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FINAL REPORT

for

THE STUDY OF THE ATTITUDE CONTROL OF SMALL SATELLITES AND RELATED SUBSYSTEMS

VOLUME I - SUMMARY

4 SEPTEMBER 1970

AVSD-0555-70-RR

Prepared under Contract No. NAS 1-10014 by
AVCO SYSTEMS DIVISION
Wilmington, Massachusetts 01887

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER
Hampton, Virginia 23365

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PREFACE

This final report presents the work accomplished in a 3 month study of the Attitude Control of Small Satellites and Related Subsystems. The study was conducted for the NASA Langley Research Center (LRC), Hampton, Virginia, under the cognizance of Howard J. Curfman, Technical Representative.

The report is presented in two volumes:

Volume I - Summary

Volume II - System and Subsystem Technical Releases

The Summary, Volume I, is a description of all of the work performed on this program, and Volume II is the collection of all of the Technical Releases (TR's) prepared during the study.

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Study Project Manager

Approved by:

Ronald J. Massa
Director, Space Programs

TABLE OF CONTENTS

THE STUDY OF THE ATTITUDE CONTROL OF SMALL SATELLITES AND RELATED SUBSYSTEMS

SUMMARY

Engineering studies and analyses were conducted by Avco Systems Division at their Wilmington, Massachusetts facilities, to support the NASA Langley Research Center (LRC) in their definition of further unmanned space flight activities. These activities concentrated on a satellite concept capable of quick response to the implementation of a single, or at most dual, experiment aboard a relatively simple and cost effective flight spacecraft system.

In the implementation of the satellite concept, primary emphasis was focused on:

(1) Attitude control subsystem concepts and component hardware definition

(2) Data handling subsystem concepts and component hardware definition

(3) Definition of a typical spacecraft system configuration to which the attitude control and data handling subsystems concepts can be applied.

In order that all major aspects of this typical spacecraft system could be defined, secondary support studies were conducted to define related subsystems, such as, power, communications and structure. A nominal amount of program planning was also conducted to establish a management framework to be used as a baseline for cost and availability trade-off comparisons of the major hardware elements in the attitude control and data handling subsystems. Weight statements and configuration sketches of the spacecraft concept, were prepared as packaging examples of typical implementation of the selected subsystem hardware, including allowance for installation of the experiment hardware.

Based on the investigations conducted during the studies, attitude control and data handling concepts are recommended which demonstrate that a small cost-effective satellite is feasible to accommodate the requirements of a majority of experiments considered.

INTRODUCTION

Status of Problem

Various Government agencies have identified several areas of experiments for which the actual space environment is required for continued development or for which operational feasibility should be demonstrated prior to commitment to a complex operational satellite program. To accommodate such various potential experiments, in a timely and cost effective manner, resulted in the need for a versatile spacecraft concept.

Present NASA LRC work was restricted to in-house feasibility studies of various concepts and their supporting subsystems. This work has identified earth pointing as a prime requirement of many potential experiments, with the resultant need for two- and three-axis attitude control. Attitude determination was also recognized as a potentially strong requirement, since some experiments may not require a precise control of attitude but would require a more precise knowledge of the satellite attitude, especially during the gathering of experiment data. The characteristics of these subsystems and the required monitoring of their status and performance, were also recognized to have a significant impact on the "on board" handling of data, the communications link and the command requirements.

Purpose of Investigation

The purpose of this study was to investigate these basic subsystems areas and define an operational concept for each of them consistent with spacecraft system requirements which could accommodate a variety of experiments on a "short turn-around time" and "cost effective" basis.

Study Procedure

In particular, Avco was to utilize the experience and capabilities developed and demonstrated for the RAE-A Satellite, the Magnetic Storms Satellite and the Small Spinning Scientific Satellite. In applying this existing technology and experience to this study, particular attention was given to the operational characteristics of the basic subsystems involved with regard to their state-of-the-art, availability, reliability, cost, versatility, and applicability to various satellite concepts of which they might be a part. As a guide to these studies, supporting reference information on potential experiment requirements was given which was illustrative of typical experiments these subsystems might have to support. Frequent technical interchange meetings were held, and copies of technical data were transmitted to LRC as it was documented. Volume II of this report contains all of the Technical Releases (TR's) which were prepared during the study.

Scope of Present Work

The scope of this present work was limited by systems level and subsystems level assumptions formulated during technical interchange meetings between the study representatives from LRC and Avco, summarized below:

(1) The spacecraft concept shall be designed to accommodate an experiment demonstration not for operational utilization.

(2) A program concept shall be formulated to be cost effective over several launches, not necessarily the first launch.

(3) Sufficient initial analyses, hardware development and qualification shall be conducted to minimize changes and redesign for later launches.

(4) The Quality Assurance program approach shall apply selective utilization of NASA specifications to maximize the probability of data return from the experiment with minimum constraint on cost and schedule considerations.

(5) The reliability program shall maintain a qualified parts selection requirement, supplemented by design analyses where qualified hardware is not available.

(6) The launch vehicle shall be a standard four (4) stage Scout with optional utilization of the elongated heatshield designed for the five (5) stage Scout or the 42 inch diameter heatshield planned for future four (4) stage missions using the Algol III first stage.

(7) The list of experiments given with the contract schedule and Statement of Work (SOW) shall be used as a guide, not as a constraint. Other categories of experiments shall be considered, where practical.

(8) Earth pointing capability shall be the primary mode of operation, with horizon scan mode as a secondary consideration.

(9) Each spacecraft shall be designed to accommodate one experiment at a time.

(10) Multiple spacecraft per launch vehicle shall be considered where light weight and low power experiments result in spacecraft weights no greater than 175 pounds.

(11) All of the Scout launch sites shall be considered (Western Test Range, San Marco and Wallops Island).

(12) Nominally circular orbits, compatible with the Scout Launch Vehicle injection capability, between 200 and 1000 nautical miles shall be considered from each of the three launch sites.

(13) Experiment life shall range between three (3) weeks minimum and three (3) months maximum; the spacecraft hardware shall have operational capabilities up to a year in orbit, but any expendables used will have a capacity equivalent to the experiment lifetime.

(14) Subsystem hardware design concepts shall have high versatility to accommodate the anticipated variety of experiment requirements and to provide additional capacity for the unexpected operation of any spacecraft subsystems and/or the experiment.

(15) Subsystems hardware design shall be based on known workable hardware concepts, not necessarily completely qualified.

(16) "Off-the-shelf" components shall be used, where practical, but shall not limit design versatility. (Over several launches, the new development and qualification of a component may be more cost effective because of its overall mission flexibility.)

(17) Maintain a simple well defined electrical and mechanical experiment to spacecraft system (including Ground Support Equipment) interface for ease of integration.

(18) Spacecraft communications, command and data handling concepts and performance criteria shall be compatible with STADAN.

(19) The primary telemetry communications link shall be S-band with a VHF link for command and tracking.

(20) The spacecraft structural concept shall accommodate a variety of experiment hardware installations without requalification.

(21) Experiment sensors which require total spacecraft output data rates to exceed 200,000 bits per second may be accommodated by special "snap shot" playout techniques to demonstrate experiment feasibility.

(22) Power subsystem designs may be based on limited duty cycle operation of those experiments which require high power.

(23) The Ground Support Equipment (GSE) concept shall provide electrical and mechanical elements which require a minimum of changes between spacecraft configurations, and shall be capable of accommodating each experimenter's GSE. As an option, the spacecraft contractor's GSE may provide the capability to function as the GSE for the experiment.

Recognition of Similar Work

The contractor study team was specifically selected and assigned to this program because it was recognized that they had background and experience in similar work, that is, the application of guidance/control and data handling disciplines and techniques to spacecraft missions. For this study, this combination of analyses and hardware knowledge was uniquely applied to establish a spacecraft concept which can accommodate multiple launches of a variety of experiment missions in a cost effective manner with a minimum of hardware changes between launches.

Significance of the Material Treated

The depth of the investigation conducted in this study was not intended to produce a preliminary design solution for the selected spacecraft concept. Rather, known subsystems techniques and demonstrated spacecraft operations were to be reviewed and traded off in order to derive a feasible satellite concept capable of accomplishing the various mission goals. All areas of investigation will ultimately require various levels of detailed analyses and development, including possible flight test in the orbital environment, before the firm design can be established. The concept developed during the study is most significant because it meets all of the mission goals with a simple vehicle design, provides versatile subsystem capabilities for a variety of experiments, and utilizes known hardware approaches.

Acknowledgement

Acknowledgement is given to the principal contributers to this Summary Report (Volume I) and the Technical Releases (TR's) in Volume II.

J. E. Mozzicato	Attitude Stabilization, Control and Determination
K. Arnesen	Data Handling, Command and Communications
R. J. Campbell	Configuration, Design and Structures
F. W. Griebel	Systems Definition and Program Planning

W. C. Hailey	Attitude Stabilization and Control
R. Litte	Attitude Determination
W. J. Kubicki	Power and Communications
M. J. Miani	Configuration and Design
D. P. Fields	Mission Analyses and Orbital Mechanics
C. K. Wilkinson	Mission Analyses
E. J. Lawlor	Attitude Stability
G. W. Pfeiffer	Attitude Control
C. J. Favaloro	Communications

TECHNICAL DISCUSSION

Description of Baseline System Concept

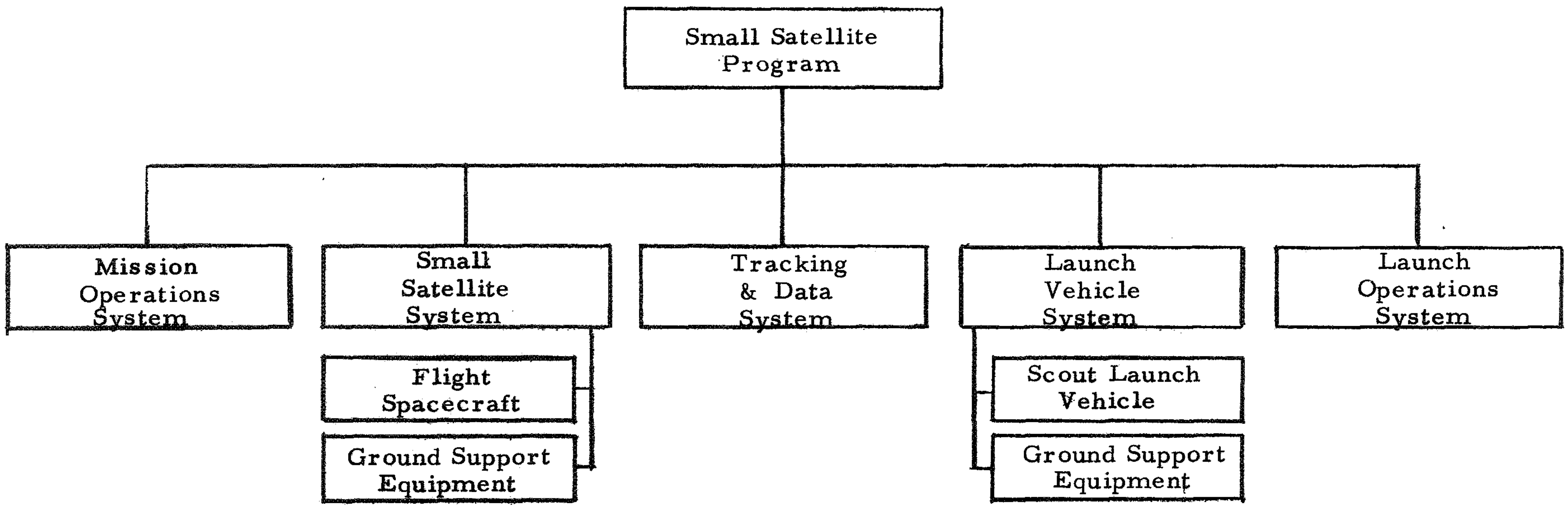
Bounded by the system and subsystem level assumptions stated above, a system definition activity was conducted. The initial step was to determine the mission requirements for the Small Satellite System. From this effort, it was established that; a) the mission elements will be as shown in Figure 1; b) the mission profile will consist of five phases, i.e., prelaunch phase, launch phase, injection phase, reorientation phase, scientific mission phase, and mission termination phase; c) the mission elements will impose major constraints on the design of the Small Satellite System. Definitions of each mission element, identification of their respective interfaces, definitions of each phase of the mission profile with a preliminary sequence of events for each mission phase, and a description of the mission constraints are presented in Volume II (Reference TR No. L910-FWG-70-5).

During the establishment of the mission requirements, continuous liaison and coordination was maintained with the technical design activities to gather the information required to develop a description of the baseline Small Satellite System concept. From this information, a hardware tree (Reference Figure 2) showing the flight spacecraft and the ground support equipment was constructed. Having identified the hardware in the reference design system, the functions of each hardware item were established, and a functional block diagram was produced to show the functional interrelationships between the various spacecraft elements and subsystems. A detailed identification of functional description of the reference system and its constituent equipments, subsystems, and assemblies are presented in Volume II (Reference TR No. L910-FWG-70-5).

In summary, the baseline flight spacecraft is a small, three axis stabilized, earth pointing satellite, designed to operate at orbits ranging from 200 to 1000 nautical miles, with polar to equatorial inclinations, for 3 weeks to 3 months. The selected spacecraft configuration is shown in Figure 3 and the functional interrelationship of the constituent subsystems are illustrated in Figure 4. The salient characteristics of this concept are as follows:

(1) The basic spacecraft will be a right circular cylinder approximately 30 inches in diameter by 36 inches long weighing between 150 and 300 pounds depending on the particular sensor and/or experiment payload and featuring a) a fixed location for the spacecraft support subsystems, b) a structural base suitable for a large, body mounted, solar array with

FIGURE 1 - MISSION ELEMENT IDENTIFICATION



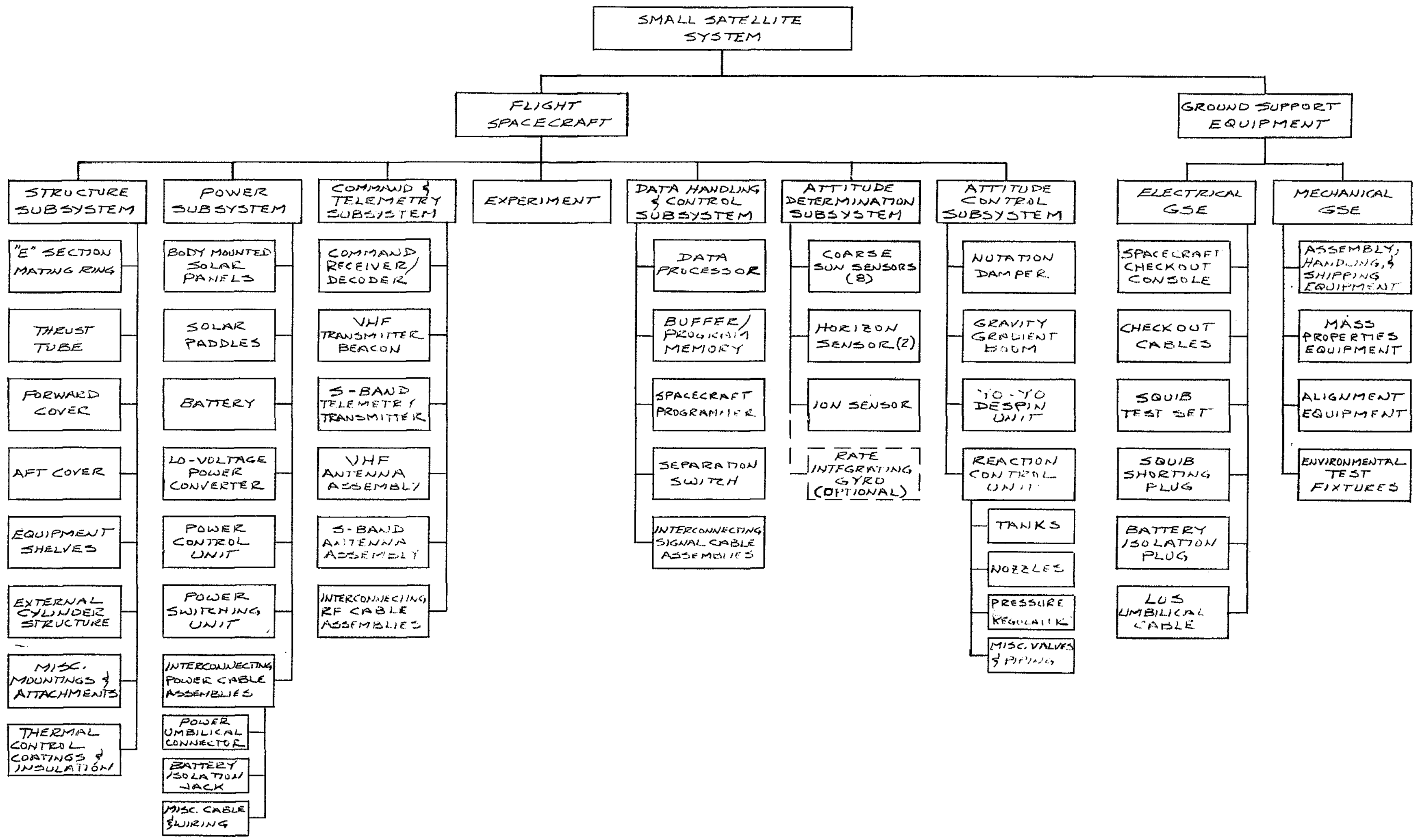
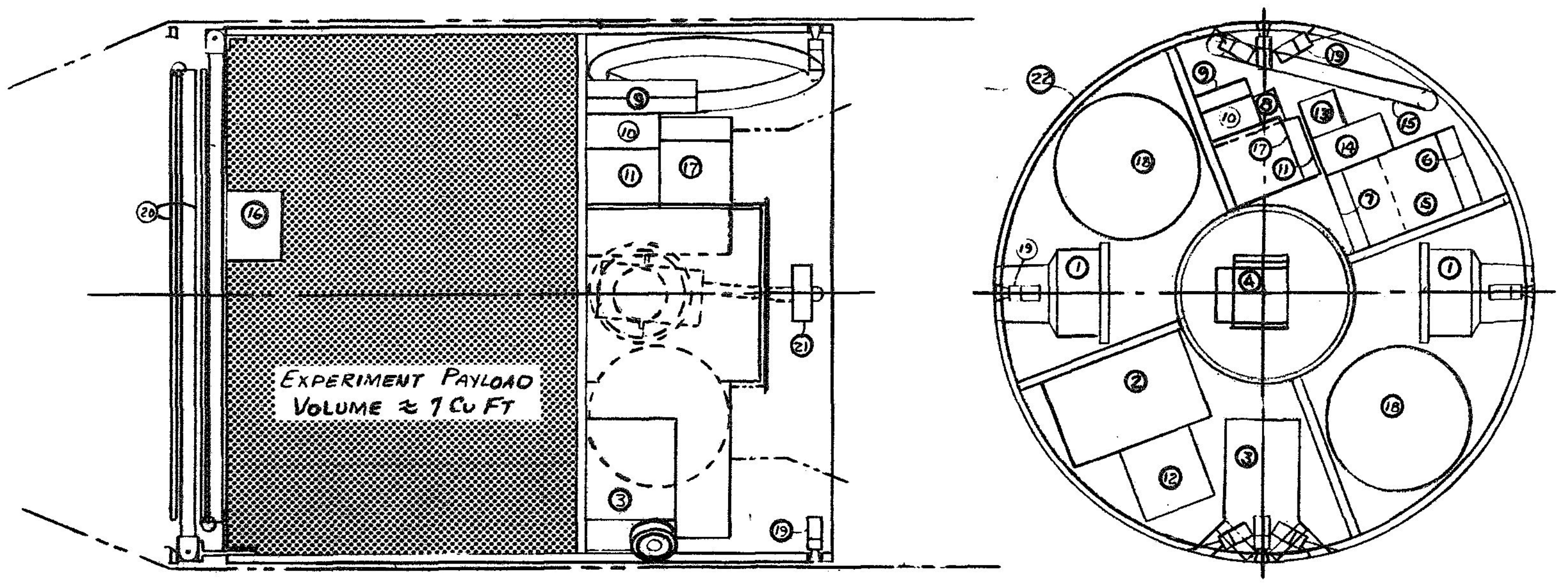


Figure 2 SMALL SATELLITE SYSTEM HARDWARE TREE

FIGURE 3 - BASIC SATELLITE WITH FORWARD SOLAR ARRAY



1. HORIZON SENSOR
2. HORIZON SENSOR ELECTRONICS
3. ION ATTITUDE SENSOR
4. GRAVITY GRADIENT BOOM
5. SPACECRAFT PROGRAMMER
6. PROCESSOR
7. MEMORY
8. VHF TRANSMITTER BEACON
9. S-BAND TRANSMITTER
10. COMMAND RECEIVER
11. COMMAND DECODER
12. BATTERY
13. POWER CONVERTER
14. POWER CONTROL UNIT
15. NUTATION DAMPER
16. S-BAND ANTENNA
17. ACS ELECTRONICS
18. ACS COLD GAS TANK
19. ACS NOZZLES
20. SOLAR PANELS
21. G.G. BOOM TIP MASS
22. BODY MOUNTED SOLAR ARRAY

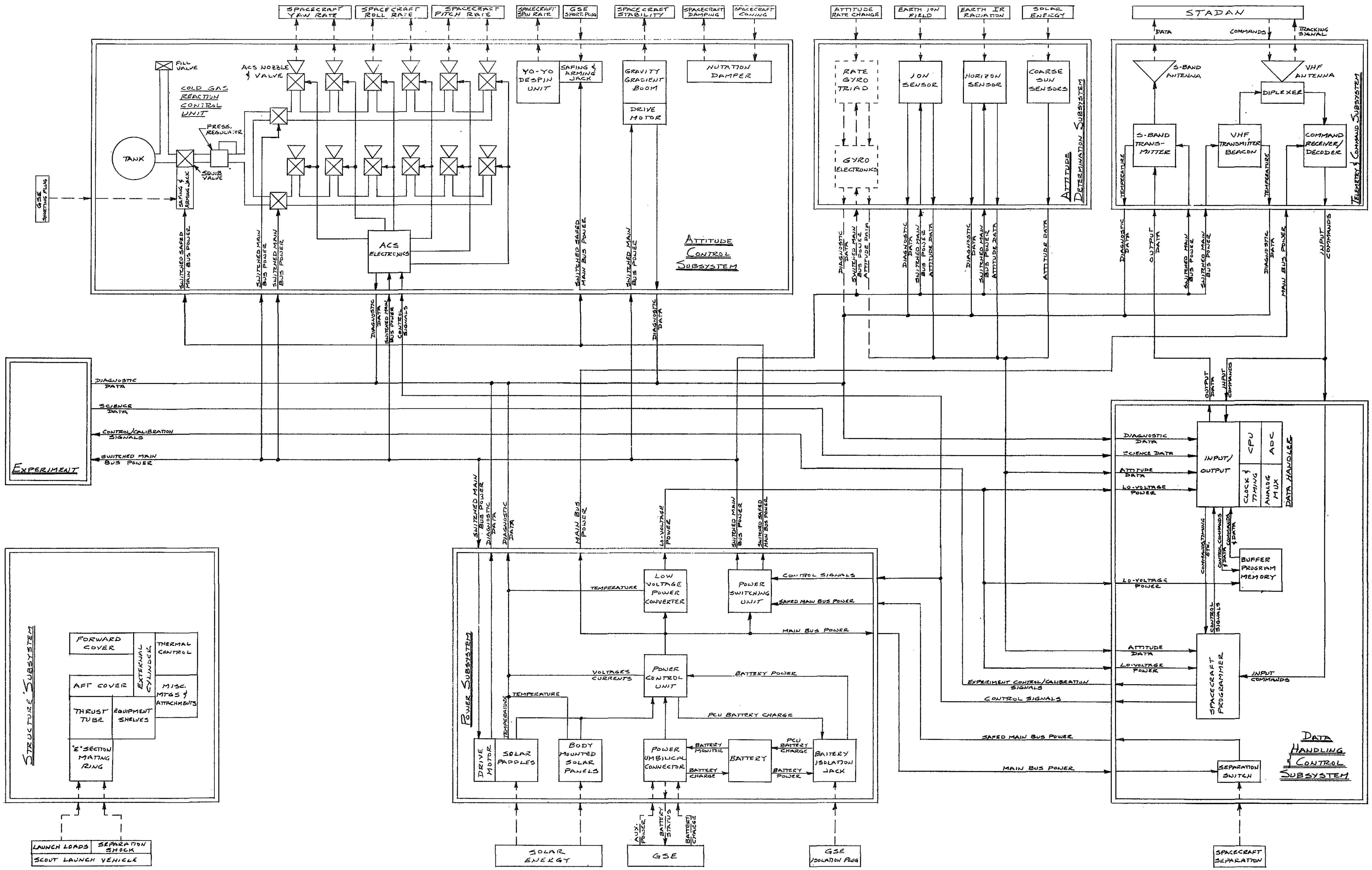


Figure 4 - FUNCTIONAL BLOCK DIAGRAM FOR A TYPICAL FLIGHT SPACECRAFT

provision for attachment of deployable solar paddles, c) approximately 7 cubic feet of unobstructed volume for "universal" mounting and integration of up to 100 lbs. of sensors and/or experiments within the structure, and d) provision for attachment of booms and/or boom mounted sensors and/or experiments as required.

(2) Spacecraft power of 20 watts minimum to 100 watts maximum will be provided by a combination of body mounted solar panels, deployable solar paddles, and Ni-Cd batteries.

(3) Spacecraft attitude will be determined by using a combination of sun sensors, horizon sensors, and an ion sensor. For missions requiring precise attitude rate information and control, a rate integrating gyro assembly will be added.

(4) Spacecraft attitude control will be accomplished with a nutation damper, a yo-yo despin unit, a gravity gradient boom, and a cold gas reaction control unit. This combination of equipments will provide an earth pointing accuracy of approximately 5° in the passive control mode, and will provide earth pointing accuracies of less than 1° and stabilization of 0.01° per second in the active control mode.

(5) Spacecraft communications will utilize both S-band and VHF frequencies. Tone digital commands from the ground network will be received at VHF frequencies. The spacecraft will have the capability to accept and handle a maximum of 70 commands. The spacecraft tracking signal will also be VHF. A common VHF antenna assembly will be used.

(6) The spacecraft data handling will have a baseline capability of processing up to 200K bits per second of data and of storing a standard 10 bit word. The utilization of modular construction will enable the spacecraft to process up to 1M bits per second and to store a 16 bit word.

(7) Spacecraft control will be achieved by a programmer which will sequence and control <u>all</u> spacecraft functions. This programmer will be completely reprogrammable and will function in any or all of the following modes:

a. Automatic sequencing to a pre-set program

b. Automatic sequencing as modified by command in flight

c. Command sequencing only

(8) Each spacecraft will be designed to carry one or, at the most, two sensors and/or experiments. A "standardized" experiment/spacecraft interface will be provided. Mechanically and thermally, the interface will be extremely flexible, such that a wide spectrum of experiment shapes,

sizes, mounting arrangements, and view ports can be accommodated. Electrically, the spacecraft will provide a) 28 volt D. C. main bus power to the experiment (Power regulation and conversion will be supplied by the experimenter), b) 6 volt D. C. control/calibration signals, c) diagnostic data circuits, and d) science data circuits. All data processing, formatting and handling will be done by the spacecraft.

The ground support equipment (GSE) will consist of the electrical and mechanical equipment required for assembly, checkout, handling, test, and transportation of the flight spacecraft during factory operations through the prelaunch mission phase. Preliminary identification of these equipments is presented in Volume II (Reference TR No. L910-FWG-70-5).

Feasibility of Baseline System Concept

The feasibility of the baseline system concept is predicated on the operability of its major subsystems, the data handling, command and communications, and the attitude control. The structural concept, although unique in application, is based on known structural design techniques and materials. Other subsystem hardware has been selected from known sources of space qualified components that have already demonstrated feasibility. The Ground Support Equipment approach is also in the same category of proven hardware for the application. Therefore, this discussion will be limited to a critique of the design approach of the major subsystems.

Feasibility of the Attitude Control Subsystem (ACS).- The attitude control concept selected as a baseline is a modular concept capable of operating in several modes. The control subsystem consists of a single boom, rigid body, gravity gradient configuration supplemented as the mission requires by an active mass expulsion system.

The gravity gradient configuration alone will be capable of maintaining the local vertical orientation of the satellite yaw axis to accuracies in the vicinity of $\pm 5°$. When supplemented with the mass expulsion system, the ACS will be capable of maintaining three axis attitude control in the local vertical orientation with accuracies of better than $\pm 1°$. The attitude accuracy and rate stability of the active attitude control subsystem is limited by the ability of the attitude determination subsystem to measure attitude and rates. For experiments where very accurate rate control (e. g. $<0.01°$/sec) is desired, a triad of rate integrating gyros used in the rate mode can be used in the ACS loop. In this fine rate control mode the system stability becomes limited by ACS thruster response and rate control of $\pm$ 0.001°/sec in yaw and $\pm 10^{-5}$°/sec in pitch and roll is easily obtained.

The acquisition phase, that is, the achievement of the local vertical gravity-gradient stabilized attitude from the conditions of the separation from the launch vehicle, can be accomplished in an open loop fashion. The satellite will be separated from the Scout launch vehicle with a high spin rate about the yaw axis. The spin vector is essentially in the orbital plane. A viscous fluid ring nutation damper, with a time constant of less than 6 minutes, will reduce the coning motion induced at separation to a negligible level. At a predetermined time in each orbit the yaw (spin) axis will be oriented very nearly along the local vertical. At this time, the vehicle will be despun with a "yo-yo" despin device. Immediately after despin, solar panels will be deployed and the gravity-gradient boom will be partially extended. A portion of an orbit later the boom will be fully extended to complete the "dead-beat" boom deployment process and leave the vehicle with the desired local vertical orientation. Figure 5 illustrates the acquisition phase.

In summary, the Attitude Control Subsystem permits the satellite to operate in the following modes:

(1) Open Loop Acquisition accomplished with a nutation damper, a "yo-yo" despin device and "dead-beat" gravity gradient boom deployment.

(2) Coarse Pointing Mode for local vertical orientation of the yaw axis to $\approx 5°$, accomplished with the single boom, rigid body gravity gradient configuration.

(3) Fine Pointing Mode to achieve three axis attitude stabilization to $< \pm 1°$ accomplished with an active mass expulsion system supplementing the gravity gradient configuration.

(4) Fine Rate Control Mode to achieve angular rate stabilization of less than $\pm .001$ °/sec in yaw and $\pm 10^{-5}$ °/sec in pitch and roll accomplished by the inclusion of a rate gyro triad in the active ACS loop.

The feasibility of this approach is demonstrated by both the successful operation of many rigid body gravity gradient satellites and is further supported by the calculations presented in the Volume II to this report. It is shown that a single 60 foot boom with a 5 pound tip weight will provide sufficient gravity gradient torque to overcome the maximum distrubance torque with an attitude error of 5°. An active mass expulsion ACS can be used to provide fine pointing control as long as the configuration is essentially rigid. The use of a boom with a diameter of from 0.5 to 1.0 inch will provide this rigid configuration. These calculations also demonstrate that a cold gas (nitrogen) mass expulsion system can provide sufficient total impulse (360 lb-sec) to achieve full three axis attitude control for the full mission length up to 90 days.

The key advantage to this attitude control concept are:

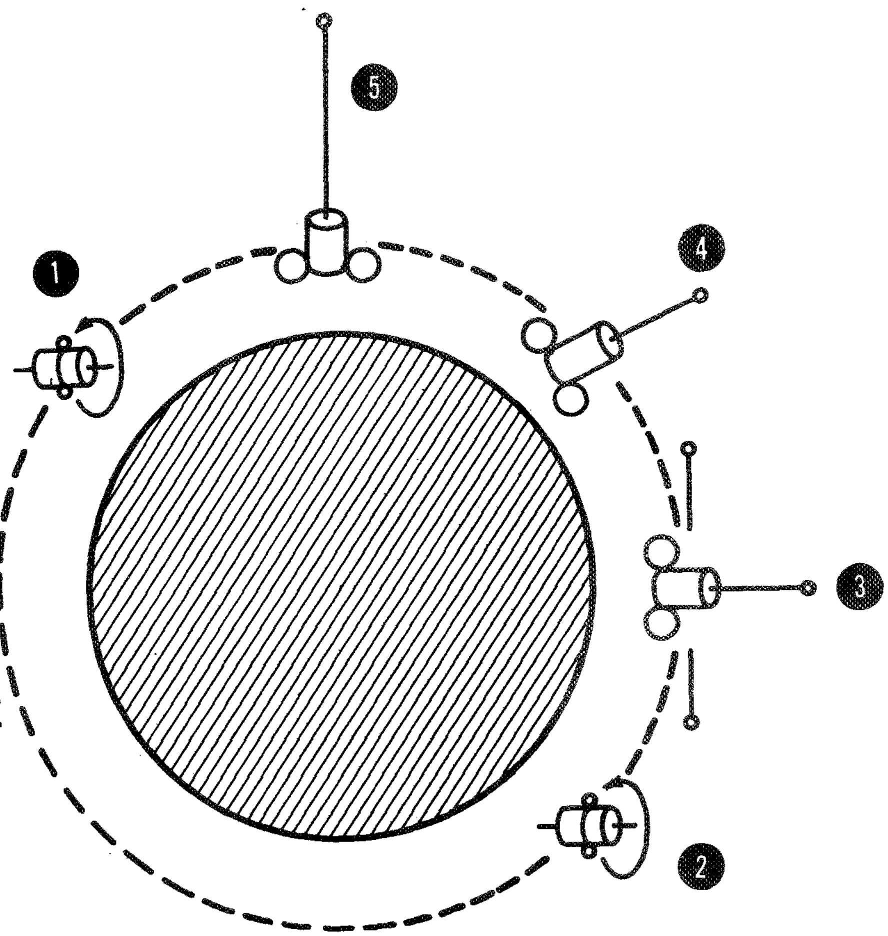
(1) The modular concept permits the selection of expendables and supporting hardware in discrete quantities compatible with the desired mission.

FIGURE 5 - OPEN LOOP ACQUISITION

1. SEPARATE FROM SCOUT
2. NUTATION DAMPING
3. YO-YO DESPIN

   DEPLOY PANELS

   50% BOOM EXTENSION
4. GRAVITY GRADIENT CYCLE
5. FINAL BOOM EXTENSION



(2) The acquisition phase may be accomplished in an open loop mode and within the first or second orbit.

(3) The gravity gradient configuration provides natural stability in the desired attitude.

(4) The gravity gradient configuration provides for a coast mode whereby the active ACS and the attitude determination subsystem may be shut down for long periods of time to greatly extend mission life and/or conserve electrical power.

(5) The versatility of the cold gas ACS permits direct ground control if modification to the mission profile is desirable in the event of unforseen circumstances.

(6) The rigid body configuration eliminates some of the instability problems which have been encountered on some flexible gravity gradient configurations.

(7) The reliability of cold gas attitude control subsystems for satellites has been adequately demonstrated.

(8) The entire control subsystem can be easily fabricated from low cost components which are well within the current state of the art, and places minimum demands on manpower, special facilities, and spares.

Feasibility of the Attitude Determination Subsystem. - The baseline attitude determination subsystem consists of a horizon sensor to determine pitch and roll attitude and an ion detector to determine yaw attitude. Coarse rate information can be obtained by differentiation of the attitude data. However, if very fine rate information is desired for rate control or to support experiment data reduction, a rate gyro triad can be added to the subsystem. Several coarse sun sensors will be mounted on the vehicle to permit determination of the solar aspect angle when the vehicle is not near the local vertical orientation.

This complement of flight instruments will provide direct on-board attitude determination with respect to the local vertical reference frame accurate to $\pm 0.1°$ in pitch and roll, and $\pm 1.0°$ in yaw. The gyro triad, when included in the system, can provide rate information down to $\pm 10^{-6}$ rad/sec.

All of the instruments used in this attitude determination concept are available and have been qualified in similar applications. The accuracies quoted represent this flight proven performance and, therefore, adequately demonstrate concept feasibility.

Feasibility of Data Handling, Command and Communications Subsystems. - The feasibility of the recommended design concept in the area of data handling, command and communications is based on the following reasons:

(1) The data handling concept utilizes tried and proven technology, which requires no technological breakthrough.

(2) Major components of the data handling subsystem are available nearly "off-the-shelf".

(3) The size, weight and power estimates for the subsystem are compatible with a small satellite concept.

The data handling design can be implemented using thick film hybrid circuits which have been available for a number of years, and have shown excellent reliability in operation. Two of the major components of the data handling unit have been implemented with hybrid circuits, the analog-to-digital converter (qualified for IMP-I (Eye)) and the memory (four have flown on satellites). In addition, very sophisticated processors, such as the autocorrelation computer which was flown on IMP-F, have been designed successfully with thick film hybrid circuits.

The command requirements for this satellite can be fulfilled using an off-the-shelf 70-command tone-digital subsystem, identical with units that have flown many times in the past.

In the communications area, more than adequate margins, calculated for a bit rate of 10^5 bits per second, can be attained at S-band using a very simple open ended circular waveguide antenna and a low power transmitter.

Similarly, more than adequate margin exists for the VHF beacon transmitter used for tracking purposes and in the command link. There are a variety of sources for space qualified hardware to accomplish this function.

STADAN compatibility has been investigated and can easily be assured for all of the data handling, communications and command subsystem approaches under consideration.

Spacecraft Subsystem Considerations

Attitude Control and Stabilization. - The attitude control studies were conducted in two major areas - control actuator evaluation and control system synthesis and analysis.

For the control actuator evaluation phase, the following set of system requirements and evaluation criteria were established to be used to compare various methods of attitude control.

(1) The satellite will be separated from the Scout launch vehicle with a high spin rate.

(2) The spin axis will be very nearly in the orbital plane.

(3) The orbits will range from 200 to 1,000 n. mi. in altitude and will have a maximum (3σ) eccentricity of 0.03.

(4) The maximum satellite weight will be 400 pounds.

(5) The maximum operating lifetime requirement will be three months.

(6) The satellite configuration will be a 30 inch diameter cylinder up to 36 inches long.

(7) The satellite shall be stabilized with the yaw axis (the axis of the cylinder) parallel to the local vertical. Yaw angle stability requirements are a function of experiment requirements.

(8) The pointing accuracy requirement of the yaw axis may range from $\approx 5°$ to $< 1°$ depending on the experiment.

(9) The angular rate stability requirement is a function of the experiment but in general it will have two ranges, coarse rate control ($\approx 0.01°$/sec) and very fine rate control (down to 10^{-5}°/sec).

All calculations were performed using the orbital reference frame shown on Figure 6.

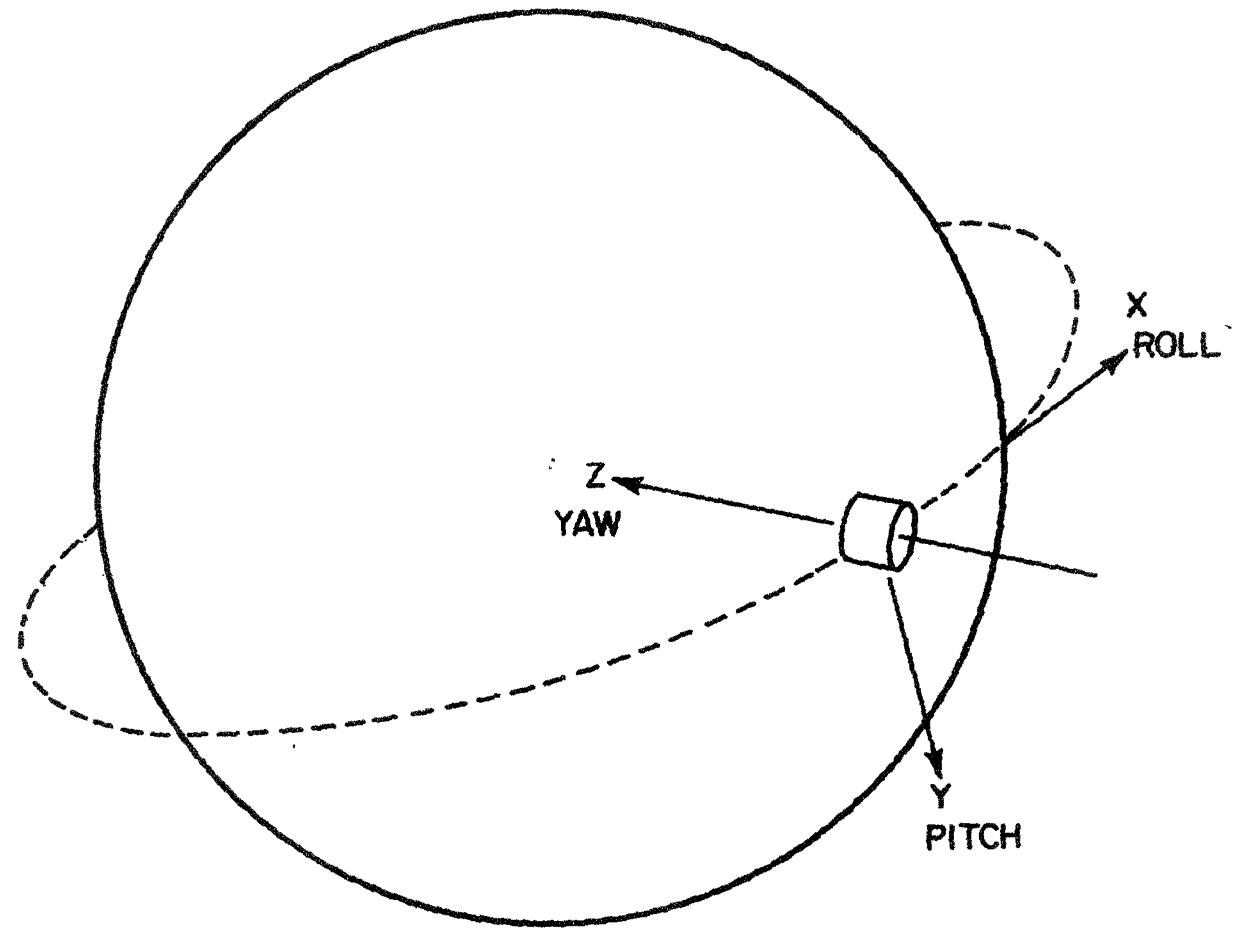
As a result of the actuator evaluation, a rigid body gravity gradient configuration with active cold gas attitude control was selected as the baseline subsystem. The subsystem can be used in a modular concept where only sufficient complexity to accomplish the particular mission need be used. Some typical configuration options are:

(1) A two axis passive gravity gradient configuration to provide $\approx 5°$ pointing accuracy in pitch and roll.

(2) A three axis passive gravity gradient configuration which is the same as the previous configuration but with inertial properties arranged to provide $\approx \pm 25°$ yaw angle stability.

(3) A two or three axis active mass expulsion configuration capable of providing pointing accuracy of $< \pm 1°$ in pitch and roll and $\approx \pm 1°$ in yaw.

FIGURE 6 - ORBITAL REFERENCE FRAME



(4) A three axis fine rate control configuration capable of providing the same pointing accuracy as the previous configuration but also capable of maintaining very fine angular rate control.

(5) A spin stabilized configuration can be easily achieved by not including (or not extending) the gravity gradient boom.

Another feature of the selected configuration is the ability to operate in a "coast mode" for long periods of time. For missions which take data for short periods of time spaced at long intervals, the ACS and the attitude determination subsystem may be deactivated during the periods when data is not being taken. This will result in substantial impulse and power savings and will greatly extend total mission life.

The selected subsystem was sized and analyzed to establish feasibility. The following technical releases, which are presented in Volume II, give the analytical results:

(1) F220-JEM-70-47, "Inertial Constraints on Gravity Gradient Stabilized Vehicles".

(2) F220-JEM-70-55, "Vehicle Distrubance Torques".

(3) F220-JEM-70-58, "Active Control of a Gravity Gradient Configuration".

(4) F220-JEM-70-60, "Sizing of Nutation Damper and Yo-Yo Despin Device".

(5) F220-EL-70-61, "Stability of Gravitational Systems".

Control Actuator Evaluation: A review of satellite attitude control subsystem actuation techniques has been performed to determine their applicability to small satellites. The results of the survey are summarized in Table I.

A significant need exists for small space platforms whose stabilization requirements can be adequately met by proven attitude control subsystem concepts. The concept of an "off-the-shelf" attitude control subsystem may never be literally achieved in practice, nevertheless, it appears that this goal can be approached for a large class of postulated missions.

Given a specific spacecraft/mission the question arises - "What type of attitude control should be used?" The obvious, and reflexive answer is - "The optimum subsystem." Unfortunately the range of potential candidate subsystems is so broad the choice is not always simple, obvious, nor invariant. Past endeavors have usually focused on the selection and

TABLE I ACTUATOR EVALUATION

PARAMETER	MASS EXPULSION	REACTION WHEELS	MOMENTUM WHEELS	MAGNETIC TORQUERS	SPIN STABILIZATION	DUAL SPIN STABILIZATION	GRAVITY GRADIENT W/DAMPING
System Weight: $\left(\frac{\text{Angular Impulse}}{\text{Actuator System Wt.}}\right)$	30 ft-lb-sec/lb system (Dry Nitrogen, Moment Arm = 1 ft)	0.083 - 0.43 ft-lb-sec/lb, Wt. range 4.8 to 19.5 lb.	0.3 ft-lb-sec/lb, Numbers vary widely - above is based upon Kearfott CMG for Agena-considered roughly representative of "available" hardware	F(coil diameter, material, power, orbital altitude, inclination, orientation, eccentricity, time, satellite attitude and position in orbit)	Very light	---	Variable-function of vehicle natural configuration, orbit, damping configuration and tip mass
Lifetime	Function of spacecraft ℓ_F/I; limit cycle amplitude and frequency; disturbance environment and spacecraft design	Reasonably good although no long lived systems known	Good	Function of power supply	Months to years	Good	>1 year possible
Precision ("Resolution")	Essentially sensor limited	Excellent/sensor limited	Excellent/sensor limited	Sensor, Environment limited	$< 1^{\circ}$ if desired WRT to inertial reference	Excellent/sensor limited	$\sim 5^{\circ}$ with (minimal augmentation) $\sim 1^{\circ}$ with (significant augmentation)
Accuracy ("System Noise Factors")	Minimum impulse bit reproducibility limited	Excellent/sensor limited	Excellent/sensor limited	Sensor, Environment Limited	Limited by mass property variances	Excellent/sensor limited	Design and environment limited
Dynamics Interaction	Minimum cross coupling although products of inertia and/or thruster misalignment may cause problems in high precision applications	Extensive cross coupling possible; degree of problem is a function of reference, orientations required, inertial properties. Fixed inertial orientations reduce severity of problem, e.g., OAO	Weak to strong (configuration)	Control axis cross coupling due to nature of control (field)	Weak to strong (payload may have flexible booms)	Medium to strong (payload and solar panels)	Relative to other techniques, more extensive analysis required initially
Orbit/Environment Interaction	None	None	None	Strong orbit interaction	Can despin (Eddy currents, radiation pressure)	Weak coupling-same similarity to plain spin	Strong-couples with orbit e, thermal, radiation and aero environment
Experiment Interaction	Gases expelled may cause problems in some experiments - solenoid valves have fields	Motor fields, primary mission attitude selected can affect duty cycles	Motor fields, primary mission attitude can affect duty cycle	Can be great in which case may not be applicable	Provides inertial orientation primarily	Motor fields	Provides local vertical orientation primarily
Acquisition Performance	Very good	Fair to good-depending upon separation conditions and satellite size	Fair to good-depending upon separation conditions and satellite size	Will align with $\vec{B}$	F (of Booster)	Fair, F (Booster)	Can be best of all systems
Steady State Performance	Excellent	Excellent	Excellent	---	Excellent	Excellent	Variable: Good to Excellent
Reliability	Substantially improved over "early days of ACS" - operation over many months now possible (Mariner)	Reasonably good	May be slightly better than reaction wheels (constant speed wheels)	Long life proven high	High	Reasonably good	Component reliability extremely high but system performance disappointing to date in some applications
Power	Low (Solenoids)	Motors (part time), desaturation system	Motors (continual), desaturation system	Torquing coils-high peak; average power is F(spacecraft, orbit, and mission)	Low	Motor (continual) plus desaturation system	Low
Miscellaneous Special Considerations	Force levels easily adjustable to attain desired torque/inertia	Desaturation system required	Desaturation system required	Desirable torque levels and direction may not always be available	Limited orientation (pointing) capability; requires torquers or synchronization	Must be considered as an Active Control System	Configuration is sensitive to orbit environment perturbations: Aero, Mag., Thermal, etc.
Torque Equation	Torque = $\dot{m}\, \ell_F V_E$	Torque = $I_R \dot{\omega}_R$	Torque = $d(I_r\, \omega_r)/dt$	Torque = $\vec{M} \times \vec{B}$	---	---	$T_y = 3\,\Omega_o^{\,2} (I_z - I_x) \sin\theta$ (small angles)
Typical Application	Mariner, Lunar Orbiter	OAO, Nimbus, OGO	ATM	Tiros	Explorer I, Syncom, IMP, Alouette I, Tiros	Tiros-M, SAS, OSO	1963-22a, Dodge; RAE, AF
Cost	Low	High	High	Low	Low	High	Low to Medium

design of an ACS for a specific mission rather than a class of missions; hence the measure of optimality has always been within well defined bounds. For the purpose of this program, it is desirable to consider the various criteria that might reasonably be employed in the determination of the optimality of an ACS in a general sense and then apply these criteria to the evaluation of techniques under consideration for a specific class of missions. Subsequent sections of this document present and discuss criteria for ACS evaluation; particular emphasis is then placed on these criteria as applied to a wide variety of candidate actuators for small satellite applications.

1. Criteria used for ACS Evaluation - Comparison of candidate ACS techniques requires the definition of terminology and standards to be used in evaluating the respective merits of each subsystem. The fundamental criteria for ACS evaluation employed in this study are defined in this section.

(a) Weight. Booster capability and the orbit desired set injected payload limits and, within this limit, weight allocations are assigned to the various subsystems. As a philosophical comment, attitude control subsystems are always too heavy hence the lightest subsystem meeting system requirements is usually preferred. However, this evaluation must be done in total system context and light weight per se is not a sufficient criterion. For example, light weight subsystems that consume significant fractions of available power may be deemed undesirable when evaluated in system context.

It would be fortunate if a single figure of merit were available to compare the efficiency of various actuators as a function of their total subsystem weight. Since no simple technique exists conducive to overall application, data are presented, where possible and applicable, for representative hardware some of which might be characterized as "off-the-shelf".

(b) Lifetime. As a minimum, ACS lifetime must equal that of the mission experiment. In spacecraft system operation, it must exceed that of the experiment, since during initial alignment, experiments may not be operational. Systems dependent upon mass expulsion as a primary actuation technique, or for desaturation, have exhibited good lifetime capability, for example, Mariner; however, passive techniques, for example, gravity gradient, inherently possesses the capability for extreme lifetimes. Possibly the best criterion for lifetime is simply whether or not an ACS is "adequate" to meet mission requirements.

Lifetime is strongly affected by satellite experiment duty cycle. Many experiments do not require accurate ACS during significant fractions of the satellite orbit simply because the phenomenon under observation is not within instrument view. The capability of an ACS to go into an off mission mode of performance can do much for increasing lifetime, optimizing system cost, reducing power subsystem requirements, etc.

(c) Volume. In a general sense, spacecraft design problems do not normally arise from ACS "bulk". Two possible exceptions to this statement can exist: 1) ACS employing mass expulsion can have a relatively low packaging efficiency factor, 2) the second exception arises from volumetric constraints placed on piggyback satellites because of volumetric limit considerations on the launch vehicles.

(d) Precision/Accuracy. Precision can be defined as the basic resolution capability of a subsystem and accuracy as the error associated with the attainment of that resolution. For example, in cold gas subsystems the irreducible minimum pulse size associated with a thruster affects the precision attainable in a subsystem incorporating that unit as an actuator; the pulse to pulse reproducibility of the impulse contained in a single pulse then influences the accuracy of the subsystem.

(e) Dynamics Interaction. The dynamic aspects of control may effect deleterious structural excitations (sloshing fuels, flexible appendages or bodies) and - almost always to some degree - entail cross axis coupling, for example, a control torque generated about a given body axis results in undesirable perturbative angular accelerations about other axes. These perturbative accelerations may arise from the presence of finite and unavoidable manufacturing and design tolerances such as misalignment of gas nozzles or products of inertia. The actuation technique employed may inherently contain or amplify cross-coupling effects. Usually these are known and may be design minimized; however, not without some penalty.

(f) Orbit/Environment Interaction. Actuation techniques such as magnetic torquing, gravity gradient, etc. depend on interaction with the orbit environment for the derivation of control torques. Schemes such as mass expulsion, inertia or momentum wheels are independent of the environment for the derivation of these torques; however, the actuator configuration, performance and capacity are still influenced by orbital parameters such as altitude, eccentricity, inclination, and orientation.

(g) Experiment Interaction. Missions involving fields and particle measurements preclude or modify the use or design of actuators that might produce contaminating particles or effects. Mass expulsion systems and magnetic torquing systems are typical examples of actuators that might produce contaminating particles or fields. (Of course subsystems other than the ACS can also contribute contaminants, for example, structural outgassing).

The reference frame required for the experiment, that is, local vertical or inertial, is of first order importance in ACS definition since it can strongly affect the amount of actuator impulse, power, and configuration selection of the spacecraft as well as ACS.

(h) Acquisition Performance. Acquisition of the desired reference subsequent to booster separation frequently necessitates subsystems not nominally necessary for steady state control requirements. Ideally, it is desirable to integrate this subsystem capability with the ACS used for steady state; this is occasionally possible with the sensor portion of the ACS but not always possible with the actuator elements, particularly where spacecraft separation from a rapidly spinning booster occurs. Costly failures have resulted from acquisition subsystems that did not perform as intended.

(i) Steady State Performance (Response and Stability). After transients associated with reference acquisition have been damped, the steady state subsystem behavior is of interest. In ideal perturbation free environments, the steady state behavior (limit cycle frequency and amplitude) of a stable subsystem is fixed by the system features that determine precision and accuracy. When, however, perturbative accelerations are considered, steady state performance is then a strong function of both spacecraft design, the actuation technique employed and its interaction with the orbit environment and/or inter-axis coupling.

Those actuation techniques deriving control torques from interaction with the environment will potentially be most susceptible to variations in the environment. Both magnetic torquing and gravity gradient techniques are typical of such subsystems with the latter exhibiting the greatest capacity for interaction with its environment. (Inherently this is simultaneously the greatest virtue of gravity gradient techniques). Design can minimize undesirable effects; however, analysis must be careful, thorough, and complete. Attention must be directed toward error sources such as orbital eccentricity, aerodynamics, radiation pressure, thermal effects, residual magnetic moments, and structural flexibility.

(j) Reliability. Every class of actuation system known to have been flown has exhibited failures of some type. The spectrum of causes for these failures has ranged from known deficiencies in design and poor quality control to environmental unknowns. Perhaps the best index of reliability is the past performance of similar systems. In general, potential problem areas are reasonably well defined now as a result of both military and NASA space programs.

(k) Power. Although power requirements for actuators significantly differ, practically all techniques require power at some phase of operation. Passive concepts such as spin stabilization occasionally require spin axis precession; this can be effected through magnetic torquing or mass expulsion both of which require some power. Gravity gradient concepts may require power for boom extension, and if active damping is necessary, torquing coils, wheels, etc. will also consume power (See Ref. 1).

When applied as a criterion, the power requirements must be evaluated in application (system) context. Small satellites have a tendency to be power limited rather than weight limited, therefore, "large ACS power requirements" assume dominating importance, particularly if these requirements result in a quantum jump in spacecraft design cost or complexity, that is, body mounted solar arrays to deployable arrays. In this context "large ACS power requirements" may be a number on the order of five to ten watts continual.

Power cost is another factor that must be considered; one effective watt of power output from a body mounted solar cell system costs approximately \$3,000 to \$5,000 for an earth orbiting satellite when considering all system factors, such as occultation, temperature, design and test. In the preliminary cost evaluation of an ACS, power cost is frequently not considered as part of the ACS cost.

If satellite lifetimes are sufficiently short, the actuator power budget may be satisfied by batteries. Extended lifetimes will require both solar cells and batteries; this combination will probably prove cost effective for a significant fraction of this decade.

(1) Cost. Subsystem costing has always been an area of controversy. Items characterized as "off-the-shelf" and suitable for application all too frequently are found to require "slight" modifications that result in substantial price increases. Other considerations can yield cost components that frequently transcend in magnitude those attributable to optimistic hardware quotations. Typical examples are the cost of power mentioned earlier, special facilities necessary for test of sophisticated hardware, the level and availability of technical personnel and facilities necessary to analyze, design, install, maintain, checkout, and monitor flight performance of the equipment.

2. ACS Evaluation - A wide variety of ACS configurations have been flown; generically they fall into the categories of active, passive, or hybrid systems. Differentiation between active and passive is largely based on whether the control torque (or stabilizing quality) is explicitly generated by spacecraft subsystems (active) or implicitly generated (passive) by either spacecraft design, that is, gravity gradient; or by spacecraft operational mode, that is, spin stabilization. Table II summarizes typical techniques that have been employed in various systems to accomplish stabilization.

TABLE II

TECHNIQUES TO ACCOMPLISH STABILIZATION

TYPE	TYPICAL SYSTEMS
Active	Mass Expulsion, Magnetic Torquers, Wheels (Inertia, Momentum)
Passive	Gravity Gradient plus passive dampers Spin Stabilization plus nutation dampers
Hybrid	Gravity Gradient plus mass expulsion, wheels, or magnetic torquers for libration damping and attitude trim Spin Stabilization plus mass expulsion or magnetic torquers for angular momentum orientation. (Dual Spin, a special class of spin stabilization, may also employ wheels in addition, for example, TIROS-M).

(a) Mass Expulsion. For small total impulse requirements, the simplest and most widely used mass expulsion subsystems utilize cold gas, typically Nitrogen, although Argon, Freon 14 and others have also been used. Both active and hybrid ACS, for example, Mariner IV (Active), OSO series (Hybrid-Dual Spin), and Lunar Orbiter (Active), have employed mass expulsion in torquing systems. Cold gas is extremely attractive not only for its simplicity and attendant economics but also for its high reliability. In addition, power requirements are minimal.

As total impulse requirements increase, other techniques offer weight advantages over cold gas. For example, an ammonia resistojet can yield over twice the specific impulse of a cold gas; however, the system consumes 6-10 watts per millipound of thrust above that required for operation of an equivalent cold gas system (Ref. 2). In addition, the cost of the resistojet is significantly greater than a cold gas nozzle.

Another option available is the liquid storage of cold gas in which the propellant is charged and stored in the liquid phase. Heat addition is required for vaporization prior to its use as a working fluid. Clever design engineering can probably develop a significant percentage of the thermal input necessary for vaporization from the spacecraft itself, without a significant weight or power penalty. Ammonia is a typical propellant for this approach. Estimates of subsystem weight requirements for a small satellite application with a total impulse capacity of 1, 000 lb-sec are: 25-30 lb for ammonia versus 35-40 lb for a cold gas nitrogen system.

If weight or volume are critical parameters in a given spacecraft design, consideration can be given to liquid storage cold gas or resistojet subsystems; however, for the small satellite under consideration in this program, with total impulse requirements less than 1, 000 lb-sec, cold gas systems appear to offer substantial advantages.

(b) Reaction Wheels. Reaction wheels may be used for single axis or multi-axis control (Ref. 3, 4). The fundamental actuation principle arises from the ability to change the magnitude of the rotor wheel angular momentum vector. Momentum wheels, effect control torques by controlled precession of their angular momentum vectors. Simply summarized, the difference between reaction wheels and momentum wheels is that the former technique employs variable speed rotors.

A hybrid technique, referred to as momentum bias, is presently receiving attention particularly for application to dual spin vehicles. Essentially the momentum bias wheel is a reaction wheel designed to operate about a specific wheel speed, other than zero. The wheel speed may then be varied about the operating speed point to effect torques in a limited wheel mode. The combined capability of the hybrid subsystem permits two degrees of control freedom - that associated with spin stability and that of a single axis reaction wheel. The problem of inter-axis coupling bears similarity to a plain reaction wheel.

A further sophistication of the momentum bias approach integrates a horizon sensor with the spinning rotor. The optical/mechanical horizon scanning function is provided by the spinning rotor. A significant drawback to the technique is that it strongly limits maximum rotor speed since the optical detector electronic time constants are too large to cope with signal frequencies associated with high rotor speeds. In effect this limits the momentum storage capacity of the subsystem, resulting in more frequent desaturation than might otherwise be necessary.

A variety of "off-the-shelf" reaction wheels are available. Representative wheels manufactured by Bendix have an angular momentum storage capacity of 0. 083 to 0. 43 ft-lb-sec per lb for single axis units, with corresponding weights of 4. 8 lbs to 19. 5 lbs, excluding power (Ref. 5). Power requirements for a three axis actuator system can span five-ten watts for average steady state operation to peak (short term) requirements of fifty watts.

When a three axes reaction wheel subsystem is used to stabilize an orbiting satellite along the local vertical, the net system angular momentum vector - in the absence of external torques - must remain constant with respect to inertial space. If some momentum is stored in each of the wheels, wheel speeds must be changed continuously to maintain the system angular momentum vector constant with respect to inertial space. If significant secular torques are encountered, for example, aerodynamic, the subsystem becomes saturated and desaturation is necessary to unload the wheels; possible candidates for desaturation are: mass expulsion, gravity gradient, or magnetic torques.

A typical example of the ideal application of reaction wheels is the Orbiting Astronomical Observatory. OAO is subjected to minimal amplitude environmental perturbations, its orientation requirements are inertial and the secular torques are low; these result in low amplitude - low frequency duty cycles and infrequent desaturation for the reaction wheels. Extreme pointing precision performance is attained with star trackers as attitude sensors. However, for a small satellite application such as this study, cost, weight complexity, reliability and other considerations combine to make reaction wheels unattractive.

(c) Momentum Wheels. Early Avco programs in ACS research (Ref. 6) covered both theoretical studies and the development of momentum wheel subsystems; comparisons were made between reaction wheels and a double-wheel precessible momentum subsystem. The ideal application for momentum wheels, the studies indicated, would be "...... future systems with rotating machinery, rotating telescopes, changing center of mass, etc.". This early conclusion is still applicable.

An extensive discussion of momentum wheels for three axis control of small satellites is unjustified since these systems are generally heavy, consume more power, and are more expensive than reaction wheels. The momentum bias approach is considered a hybrid system rather than a momentum wheel and its application is discussed under dual spin stabilization.

(d) Magnetic Torquers. Spacecraft interaction with the magnetic field of the earth can produce torques of useful levels. The principle of operation is simple; a current carrying coil in a magnetic field seeks that orientation which will maximize the flux through it.

Although use of the field as a primary actuation technique for real time active control appears attractive, a major problem eliminates it from serious consideration except for two unique cases: 1) a synchronous orbit satellite for which the magnetic field vector, ($\vec{B}$), is invariant relative to a local vertical reference frame, and, 2) a particular mission where continual alignment with the $\vec{B}$ may be necessary or desirable. For the orbital conditions (< 1,000 n.m.) and reference frame (local vertical) of primary interest in this program, the magnetic field vector when expressed in the rotating local vertical reference frame, will exhibit significant periodic variations that are a function of orbital altitude, inclination, orientation, eccentricity, and time (since the magnetic field axis is not coincident with the spin axis of the earth). It is conceded that knowledge of $\vec{B}$ may be transmitted to the satellite or determined by satellite instrumentation; however, the availability of a "useful $\vec{B}$" at the instant it is required, e.g. to offset a steady state disturbance during a fine pointing phase of the mission, would be purely coincidence. "Useful $\vec{B}$" in this context implies both orientation and magnitude of the $\vec{B}$, that is, a $\vec{B}$ orientation that will permit a torque to be generated about the desired axis with minimum inter-axis coupling and simultaneously a $\vec{B}$ magnitude that will not necessitate extreme amounts of power to generate an adequate torque level. As a result of these disadvantages, magnetic torquing as a primary actuation technique for real time active control, was not considered for this program concept.

(e) Spin Stabilization. The principle of spin stabilization has been employed for a number of satellites whose performance has done much to achieve public recognition of the importance and utility of satellites. The TIROS/TOS and Syncom series typify such spacecraft wherein the ACS is characterized by what may be deemed as stark simplicity. Although the ACS employed for these spacecraft completely satisfied their mission requirements, they can not adequately meet the earth pointing requirements of this program.

(f) Dual Spin Stabilization. Dual spin stabilization is receiving increased attention for earth pointing satellites. The concept is not new; the OSO series (Ref. 7) was an early example of the eminently successful application of the principle, but its mission objective was solar orientation and not earth orientation. A significant and critical difference exists between spin stabilization as employed for TIROS/ITOS and the dual spin systems of OSO or TIROS-M. TIROS/ITOS and Syncom satellites were spun as entities and no differential spin rates existed for separate parts of the spacecraft. The TIROS/ITOS and Syncom satellites were truly passively stabilized whereas dual spin spacecraft must be classified as actively stabilized, with all of the requisite subsystems for active stabilization, such as error sensors and high duty cycle control actuators. The control actuators on TIROS/ITOS and Syncom were necessary for spin axis precession and not for ACS stability; the associated duty cycle requirements are extremely low when compared with OSO or TIROS-M.

The TIROS-M class of dual spin stabilization has several disadvantages which preclude it from being recommended as the preferred approach for this program at this time. These are: 1) Power requirements for control are high-momentum bias maintainence and torquing necessitate several watts continual power, 2) A three axis auxiliary actuator is required to precess the spin vector and to desaturate the rotor wheel, 3) If a horizon scanner is integrated with the momentum wheel, maximum rotor speed is strongly limited (scanner electronic time constant) and a larger rotor moment of inertia (ramifications - size and/or weight) is necessary to achieve the necessary angular momentum capacity, 4) Steady state performance, that is, response to disturbances in low altitude orbits, particularly for the roll and yaw axes, will be determined by the desaturation technique - if mass expulsion is used the performance will be good - if magnetic torquing is used the subsystem will not have a "real time" capability, 5) Subsystem cost will be high, 6) No strong possibilities of fail safe modes of operation, and 7) Acquisition performance is complicated.

(g) Gravity Gradient. The gravity gradient (GG) technique for satellite ACS has been effectively employed on both military and NASA satellites. Early applications (military) involved the use of appropriately designed rigid body satellites. Subsequent to Kamm's proposal (Ref. 8) to use extendible booms to attain the necessary moment of inertia configurations, increased inpetus was given to the GG stabilization of small satellites. The resultant performance record of satellites stabilized with extendible booms has been spotty and, on occasional instances, disappointing, that is, considering the fact that this technique was held, by many, to be the universal solution to the ACS problem for local vertical oriented satellites.

In fairness to the technique and with the benefit of some hindsight, a review of the literature, Ref. 9, 10, and 11 will reveal that both analytical and design problems did not always receive the amount of careful attention necessary to achieve success. For example, as early as September 1963, Kershner, Ref. 12, cited thermal excitation of booms as a source of potential problems based on flight data from TRAAC (1963 22A) yet subsequent poor performance of later systems incorporating booms can apparently be directly attributed to thermal excitation.

The employment of GG for ACS does require some control augmentation, usually in the form of dissipative damping, to achieve satisfactorily small attitude oscillatory envelopes. Rigid body GG stabilized satellites have employed a combination of wheels and mass expulsion to provide both damping desaturation, and a moderate attitude trim capability in the presence of small secular torques. Flexible body GG stabilized satellites, flown and proposed, incorporate a broad spectrum of "dampers" ranging from the simple lossy spring of TRAAC, Ref. 12, to the SAGS system, Ref. 1, developed by TRW Systems for the Goddard Space Flight Center. Between these two extremes are to be found many techniques, such as 1) the General Electric MAGS system, Ref. 13; it dissipates excess oscillatory energy through a viscous media interface between the satellite proper and a permanent magnet aligned with and locked onto the magnetic field vector, and 2) variations on Kamm's cross axis damper, Ref. 9; proposed by Tinling, Hartbaum, et al, Ref. 14 and 15.

Probably the most successful flexible body GG three axes stabilized satellite is RAE-A. Avco's primary participation in that program was to formulate the digital simulation program, Ref. 16, 17, 18 and 19, which led to the accurate and extensive design and performance analysis requisite to program success. However, even RAE-A has not exhibited performance of less than one degree pointing accuracy although its performance has transcended initial expectations.

Passive damping alone, will not yield adequate system performance for the selected program requirements. The effects of orbital eccentricity and aerodynamics constitute forcing functions that yield both attitude errors and attitude rates which exceed system tolerances and which passive damping can not adequately counter. Active damping techniques are necessary and, with proper system design, will yield the desired performance of less than 1°. The spacecraft dynamic configuration should, to first order, resemble a rigid body to minimize dynamic interaction of the active "damper" with the boom(s). In addition, the boom(s) should be of adequate bending and torsional stiffness, plus include design features that will minimize thermal gradients and their attendant effects (Ref. 20). None of these requirements is particularly difficult to obtain. The RAE-A boom design was successfully aimed at attainment of two of these features, torsional stiffness and thermal gradient minimization.

The weight of an augmented GG system will depend on the satellite configuration and orbital environment, with the latter consideration of dominating significance. The major problem, for low altitude orbits $\sim$200 n. m., is the attainment of neutral aerodynamic stability, so that the secular torques resulting from aerodynamic forces will not necessitate large control torques for the maintainence of desired trim attitudes. Minimization of these torques will simultaneously reduce the total impulse thereby reducing the size of the system required for desaturation or control.

Control Subsystem Syntheses and Analyses:

1. Nature of Disturbances - Earth orbiting satellites are subject to disturbance torques which tend to perturb the vehicle motion. An analysis of these disturbances as presented in Volume II, F220-JEM-70-55, results in the following findings. Aerodynamic pressure is the most significant disturbance at low altitudes. For orbital altitudes less than 350 n. mi., it is highly desirable to aerodynamically balance the vehicle; that is, the external configuration of the vehicle should be such that the nominal location of the center of pressure coincides with nominal location of the center of mass. At altitudes above 400 n. mi., solar pressure and magnetic torques become more significant than aerodynamics, but they are not significant in an absolute sense. Orbital eccentricity is a significant disturbance at any altitude for a gravity gradient stabilized system. The

assumed maximum eccentricity of 0.03 3 sigma will induce a pitch error of ± 1.7° 3 sigma in a passive gravity gradient configuration. Another disturbance which is peculiar to vehicles which have long booms is the deflection or flutter induced by thermal gradients in the boom. The use of booms with interlocked edges, coatings, and perforations has greatly reduced this problem. The short (60 ft) boom suggested for this application should not exhibit strong tendencies toward thermal deflection.

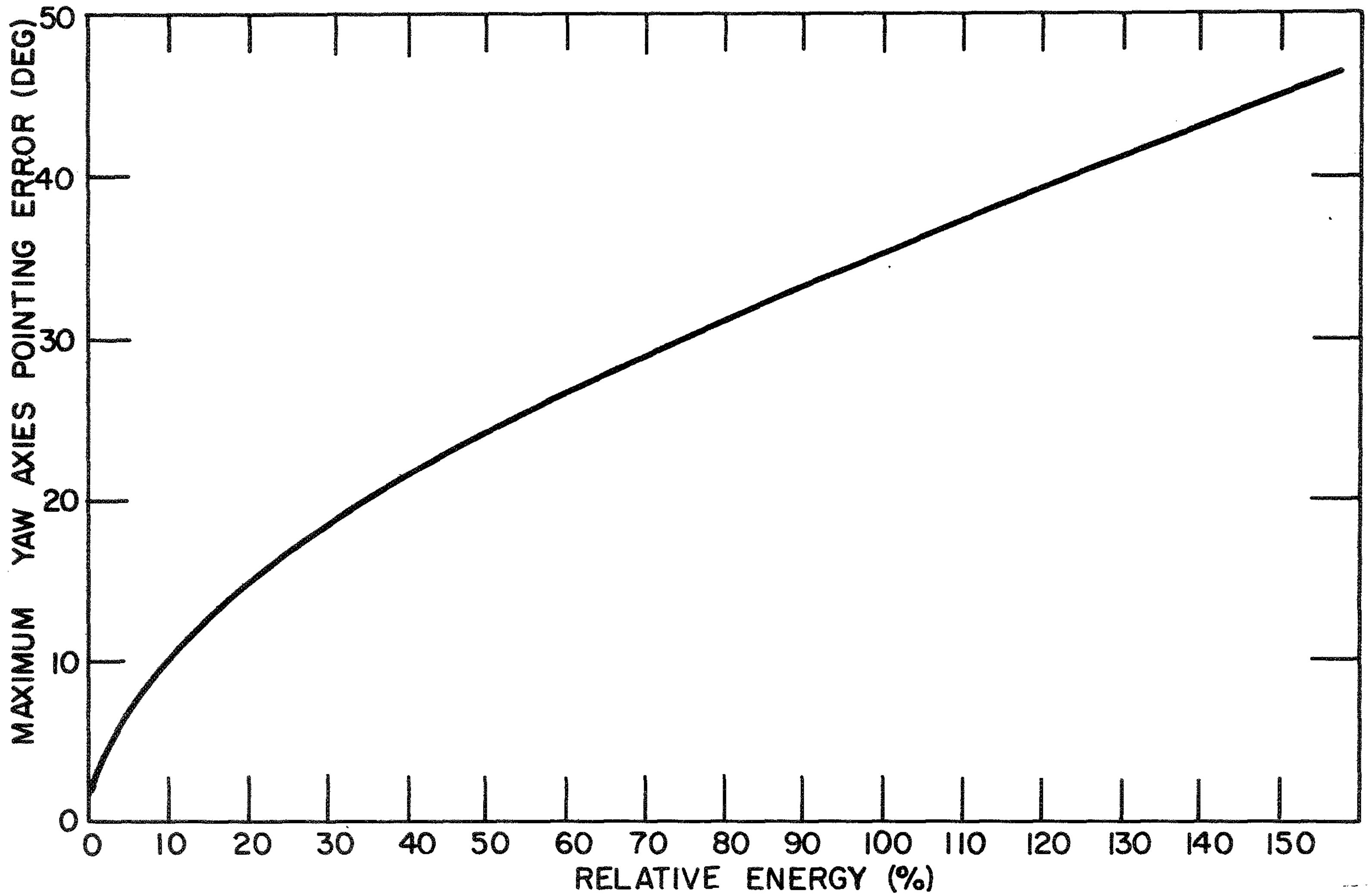
2. Passive Stability of Gravity Gradient Satellites - The development of the equations for gravity gradient torque and the inertial constraints for system absolute stability are presented in Volume II, F220-JEM-70-47. An analysis of the stability of gravity gradient systems is presented in Volume II, F220-EL-70-61. In the discussion of the disturbance torques (F220-JEM-70-55), it is shown that the maximum aerodynamic torque can be countered with the gravity gradient torque produced by a 60 ft boom with a 5 pound tip weight at a pitch angle of 5°.

The passive gravity gradient satellite will exhibit several types of attitude error. The static trim which is a constant bias of one axis, for example, a pitch trim of 5° or less, required to offset the secular aerodynamic torque. The transient oscillation is a result of energy which has been introduced into the system as the result of some previous event. This oscillation will be damped either by the natural damping of the body, by some intentional passive damper, or by an active damping subsystem. The steady state oscillation is the result of some continuous cyclic stimulus. Oscillation of this type is illustrated by the ± 1.7° pitch oscillation induced by the 0.03 orbital eccentricity.

The only significant secular torque encountered by the baseline satellite is the low altitude aerodynamic torque. This torque has been used to establish the moment of inertia required for a system with a 5° static trim in pitch. The static trim angle is essentially inversely proportional to the maximum moment of inertia so that increasing the moment of inertia by a factor of two would reduce this static trim to 2.5°.

The causes of transient oscillations are the initial conditions at gravity gradient acquisition or occasional random inputs such as meteorite impact. The analysis shows that meteorite impact is a very unlikely disturbance; so the magnitude of the transient oscillations is dependent virtually on initial conditions and the damping available. The passive stability analysis relates the size of satellite oscillations to the energy of angular rotation. For example, if the satellite has zero angular velocity and zero attitude error relative to the orbital reference frame, its relative energy is said to be zero; and if the satellite has zero angular velocity with respect to an inertial reference frame and zero attitude error with respect to the orbital frame, its relative energy is said to be 100%. Figure 7 shows the maximum oscillatory pointing error of the yaw axis as a function of relative energy. This curve shows that if the satellite were inertially stabilized and allowed to acquire the gravity gradient orientation from that condition, it would have a relative energy of 100% and would have potential maximum pointing errors

FIGURE 7 - OSCILLATORY POINTING ERRORS



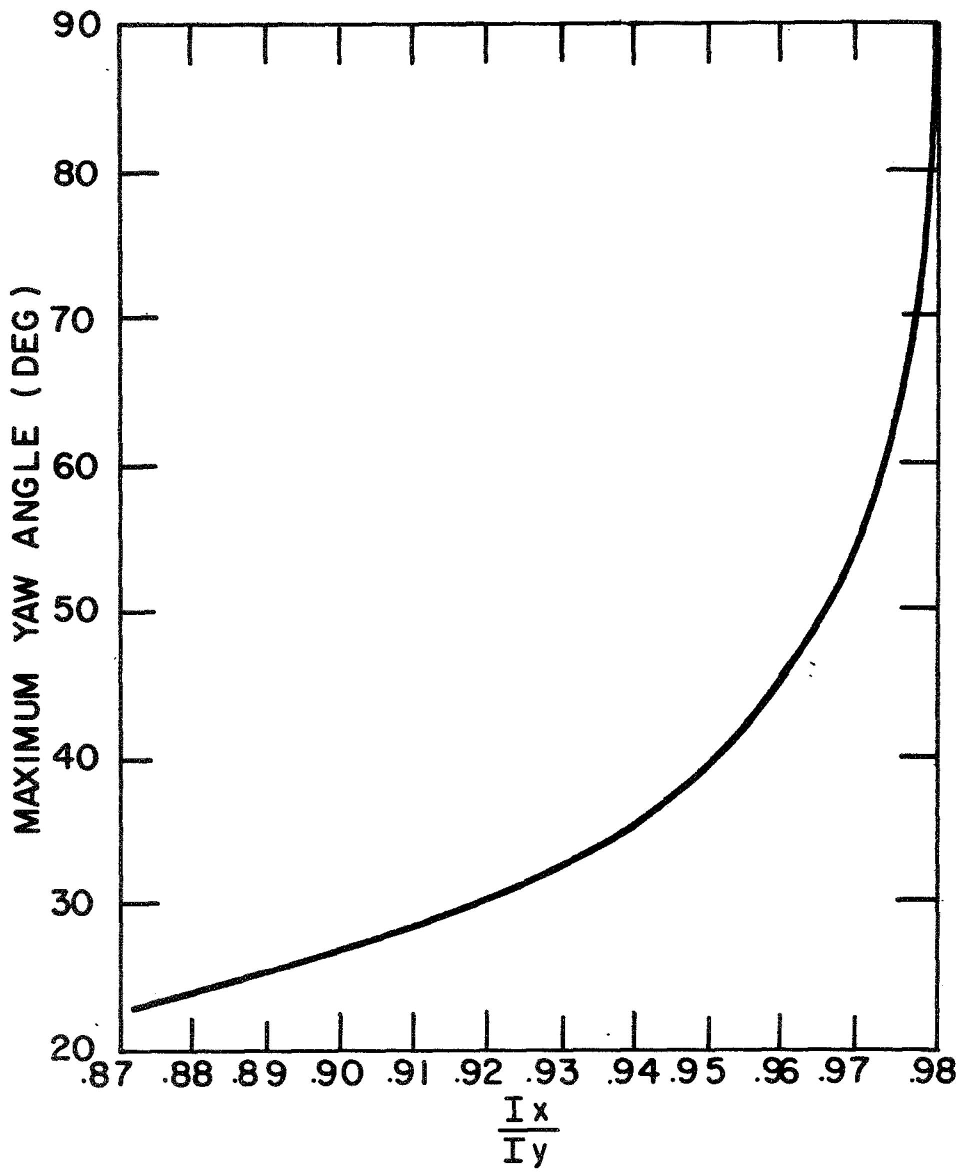
of 35°. If, however, a boom deployment technique could be used which results in a lower residual energy level, the resulting angular error would be lower. A boom deployment technique known as "dead-beat" deployment was used on RAE-A satellite. This technique resulted in very little residual relative energy and, therefore, produced very low pointing errors ($\approx$5°). In "dead-beat" deployment, the boom is extended in two or more steps phased to yield a final condition of zero relative energy. This procedure was used three times on RAE-A with very good results. It is estimated that for the rigid body configuration proposed here, this deployment method could reduce the relative energy to less than 10%, thereby producing a maximum yaw axis pointing error of 10°. It is also estimated that using analytical and simulation techniques developed for RAE-A; taking into consideration the uncertainty in initial conditions and the effects of aerodynamic disturbances during the deployment, a relative energy level as low as 2% could be achieved by a pre-established multiple step boom deployment technique, thereby yielding a maximum yaw axis pointing error of 5°. Any greater accuracy requirement would require an active ACS.

A configuration which is symmetric about the yaw axis (I_x = roll moment of inertia = I_y = pitch moment of inertia) has no (or neutral) yaw stability; that is, it may drift about the yaw axis or take on any yaw attitude. Yaw stability can be induced by making the vehicle unsymmetric about the yaw axis such that $I_y > I_x$. Figure 8 shows that it would be difficult to achieve passive yaw stability better than +30° and that if very good yaw stability is desired, an active ACS would be required on the yaw axis.

3. Active Attitude Control - The problem of active attitude control of a gravity gradient stabilized satellite is treated in F220-JEM-70-58 (See Volume II). The results of the analysis show that, in order to control the vehicle in the presence of the maximum expected aerodynamic torque and still maintain a configuration which can be considered a rigid body, a boom diameter of one inch is desirable. For a mass expulsion ACS, thrust levels of 0.001 to 0.002 pounds are desirable and a total impulse capability of 360 lb-sec will be adequate to maintain full active attitude control for a 90 day mission. Similar systems have been space qualified for OAO and DISCOS/TRIAD spacecraft.

The most significant factor to influence the ability to actively control the gravity gradient vehicle is its rigidity. The rigidity in this instance can be characterized by the ratio of the natural frequency of the bending mode to the natural frequency of the gravity gradient mode. In Volume II, F220-EL-70-61, these frequencies are established for a 60 ft long, 1/2 inch dia. boom. The pitch bending natural frequency is 164 times greater than the pitch gravity gradient natural frequency, and the roll bending natural frequency is 143 times greater than the roll gravity gradient natural frequency. This configuration can be considered rigid with respect to active control near gravity gradient frequencies. If a one inch diameter boom were used, these numbers would increase by a factor of 2.8 to 1 and the configuration would be significantly more rigid.

FIGURE 8 - YAW STABILITY FOR 2% RELATIVE ENERGY



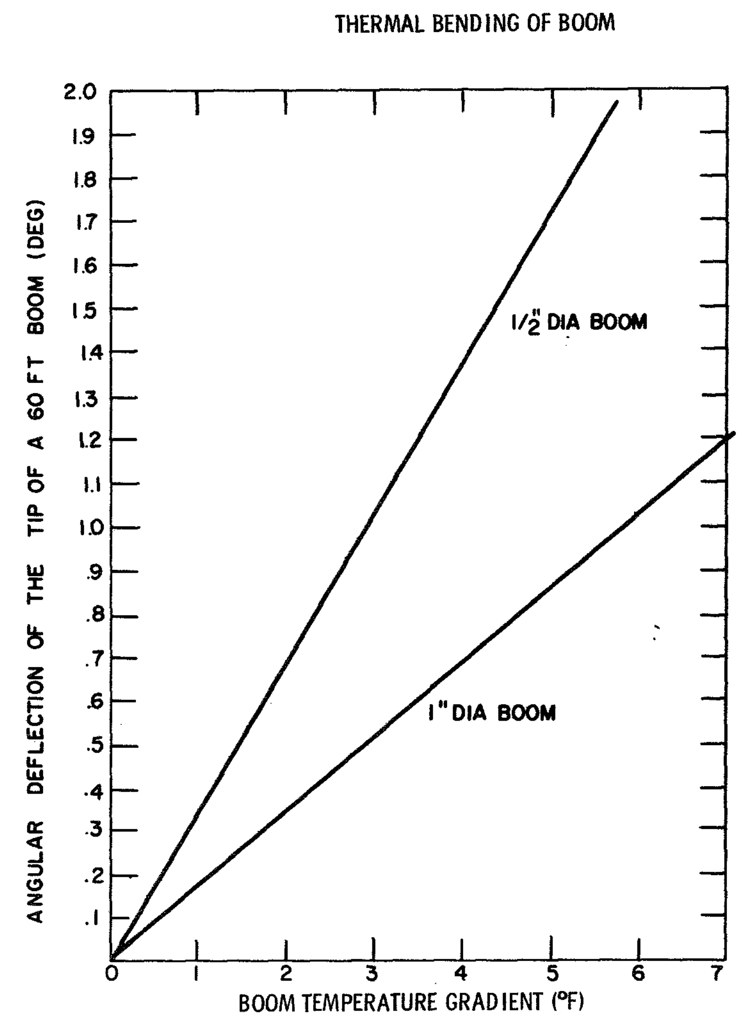
4. Acquisition Phase - The acquisition phase of the mission is defined as the events which occur between separation from the launch vehicle until the boom is fully extended. Upon separation from the Scout launch vehicle, the vehicle has a high rate ($\approx$180 RPM) about the spin axis (Note: The spin axis is the same as the yaw axis). Also, the spin axis will be very nearly in the orbital plane. During some portion of an orbit the viscous fluid nutation damper described in Volume II, F220-JEM-70-60, will damp the separation coning angle to a negligible level. To insure that the cone angle is convergent rather than divergent the satellite must be designed so that at this time the spin axis is the axis of maximum moment of inertia. The nutation damper is sized so that significant damping takes place in one quarter of an orbit. Three quarters of an orbit ($\approx$270°) after separation the spin axis of the satellite will be very nearly aligned with the local vertical with the thrust tube end of the vehicle pointed away from the Earth. At this time the vehicle will be despun by a "yo-yo" despin device, the solar panels will be deployed, and the first boom extension of the "dead-beat" deployment sequence will be made. It is expected that this sequence can be accomplished in an open loop fashion. That is, it can be accomplished as the direct result of on-board commands given at predetermined times. If, however, the uncertainty in orbital position is significant, the procedure can be accomplished by simply allowing the vehicle to coast in the spinning state until sufficient orbital information is gathered to establish an initiation time for this sequence. A ground command can then be given to initiate the sequence at the proper future time and the sequence will be conducted in an open loop fashion from that point foreward.

The "dead-beat" deployment phase can be accomplished in a variety of ways. The simplest method is to use a sequence which consists of two boom extensions. At the first boom extension the vehicle achieves 50% of its final inertia. One half of a gravity gradient cycle later ($\approx$104° in orbit) the vehicle will again be aligned with the local vertical and will have a pitch rate which is twice the orbital rate. At that time, the inertia is doubled by completing the boom extension which reduces the pitch rate to the orbital rate required.

To accomplish a good "dead-beat" deployment, as demonstrated by RAE-A, the process must be analyzed thoroughly and simulated accurately taking into consideration such factors as the effects of the initial conditions of rate and attitude, the effects of external disturbances during the process, the finite time required for boom extension, the effects of mass property uncertainties, and the effects of boom deflection.

5. Thermal Deflection of Boom - For any system with booms, thermal bending must be given consideration. Thermal bending is a result of the differential expansion of the two sides of the boom caused by the existance of a thermal gradient. Figure 9 shows angular tip deflection for a 60 ft long boom as a function of temperature gradient. The coating methods and

FIGURE 9 - THERMAL BENDING OF BOOM



perforation techniques used on the RAE-A booms reduce the temperature gradient to less than 1°F. If these techniques are used on the boom described here, the thermal bending will be only a small fraction of a degree.

6. Hardware Considerations - Many types of mass expulsion subsystems are possible. The hardware described here is probably the most common and, therefore, in the highest state of development. To demonstrate the feasibility of the mass expulsion subsystem, weight and size estimates have been made using nitrogen gas stored at a high pressure (≈5000 psi). The subsystem weight and size estimates in Table III has a total impulse capability of 360 lb-sec and has a full complement of 12 nozzles.

TABLE III

MASS EXPULSION WEIGHT AND SIZE ESTIMATE

	Weight (pounds)	Size (inches)
Tank (2 Spherical or Toroidal)	6.0	8 Dia
Gas (N_2)	6.0	---
ACS Electronics	1.0	4 x 4 x 4
Fill and Vent Valve	0.3	1 x 3/4 Dia
Squib Valve	0.4	1.4 x 1 x 1
Pressure Regulator	0.8	2 x 1 1/4 Dia
Shut Off Valves (2)	0.8	1.5 x 1 x 1 each
Nozzles (12)	3.6	2.5 x 3/4 Dia each
Piping	0.5	1/8 Dia
Total	19.4	

Figure 10 shows a schematic diagram of a 12 nozzle cold gas attitude control subsystem and its interrelation with the attitude determination subsystem and the data handling subsystem. The concept shown is a full 12 nozzle configuration which permits the application of pure couples in each direction about each axis. The nozzles are arranged in two banks of siz nozzles each. Each bank contains its own shut-off so that if a malfunction such as a leak develops in one bank, it may be shut-off and the mission can be continued using the remaining six nozzles. With one bank shut down the spacecraft still has the full ACS capability. Many examples of every component of the cold gas ACS have flown successfully in space satellites. Also, these individual components are so readily available and so easily combined to form a subsystem that it makes little sense to strive to find some "off-the-shelf" subsystem which meets all the requirements.

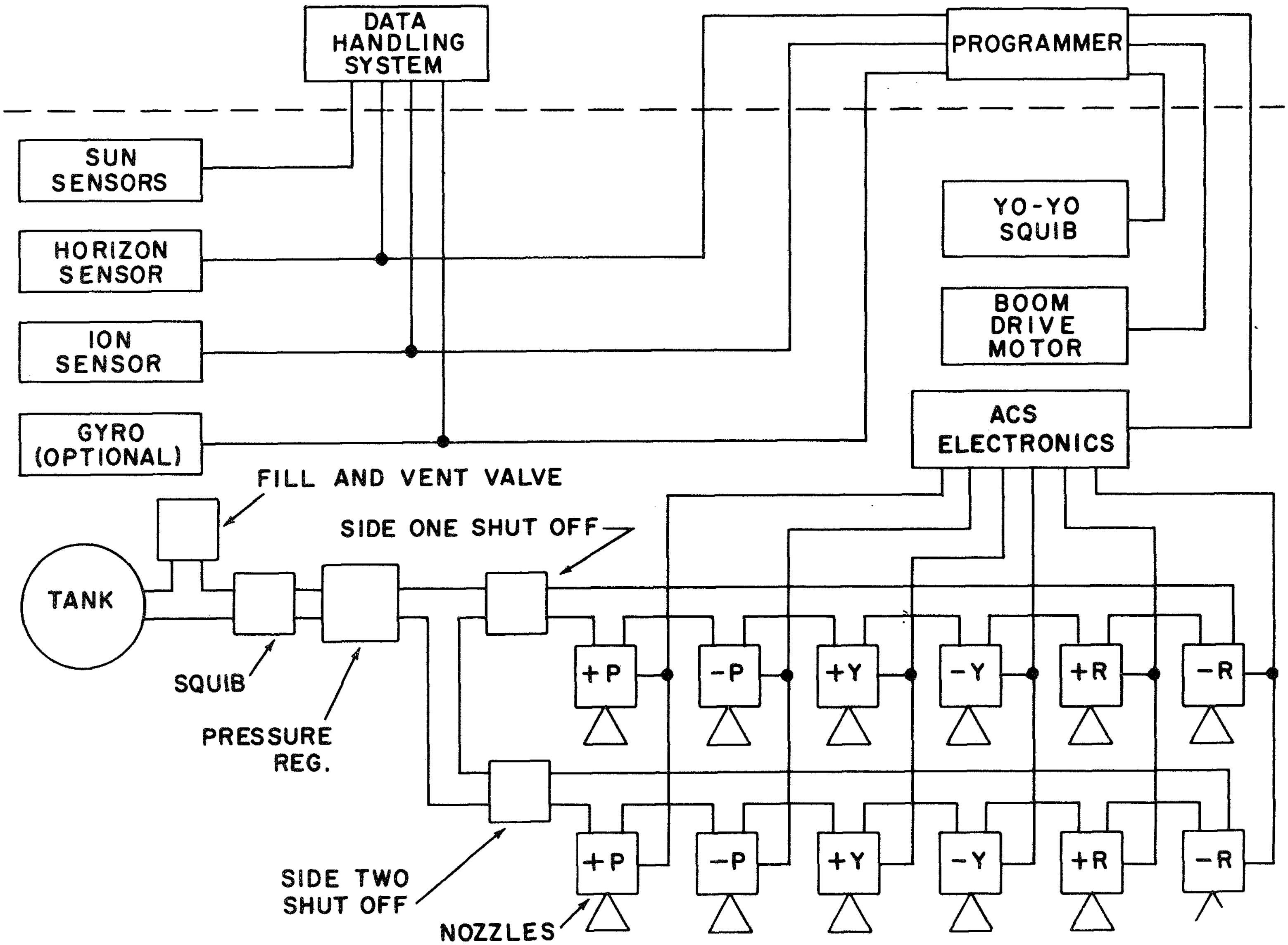


FIGURE 10 - ATTITUDE CONTROL SUBSYSTEM SCHEMATIC

The miscellaneous remaining items attributed to the attitude control function are the nutation damper, the gravity gradient boom and tip mass, and the "yo-yo" despin device. Table IV is a size and weight estimate for these items.

TABLE IV

MISCELLANEOUS ACS HARDWARE WEIGHT AND SIZE ESTIMATE

	Weight (pounds)	Size (inches)
Nutation Damper	1.0	13 Dia Ring
Boom and Mechanism	2.1	4 x 3 x 7
Yo-Yo	2.0	2.8 x 2.8 x 2.8
Tip Mass	5.0	Not Critical
Total	10.1	

The nutation damper is a toroidal aluminum tube filled with a viscous fluid which damps the coning motion of the spinning vehicle during the first part of the acquisition phase.

The "yo-yo" despin device consists of two one-pound weights each on the end of a 10 ft cord. The cords are initially wound around the exterior of the spinning satellite. The weights are held in position by squib operated devices. When despin is desired, the squibs are fired and the weights are allowed to unwind cord from the vehicle. This unwinding action produces a torque to despin the vehicle. The ends of the cords are not fixed to the vehicle and when the cords are completely unwound the weights with cords attached are allowed to fly off into space. If the weights and cord lengths are properly sized, the vehicle will be left with zero or very nearly zero spin rate. The "yo-yo" despin was chosen because it is a very simple and reliable method of despinning a vehicle demonstrated many times in the past. F220-JEM-70-60 in Volume II contains a sizing analysis for the device; it also shows that the despin process requires less than 0.5 seconds.

The tubular deployable boom is a thin wall ($\approx$.003") Beryllium-copper tube which can be obtained in a variety of sizes. Lengths of from 6 ft to 750 ft have been flown. By far the most popular diameter at present is 1/2 inch; however, booms with diameters of from 1/4 inch to 2 inches have been fabricated. The 60 ft long 1 inch diameter boom suggested for this application can be considered to be well within the "state-of-the-art" at this time.

7. ACS Command Sequences - The following control sequences are typical of commands which could be established and stored in the programmer to demonstrate the versatility of the proposed attitude control concept.

(1) Acquisition Sequence

- Initiate Yo-Yo Squib
- Deploy Panels
- Boom Extension No. 1
- Boom Extension No. 2

(2) Initiate ACS Sequence

- Activate Attitude Determination System
- Activate ACS Electronics
- Fire Cold Gas Squib

(3) Deactivate ACS Sequence

- Deactivate ACS Electronics
- Deactivate Attitude Determination System

(4) Initiate Rate Control Sequence

- Power to Gyros
- Command Rate Mode

(5) Deactivate Rate Control Sequence

- Deactivate Rate Mode
- Shut Down Gyros

(6) Commands for ACS Failure

- Shut Down Side 1
- Shut Down Side 2

(7) Ground Operation Mode

- \+ Pitch
- \- Pitch
- \+ Yaw
- \- Yaw
- \+ Roll
- \- Roll

(8) Manual Boom Operation

- Extend Boom
- Retract Boom

Should the ground control station desire to modify the mission profile at any time, it may do so by calling up any of these sequences at any time or by calling up any individual command in any sequence.

Attitude Determination. - Determination of attitude in a three axes reference frame requires knowledge of a minimum of two measurement vectors. The two vectors need not be related to the same phenomena; that is, one could be the direction of the sun line while the other could be the direction of the Earth's magnetic field. It is clear that as the angle between the two measurement vectors becomes small, knowledge of rotation about the measurement vectors becomes inaccurate and is completely indeterminate when the two vectors coincide. This effect is illustrated in Figure 11 for three different accuracy requirements.

Candidate Sensors: The types of sensors listed in Table V have all demonstrated the feasibility of determining spacecraft attitude. The table includes a range of accuracy capability where an attempt is made to indicate capability from the simplest version of each sensor, through existing sophisticated versions, to predicted capability in the next few years. The difference in accuracies is attributable to varying degrees of instrument complexity as well as varying amounts of data processing.

TABLE V

SPACECRAFT ATTITUDE DETERMINATION SENSORS

	Measurement Accuracy Capability		
Sensor	Simplest Version	State of Art	Predicted
Horizon	5°	0.1°	0.05°
Star	0.1°	0.01°	0.005°
Solar	10°	0.25°	0.06°
Magnetometer	7°	0.1°	---
Ion Sensor	3°	1°	0.1°
Gyroscope	5°	0.01°	0.001°

The table infers performance of sensors on non-spinning vehicles and excludes the use of gimbals to orient the sensors. Otherwise, significantly better numbers could be quoted, particularly for the star and solar sensors. The gyroscope is included in the table even though it fundamentally does not measure an absolute direction. However, in a gyrocompass mode it can measure the spacecraft yaw angle. Further, the gyroscope is useful for measuring attitude increments with high precision, a fact which might be of great value to the experimenter. When the gyro is included in a closed loop ACS system, very precise rate stability can be achieved and accurate angular increments can be commanded.

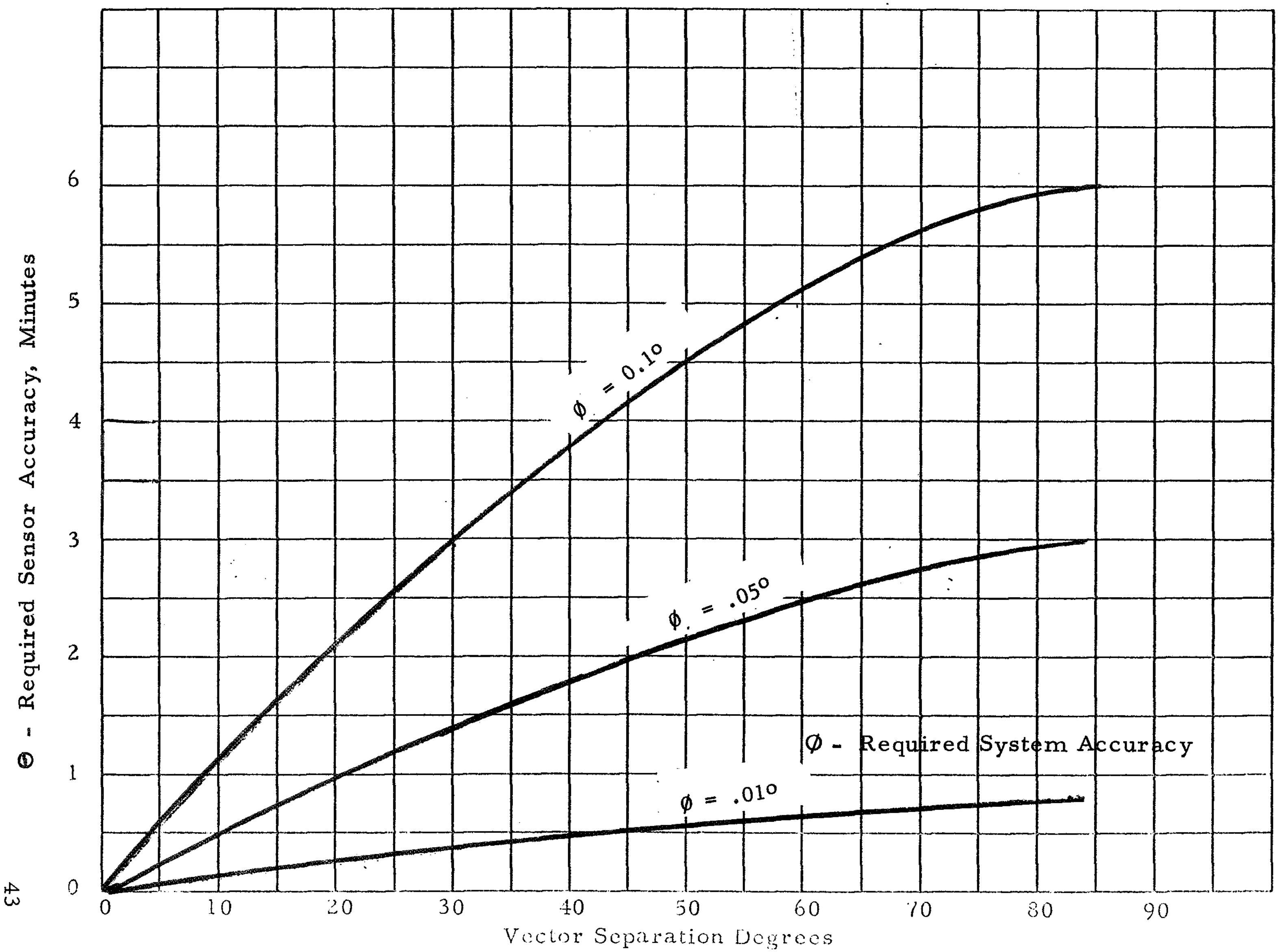


FIGURE 11 - RESOLUTION REQUIREMENTS AS A FUNCTION OF VECTOR SEPARATION

The matrix shown in Table VI illustrates the various combinations of sensors that can be used to establish the two measurement vectors. The accuracies shown are in general the best that the state-of-the-art permits with no restrictions on data processing. The costs listed are illustrative and do not include nonrecurring fees that are so often associated with procurement of spacecraft components.

Some of the matrix combinations are not feasible; for example, any number of magnetometers would be used, but without some other reference only two axis information is obtainable; that is, rotation about the magnetic vector is indeterminate.

The complexity of some of the combinations is not to be underestimated. The magnetometer in conjunction with the sun sensor leads to a small inexpensive system with good accuracy capability. However, to achieve the high accuracy potential, a complex model of the Earth's field must be used (spherical harmonic model with at least 25 terms), as well as the emphemeris of the sun and knowledge of orbital position. TR F220-RL-70-115 in Volume II discusses the sensors listed in Table VI and sets criteria for the selection of the most promising combination.

Selected Attitude Determination Concept: The nature of the proposed experiments suggests that attitude should be determined in the Earth coordinates. An examination of the combinations in the matrix indicates that an attractive combination is the ion and horizon sensors.

The horizon sensor indicates local vertical while the ion sensor indicates the yaw angle. Since the ion sensors input axis is nominally along the velocity vector, the two measurement vector are nominally orthogonal regardless of orbital position. The proposed system has a high degree of flexibility, particularly with respect to accuracy. If the mission requirements are extensive, the attitude can be determined to a high accuracy, proportional to the amount of data processing the spacecraft contractor is willing to perform.

Prior to boom deployment in the gravity gradient ACS, approximate solar aspect angles can be made with a group of coarse solar sensors.

A triad of gyroscopes is recommended to complete the attitude determination sensors. The gyros add a great deal of capability and flexibility to the system.

A block diagram of the proposed attitude sensor system is shown in Figure 12 and the pertinent parameters of typical available hardware in Table VII.

TABLE VI - COMBINATIONS OF SENSORS FOR ATTITUDE DETERMINATION

	Horizon	Sun	Star	Magnetometer	Ion
			Accuracy (Degrees)		
Horizon	N. F.	.42	.12	.14	1.0
Sun	.42	N. F.	.42	.18	1.1
Star	.12	.42	.07	.11	1.0
Magnetometer	.14	.18	.11	N. F.	1.0
Ion	1.0	1.1	1.0	1.0	N. F.
			Total Volume Inches^3		
Horizon	---	380	420	390	570
Sun	380	---	60	30	210
Star	420	60	100	70	250
Magnetometer	390	30	70	---	220
Ion	570	210	250	220	---
			Cost K Dollars		
Horizon	---	55	100	60	80
Sun	55	---	55	15	35
Star	100	55	100	60	80
Magnetometer	60	15	60	---	40
Ion	80	35	80	40	---
			Complexity of Computation High, Med., Low		
Horizon	---	M	M	H	L
Sun	M	---	M	H	M
Star	M	M	M	H	M
Magnetometer	H	H	H	H	H
Ion	L	M	M	H	---

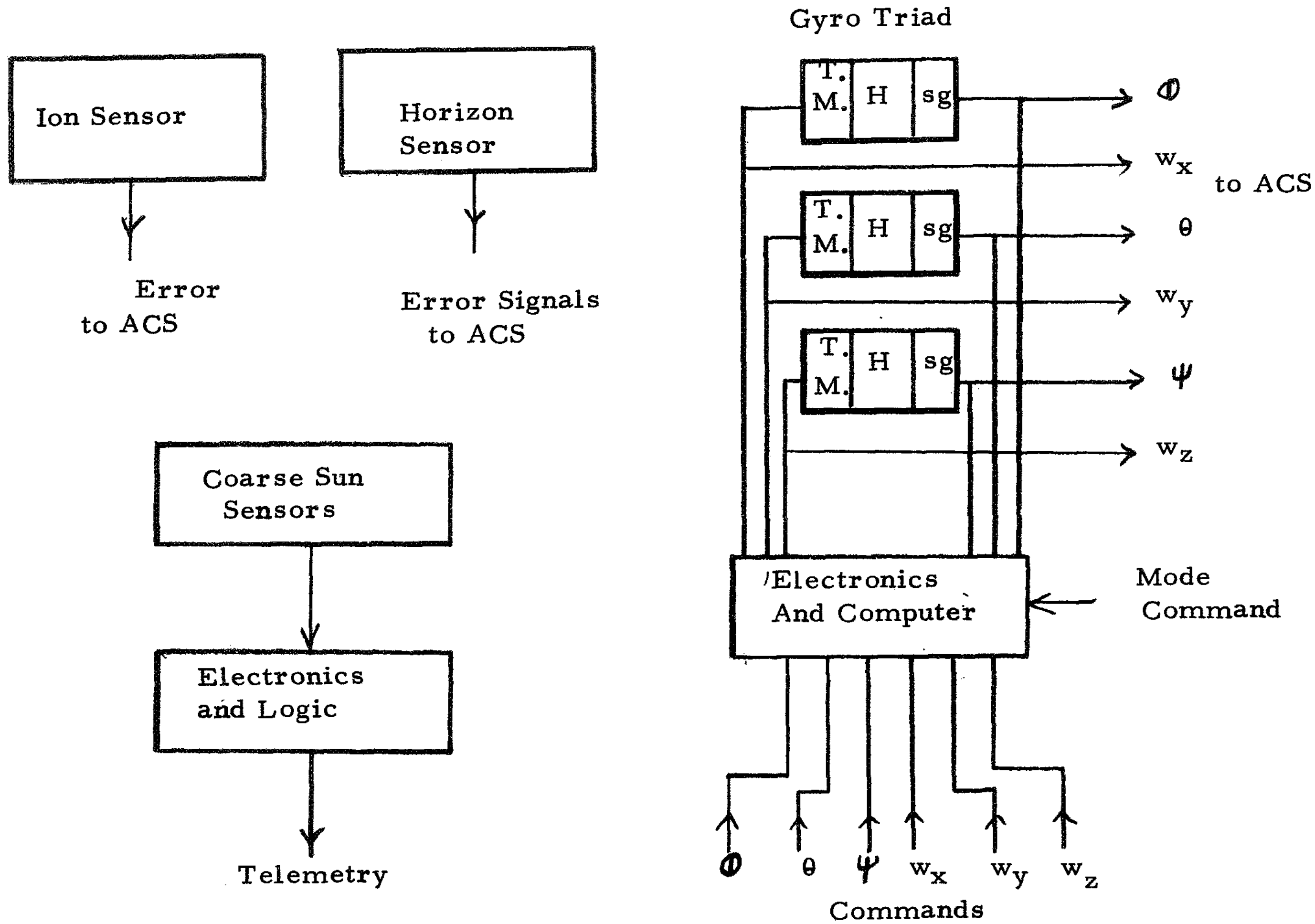


FIGURE 12 - ATTITUDE SENSOR SUBSYSTEM BLOCK DIAGRAM

TABLE VII - SENSOR PARAMETERS

Sensor	Possible Vendor	Model No.	Weight (lbs)	Volume in^3	Power Watts	Accuracy*
Horizon	Barnes	13-151	17.5	443	10	0.1 degrees
Ion	Avco	IAS	5	200	3	1.0
Sun	Ball Brothers	C105	.25 (total)	2 (approx)	0.5	2.0
Gyro**	U.S. Time	RIG-30	1.0 (triad)	20 (triad)	9 (triad)	Ang. inc. 0.1% Rate .001°/sec

* This is instrument accuracy, net accuracy is a function of degree of data processing.

** Gyro chosen for miniature size and lack of heating requirements. Gyros available with significantly superior performance.

Data Handling and Control. - The major considerations which were kept in mind while studying the requirements for the data handling and control subsystem were the following:

Design Approach: The recommended design approach must meet the goal of providing sufficient versatility to permit a complete and thorough orbital test of the experiment to be flown. In order to accomplish this goal, it was determined that a flexible concept should be capable of variations in word length, bit rate, format, programming, etc.

It became apparent that a truly flexible design had to be supplied as part of the basic spacecraft, although it was not unreasonable in this case to want to give the data handling responsibility to the experimenter.

In general, it has been taken for granted that the spacecraft contractor should assume the responsibility for the data handling unit, because practically all satellites have carried more than one experiment, although some portion of the processing was quite often done as part of each experiment. In some cases, this processing was very sophisticated, such as the autocorrelation computer that was flown on IMP-F. Other spacecraft generally leave very little processing to the experiments, such as IMP-I (Eye) and S^3.

On the other hand, SAS tends to leave the data handling chore to the experimenter. In those cases where a standard spacecraft will be used for a number of widely different experiments, with the experiments being flown one at the time, this approach might make sense. The result will be that the cost of the standard spacecraft will appear low. On the other hand, the cost of the experiment will be very high. In the case of SAS-B, the cost of the data processor ran close to $400, 000. The cost of the data handling unit will more than likely remain high for each and every experiment.

Flexibility, if left to the experimenter, tends to become very expensive, and a single, versatile unit designed to accommodate many experiments by the spacecraft contractor is the selected choice.

Cost Analysis: Cost effectiveness is a second major consideration, and after considering four different design approaches, the one with the greatest versatility and flexibility was the most cost effective according to the following ground rules:

(1) The primary goal is to have a total mission cost effectiveness not necessarily one or two low cost subsystems.

(2) Selection of a subsystem because of its low initial cost does not necessarily mean that the overall system cost will be low.

(3) The cost comparison will be based on the "block" concept. Four flights were typical of a "block", each carrying experiments that have been selected because of some similarity in their requirements. The similar requirements may be in the area of attitude control and determination, data handling, power, communications, or any combination of these areas. Thus, "similar requirements" do not necessarily mean similar data handling requirements.

The comparative costs discussed in Volume II, TR KA-70-22 "Cost Considerations" show that, although the initial costs of the selected data handling concept are high, the costs for follow-on flights drop off very rapidly. The average cost for the first block of four flights shows the most flexible design to be the least costly. The second block, if shown, would have indicated even more dramatically the cost advantage of this option. Thus, cost considerations lead to the conclusion that the most cost effective data handling option is the most sophisticated and flexible one, assuming that a memory is required in all cases.

Other Considerations: The selected data handling subsystems option was a "universal" design where changes from mission to mission take place in software. A number of other considerations tend to strengthen the conclusion that the selected option will result in the maximum return per dollar of investment.

The satellite concept under study is supposed to serve as a test bed for each given experiment, which implies that the quality, quantity and resolution of the data to be taken during orbital testing exceeds that required under operational conditions. In addition, the possibility exists that unforseen circumstances or unpredictable phenomena may lead to a desire to reconfigure the experiment in orbit. Ultimately, it would seem that the object of the orbital test is to determine not only the characteristics and performance of the instrument itself, but also to determine the best data handling concept which will be required under operational conditions. It is hard to imagine how such goals can be attained without a very sophisticated and flexible data handling approach.

Another advantage of the "universal" data handling unit becomes apparent when testing and qualification of experiments is considered. If several experiments are being developed in parallel, their compatibility with the spacecraft data handling subsystem can be determined with very little turn-around time required, using a single flight unit or flight spare spacecraft with several software configurations. Fast turn-around time implies even further cost savings.

Further, simulation of the data handling unit can be attained using a small general-purpose computer with an input/output buffer having circuits identical to those used in the data processor. This implies that universal ground support equipment can be designed, with quick turn-around time between experiments. Again, this reflects in lower costs per mission, and supports the concept of a flexible data handling unit.

STADAN Compatibility: The data handling and control subsystem compatible with STADAN was discussed in TR KA-70-24, Volume II, and it was concluded that STADAN compatibility can easily be satisfied with the selected flexible programmable data handling unit.

Recommended Design: The recommended design is based on off-the-shelf technology, although it was realized that off-the-shelf hardware for a large portion of the subsystem was not available. Four different design options were considered (TR KA-70-20, Volume II) and the fourth option, describing a "universal" data handling subsystem, where changes from mission to mission take place in software, was selected as the recommended approach. This design can be implemented with present-day technology, and there is no requirement for a technological breakthrough.

To determine which of the many available circuit technologies to recommend for this satellite concept, the following criteria were considered:

(1) A primary design goal is to make as much size, weight and power as possible available to the experimenter.

(2) Although the lifetime of the mission is generally only a few weeks, some experiments might want the operating time to occur in the form of short segments spread over a long period of time.

These criteria resulted in the following ground rules for the data handling unit:

(1) Size and weight should be kept reasonably low.

(2) The power consumption should be kept low, in particular for those missions where only body-mounted solar cells are used.

(3) Despite the generally short duration of the missions, the data handling unit should be designed to operate for at least a year in orbit. This includes degradation which can be attributed to radiation and other sources.

The technologies which can be considered for the data handling and programmer subsystem, compatible with the above ground rules, are the following:

(1) Monolithic bipolar integrated circuits.

(2) Thick or thin film hybrid circuits.

(3) P-channel or complementary MOS integrated circuits.

Each of these technologies has been used onboard spacecraft, and each has its own peculair characteristics that must be evaluated before a choice can be made. Evaluation of the three technology options, resulted in the recommendation that hybrid electronics be considered for the data handling unit because low power and high-speed capability can be combined with small size and low weight. In addition, these circuits have proven highly reliable in space, and have shown excellent resistance to radiation damage.

Command and Telemetry. - The satellite command and telemetry subsystems were defined and reviewed for compatibility with the STADAN requirements as specified in X-530-69-109, "Space Tracking and Data Acquisition Network Manual", August 1969. In addition, the predicted STADAN S-Band availability and compatibility for the 1974 time period was investigated. It was confirmed that command and satellite tracking in the VHF band, and telemetry in S-band, will be compatible with NASA Aerospace Standards and IRIG standards.

Major elements of the satellite communications subsystem include: 1) antenna, both S-band and VHF; 2) telemetry transmitter; 3) beacon transmitter; and 4) command receiver/decoder. A summary of important parameters of the communications design is presented in Table VIII. Detailed design parameters and analyses of the command and telemetry subsystems appear in Volume II of this report, ESDT/R-F440-4097.

The S-band antenna is an open ended circular waveguide, mounted to the periphery of the cylindrical reference design in an earth pointing position. The basic design objective was to provide an antenna pattern with maximum gain along the major axis with a beam width sufficient to provide full earth coverage at a circular orbit altitude of 100 nautical miles. Gain at the beamwidth edges was not sacrificed to achieve these objectives.

The VHF antenna is a turnstile configuration with four quarter-wavelength whips in the same plane extending perpendicular to the satellite major axis. A diplexer provides the isolation required between the radiated tracking signal and the received command signals. This type antenna has a bandwidth sufficient to radiate the tracking signal in the 136-138 MHz band and receive the command signals in the 148-150 MHz band. Gain and beamwidth considerations were similar to the S-band antenna objectives.

Transmission of telemetry data may be continuous throughout the orbit with the capability of transmitter turn on/off by ground command. All communication functions will be possible whenever satellite-ground station line of sight geometry restrictions are satisfied.

TABLE VIII

COMMUNICATIONS SUMMARY

Telemetry Frequency	2200-2300 MHz band
Transmitter Power	0.10 watts nominal
Tracking Frequency	136-138 MHz band
Beacon Power	0.100 watts nominal
Command Frequency	148-150 MHz band
Antenna	
S-band	Flush mounted circular open ended waveguide
Gain	+6db boresight -4db @ 75°
VHF	Four quarter-wavelength whip turnstile
Gain	+1db boresight -3db @ 45°
Data Rate	100KHz nominal
Design Bit Error Rate	10^{-3}
Commands	
Type	Tone Digital "ON/OFF"
Number	70 maximum

Continuous communications capability does not exist because the satellite is not continuously intercepted by STADAN ground stations. Typical viewtimes for launches from Wallops Island, San Marco and the Western Test Range are presented in Volume II, ESDT/R-F440-4097.

The tracking signal for the Minitrack Interferometer in the 136-138 MHz band will be obtained from a separate beacon transmitter continuously on throughout the orbit.

The typical command subsystem selected will have the capability of receiving a maximum of 70 discrete, non-redundant ON/OFF commands in Tone Digital Format, compatible with NASA Aerospace Data Systems Standards. Protection from reception of spurious commands will be incorporated into the design. Command status monitoring of the satellite will be continuously available through telemetry. The system will be capable of receiving and executing unambiguously, any series of commands at any point in the orbit when in view of a STADAN ground station. The command subsystem will operate in a frequency band of 148-150 MHz

Table IX lists the performance margins available for each of the three communications links analyzed, i.e., telemetry, tracking and command. The margins are representative of the link parameters at maximum slant range between satellite and ground station. Transmitter power outputs of 0.100 watts for both telemetry and tracking, together with the antenna gains available, provide more than adequate margins for all links.

TABLE IX

COMMUNICATION LINK PERFORMANCE MARGIN

Link	Performance Margin
* Telemetry	+8.0db
Tracking (Minitrack)	+9.5db
Command	+26.1db (@ -3db antenna gain)

*For a data rate increase from 100 KHz to 1 MHz the resulting performance margin would be -2db, similarly if the bit error rate was decreased from 10^{-3} to 10^{-5} the performance margin will reduce by 2.7 db.

Power. - The power analysis conducted in support of the study was confined to determining the solar power available from the satellite in its gravity gradient mode of operation. The result of the analysis indicates a fixed body array of cells cannot provide the solar power required; additional solar paddles, fixed or sun seeking, will be required. Paddle(s) location and pointing vector will be determined after orbit definition and launch parameters are established. This section will summarize the function of the major elements in the power subsystem. Detailed values of the solar power available and variation in orbit can be obtained by referring to ESDM-F440-4086 in Volume II.

A simplified block diagram of the satellite power subsystem is shown in Figure 13. The power subsystem consists of the following components:

(1) Solar Array; body mounted and/or paddle(s)

(2) Battery

(3) Power Control Unit (PCU)

(4) Power Switching Unit (PSU)

(5) Lo-Voltage Power Converter

(6) Power Umbilical Connector

(7) Battery Isolation Jack

The subsystem is designed to provide full operation of the satellite during sunlight and sun occultation. During sunlight operation, the solar array is the prime supplier of power. During peak loads, the battery may be switched to assist the solar array, as necessary, to provide the power required. The battery provides the necessary power during sun occultation periods. In addition, the power control unit provides battery charging capability, provision for switching the battery on and off the main bus, solar array over-voltage protection and under-voltage detection. The power switching unit switches the various satellite loads on command, and the power converter provides the required low voltage and regulation to the support electronics. Experimentor low voltage will be his responsibility. To clarify a typical operation of the system, a description of each component follows.

Solar Array: The solar array is the primary power source for the satellite. The body mounted array consists of sixty strings of solar cells, 88 cells per string, bonded to the peripheral surface area of the right cylinder design, generating approximately 56 watts of raw solar power at normal sun incidence. Power availability is based upon an array temperature of +105°C while in the gravity gradient attitude. Power variation within specific orbits is presented in ESDM-F440-4086, Volume II. A single 29" diameter paddle with 12 strings of solar cells, 64 cells per string, provides 38 watts normal sun incidence at an assumed array temperature of +45°C.

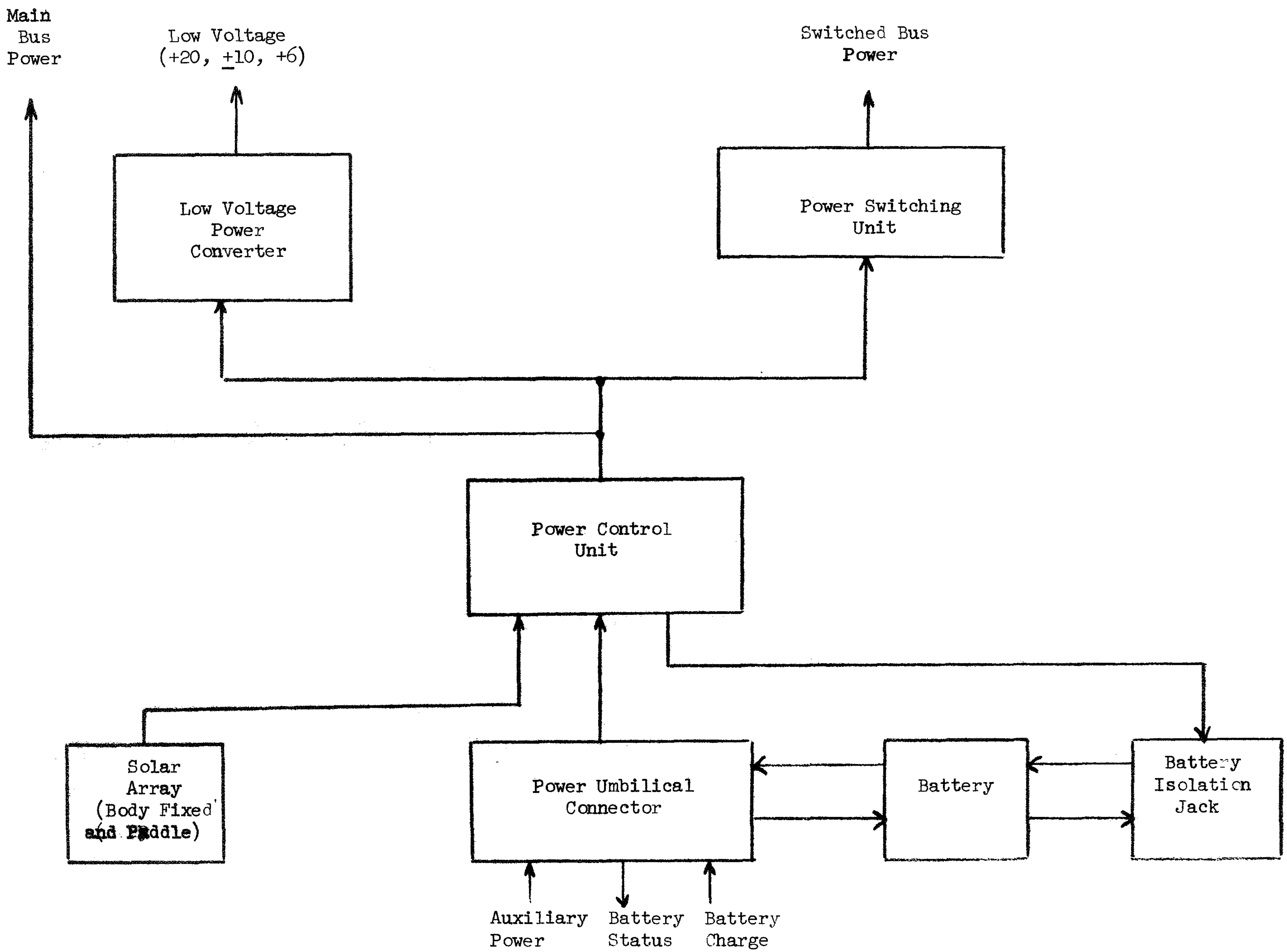


FIGURE 13 - POWER SUBSYSTEM BLOCK DIAGRAM

Battery: The battery recommended is a sealed nickel cadmium type. A one ampere hour rating will provide a maximum discharge depth of 50%.* The watt hour rating of 24 is based on supplying 20 watts continuous for a maximum occultation period of 35 minutes. Satellite power is supplied by the battery during periods of sun occultation, during any dips in solar array power output, and during any satellite high power transients. A battery weight of less than three pounds is based upon a typical conservative energy density of 12 watt hours per pound.

Power Control Unit (PCU): The power control unit performs several functions in the power subsystem. First, it switches battery power to the main bus when necessary, to maintain the bus voltage above the 24 volt lower limit. Second, the PCU provides for battery charging when excess solar array power is available. Third, zener diode limiting is provided so that the solar array voltage is limited to 32 volts. Fourth, under voltage sensing and output signals are provided to indicate a discharge battery condition.

Power Switching Unit (PSU): All spacecraft power switching functions nominally are performed in the PSU. Latching relays shall be utilized for the switching functions which route power throughout the spacecraft. Switching will be performed on command from the Data Processor. An additional functional requirement of the PSU is current overload detection for the Low Voltage Power Converter.

Low Voltage Power Converter: The low voltage power converter supplies highly regulated voltage, typically +20, ± 10 and +6 volts dc, through the power switching unit to those users requiring low voltage, regulated to within ± 1.0%.

Power Umbilical Connector: The power umbilical connector will provide access to the power subsystem during factory and pre-launch checkout. Auxiliary power will be supplied through the connector; battery status monitoring and battery charge capability will be permitted via the connector. During ground checkout, these auxiliary power, charging and monitoring functions would be accomplished by GSE.

Battery Isolation Jack: The battery isolation jack will provide the capability to positively remove the battery from any loads within the satellite, whenever desired. It can be designed as an integral part of the power umbilical connector.

Typical Subsystem Operation: During ground checkout and operations, the satellite will function on external ground power and the battery can be charged by a battery charger located within GSE.

During flight, the battery charger within the PCU boosts the solar array voltage to the proper voltage to charge the battery. The charge rate is controlled such that safe charge rates are never exceeded and so that the

*Based on a more conservative discharge depth, say 25-30% typical of long life missions, a larger battery would be required. Table XI, page 70 indicates a weight capability for batteries of ~10 pounds.

main bus voltage does not fall to the battery sharing point. Full charge is determined by a combination of battery voltage and temperature. The battery is switched onto the main bus by a transistor switch which adds only enough battery power to maintain the bus voltage above 24 volts. During occultation, however, the battery will be fully switched onto the main bus. Should it be desired, the battery can be removed from the system through a battery disconnect switch upon command. Control is provided to maintain the bus voltage limits between the 24 and 32 volt extremes. Shunt zener diodes are used to limit the solar array voltage to 32 volts during periods of light loading and/or low temperature. If the battery should become discharged during a peak load sharing condition or during occultation, and the main bus voltage falls below 24 volts, an under-voltage detector function is provided to sense this condition and provide a signal to the data processor. If the battery is further depleted, the PCU provides a loads-off signal to the data processor. All other power switching functions are conducted in the PSU on command from the data processor.

Structure. - The design concept of the vehicle and structure considered in this study was derived from the requirements for a basic vehicle capable of accommodating a wide range of potential experiments, attitude control systems, and spacecraft support subsystems. The spacecraft structure consists of three basic subassemblies; 1) the support system platform, 2) the exterior cylinder, and 3) the forward cover (see Figure 14). The individual components, and their assembly into a spacecraft are described below.

The advantages of the selected design concept lie in the flexibility to accommodate the great variety of potential experiments. Some of those advantages are listed below.

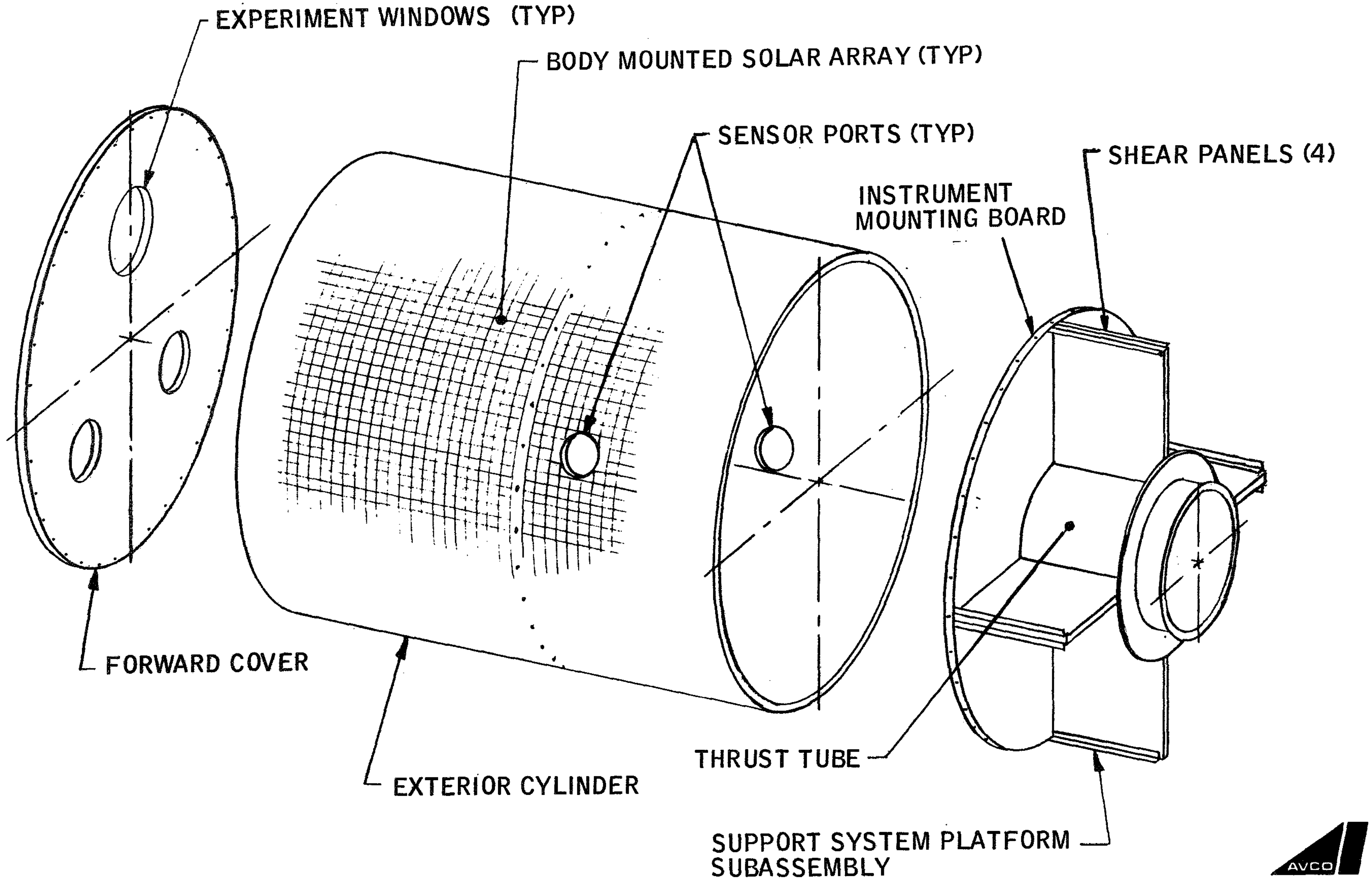
(1) Flexibility is achieved through the provision of a single large experiment volume of cylindrical shape.

(2) The spin moment of inertia can be maximized by placing a major portion of the structure at the maximum radius and by providing for placing of experiments as far outboard as possible, to provide for maximum spacecraft stability during that portion of the mission when the spacecraft is spinning.

(3) The capability for securing instruments to any part of the structure provides flexibility and minimizes integration problems.

(4) The "major subassembly" approach allows the various elements of the spacecraft to be assembled and checked out separately, and later assembled as a unit. This will minimize the quantity and complexity of interfaces, and will reduce the time required to build and check out the spacecraft.

FIGURE 14 - SPACECRAFT STRUCTURAL CONCEPT



The concept is particularly effective when applied to the class of satellites carrying experiments pointing in the direction of a particular object in space, such as a gravity gradient stabilized, earth pointing spacecraft. The complete forward surface of the spacecraft is available for the installation of experiments. The attitude control system, in this case the gravity gradient boom, is conveniently located at the aft end of the spacecraft in the center of the thrust tube. The other spacecraft support subsystems are also located in a group separate from the experiment area, minimizing the potential for interference with the experiments.

The allowable envelope for the spacecraft is defined by the heatshield of the Scout launch vehicle. Three payload envelopes have been considered (See Figure 15).

The basic spacecraft is designed to fit within the standard 30 inch diameter four stage heatshield. However, under certain conditions the alternative heatshields are attractive; such as when large deployable solar arrays or booms may be required, for which space is available between the spacecraft and the 38 inch diameter envelope, or when the additional length of either of the two longer shrouds will permit the installation of a light weight piggyback satellite.

A number of possible spacecraft configurations, all built on the basic design, are shown in Figures 16 through 19; including one with a typical experiment installed to illustrate the flexibility which is available with the concept which has been considered.

Supporting System Platform Assembly: The support system platform assembly consists of six basic parts; the thrust tube, the instrument mounting shelf, and 4 shear panels, assembled as shown in Figure 14. Details of the assembly are shown in Figure 20. This platform is treated as a subassembly which can be assembled separate from the remainder of the spacecraft, and on which all, or nearly all of the support system hardware can be mounted prior to assembly with the remaining elements of the spacecraft.

The thrust tube will provide the interface with the fourth stage of the Scout - the E section - from which it will be separated by means of the standard pyrotechnically operated separation clamp. The thrust tube, and the attachment of the thrust tube to the remainder of the support system platform will be designed to withstand the acceleration and vibration loads associated with the heaviest potential vehicle ($\sim$ 350 pounds). The thrust tube will be machined from a forging of a high strength aluminum alloy. Since weight does not appear to be a problem the added cost associated with the use of other materials, such as titanium or beryllium, does not appear to be justified.

FIGURE 15 - SCOUT VEHICLE PAYLOAD ENVELOPES

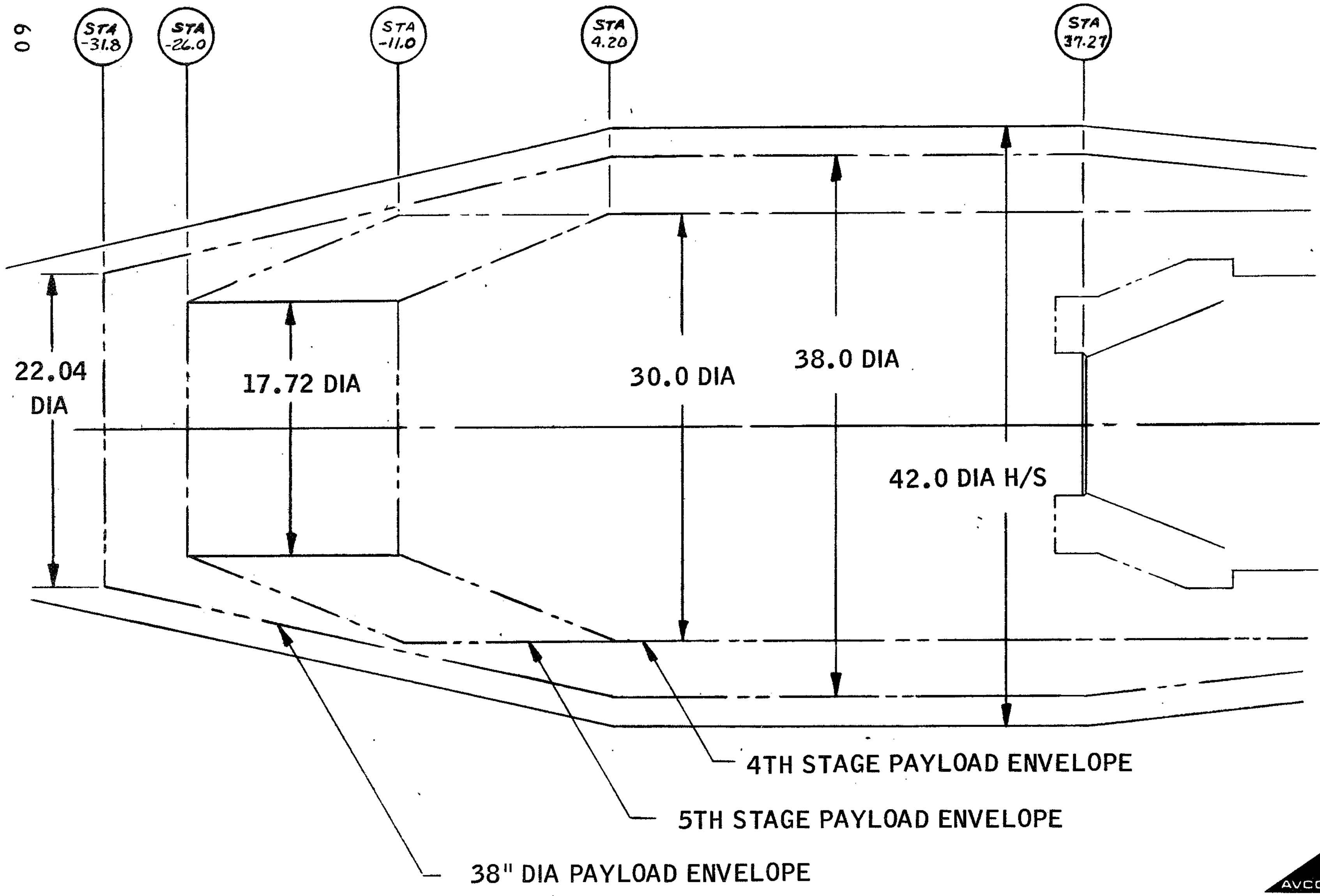


FIGURE 16 - BASIC SATELLITE WITH FORWARD SOLAR ARRAY

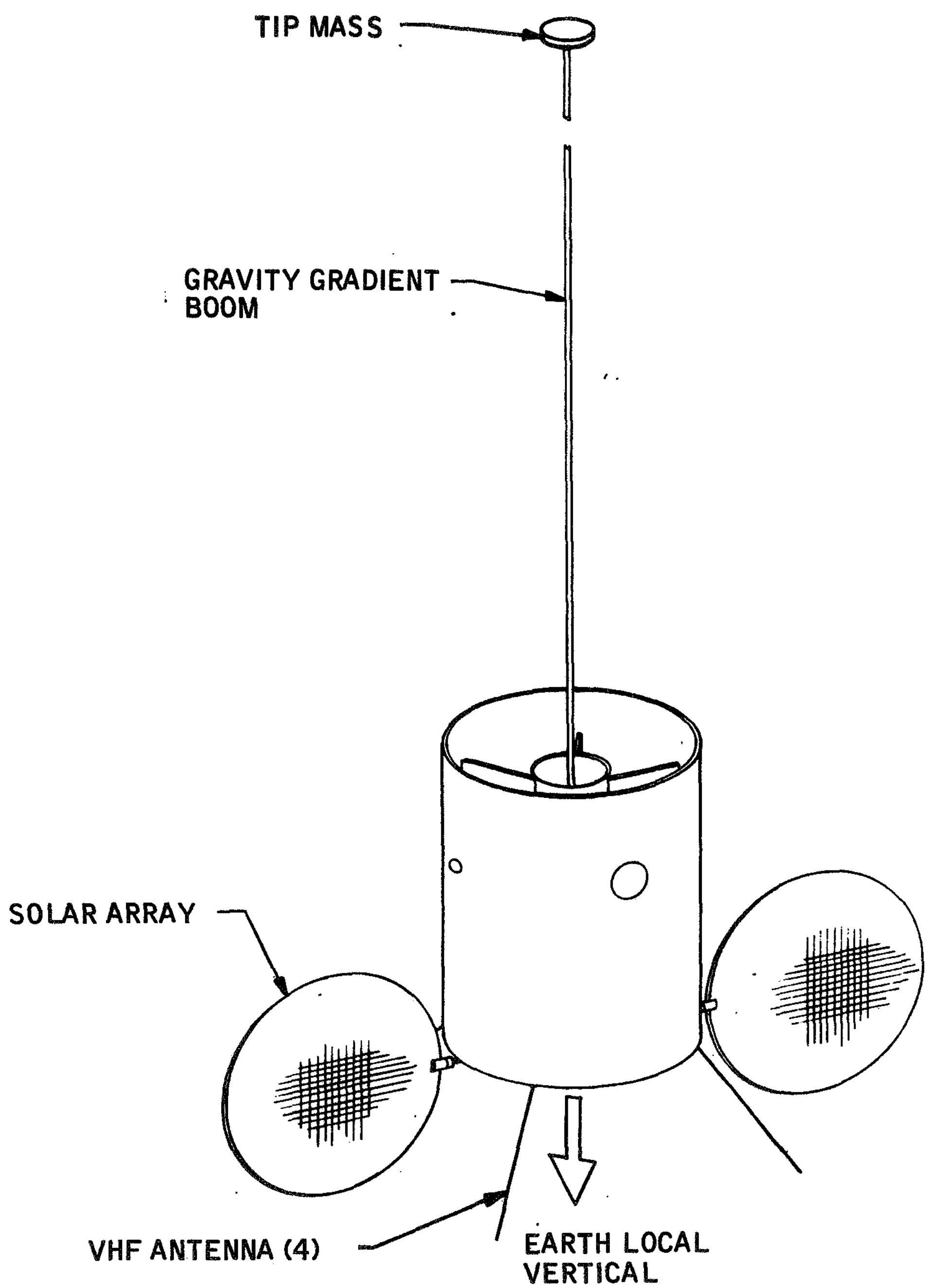


FIGURE 17 - BASIC SATELLITE WITH AFT SOLAR ARRAYS STORED

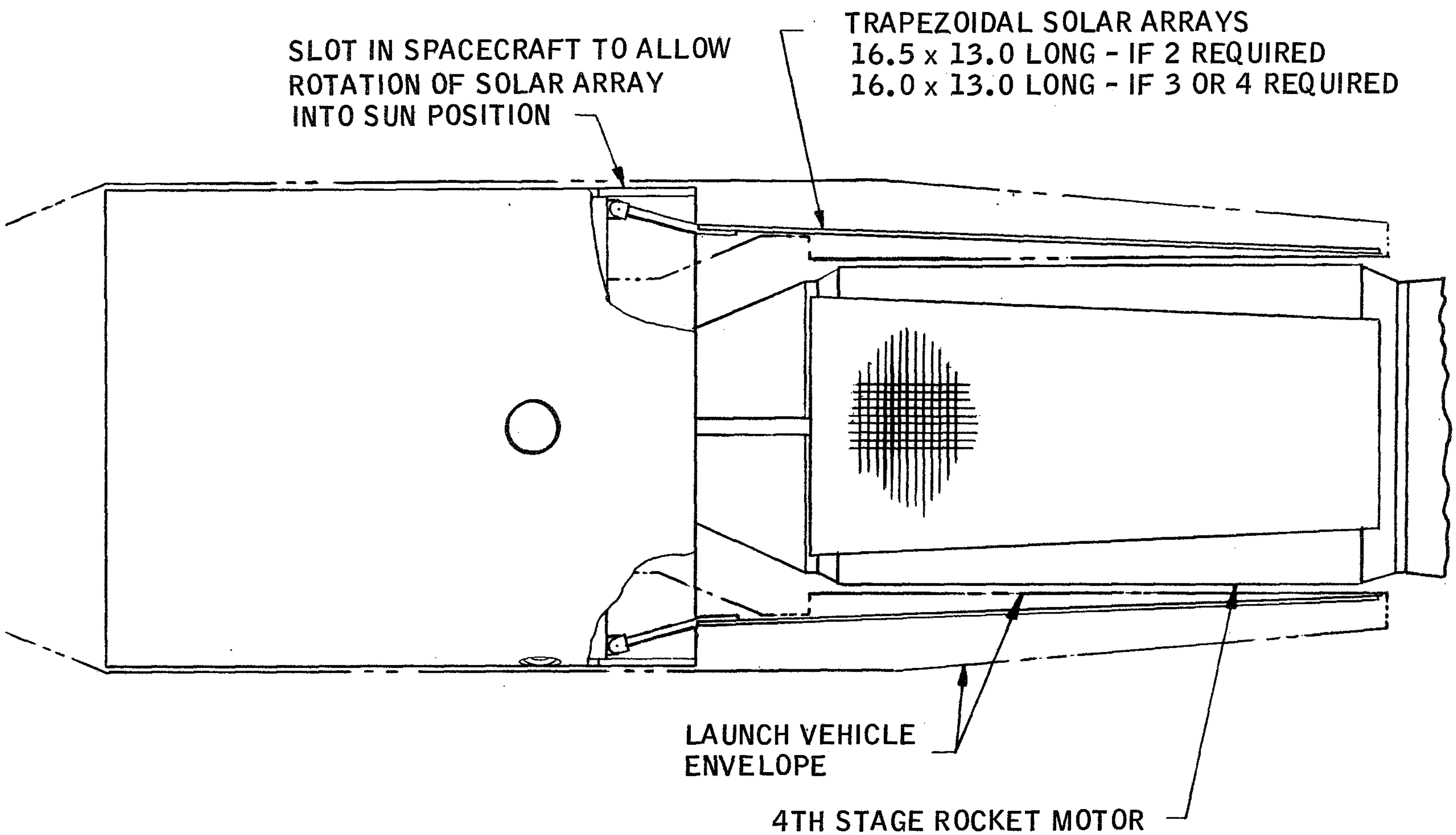


FIGURE 18 - PIGGYBACK SATELLITE CONCEPT

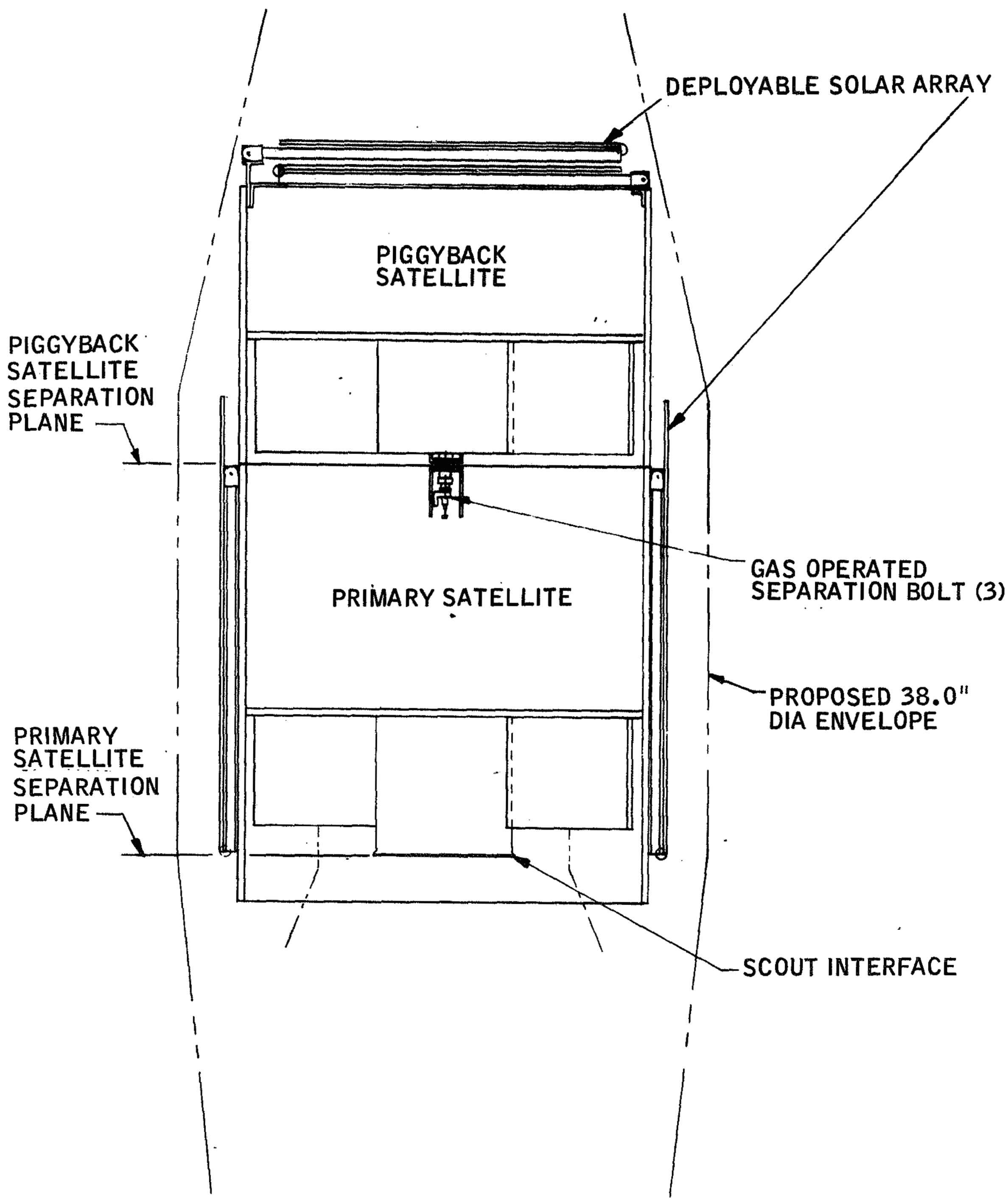


FIGURE 19 - SATELLITE WITH TYPICAL EXPERIMENT INSTALLATION

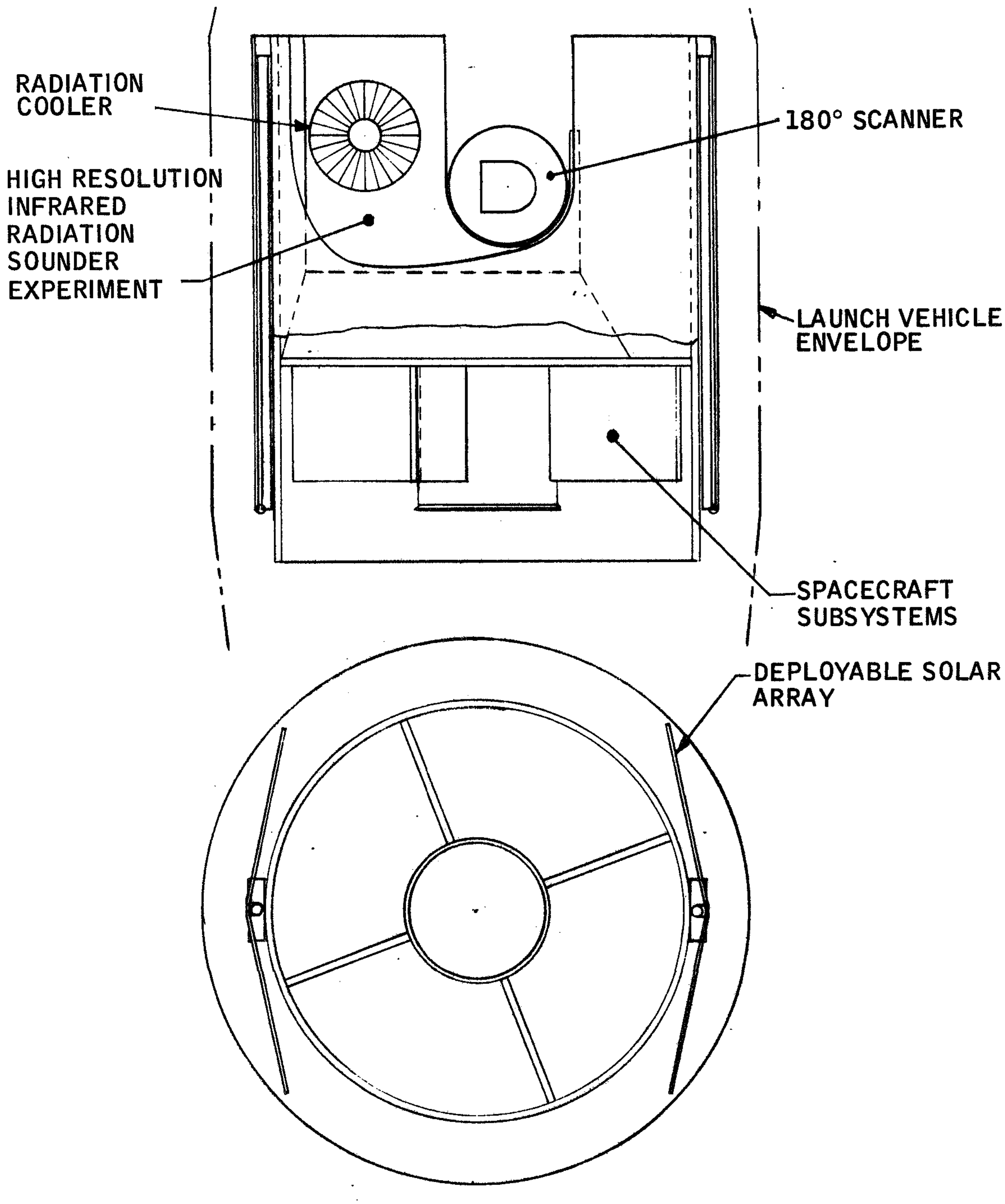
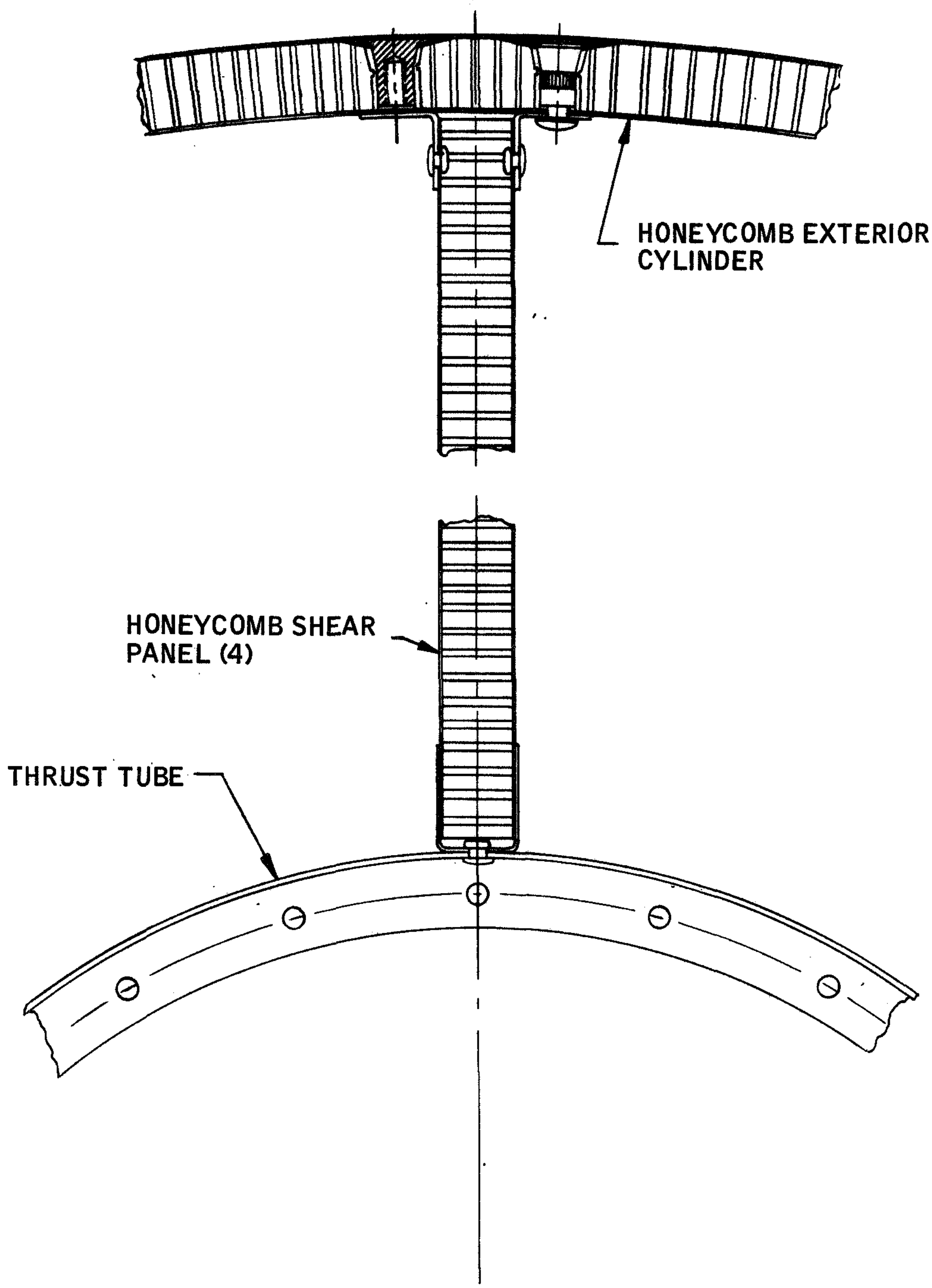


FIGURE 20 - STRUCTURAL DETAILS OF SUPPORT SYSTEM PLATFORM SUBASSEMBLY



The instrument mounting shelf, along with the shear panels, will serve as the mounting platform for the various spacecraft subsystems. The shelf also acts in shear to transfer the torsion loads resulting from spinup and despin of the spacecraft, and the lateral loads resulting from lateral accelerations and vibration experienced during launch. The shelf will be fabricated from an aluminum honeycomb sandwich panel which, by its nature, has extremely high strength and stiffness in relation to its weight. A thickness of 0.5 inch was selected with face sheets consisting of a high strength aluminum alloy such as 7075-T6. This combination will provide adequate support for any packages which might be mounted on it, and is extremely flexible from the standpoint of location of packages, since light weight threaded inserts can be installed almost anywhere after the panel is fabricated. The shear panels, like the mounting shelf, will also serve as the mounting point for spacecraft equipment. In addition, they will transfer the vertical loads and bending moments from the exterior cylinder to the thrust tube. Equipment mounting flexibility and strength will be attained through use of similar honeycomb material combinations for the shear panels.

The Exterior Cylinder: The exterior cylinder will consist of a single monocoque cylinder of aluminum honeycomb sandwich construction. This cylinder, in addition to carrying a large portion of the total spacecraft weight throughout the launch environment, will perform the following functions.

(1) Provide the substrate for any body mounted solar array.

(2) Serve as a spacecraft cover and thermal control barrier.

(3) Provide internal mounting hardpoints for instruments and subassemblies.

(4) Provide external mounting hardpoints for booms, deployable solar arrays, antennas, or a yo-yo despin device.

(5) Provide internal or external mounting hardpoints for a piggyback satellite separation system.

The large exterior surface has sufficient area for a solar array to provide power to spacecraft subsystems during the acquisition phase and operational phases, and to provide low power for experiments with modest requirements during the operational phases, without resorting to deployable arrays.

The very high strength and stiffness of the honeycomb cylinder permits mounting of experiments or other sensors directly without the necessity of providing complex bracketry, and permits viewing ports of substantial size without compromising structural integrity.

A piggyback satellite of construction similar to the basic spacecraft can be mounted through a separation system attached directly to the cylinder (See Figure 18). Such an installation eliminates the necessity for providing structure to carry the loads into the center of the spacecraft, as would be required if a Scout type separation system were used to separate the piggyback satellite.

Forward Cover: The forward cover, like most of the rest of the structure, is fabricated from aluminum honeycomb sandwich. The forward cover is simply bolted to the forward end of the cylinder, in which appropriate inserts are provided. Access to the interior of the spacecraft is provided by removal of the cover. The forward cover can provide the structural support for mounting of earth looking experiments of other packages of moderate size. Very large experiments will be mounted on the inside of the cylinder, on the mounting shelf, or on simple supports attached to those elements.

In addition to the basic components described above, the flexibility of the design accommodates a variety of elements which might be necessary to meet the special requirements of a particular experiment. For example, an intermediate shelf system mounted within the cylinder would provide mounting surfaces for an experiment which required a large mounting area.

Structural Design: A preliminary analysis has been performed to determine the sizes and weights of the various structural elements of the spacecraft. The results of that analysis indicate that a single structural design concept can accommodate the loads which will exist for various spacecraft gross weights, without significant weight penalty.

It is anticipated that the thrust tube will be machined from a forging of high strength aluminum alloy, with a nominal wall thickness of 0.050 inch, limited by machining capability. However, the use of alternate materials, and alternate designs, can be considered in the detailed design.

All of the honeycomb components will be fabricated using 0.010 inch thick high strength aluminum alloy face sheets and a total thickness of 0.50 inch. The 0.010 face sheets thickness is considered a minimum for supporting attachments and for resisting damage during handling. The stresses in the face sheets will generally be low, except when a large panel, such as the forward cover is loaded in bending if a very large package is attached to it. In such cases a minor increase in core or facing thickness can easily be made to accommodate such unusual conditions.

Since the three major subassemblies can be easily separated, the construction of the spacecraft, and the integration of components, can proceed on a parallel basis. The approach to building the spacecraft will proceed as follows:

(1) The support system platform will be fabricated and assembled for integration of the supporting subsystems (attitude control, communications, etc.). The design can accommodate a wide range of package positions and mounting arrangements, which can be installed after the fabrication and assembly of the platform is complete.

(2) The exterior cylinder will be fabricated complete with attachment points for the other components, and with viewing ports for attitude control sensors, as required. Provision will be made for attachment of experiments by providing a pattern of inserts. Requirements for special mounting features and viewing ports may be defined after the cylinder is fabricated, if necessary, but major modifications should be made prior to installation of the solar array.

(3) The forward cover can be fabricated and, if desirable, provided to the experimenter for mounting of experiments and equipment. Alternatively, the experimenter can be provided with a forward cover which will be a duplicate of the one which will be flown, allowing him to work out many of his integration problems. Any intermediate shelves to be installed in the experiment cylinder area would be treated in the same manner as the forward cover.

(4) The completed spacecraft subassemblies can be assembled into a spacecraft with a minimum of effort, and a minimum of potential conflict. The interfaces involved in such a system would be few and simple.

While the foregoing description is oversimplified due to lack of the detail of a specific application, it is clear that the concept promises to greatly reduce the problems which would exist if one were to try to design a "universal" spacecraft in such a way that the interface conflicts of each experiment, each subsystem, and each structural detail had to be treated for each possible application.

Mass Properties: The structural weight summary for a basic spacecraft of the type shown in Figure 16, and for a possible piggyback satellite of similar design shown in Figure 18, are summarized in Table X. Spacecraft weights using these structural weights, are summarized in Table XI.

TABLE X

STRUCTURE WEIGHT SUMMARY

Basic Spacecraft

Cylinder	0.33 lb/inch x 36 inch	12.0
Fwd Cover		2.5
Aft Mounting Plate		2.5
Instrument Shelves (4)		1.5
Interface Ring		1.5
Hardware (Inserts, Fasteners, Brackets)		3.0
		23.0 pounds

Piggyback Spacecraft

Cylinder	0.33 lb / inch x 18 inch	6.0
Fwd Cover		2.5
Aft Mounting Plate		2.5
Instrument Shelves (4)		1.5
Interface Ring		0.7
Hardware (Inserts, Fasteners, Brackets)		3.0
		16.2 pounds

The available weight for experiments is greatly dependent on the selected orbit and on launch conditions, but under favorable conditions a maximum launch weight of approximately 350 pounds can be accommodated. It is clear that substantial reductions in weight could be made if certain elements such as deployable solar arrays or portions of the attitude control subsystem were not required for a particular mission.

Several combinations of spacecraft and experiment payloads can be postulated from the weight summary presented and the maximum launch weight available for the orbits considered.

(1) The basic spacecraft weight can be combined with experiments in excess of 100 pounds and still include a contingency in excess of 100 pounds.

(2) The basic spacecraft and a piggyback combination can include two experiments (one each) at approximately 25 pounds each, and still include a contingency in excess of 50 pounds.

(3) Selected orbit altitudes and reduced subsystem capability will allow even more margin.

TABLE XI

SPACECRAFT WEIGHT SUMMARY

Basic Spacecraft	
Structure	23.0
Power Subsystem	14.5
Body Mounted Solar Cells	10.0
Deployable Solar Array	12.0
Command and Telemetry Subsystem	7.0
Data Handling Subsystem	9.1
Attitude Determination Subsystem	22.8
Passive Attitude Control/Stabilization Subsystem	10.1
Cold Gas System and Sensors	19.4
Sub Total - Less Experiment	127.9 Pounds
Piggyback Spacecraft	
Structure	16.2
Power Subsystem	14.5
Body Mounted Solar Cells	5.0
Deployable Solar Array	6.0
Command and Telemetry Subsystem	7.0
Data Handling Subsystem	9.1
Attitude Determination Subsystem	22.8
Passive Attitude Control/Stabilization Subsystem	10.1
Cold Gas System and Sensors	19.4
Separation System	3.0
Sub Total - Less Experiment	113.1 Pounds
E-Section Adapter	18.2
Total - Less Experiments	259.2 Pounds

Program Planning

Critical Issue. - To establish the proper perspective between the major subsystems under study and the balance of the spacecraft system within which these subsystems must operate, a nominal amount of program planning was undertaken. This planning considered the critical issues of how to determine when a given experiment is ready for integration with a small satellite and what effect the accommodation of a variety of experiments will have on these major subsystems.

In order to determine when an experiment is ready for spacecraft integration, test and launch, a clear understanding of the development status of the experiment hardware is necessary. Therefore, any integration planning has to provide techniques and methods by which a spacecraft contractor can evaluate an experiment for launch readiness. In addition, such a program plan should provide a "check list" of detailed program items that an experimenter uses in his development planning, in addition to the usual "mechanical/electrical" interfaces identified in a design and development effort. Effectively, this "check list" would be a matrix of interface items that will need general identification, detail design definition, test criteria, coordination, and several levels of hardware and software integration between the experiment and the spacecraft. Typical interface items to be considered would include but not be limited to the following:

(1) Electrical (power, voltage, regulation, plug/connector power level, etc.).

(2) Electromagnetic (communication interferences, RFI, etc.).

(3) Magnetic (interference, cleanliness, etc.).

(4) Radiative (nuclear and particle fluxes, solar, etc.).

(5) Mechanical (structural, attachment, Moments of Inertia, shock, vibration, transmissivity, etc.).

(6) Thermal (heat flow, temperature, attachment, emissivity, absorptivity, dissipation, etc.).

(7) Spatial (view factors, attachments, volume, positioning, alignment, etc.).

(8) Information (word format, bit rate, storage, programming, calibration, mode selection, etc.).

(9) Biological (cleanliness, contamination, evaporative deposits, etc.).

(10) Procedural (operating plans, checkout modes, assembly, inspection, qualification, test, handling, safety, etc.).

Management Approach. - Probably the best way to identify these interface requirements between the experiment and the spacecraft, would be to provide a "standard" format for the preparation of the proposals for each of the anticipated experiments. Such a standard would not only provide a frame work to assure that all of the experiment to spacecraft interface areas have been appropriately considered, but would also be beneficial to any comparison necessary to establish which experiments might be acceptable for a given series of launches. After this acceptability has been established, a "standard" format for an experiment to spacecraft compatibility questionnaire or specification would be "filled in" as a further tool to survey the experiments and determine development status and hardware availability. Figure 21 is a flow diagram which summarizes a typical small satellite program planning approach, from the end of the Phase B preliminary design through launch readiness for the first spacecraft and experiment payload.

The approach shown, assumes NASA will establish a small satellite project office as the NASA management control function, identified as the "NASA Project Office". This project office would be supported by a science/experiment oriented function identified as the "NASA Science Committee". An industrial contractor would be selected for the "Spacecraft Design and Integration" function, supported with the usual Scout launch vehicle contractor and NASA launch operations functions.

As shown, the planning sequence will begin with the completion of the spacecraft system Phase B definition, followed by a review of experiment proposals from the scientific community, and the selection of blocks of acceptable experiments. The spacecraft contractor will schedule the experiments according to the priority established as a result of a survey and evaluation to determine the status of the experiment hardware. The balance of the program activities are typical of the coordination and integration functions usually associated with a spacecraft system program.

Schedule. - The schedule shown in Figure 22 time phases the activities identified in the planning approach diagram, extended through twelve spacecraft launches in blocks of four each. The choice of four units in each block was arbitrarily selected to demonstrate the effectiveness of a concept to procure several sets of common hardware for similar spacecraft and spares requirements in each block. Not only will the purchase price be lower, but test and handling costs will be less to process hardware in lot quantities instead of individually. Also, the availability of extra hardware for succeeding flights reduces the need to stock spare long lead subsystem hardware.

Another feature of the program planning approach shown in the schedule, is the use of a Structural/Thermal Model (STM) and an Engineering Test Model (ETM). Normally, use of extra models of spacecraft in the development of a single flight article is expensive. With many spacecrafts involved, these model costs can be distributed over many units. These models will be used in the design and development of the first spacecraft,

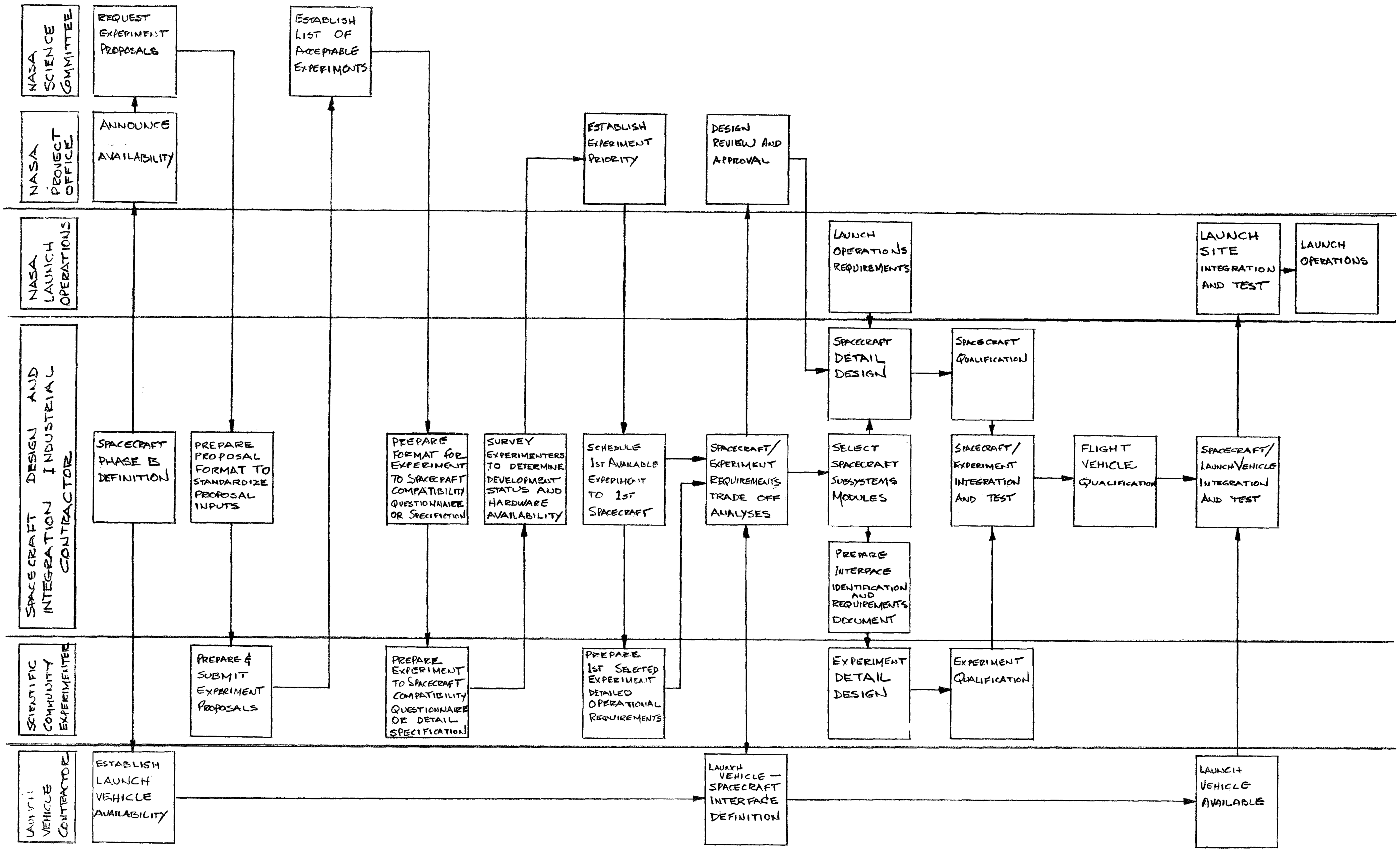


Figure 21 TYPICAL SMALL SATELLITE PLANNING APPROACH

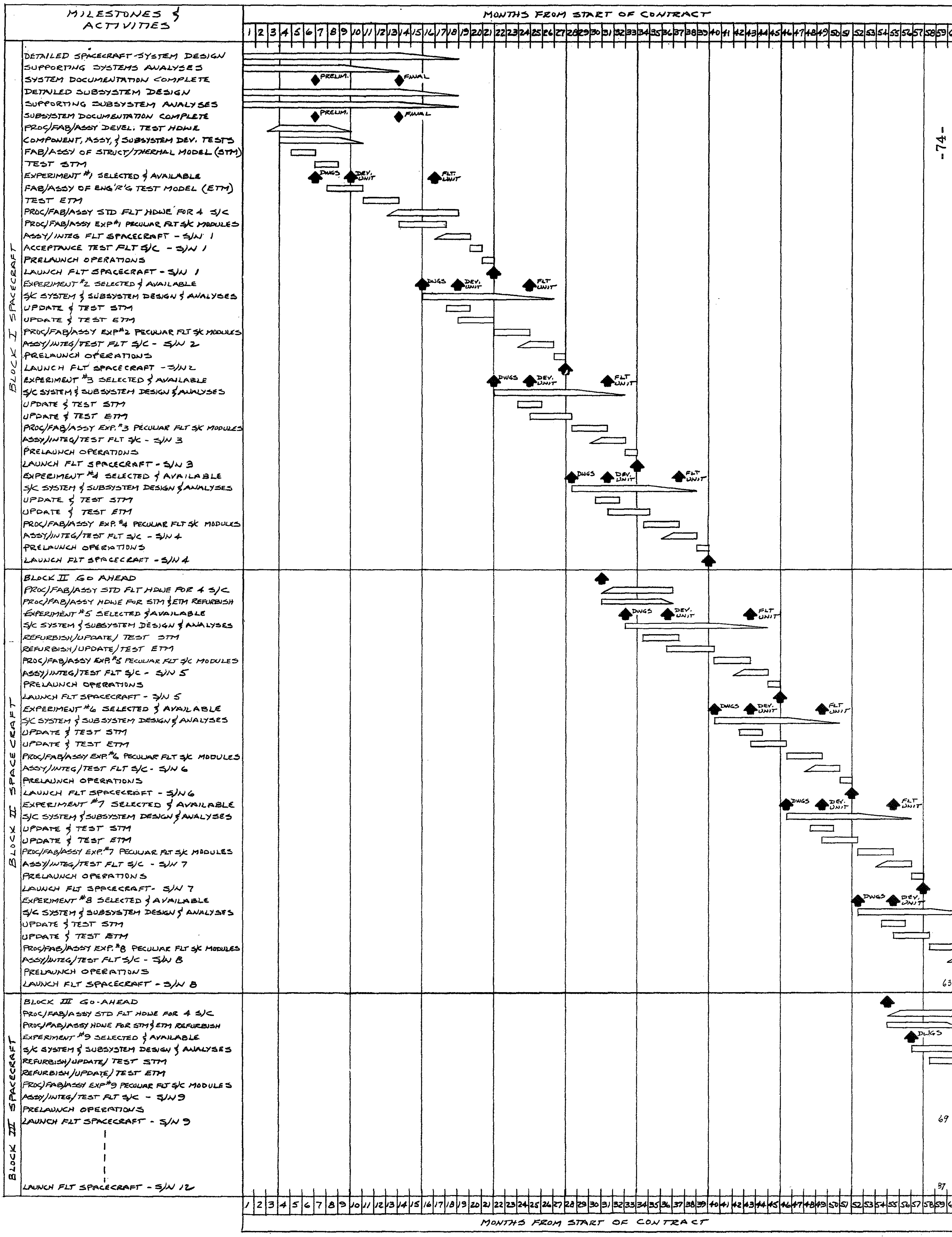


Figure 22 TYPICAL SMALL SATELLITE PROGRAM SCHEDULE

and will be available for all of the spacecraft which follow. It is expected that the ETM and STM will require minor modifications between each experiment installation, and possibly more extensive modification with each new block. Use of these models, not only for the initial development but for each follow on spacecraft, will also reduce integration time and cost.

The six month interval between launches, was selected as a convenient scheduling interval. There is no reason why three or four months between launches could not be accommodated. The time interval between the hardware program (Phase C/D) "go ahead" and the first launch, is more constraining. Similarly, the time interval is critical between the "go ahead" for each block and the first launch within the block. No schedule for the planning and preliminary design phases (A and B) is included because these two phases would take only a few months each, once they are funded. The scheduling of these early phases will depend on the priority established within NASA for a program of this type.

Cost. - Spacecraft systems costs were estimated for the first two blocks of four SC each, and for the "X"th block, also with four spacecraft. Cost shown in Table XII includes all of the design, development, and qualification cost for the spacecraft and its ground support equipment, plus integration with both the experiment and launch vehicle. Experiment and launch vehicle costs are not included.

S/N 1 costs include all of STM and ETM costs, plus qualification costs for anticipated new hardware developments. Even with new development problems and adequate model hardware, the costs shown are comparible to some of the lower price satellites today. These moderate costs, not only for the first unit, but even more dramatically shown for the first block of units, are a result of the recommended design concept, where known concepts and principles are combined in a versatile, cost effective space vehicle.

The effects of the "learning cycle" are seen by the ever decreasing costs of the SC in the first two blocks, even though economic growth factors would tend to increase costs. By the end of the second block, cost per copy should stabilize. The first SC in blocks II and X reflects some added costs for anticipated updating of the STM and ETM hardware.

TABLE XII

TYPICAL SMALL SATELLITE COST ESTIMATE SUMMARY
SC SYSTEM COST ONLY

Block I	Block II	Block "X"
S/N 1 - 2.7*	S/N 5 - 1.5	S/N - 1.3
S/N 2 - 1.9	S/N 6 - 1.3	S/N - 1.1
S/N 3 - 1.7	S/N 7 - 1.2	S/N - 1.1
S/N 4 - 1.5	S/N 8 - 1.1	S/N - 1.1
7.8	5.1	4.6

*Millions of Dollars

CONCLUSION AND RECOMMENDATION

As a result of the studies conducted and reported herein, it has been concluded that:

(1) A relatively simple, cost effective, small satellite is feasible for "quick response" implementation to evaluate a series of single experiments which are scheduled for utilization on future unmanned space flight programs.

(2) A hybrid attitude control subsystem, utilizing gravity gradient techniques, can be readily developed from demonstrated concepts to provide stable, three axis control and earth pointing of a small satellite.

(3) A complement of accurate sensors is available to; a) precisely measure spacecraft rates and position, and, b) supply control data to the attitude control subsystem during the experiment evaluation phase of the flight sequence.

(4) A versatile data handling and control subsystem can be implemented which requires only minor changes in software programming to accommodate various experiment requirements from one mission to the next.

(5) This same data handling subsystem will be capable of being reprogrammed in orbit to accommodate unforeseen circumstances or unpredictable phenomena which occur during the experiment operation phase.

(6) Command and communication subsystems are currently available for the small satellite and are compatible with STADAN in the 1970 time period.

(7) The conceptual design of the small satellite can accommodate an experiment volume in excess of seven cubic feet with a structural concept which allows hardpoint attachment at almost any point on the inside or outside of the spacecraft shell.

(8) All of the operating components in the recommended system are available either as qualified hardware items or through the application of state-of-the-art design approaches.

(9) The first of the small spacecraft can be available in less than eighteen months after the go-ahead of a Phase C/D design and development program.

It is recommended that funding be made available as soon as possible for follow-on phases of design and development, to furnish the scientific community with the small satellite "test bed" for a variety of earth pointing experiments. Many of these experiments are scheduled for operational flights in the mid 1970 period; for example, Nimbus, ITOS, and ERTS all plan flights starting in 1972, and will be operational throughout the 1970's. Availability of a "test bed" small satellite in 1972 is possible for support to these programs, as well as other operational programs thereafter.

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