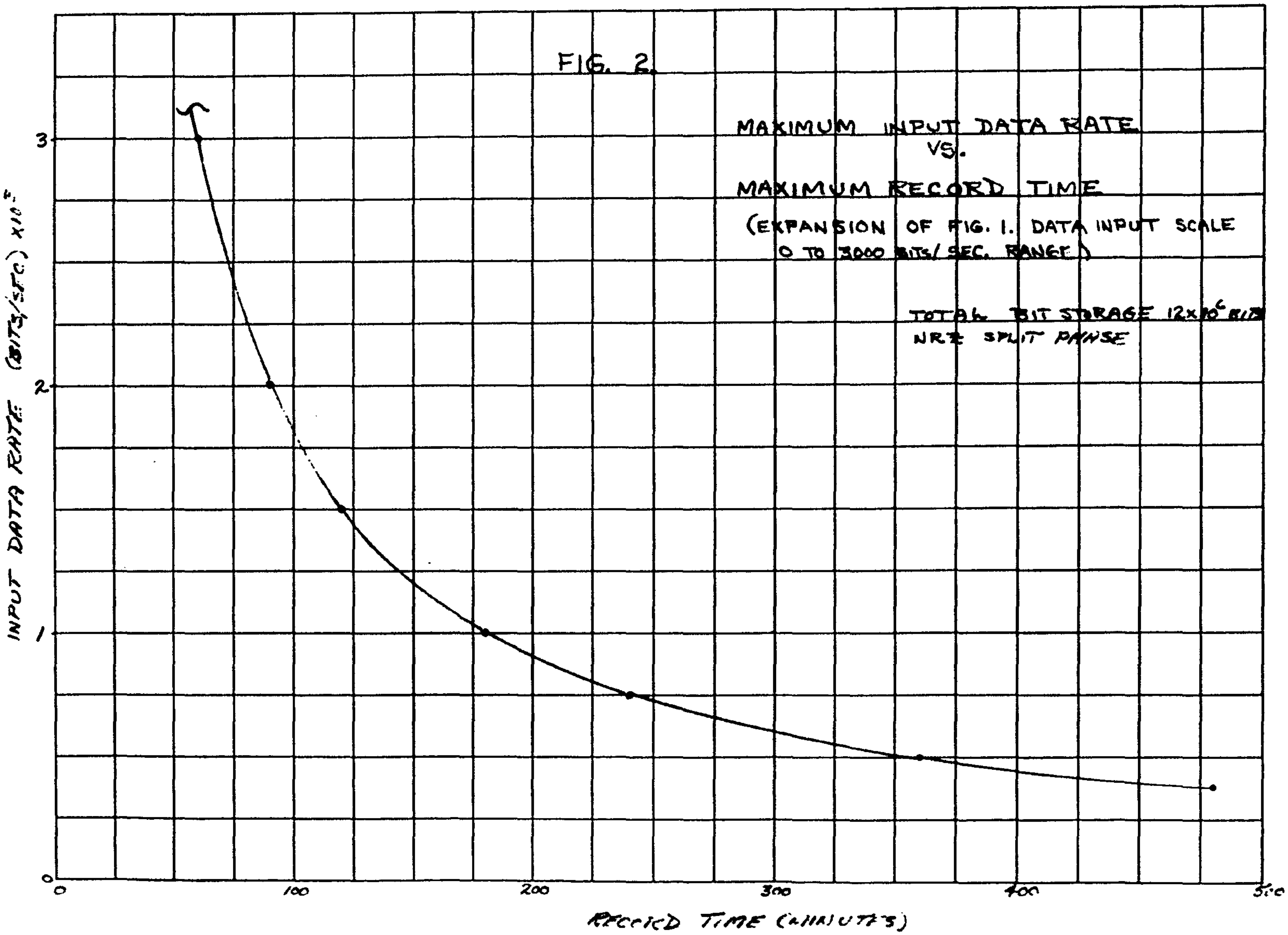


FIG. 2.
MAXIMUM INPUT DATA RATE
VS.
MAXIMUM RECORD TIME
(EXPANSION OF FIG. 1. DATA INPUT SCALE
0 TO 3000 BITS/SEC. RANGE)
TOTAL BIT STORAGE 12x10^6 BITS
NRZ SPLIT PHASE
INPUT DATA RATE (BITS/SEC.) x10^3
0
1
2
3
0
100
200
300
400
500
RECORD TIME (MINUTES)

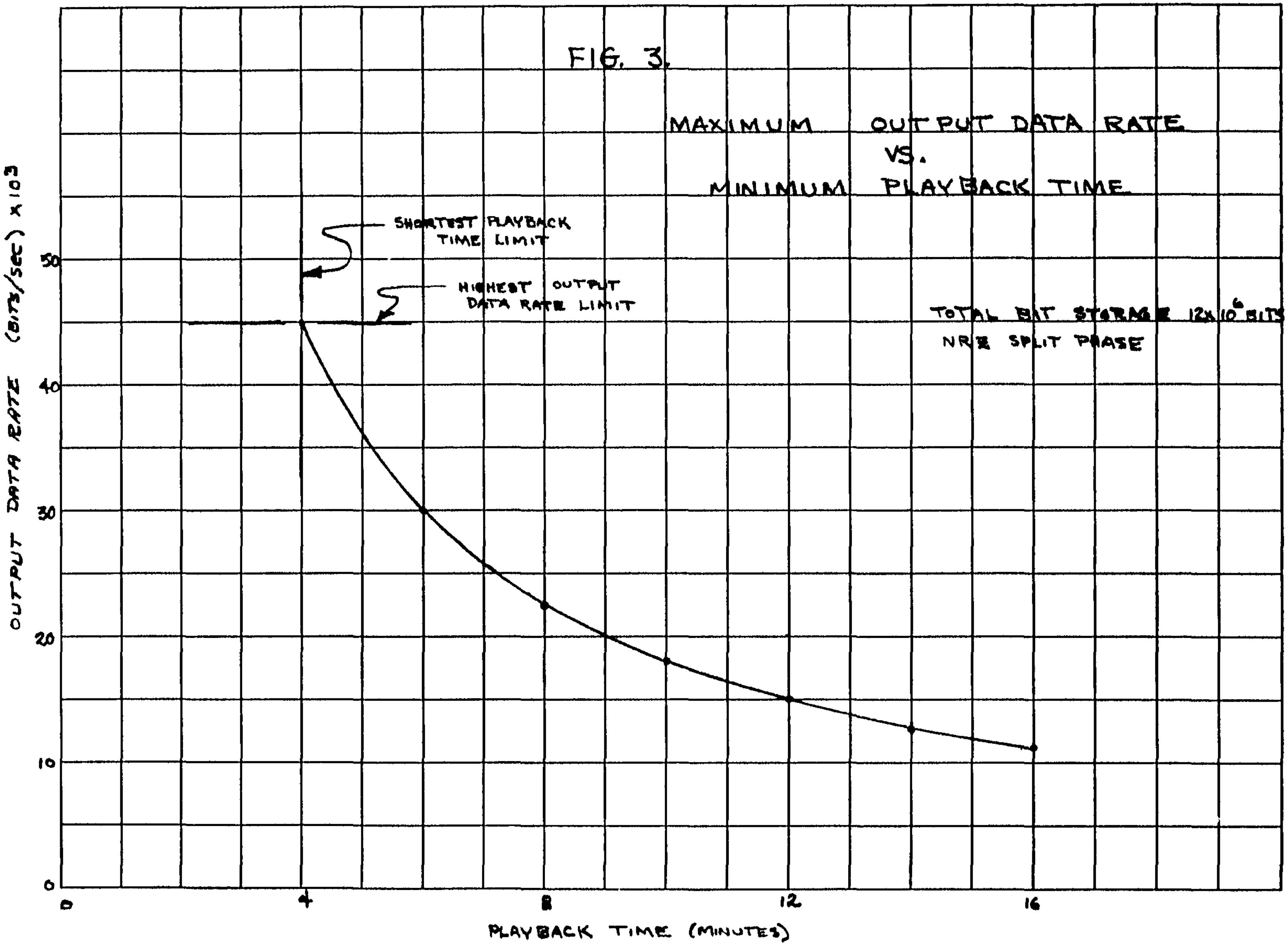


FIG. 3.
MAXIMUM OUTPUT DATA RATE
VS.
MINIMUM PLAYBACK TIME
SHORTEST PLAYBACK TIME LIMIT
HIGHEST OUTPUT DATA RATE LIMIT
TOTAL BIT STORAGE 12x10^6 BITS
NRZ SPLIT PHASE
OUTPUT DATA RATE (BITS/SEC) x 10^3
0
10
20
30
40
50
0
4
8
12
16
PLAYBACK TIME (MINUTES)

OPTIONAL FORM NO. 10
MAY 1962 EDITION
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APPENDIX H

UNITED STATES GOVERNMENT

Memorandum

TO : G. W. Longanecker, Project Manager
Small Standard Satellite

DATE: Feb. 28, 1966

FROM : A. B. Malinowski
Flight Data Systems Branch

SUBJECT: Data Processing System for the Small Standard Satellite

The enclosure is a write-up for the data processing system portion of the PDP on the Small Standard Satellite.

A. B. Malinowski
Telemetry Section
Flight Data Systems Branch

Enclosure

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Data Processing System

The data processing system can best be described by looking at some of the functions to be performed on the incoming data from the experimenter's sensors. Figure 1 shows a list of these functions suggested by the experimenters. Many of these same suggestions were implicit in the answers of some experimenters. The more important ones can be considered to fall into six broad categories as follows:

1. High time resolution sampling.
2. Tape recorder system functions.
3. Data collection; non-synchronous with telemetry rates.
4. Programmable telemetry format.
5. Particle counting and compression.
6. A/D converters (7 to 10 bit accuracies).

In addition, the following more specific data system functions were mentioned:

1. Data summary or averaging.
2. Simultaneous sampling of multiple outputs.
3. VLF data handling.
4. Capability of doing pulse height analysis.

A data processing system that will provide the above functions will require, in addition to the normal components found in telemetry systems, the use of the following:

1. Programmable telemetry format memory (with a N% in-flight write-in capability).
2. Data buffer memory.
3. Tape recorder system with start-stop and speed-up capabilities.

The programmable telemetry format will have a 100% write-in capability on the ground but only a small fraction will be changeable in-flight (N%). This condition is due to the fact that the write-in circuitry will be contained in the ground support equipment, except for the small fraction to be included in the flight system to provide the in-flight write-in capabilities.

It is intended that these components (plus others) be developed as standard modules easily integrated into the particular data system, unique to each spacecraft mission. It is felt that a general data processing system developed as a standard unit could not efficiently utilize the available space, weight, and power. A general data processing system suggests that excess processing capacity would be flown on each mission, a luxury that can be little afforded on small spacecraft. On the other hand small satellites generally have the greatest need for processing because of the restrictions of power, and therefore bandwidth.

Therefore, to satisfy the inherent need for data processing on the small satellite and provide the flexibility required for different missions the data system will be developed along a modular concept whereby a series of standard modules may be easily integrated to suit the needs of a particular data system. This modularity will give the experimenter the flexibility of trading data handling capability for additional experiment weight and power--a trade off that is highly desirable for a small satellite that is to satisfy a large number of different missions. The following paragraphs describe briefly some of the data system function characteristics.

The high time resolution sampling requires a data buffer memory to store the incoming high data rate which is read out later at the telemetry rate. In a fixed telemetry bit rate system the high density sampling is done on a fixed duty cycle basis.

Often, however, the high density sampling would be required for longer periods of time when the satellite experiences a sudden commencement in data activity. Examples of sudden bursts of activity are:

1. Passing through the auroral zone.
2. Passing through the magnetospheric boundary.
3. Solar radio bursts.
4. VLF emissions - whistlers.
5. Magnetic storms.

In this case the small data buffer memory could not handle the increased activity. In this situation a speed-up of tape recorder record rate would be required to handle the increased bit rate. Upon passage of the sudden activity the tape recorder would revert to its normal operating mode. It should be mentioned that present tape recorder drive capabilities limit the speed-up factor to approximately six (a maximum figure).

On a spin stabilized spacecraft much of the experiment output data is spin modulated due to the directionality of the physical process, and in general results in a case where the telemetry rate does not match the data rate. It may be preferable to sample in synchronism with the spin rate, removing the spin modulation. In this manner only those points which are of use to the experimenter would be transmitted. This situation is accomplished through use of a buffer memory which collects data in synchronism with the spin rate and reads it out in sync with the telemetry rate.

Sampling of data in the manner in which the data is being presented is a prerequisite to further data processing. For example, in the case of the spinning satellite, particle counting can be done over selected angular sectors of the satellite azimuth. Multiple readings can be collected from the identical sector and then sent to telemetry as a sum of several readings. Arithmetical functions such as this example can easily be carried out.

The programmable telemetry format concept allows the data system designer to complete his design early in the program and still allow the experimenter last minute changes in his interface with the data system. This situation should allow considerable reduction in lead times and program slow-downs due to a change in the experimenter's plans.

Format Standardization

In order that all spacecraft missions be capable of being processed through the same ground data processing system a standard telemetry format will be set up. The only changes from mission to mission should be in the computer software area. A proposed format is shown in Figure 2. The numbers given here serve to indicate the order of complexity and are subject to change as more work is done on the format organization. The format consists of a sequence (i.e., one complete subcommutator sequence) of 128 major frames. Each major frame is subdivided into 8 minor frames. The minor frame is 32 channels long, each channel being one (4 bit) byte. The format will contain such non-data parameters as: frame synchronization, format identification, sequence and frame counter, frame parity check, S/C configuration identification, and performance parameters. The channel byte concept allows words of various lengths to be handled more efficiently. A byte size of 4 bits allows data words to be fitted into one of 4 possible lengths: 4, 8, 12, or 16 (thus requiring 2 bits to describe the length of the word). Investigation is continuing in determining the optimum byte size but at present 4 seems ideal, it being binaryly related, and easily handled in the data system. The sequence length of 128 frames is made long to accommodate events (such as a read out of buffer stored data) occurring over long periods of time. For example, at a telemetry bit rate of 200 bits/sec. one sequence interval is approx. 65 secs. This technique is done to simplify ground data processing. In this case each data point in a sequence of 256 channels x 128 frames is uniquely identifiable with one and only one data source and this condition exists from one sequence to the next.

Data Processing System Configuration

Figure 3 shows a block diagram of a modularized data processing system for the small standard spacecraft. The drawing is intended to show the major components, some of the auxiliary components, and their interrelations. The diagonal line in the upper right hand corner of some blocks indicates a component to be developed as a standard module. The important components of this system are the following: programmable format memory, data buffer memory, servo control tape recorder, and data encoding.

Figure 4 contains an example of how a typical data system would function. In synchronism with the telemetry channel rate, words are sequentially read from the programmable format memory. These data words instruct, (1) which experiment output (or non-data words such as synch, or data words from the buffer memory) is to be sampled, (2) how long the word will be (no. of bytes) and, (3) where the sample word is to be routed (e.g., to telemetry, to the data buffer, from the data buffer to telemetry, etc.). The experiment output is selected by the M bits of the instruction word. If M were equal to 6 binary bits, the possibility of selecting one of 64 possible outputs would exist, and each combination of the 6 bits would constitute an address for a particular experiment output. All of the addressable switches are contained in the multiplexer section of the data system. The word size is determined by the J bits. Making J equal to two binary bits allows data words of four different lengths to be handled. In the example of Figure 4, memory word No.1 is read out at channel 1. In this case a synch word (4 bytes in length) is to be inserted into the first 4 channels of telemetry (one byte per channel). It is noted that less memory words than telemetry channels will be required, i.e., approximately 128 memory words will be required in a 256 chanel frame. This condition exists due to the fact that data word lengths will, on the average, be equal to 8 bits (2 four bit bytes).

Except for the data collection synchronizer, all timing functions are in synch with the main clock and timing functions. As an example of non-synchronous data collection consider sampling in synchronism with the satellite spin rate. In this case a sampling sequence is initiated by a command from the programmable format and a set of readings are loaded into the data buffer memory. The previous set of readings are concurrently read out of the data buffer into the telemetry format.

Performance Parameters

Normal performance parameter functions will be included in the data system. Some of these are: temperature and system voltage monitoring, satellite aspect information, various charge and discharge currents, calibration of A/O converter, S/C configuration or flag identification data. Additional functions will be implemented to check for proper operation of various components especially the programmable format and data buffer memories. These self-check operations will provide valuable information (to help localize faulty operating equipment) when discrepancies arise during S/C integration.

Standard Components

The following components of the data system will be developed as standard modules capable of incorporation into any data system for a specific mission:

1. Programmable format memory
2. Data buffer memory
3. A/D converter (10 bit)
4. S/C spin synchronizer
5. Word compressor
6. Particle counting accumulators
7. Adder module

The tape recorder system and encoding techniques are discussed elsewhere in this report.

It is intended that these modules be completely specified in their electrical and physical parameters so as to be easily reproduced.

The buffer memory will be a magnetic core memory organized on a 4 bit word basis with a total of 8192 bits. It will contain all of the necessary read and write electronics but input/output control will have to be handled in a different module. This module will be quite different from one data system to the next.

The programmable format memory will be a NDRO memory (non-destructive read out) capable of being altered by electrical signals through a separate memory access plug not associated with the normal system operation. The memory will be capable of quick and easy program changes at any time during S/C design and integration operations. The possibility of reserving 10% of the total format for writing in program changes while in-flight is still being studied both from a reliability aspect and whether, in fact, it would prove useful if this capability existed.

The size of the memory is anticipated to be in the order of 128 words of 14 bits each. This length of instruction word will make provisions for handling 64 prime monitor points, data word lengths of 4, 8, 12, or 16 bits, and routing of data words (also sectioning of the data buffer memory). Approximately twelve of the 64 prime monitor points will be used for subcommutation.

Writing into the programmable memory will be accomplished through the use of GSE, containing the necessary write electronics. This method considerably reduces the weight and power requirements of the memory and precludes accidentally writing into the memory by some transient or other unknown cause while operating in the S/C system. Only write wires will thread the memory cores for writing purposes and these wires will be electrically isolated from the system electronics.

The data collection synchronizer will supply the required sampling pulses to sample the experimenters output in synchronism with its natural data rate. As an example the spin period of the S/C might be divided into 64 equal segments corresponding to 64 sectors of sensor look angle through which the sensor would collect data. This process contains the basic ability of being expanding further such that each of the 64 sectors could further be divided into 64 sectors.

Power and Weight Estimates

It is estimated that a typical data processing system shown on the Block Diagram (Figure 3) would weigh roughly up to 6 lbs. and have a power consumption of approximately 4 watts. The space required would be that of a typical delta pack up to 5 inches in height.

List of Suggested Data System Functions

		No. of Times Requested
1.	High time resolution sampling	10
2.	Programmable telemetry format	6
3.	Simultaneous sampling of multiple outputs	4
4.	Data collection synchronized to S/C spin	6
5.	Data summed over several S/C spin periods	2
6.	Particle counting using accumulators	6
7.	Compression of accumulator outputs	5
8.	Tape recorder start-stop or speed-up operations	4
9.	VLF, wideband	3
10.	Spectral analysis for wideband experiments	5
11.	High accuracy A/D converter (10 bits)	2
12.	Capability of doing pulse height analysis	4
13.	Continuous orbit coverage	2

Figure H-1

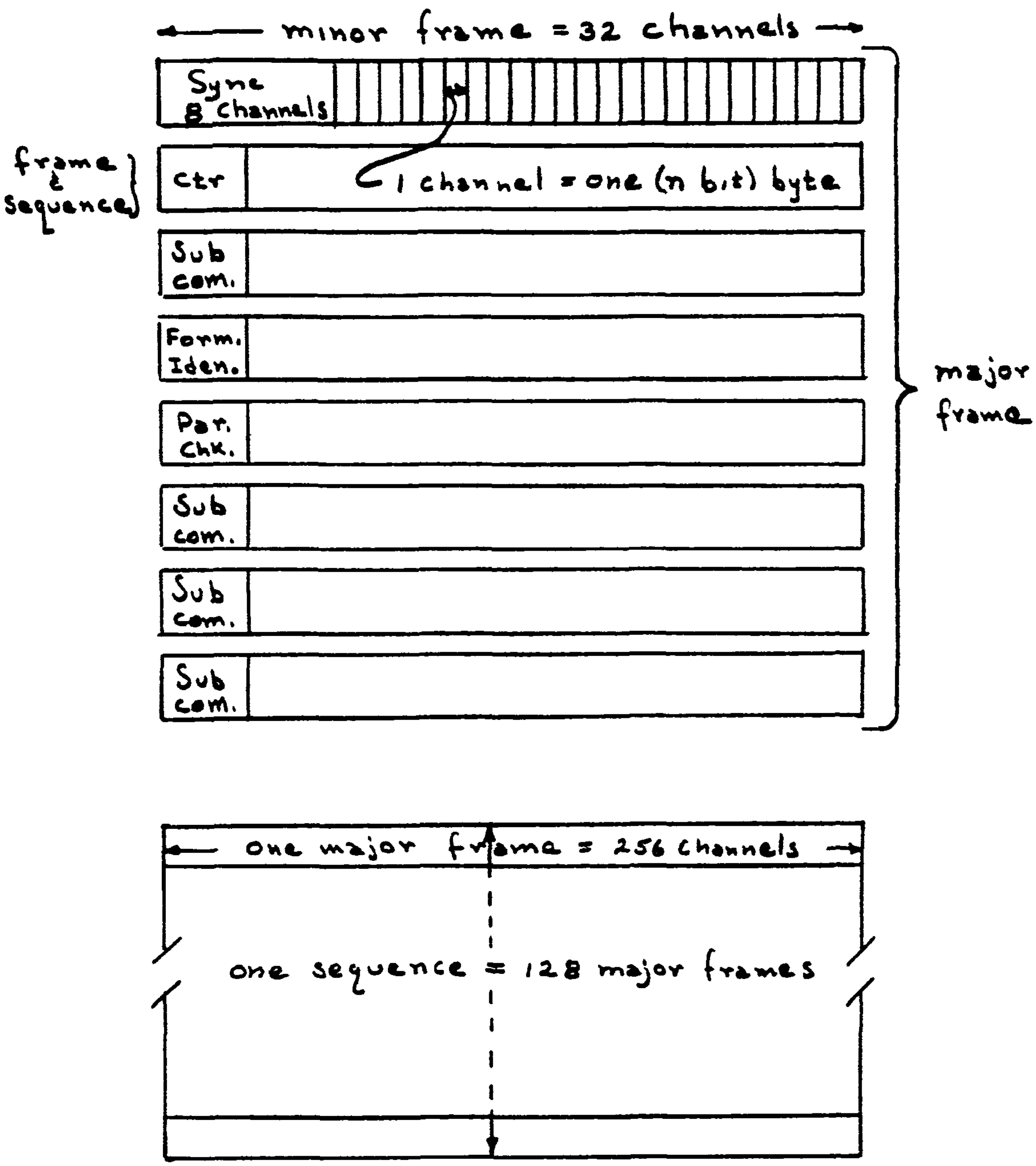

Figure H-2—Small Standard Satellite Telemetry Format (Proposed)

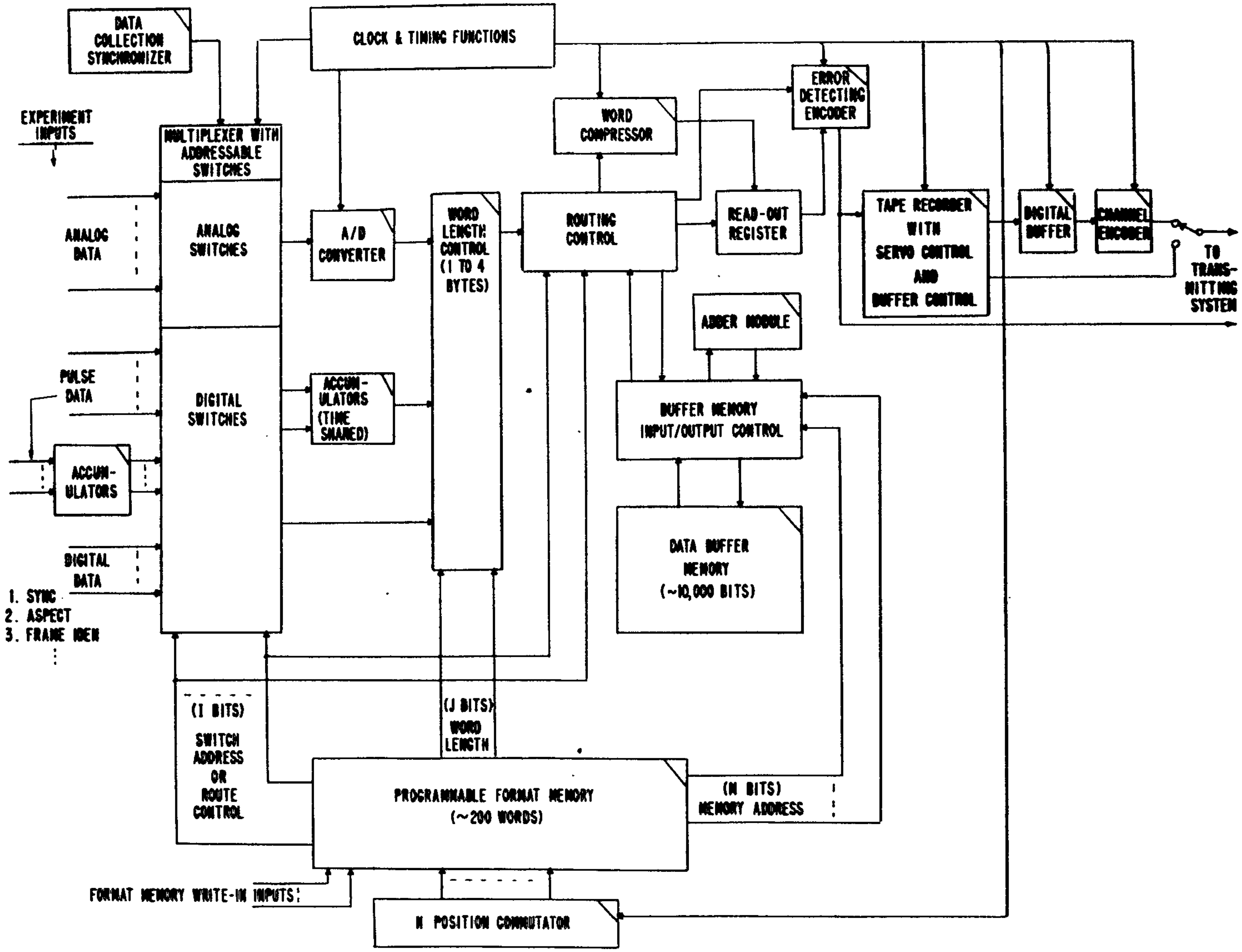

Figure H-3—Block Diagram of the Data Processing System

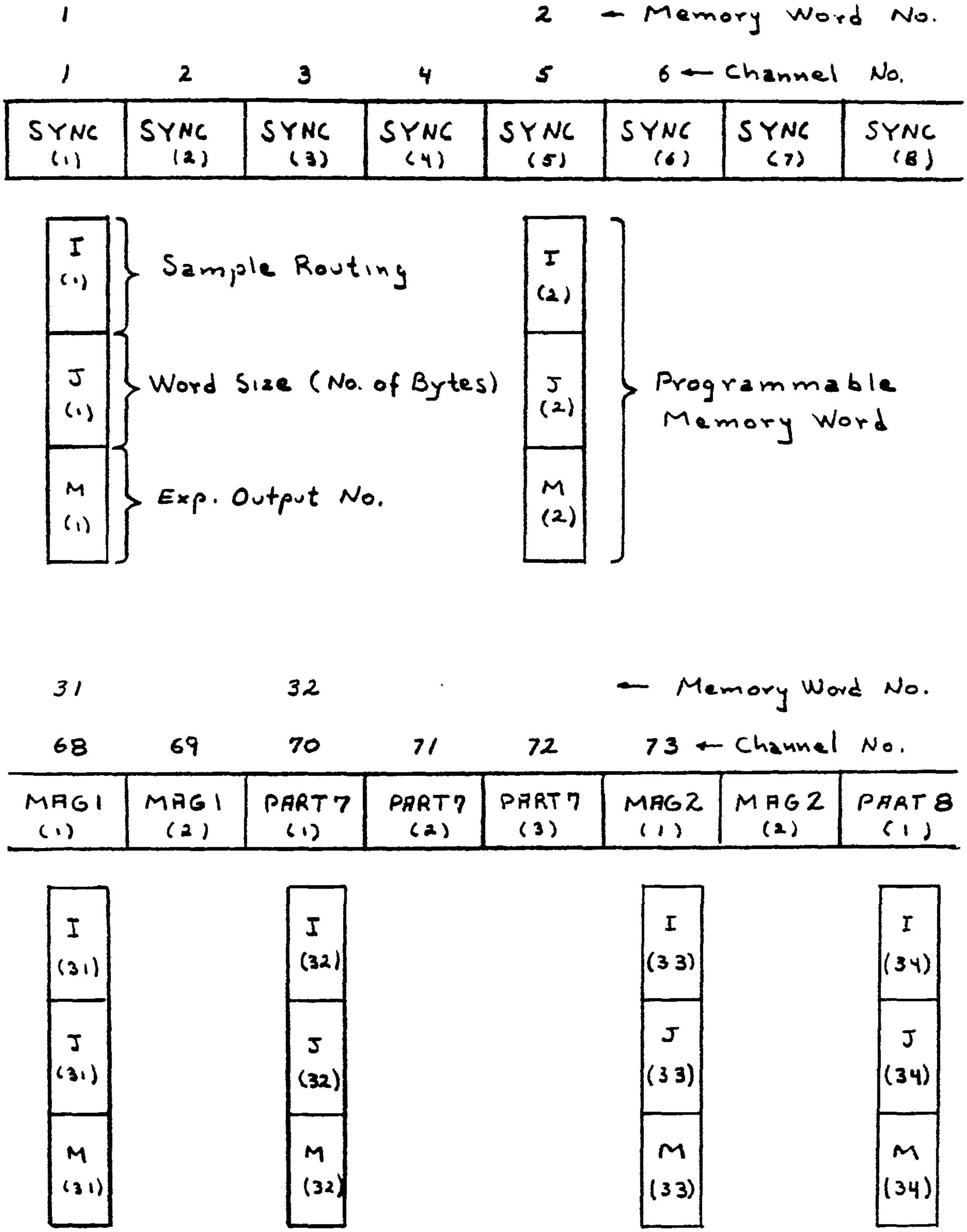

Figure H-4–Programmable Telemetry Format Organization

APPENDIX I

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UNITED STATES GOVERNMENT

Memorandum

TO : Mr. Gerald W. Longanecker DATE: March 23, 1966
Project Manager, Small Standard Satellite

FROM : Telemetry Computation Branch
Information Processing Division, T&DS

SUBJECT: Preliminary Data Processing Plan for the Small Standard Satellite

The enclosure is a report which discusses the Information Processing Division's preliminary plans for the ground reduction of SSS telemetry data.

Michael Mahoney

Michael Mahoney
Head, Project Computation Section

Enclosure

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APPENDIX I

INTRODUCTION

This report contains the Information Processing Division's preliminary plans for processing experimental and housekeeping data telemetered from a small standard satellite (SSS), together with ground rules relating to modes of operation. The objective is to provide the SSS project with a single data-processing capability (equipment, programs, and operations) which will meet the requirements of the entire satellite series. Recommendations given here on data content and format are a composite of the knowledge and experience acquired in processing data from many satellites. Each recommendation either eliminates a problem area previously encountered, permits more efficient use of ground processing equipment, or increases confidence in the integrity of the data. Without unduly increasing satellite complexity, power, weight, or cost, these recommendations allow for wide flexibility in varying the telemetry system both for a given satellite and for the different satellites of the SSS series. The report describes the flow of telemetry data-processing, as well as equipment and computer programming support.

SUGGESTED DATA FORMAT

The following recommendations are preliminary results of project thinking. A preference has been indicated for 4-bit modular word (or channel) structure, contained in a 256 - 4-bit byte-frame. Other features discussed include programmable format memory, high-speed data buffer, and on-board tape recorder. Considerations of spacecraft engineering and concepts of increased data confidence and efficiency of operations for a ground data-processing facility have led to the following proposals for data format:

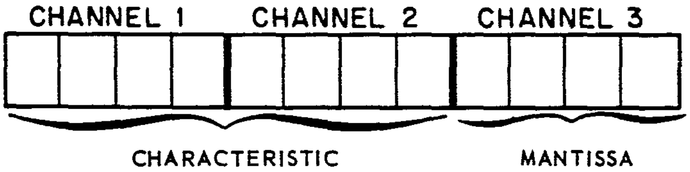
1. Encoding – Split-phase PCM, which lends itself to the use of error-detection and correction codes

2. Channel Resolution – 4 bits

   a. Ample (but not wasteful) event marker and scale factor resolution

   b. In combination with another 4-bit channel, provides ample (but not wasteful) analog resolution of 0.4 percent

c. In combination with two other 4-bit channels, provides comparable pseudo-binary floating point counter resolution of 0.4 percent, along with a much extended range of these counters



(The term channel is used here to mean an integral 4-bit byte, and the term word to indicate a meaningful sequence of channels.)

d. Lends itself well to use with a 24-, 32-, or 36-bit per word computer. The CDC 3200 STARS Phase II system uses a 24-bit word; the IBM 360 system uses a 32-bit word; the Univac 1108 uses a 36-bit word. This bit configuration is also compatible with the new 9-track tape convention.

3. Channels per Frame – 256, an almost arbitrary selection, but a good choice in that it is a binary number, is binarily related to bits per channel, and is not so long that it violates Goddard's telemetry standards.

4. Number of Subcoms – At least one devoted to experimenter use and one devoted to subsystem use. If a short subcom sequence is chosen, perhaps additional subcoms should be provided. All subcoms should be synchronized with respect to each other.

5. Synchronization Word –

a. Should be equal to or less than 32 bits in length

b. Should be one continuous string of bits

c. Its position in the frame should not interfere with the cyclic behavior of subcoms.

d. The synchronization pattern should have characteristics similar to Barker codes.

6. Spacecraft Clock Word – Should be a subcom sequence counter updated in a known position within a particular frame of the sequence. If multiple bit rates are possible, it is a multiple sequence counter. If this clock is read out in another equipment group, the equipment group timing should be slaved to the same source as the clock, rather than related in an obscure way. This readout,

combined with the frame counter and channel number, will provide timing resolution equal to a channel-readout time interval. The clock should have a number of bits sufficient to prevent recycling in short time intervals. As a lower limit on this, the clock should never be of such a length that it would recycle more than once during the period of a tape-recorder sequence.

7. Bit Rates – If more than one bit rate is possible, the rates should be related in a binary manner for practical consideration in playing back data on the ground equipment at reduced or accelerated rates.

8. Playback Rate of On-board Tape Recorder – Playback rate should be binarily related to real-time rate. This, again, is a practical consideration. The storage capacity and playback rate should be chosen so that a playback sequence will not overlap two or more ground-station tape recorders.

Experience has shown that erase-after-read electronics in tape recorders provide the most trouble-free operation. Situations have seldom arisen in which any gains were foreseen by dumping the tape recorder a second time in an erase-before-write (or saturated-write) system; when they have, the gains were not realized. Meanwhile, the overhead wasted in detected "old" data during each dump has far outweighed the possible advantages.

Other considerations include the operation of the on-board tape recorder itself. Segmenting recording sequences on one tape recorder is not generally a good idea, especially if two on-board tape recorders are used in combination. The difficulties inherent in unscrambling tape-recorded sequences that jump from one recorder to another in real time should be avoided if possible.

Of course, when a storage device such as a tape recorder or an on-board core memory is used, a spacecraft clock must be used. It is also mandatory that synchronization, identification, frame-counter, and mode data be multiplexed in the data stream at the time of entry from the storage device, not at playback.

9. Frame Counter Word – Main-frame word indicating the current frame number in subcom sequence.

10. Identification Word – Main-frame word indicating type of data (satellite, bit-rate, tape-recorder, buffer, or real-time).

11. Mode or Configuration Matrix Word – Contents of the relay matrix controlling the various modes, spacecraft equipment, and experiments are read out in the mode word. This channel indicates the states (ON - OFF - MODE - SCALE) of the spacecraft equipment and experiments. As an example, this word makes it possible to determine what experiments are turned on or off; what the scale

settings of the various detectors are; what mode of operation the particular experiment packages are in; etc.

12. Format Word – Main-frame word indicating the data format of the main frame and subcommutators. This word defines the source of the sample and the channel in which it appears. For the SSS, the content of the format word defines the individual location and content of each location in the format-control memory. The channel or position in the format-control memory could be indicated by the frame-counter channel, in which case the frame counter would fill a double role as a frame-position indicator as well as a channel-position indicator in the format-control memory.

13. Check Sum Channel – A channel devoted to error-detecting of data bits occurring in the previous frame or fraction of a frame.

The following characteristics should also be considered:

14. Standard Index Gates – Experimenters should have a standardized set of control pulses to drive their experiment packages. These standardized index or control gates would have the following characteristics:

a. Main-frame channel related

b. Subcom channel related

c. Spin related

d. Sun-crossing related

e. Earth-crossing related

f. Velocity vector related

g. Clock related

h. Data relating to the time of incidence of items d, e, and f should be read out in telemetry.

15. Standardized output formats for attitude and housekeeping.

16. Housekeeping data reflecting critical parameters aboard the spacecraft (the ACS system, as an example) should be capable of being read out at a rate higher than the normal subcom rate.

17. A "safe" mode of operation should be provided into which the spacecraft can be commanded in the event of an ACS failure.

18. Any command which could adversely affect spacecraft performance if sent unintentionally should be a duplex command.

19. Standardized attitude sensors.

20. Sequence integrity should be maintained; that is, it should be logically impossible during a subcom sequence to change the spacecraft telemetry configuration as indicated in the format word, identification word, and mode word.

SAMPLE FORMAT

Figure I-1 shows a possible application of the suggestions in the previous section to the design of a telemetry data format.

The main components of the suggested data-processing system:

Equipment – the physical facilities supporting the overall data-processing effort.

Programs – the internal machine operation, configuration, or stored program sequence to which the individual physical elements of the facility will respond

Operations – the overall operational sequence and functions imposed on the incoming data, ancillary equipment, and program systems in processing satellite telemetry data.

EQUIPMENT

The SSS data-processing system will use STARS Phase II equipment for real-time and analog-tape data processing; the 1108 computer, for experiment and subsystem analysis; and the Stromberg-Carlson 4020 microfilm printer/plotter, for data display and presentation. Special prelaunch and postlaunch processing may use any other equipment available in the Information Processing Facility, such as strip chart recorders or film readers.

STARS, PHASE II

The Satellite Telemetry Automatic Reduction System, Phase II (STARS II), is a general-purpose system designed to handle large volumes of PCM telemetry data. Its primary function is to perform preliminary processing of PCM telemetry inputs, recorded on analog magnetic tape, from earth satellites such as OAO and OGO.

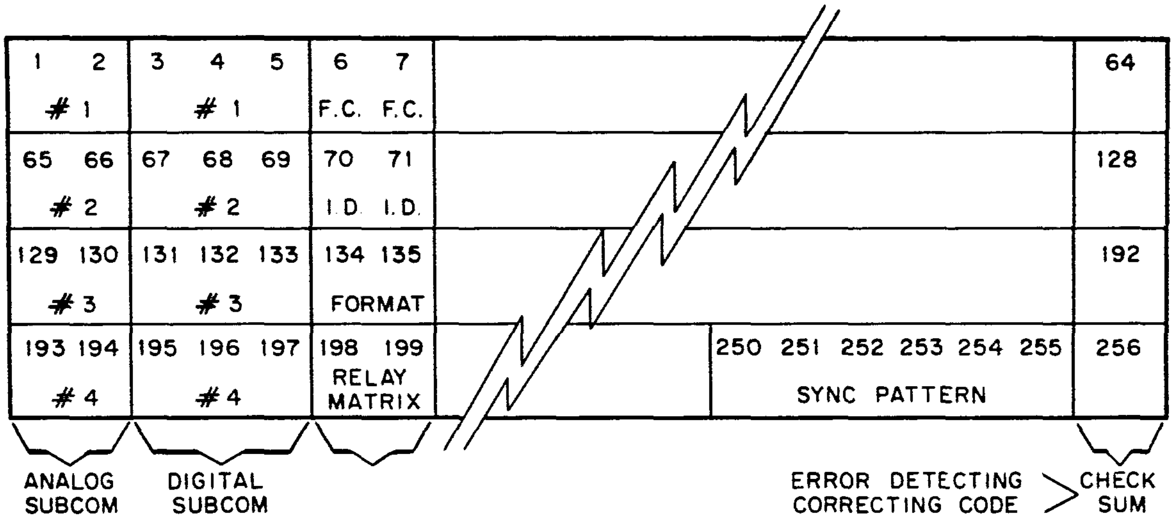


Figure I-1–Sample Format

F.C. = Frame counter

I.D. = Identification of data (satellite - bit rate - tape recorder, buffer, or realtime)

Format = Contents of format control memory

Relay Matrix = Contents of command matrix (on-off-mode)

Spacecraft Clock is a sequence counter. It counts the number of subcommutator sequences that would occur at the lowest bit rate. It is read out in the digital subcom of Frame #1.

Bit Rates are related in a binary manner.

Check Sum is a form of error detecting for data occurring in the previous 63 channels. On the ground, this channel will be checked against the previous 63 channels and in conjunction with bit errors in sync word will be used to determine the continuous level for data. This information will be passed on to experimenters.

The organization of the system makes it applicable to all common types of PCM telemetry inputs, either real-time or tape-playback. The modular structure of the system interfaces permits its adaptation to virtually any signal-conditioning and formatting device to handle other types of data inputs.

The STARS Phase II system is fully automated under computer control, including control of the simulation equipments for fully automatic system, setup, checkout, and diagnostic operation. Total or partial manual control may also be exercised from the central telemetry console.

The primary source of data is analog magnetic tapes recorded at satellite-tracking stations and containing telemetry data received during a satellite pass over the station, plus time generated in the station. The STARS II system is designed to process data from the recorded tape over a range of playback speeds from 1/8 to 32 times real time. The system can also receive and process inputs from a real-time data link and a local time code source.

The primary system output is a master computer magnetic tape containing all telemetry data merged with time and quality information generated during the run. This tape will usually receive further processing on other off-line computing equipments. However, the STARS II systems have the capability and capacity to perform auxiliary functions, such as scaling, conversion, or sorting, on either single- or multiple-pass operations. The multiple-playback ratio capability provides considerable flexibility in matching the input data rate to the computing load imposed upon the computer and output devices by the number of auxiliary operations performed.

SYSTEM ORGANIZATION

Figure I-2 shows the functional organization of the STARS II system. The CDC 3200 general-purpose computer – the basic control element for the assembly of signal-conditioning, conversion, control, simulation, and computer peripheral equipments – can perform both on- and off-line functions. The computer is provided with 16,384 24-bit words of memory, magnetic tapes, card equipment, and a high-speed line printer.

The input subsystem consists of two analog magnetic-tape reproducers, plus switching elements for automatically routing inputs to the proper telemetry subsystem. The input subsystem can receive telemetry inputs from a real-time data link and time from a local timecode generator, in addition to those from the tapes and simulator.

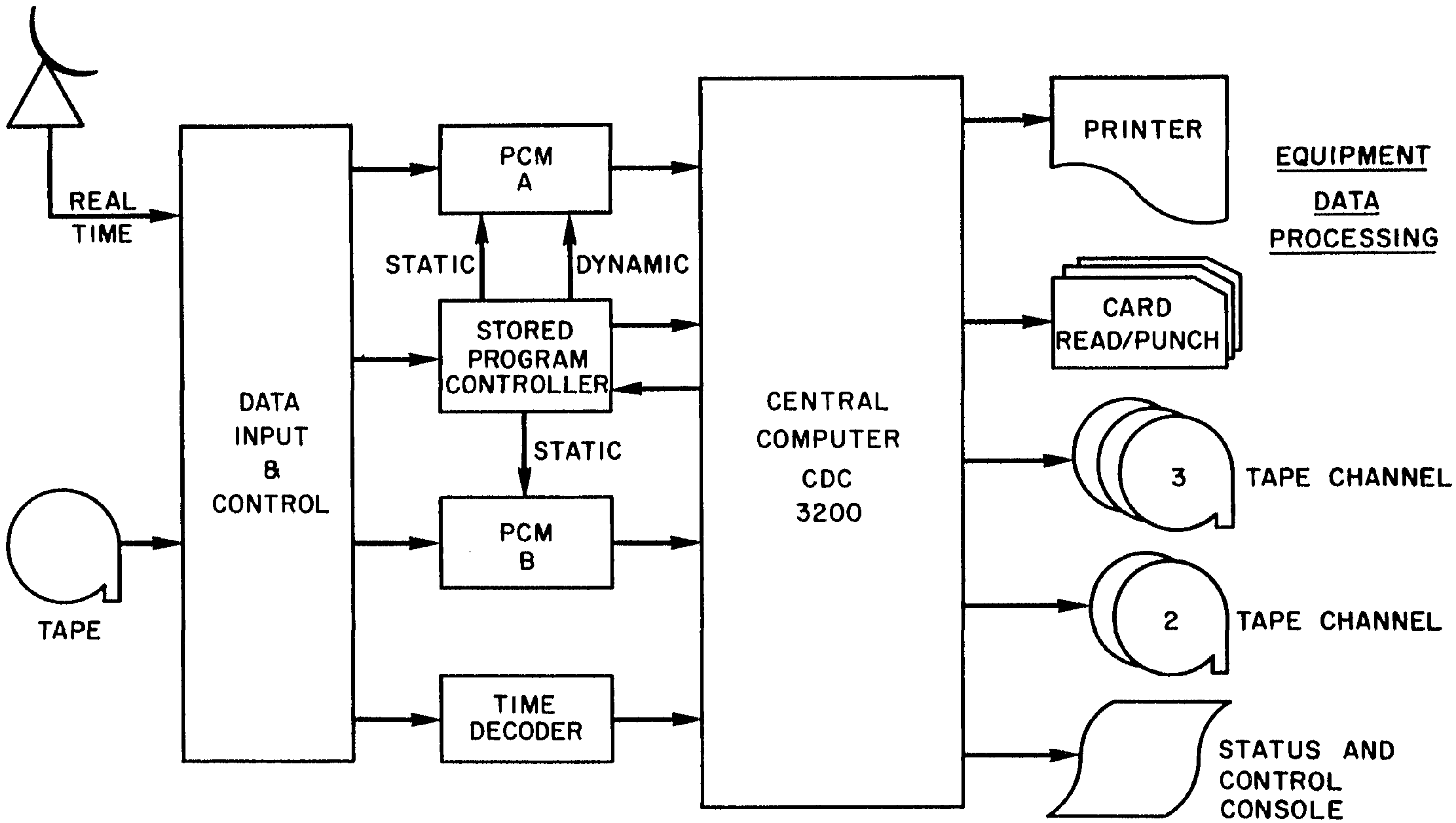


Figure 1-2–STARS II System, Functional Diagram

Two independent channels in the PCM subsystem can process PCM telemetry. Channel A contains a core memory and exercises dynamic stored-program control of the telemetry-conversion operations. Its program may be loaded and verified by the computer. Channel B contains no core memory, but can be set up automatically from a program stored in the Channel A memory. Data from two telemetry links (recorded on the same tape) can be processed simultaneously, using both channels.

The timecode subsystem converts time signals accompanying the telemetry data into binary time-of-day and day-of-year information. Normally, three separate time signals (BCD, serial-decimal, and a reference frequency) are received and used to derive a "best estimate" of the true time instants. The binary time output is continuously available on demand to the computer for correlation with the data samples. Time-freeze buffers on each PCM telemetry channel preserve proper time correlation over the range of playback ratios. The display subsystem provides analog outputs, derived from either the PCM subsystem or the computer, which may be displayed in time-history form on a recording oscillograph.

The computer subsystem consists of a CDC 3204 basic processor, 16,384 words of core memory, plus magnetic-tape, punched-card, and printer peripherals.

The central telemetry console of the STARS II system contains all controls and displays associated with the telemetry-processing and simulation equipments. From this central location, the operator can continuously monitor signal waveforms and the operating status of the telemetry subsystems, and can manually override any or all functions normally set up by the computer. In general, the data-tape output from STARS II will be used as input to the Univac 1108 computer system.

THE UNIVAC 1108 COMPUTER SYSTEM

The Information Processing Division's UNIVAC 1108 computer system is a high-speed general-purpose stored-program computer system with hardware specially designed to meet overall requirements for processing scientific satellite telemetry data.

In addition to computer word-byte instruction capability and shift-matrix instructions, the system has unusually large random-access storage. This includes a hierarchy of on-line high-capacity random-access storage devices. Table I-1 demonstrates this range in capability.

Table I-1
UNIVAC System Capabilities

Unit	Capacity (words)	Avg. Access Time (sec)	Transfer Rate (words/sec)
Control[1] unit	6.0×10^1	1.25×10^{-8}	3.0×10^7
Core storage	6.5×10^4	3.75×10^{-8}	2.7×10^8
FH 432 drum	1.6×10^6	4.25×10^{-3}	2.4×10^5
Fastran II drum	2.2×10^7	9.2×10^{-2}	2.5×10^4
Tapes[2]	4.7×10^7	1.2×10^2	1.6×10^4

[1]Random access, but not high capacity

[2]High capacity, but not random access

SYSTEM ORGANIZATION

Figure I-3 shows the functional organization of the UNIVAC 1108 computer system. The computer is provided with 64K 36-bit words of memory with an effective cycle time of 375 nanosec, and a control unit composed of:

- Sixteen arithmetic registers
- Sixteen index registers
- Sixteen access-control registers
- Four special-purpose registers, each with a cycle time of 125 nanoseconds

The input/output systems include the following units:

(a) Central processor with 65K core memory and 12 input-output channels

Eight of these channels are connected to the following equipment:

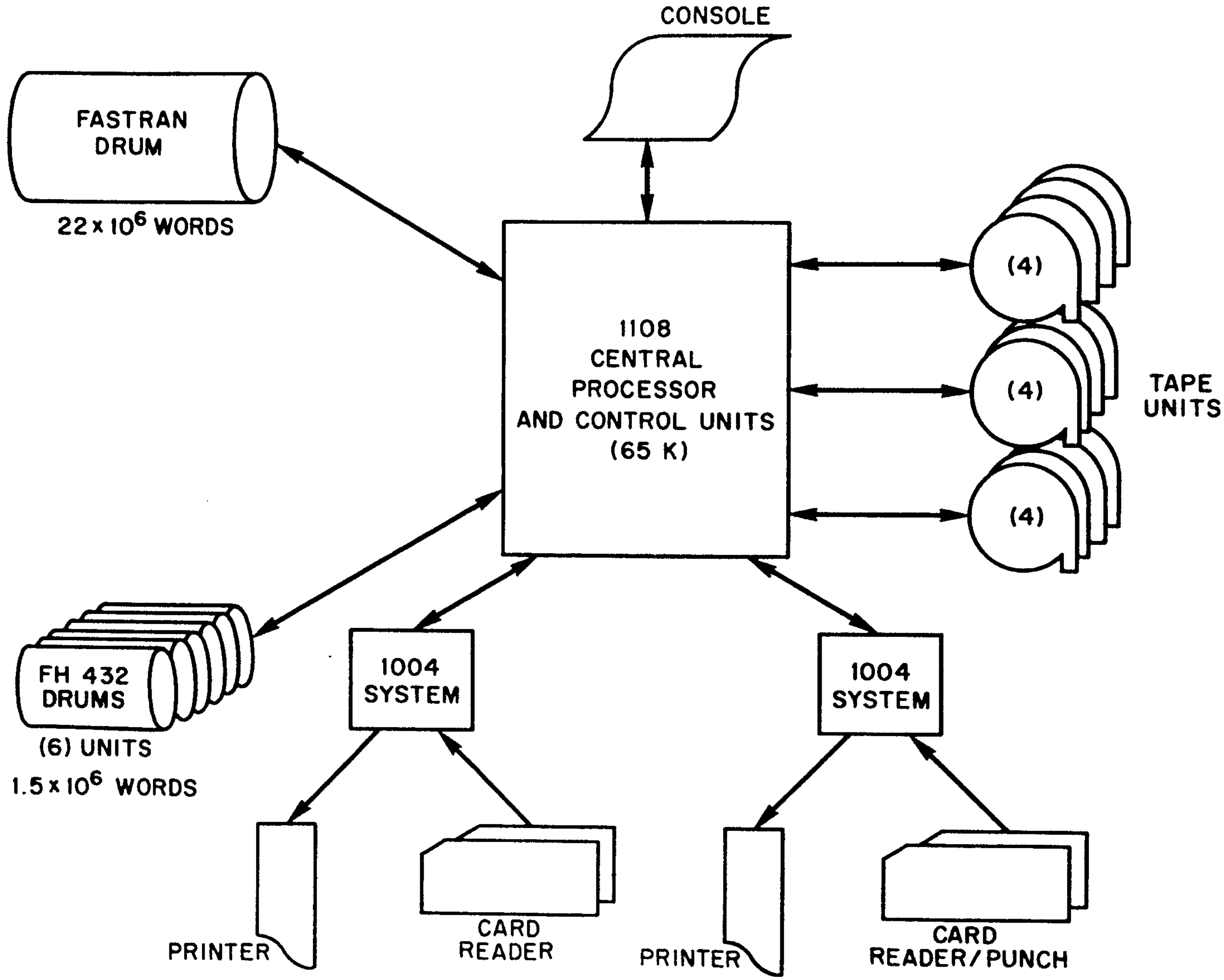


Figure 1-3–UNIVAC 1108 Equipment Layout

(b) A Fastran II magnetic-drum storage system with a capacity of 22 million words of random-access storage. Average access time is 92 milliseconds, after which words can be transferred at a rate of 25.15K words/sec.

(c) Six FH 432 magnetic-drum storage units with a total capacity of 1.6 million-word random-access storage. Average access time is 4.25 milliseconds, after which words can be transferred at a rate of 240K words/sec.

(d) Twelve Uniservo VIII-C tape units (7-track) on three channels (4 tape units per channel). These tape units read or write 200, 556, or 800 bpi, at a transfer rate of 24.0/66.72/96.0K characters per second.

(e) Two Univac 1004-II-0C satellite units, each controlling a 600-line-per-minute printer and a 615-card-per-minute card reader. One of the 1004 systems controls a 200-card-per-minute punch.

(f) Console typewriter.

DATA-PRESENTATION EQUIPMENT

The system will use high-speed data-reproduction and display devices such as the SC-4020 microfilm printer/plotter.

The 4020 computer recorder (Figure I-4) is an electronic system capable of accepting digital magnetic-tape signals and converting binary or binary-coded-decimal codes into combinations of alphanumeric printing, curve plotting, and line drawings. Recording the information on both microfilm and photorecording paper, the system translates coded data into complex annotated graphs and drawings. The heart of the system is a charactron shaped-beam tube. Basic units of the system include a typewriter simulator, vector generator, variable stop-point axis generator, forms projector, rotatable tube mount, programmable expanded image, and a recording camera.

The 4020 computer recorder uses a charactron shaped-beam tube to generate characters, lines, or curves. The heart of the tube is a stencil-like matrix – a thin disc with alphanumeric and symbolic characters etched through it – located within the neck of the tube in front of an electron gun. The stream of electrons emitted from the gun passes through the matrix, cutting the desired character from the beam. When the shaped beam impinges on the phosphor-coated face of the tube, the character is reproduced.

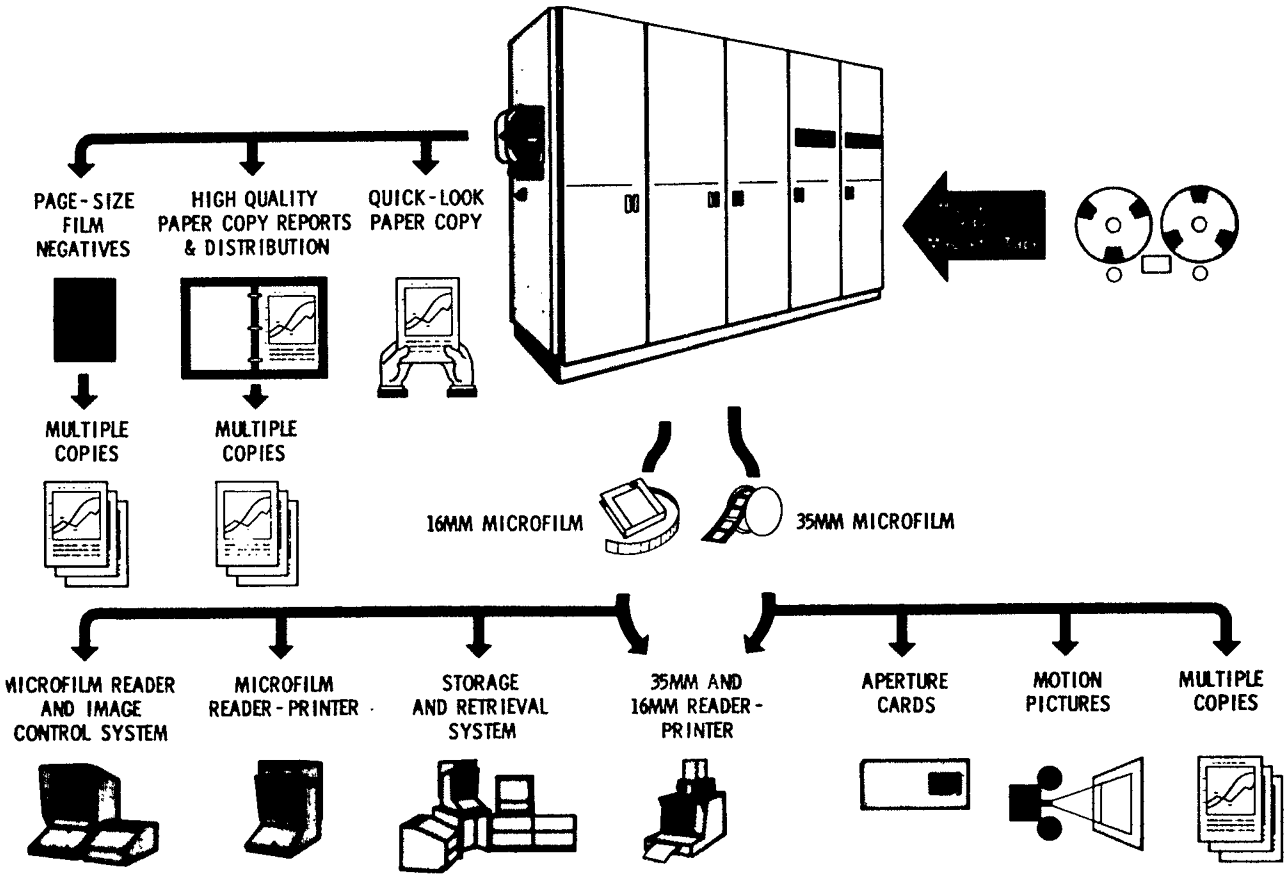


Figure 1-4—4020 Computer Recorder

This device also has extensive plotting capability, including the ability to draw vectors, and a variable-intensity plot mode which allows for an individual plot-print intensity of one of 16 levels of grey. This variation in levels of grey makes it possible to produce pictures containing three degrees of freedom, x, y, z; x and y relate to the position on the surface of the plot, and z relates to the value at that point as represented by its grey level.

The UNIVAC 1108 has an extensive library of subroutines which can be used with this plotting device to produce a large variety of plots.

Figure I-5 is an example of the types of output plots which may be produced.

PROGRAM DEVELOPMENT

Efficient development of a program for the SSS series depends greatly on establishment of a standardized telemetry format; such a format will permit development of a unified program system for the CDC 3200 and UNIVAC 1108. This system would be dynamic, in that it would automatically adjust itself to the incoming data format; it would sense any change in data format by virtue of the contents of the frame counter, identification format, and data mode words. This ability would allow for format changes after launch, and indeed in real time, without the perturbations usually experienced.

A system of this type has the added advantage of making known (by definition) the individual experimenter data-tape output formats as soon as the inputs to the system are known; once the telemetry format has been assigned, the output tapes from system are defined. Furthermore, if standardized housekeeping and attitude sensors, calibrations, and formats are used, then a standardized output is possible; this has the added advantage of providing spacecraft subsystem engineers with comparable data from satellite to satellite in a format to which they are accustomed.

Also, since the mode word identifies the configuration of the spacecraft and experiments, the laborious decoding, verification, and chronological ordering of command data necessary in past satellite projects will not be necessary. Data content, mode, and format will be an integral part of the telemetry system.

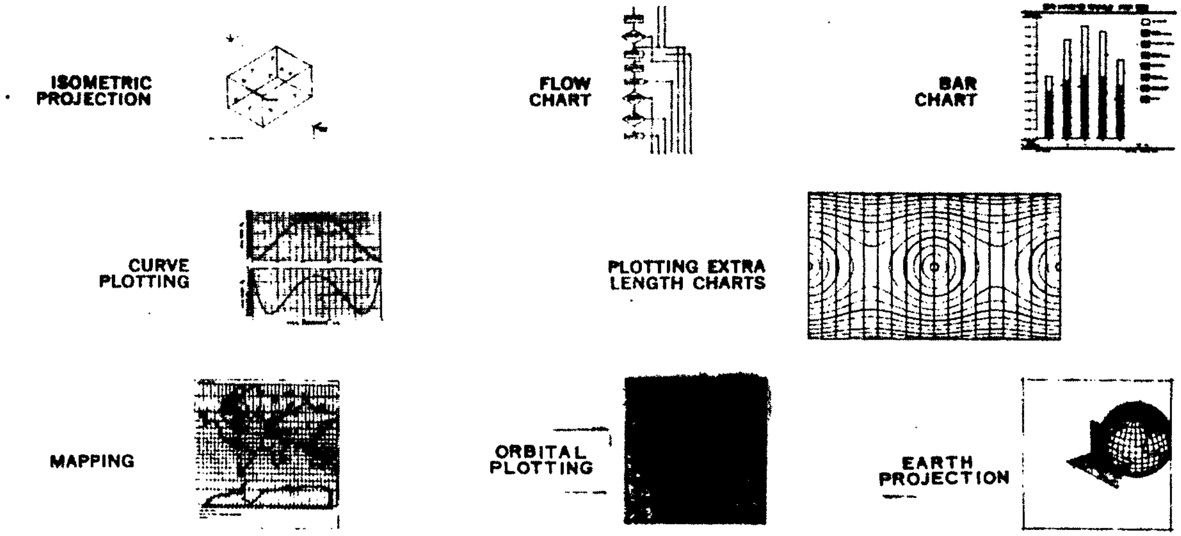


Figure I-5—Sample Plots

OPERATIONS

1. Prelaunch Experimenter and Spacecraft Engineer Support

This support is provided through two facilities: the data interpretation laboratory, and the data processing readiness trailer.

Present plans include a facility where an experimenter or subsystem engineer can perform special processing of his data. Data may be in an analog form on magnetic tape, at the output terminals of his equipment, or multiplexed into proper channels in telemetry. The facility will contain equipment such as strip-chart records, real-time data-display devices (CRT's, etc.), and hard-copy presentation of plots in a matter of seconds. A tremendous advantage will be realized in that no programming is necessary on the part of the user; the program will be arranged by a simplified dialog with the computer.

Data-Processing Readiness Trailer

The purpose of the readiness trailer is to generate analog-data tapes containing telemetry and associated signals in the format to be used by the tracking stations, as a particular satellite and its experiments are exercised. The resulting analog tapes are returned to Goddard Space Flight Center for use by the Branch in checking out the data-processing lines to be used for that satellite, and to verify the operational parameters set into those lines. The digital-tape outputs of the processing lines are, in turn, used to check out and verify the many computer programs which will be used in production processing. The final computer-output tapes are shipped to the experimenters. Production techniques, tape flow, and record-keeping are checked out during processing. The experimenters use tapes to determine that the format is acceptable and that their programs and processors are valid, and to check the operation of their experiments.

Experience has shown that the above checkout procedures are necessary to confirm the proper operation of the satellite telemetry system and data-processing equipment, as well as the validity of the computer programs.

Figure I-6 is an artist's conception of the readiness trailer.

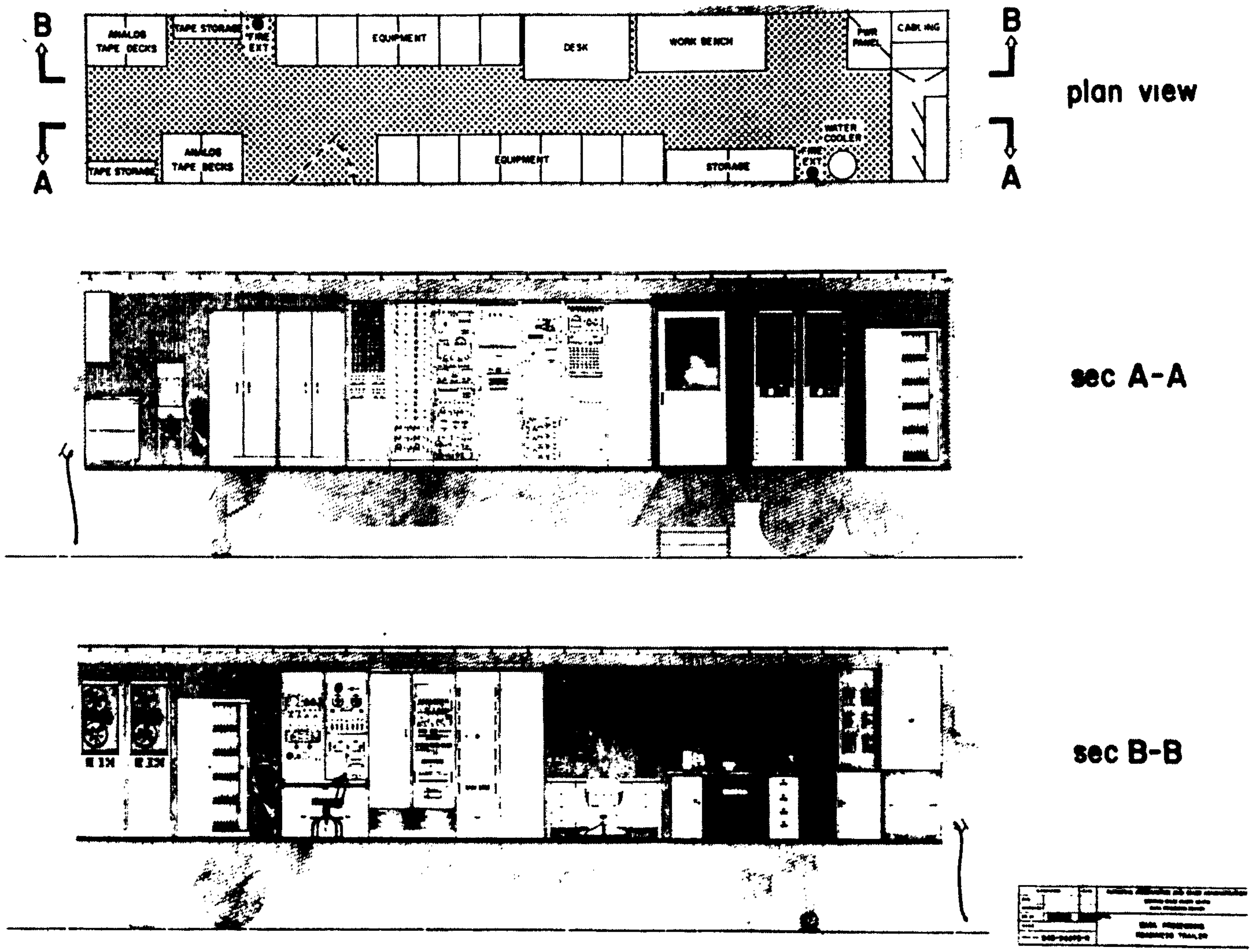


Figure I-6–Readiness Trailer, Artist's Conception

2. Early Postlaunch and Periodic Quick-Look

A program system will provide the experimenter and spacecraft subsystem engineers a near-real-time abbreviated look at selected data channels. This operation is limited in scope and is provided at launch and scheduled intervals thereafter (Figure I-7).

The output of this program system will be printouts and plots of data channels of interest to project engineers and experimenters.

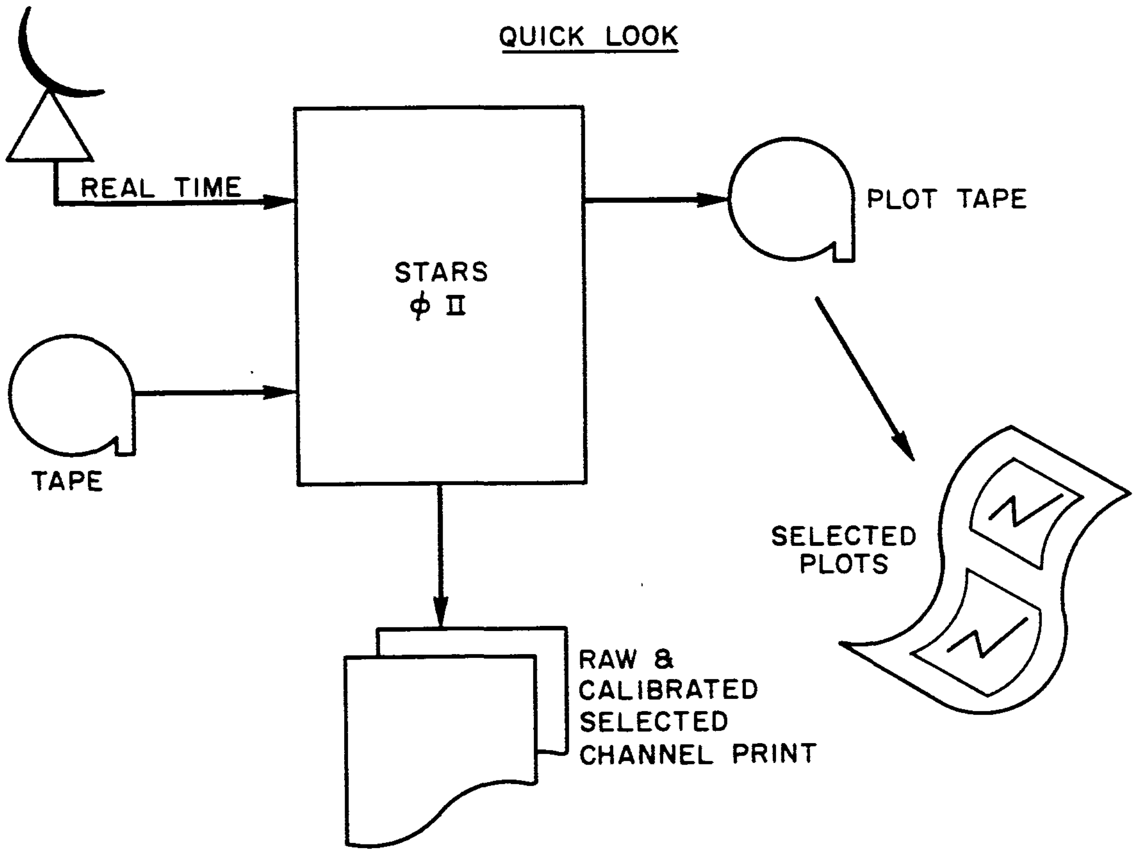


Figure I-7–Quick-Look System

3. Spacecraft Behavioral Analysis

This procedure, which provides spacecraft subsystem engineers with housekeeping parameters in graphic and printed form, is done on an interval basis, and on selected data, upon request (Figure I-8).

The processing provides to subsystem engineers in-depth analysis of housekeeping data, including calibration, scaling, and correlation of these data with other parameters such as orbital position and spacecraft attitude. The process differs from the quick-look system, in that two passes are required: Pass I uses STARS II to digitize and edit spacecraft housekeeping data, producing a tape which is used as input to Pass II; Pass II uses the UNIVAC 1108, requiring as input the output tape from Pass I and an orbit tape. Outputs from Pass II will include listings, plots, and data tapes which may be used for further analysis.

4. Regular Processing

All data will be processed in chronological order, using the CDC 3200 STARS φ II and the UNIVAC 1108 computer systems. Processing will proceed in two passes (Figures I-9 and I-10). Pass I, using the CDC 3200 STARS φ II system, will perform the following:

a. Chronological ordering

b. Identification of various indicators in data, such as format, mode, and identification channels, to set up equipment and software for processing file

c. Produce standard output edit tape for input to 1108 analysis system

d. Produce set of standard statistical data-quality indexes for evaluation, and as a guide to selecting data for further processing

e. Keep running account of data quality to allow for immediate rerun of trouble files; also, separate reruns may be helpful.

f. Monitor bit-error rate as a means of adjusting frame synchronization strategy to achieve optimum recovery for the bit-error rate experienced.

Pass II of the regular data-processing procedure will be tailored to the individual SSS requirements. The program system will contain and make use of subroutines of a general nature used in common by all small standard satellites: input-output routines; plotting routines; printing routines; attitude-determination, subsystem, and special scientific subroutines. Special programming requirements peculiar to a given mission will be written and combined with the more general routines to form an experimenter and subsystem analysis package for each individual satellite.

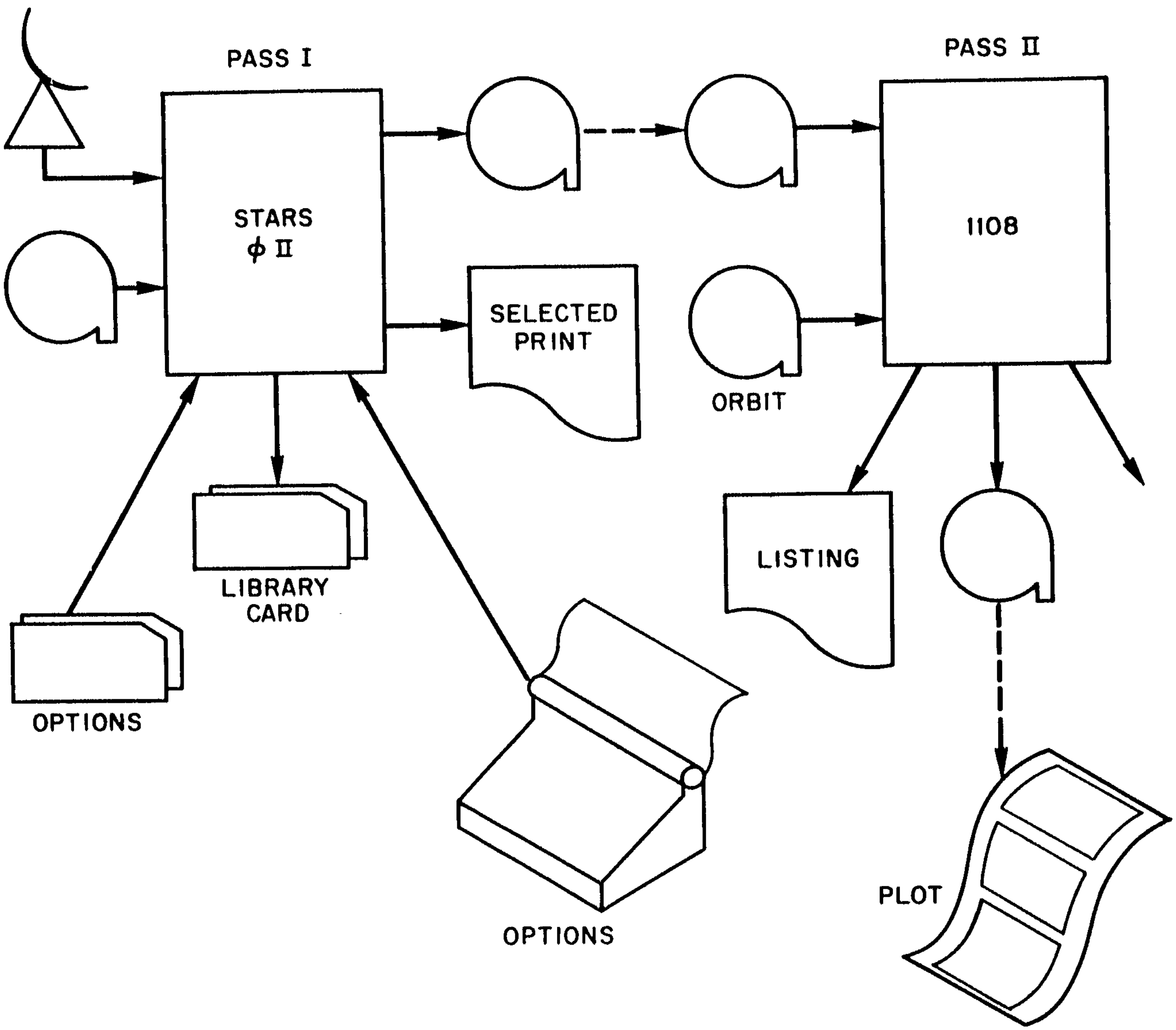


Figure I-8—Spacecraft Behavioral Analysis

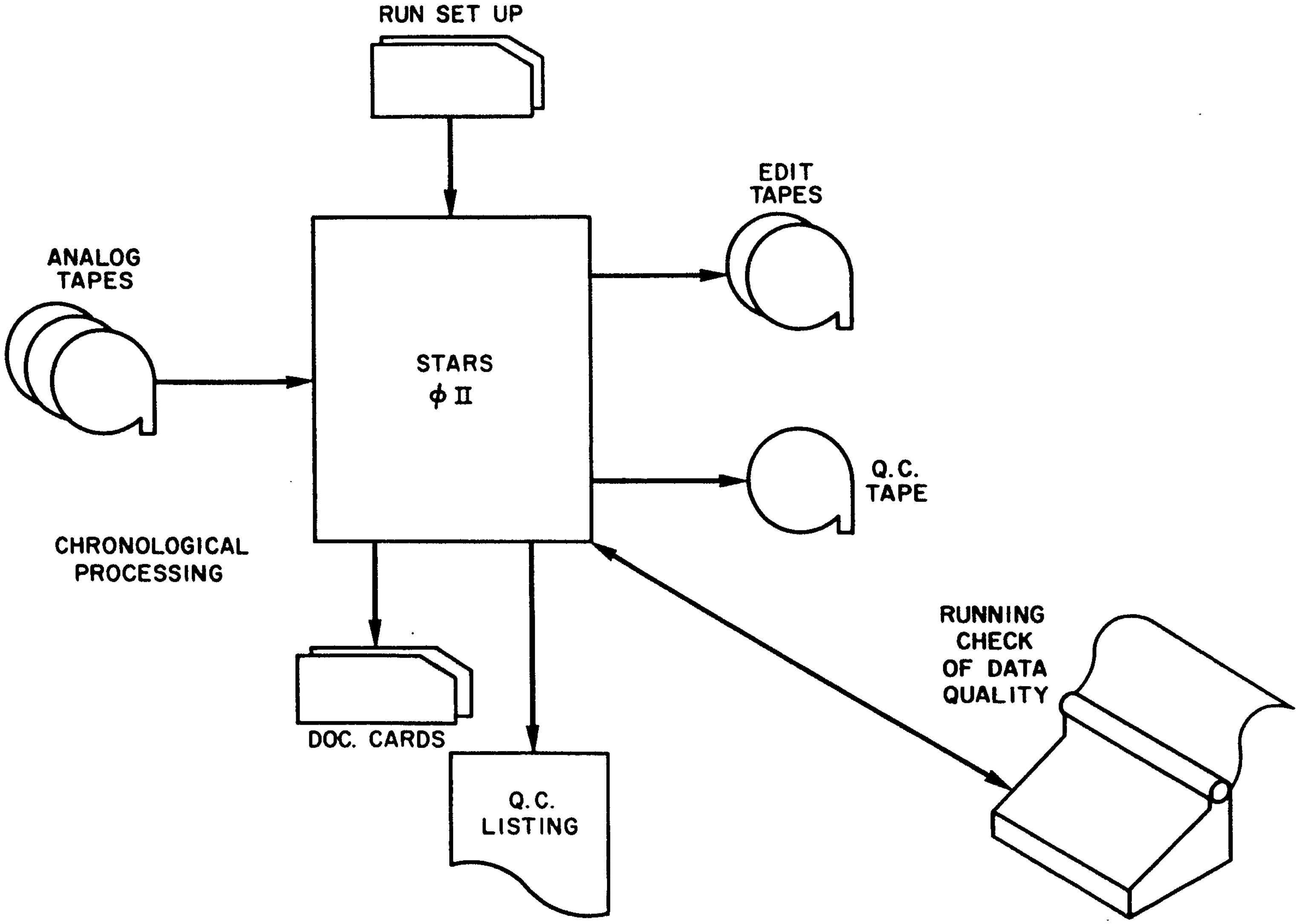


Figure 1-9–Regular Data Processing, Pass I

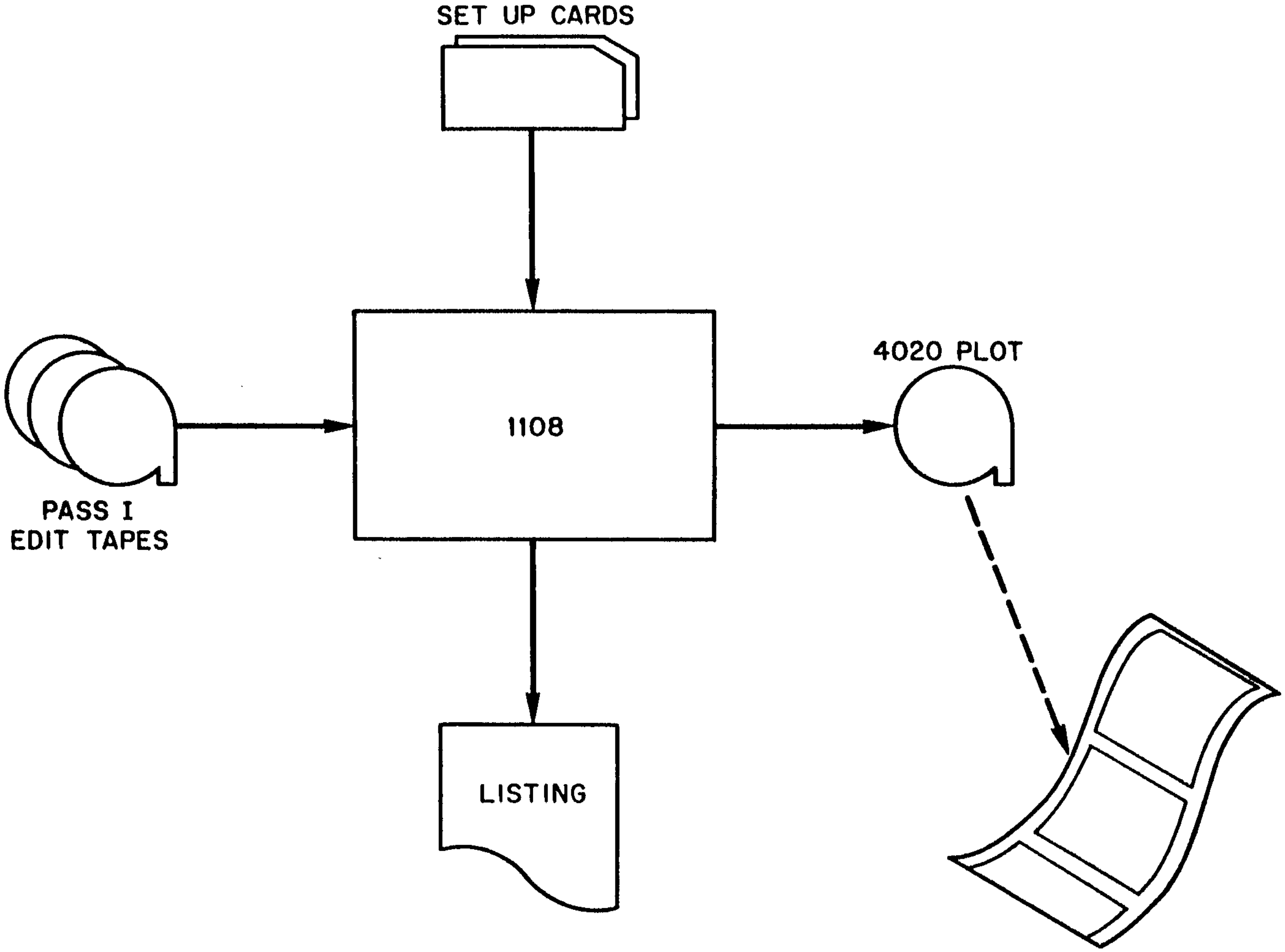


Figure 1-10—Regular Data Processing, Pass II