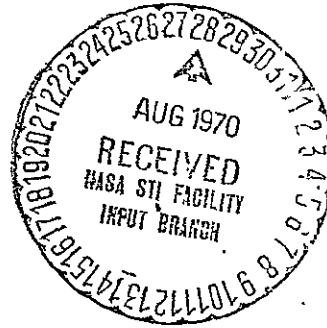


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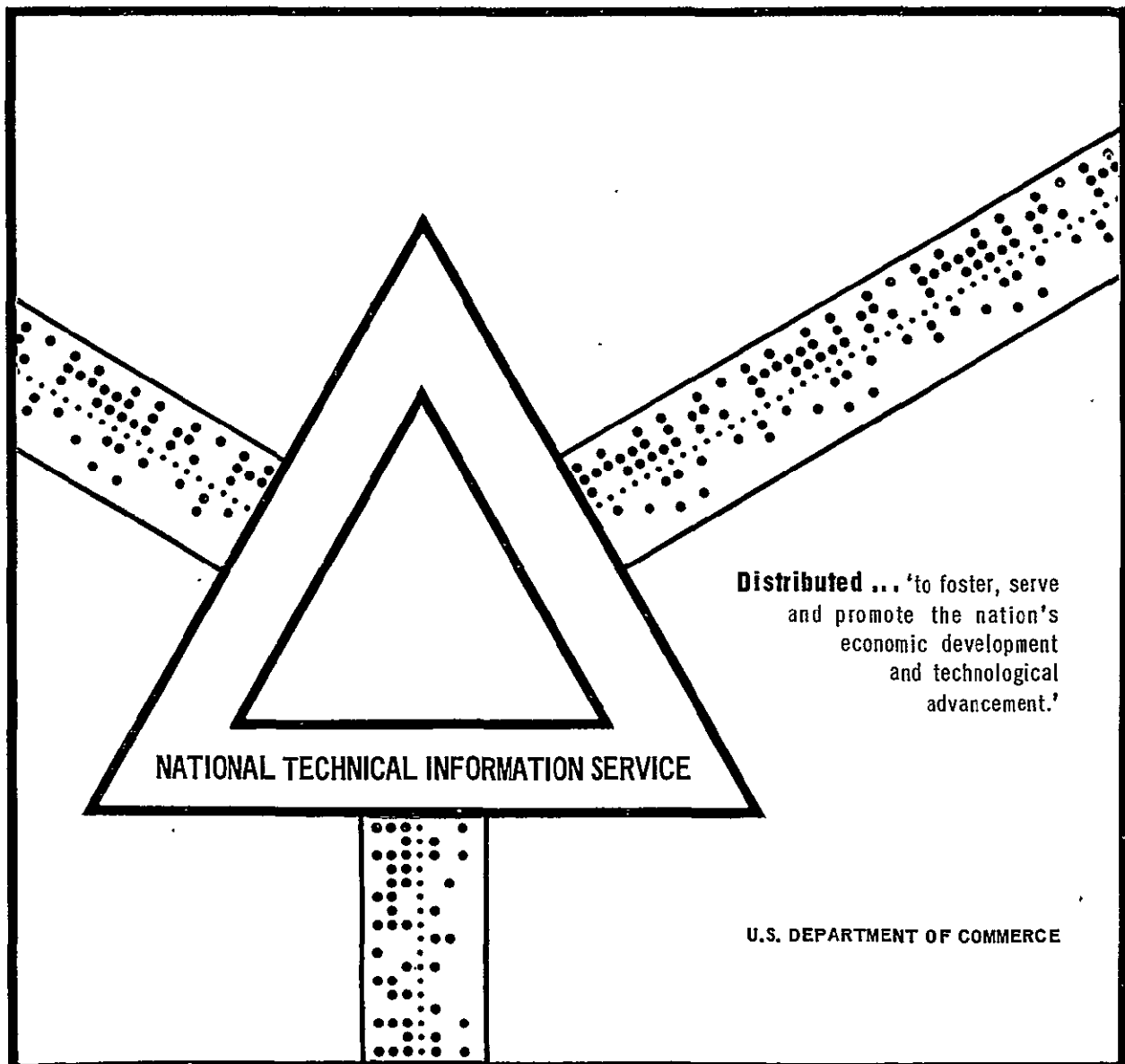
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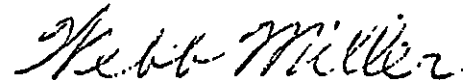
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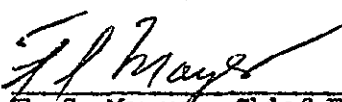
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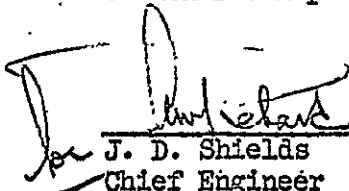
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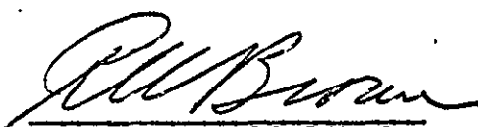

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ABSTRACT

A study was made of attitude control systems to provide several modes of control for the Saturn S-IVB Orbital Workshop to be used for the second Apollo Applications Program mission (AAP-2). Areas studied included control sensors, control logic electronics, reaction control propulsion, and thermal and environmental control systems. The control modes studied included gravity-gradient capture, local vertical hold, rate stabilization, and solar orientation. Solar panel electrical power was determined for several methods of solar orienting the solar panels; and the control impulse for these methods was determined as a function of attitude control gains and the angle between the orbit plane and the sun line. The recommended system utilizes horizon sensors, sun sensors and rate gyros to generate attitude error signals for pulsing the reaction control thrusters. The propulsion system utilizes the Saturn V S-IVB APS modules with new engines and tanks. Dual redundant components and failure detection logic are used to improve system reliability.

DESCRIPTORS

Orbital
Workshop
Attitude
Control
Gravity-Gradient
Propulsion Systems

Solar
Redundancy
Thermal Conditioning
Electronics
Dynamics
Impulse
Orientation

PREFACE

Douglas Aircraft Company, Saturn Development Engineering, has performed a study of active attitude control systems for the S-IVB Orbital Workshop under Supplemental Agreement 1441 to NASA Contract Nas7-101.

This final report presents the results of studies in the areas of propulsion systems, thermal and environmental control systems, control electronics, and control sensors.

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1.0 INTRODUCTION

Douglas has performed studies, as defined by MSFC in reference 1, of attitude-control requirements and system-mechanization concepts for a Workshop Attitude Control System (WACS) for the S-IVB Orbital Workshop (OWS). The system is to be installed on the second Apollo Applications Program (AAP) vehicle for the AAP-2 mission. Attitude control requirements have also been specified for the S-IVB Stage of the AAP-4 mission; however, these requirements are much less severe in terms of impulse required, duration of operation, and number of modes of operation than those for the AAP-2 mission. Consequently, the study has concentrated on the AAP-2 mission on the assumption that the same system can be used on the S-IVB Stage for the AAP-4 mission.

The AAP-2 OWS is used for two missions, designated Mission A and Mission B. The configuration of the cluster used for these missions and the vehicle coordinate system used in this study is shown in figure 1-1. The elements of the OWS cluster configuration include the Airlock Module (AM), Multiple Docking Adapter (MDA), Command and Service Module (CSM), and the Lunar Module and Apollo Telescope Mount (LM/ATM).

The Mission profile is shown in figure 1-2. For Mission A, the S-IVB/OWS/AM/MDA will be launched by the AAP-2 vehicle into a 260 nautical mile circular orbit of 28.5 degrees inclination. The CSM, with the crew, will be launched by the AAP-1 vehicle into a lower elliptical orbit, separate from the S-IVB Stage of the launch vehicle, and perform a rendezvous and docking maneuver with the OWS cluster. After a period of up to 28 days in orbit, the CSM will separate from the OWS for deorbit to return the crew to earth.

Following Mission A, the OWS will be placed in a gravity-gradient orientation with the MDA down for a three to six month storage period.

For Mission B, the CSM, with the crew, will be launched by the AAP-3 vehicle, and the LM/ATM by the AAP-4 vehicle. The CSM will separate

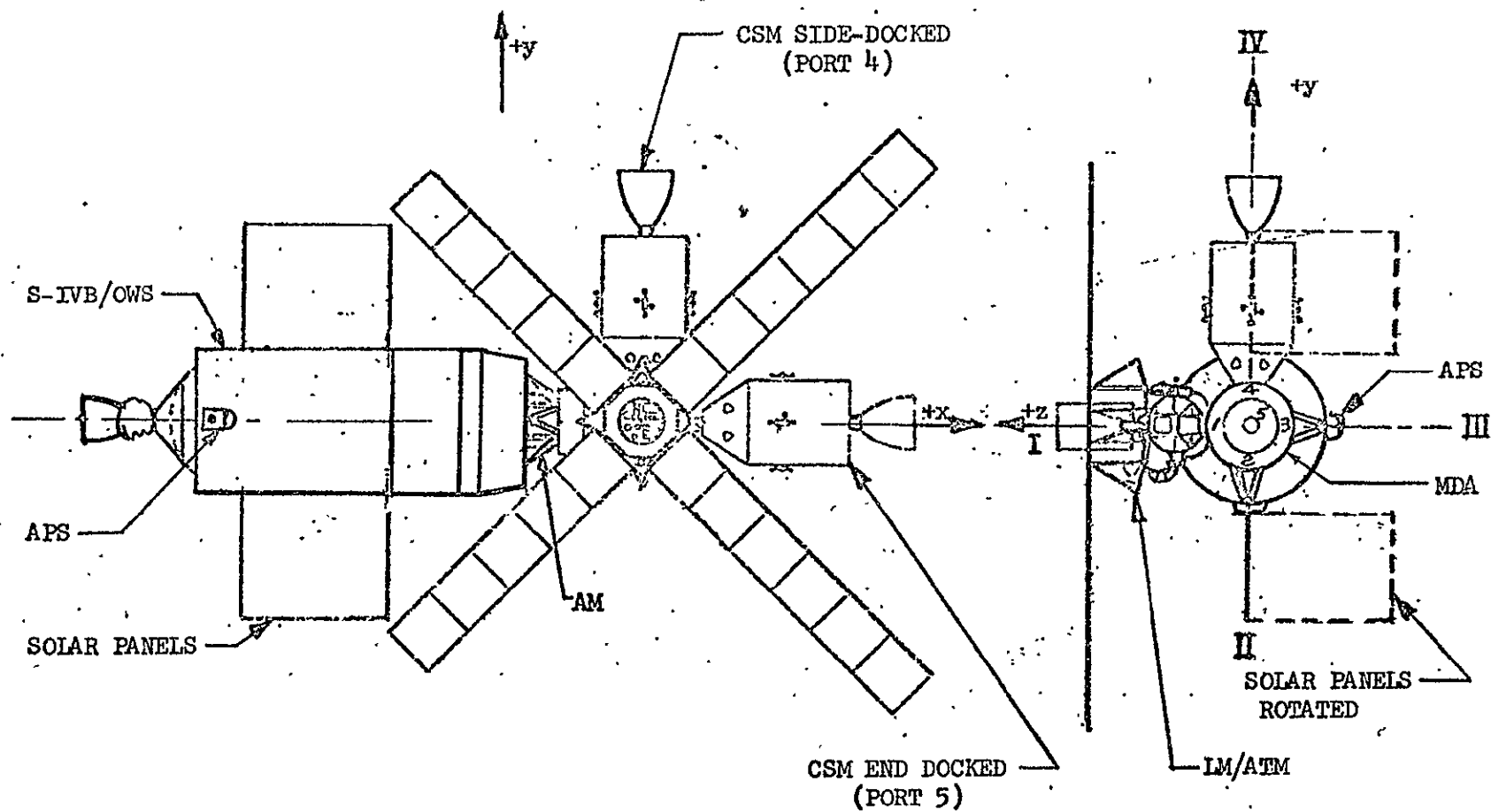


FIGURE 1-1 CLUSTER CONFIGURATION

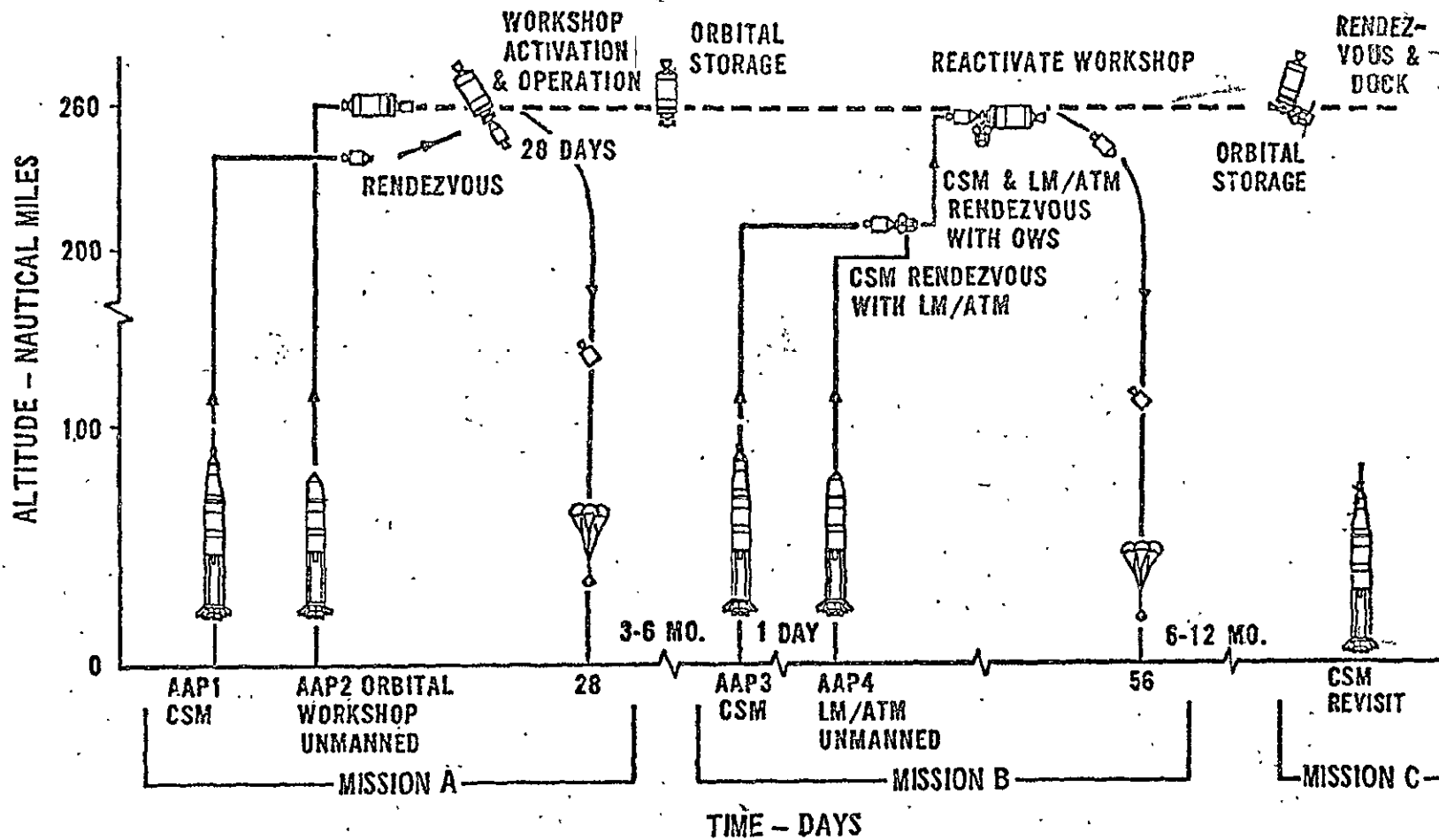


FIGURE 1-2 ORBITAL WORKSHOP MISSION PROFILE

from the launch vehicle and rendezvous and dock to the LM/ATM. After extraction of the LM/ATM from its launch vehicle, the CSM/LM/ATM will effect a rendezvous with the OWS/AM/MDA or AAP-2. The LM/ATM will then be docked to side port number 1 of the MDA, and the CSM will dock to the MDA axial port (number 5).

Following Mission B, the OWS/AM/MDA/LM/ATM will be placed in a gravity-gradient orientation with the MDA down for a possible revisit after a six to 12 month storage period.

At various times during the mission, attitude control capability is required of the OWS because of the unavailability of the CSM or LM/ATM, while at other times when attitude control is required, it could conceivably be provided by either the OWS, the CSM, or the LM/ATM.

Specific OWS attitude control requirements which were specified for the study are given in section 2.

1.1 Study Requirements

The study was divided into two phases as defined in reference 1.

Phase I considered attitude control systems to provide all of the required attitude control functions. Impulse requirements for these functions were specified by MSFC in reference 2. For the Phase I study, the WACS was required to provide pitch and yaw control to hold the gravity-gradient orientation throughout Mission A.

The areas of evaluation for attitude control were specified in reference 1 as:

1. Propulsion Systems (hypergolic, monopropellant, and compressed gas)
2. Thermal and environmental control systems
3. Control logic and electronics
4. Control sensors (gyros, horizon sensors, etc.)
5. Structural installation

Guidelines for evaluating alternate systems, as specified in reference 1, included:

1. System and component reliability
2. System weight
3. Schedule compatibility
4. Cost
5. Power requirements
6. Vehicle control system installation
7. Availability of qualified components

The study was extended in Phase II to study alternate orientations during Mission A which were specified by MSFC for increasing both solar panel power and thermal heat input to the OWS over the levels obtained with gravity-gradient orientation. The Phase II study, as specified in reference 1, included evaluation of the following for the specified orientations:

1. Control impulse requirements
2. Available solar panel power
3. Recommended implementation (hardware mechanization)
4. System weight
5. Power requirements
6. Reliability

This report includes results of both Phase I and Phase II studies. In the following sections, system requirements, control system mechanization concepts, results of attitude control dynamic analyses and simulations, solar panel power, and hardware for electronic, propulsion and thermal conditioning subsystems and S-IVB Stage modifications to accommodate this system are discussed.

The propulsion system designs presented in this report were based on a launch date which allowed a system development cycle of only 18 months. This accelerated development cycle made necessary the maximum utilization of developed and qualified hardware and as a result system optimization was not possible.

2/0 SYSTEM REQUIREMENTS

The WACS is required to provide attitude control of the S-IVB/OWS/AM/MDA which will be launched on AAP-2, and the S-IVB/LM/ATM which will be launched on AAP-4. The basic requirements for AAP-2 are for stabilization prior to docking of other modules of the cluster mission (i.e., the CSM or LM), and for restabilization following disturbances caused by unsuccessful docking attempts or by undocking of other modules. In addition, it is required to provide attitude control for the cluster at times when the LM/ATM or CSM is unable to do so because of Reaction Control System (RCS) propellant limitations or other considerations.

Attitude-control capability is required on AAP-4 to stabilize the vehicle for rendezvous and docking, in the event the CSM is unable to dock with the LM/ATM before the present launch vehicle capability is exhausted.

The attitude-control requirements for AAP-2 and AAP-4, as established by MSFC, are contained in reference 2. This section presents these requirements. More detailed subsystem design requirements, which were derived from these system requirements and mission profile, are given in section 4.

2.1 Schedule

The WACS must be compatible with the existing AAP launch schedule in terms of procurement, testing, qualification, and implementation. This requires control system accommodations installation on the AAP-2 launch vehicle during the first quarter of 1969.

2.2 Lifetime and Environment

Lifetime and environmental requirements are implicit in the specified mission profile.

2.3 Phase I Requirements

The attitude control system requirements for the Phase I study (reference 2) included a sequence of attitude control functions or maneuvers for the entire mission, the duration of each function or maneuver, the impulse required for each function, and the control system performance requirements. These requirements are discussed in detail in the following sections.

2.3.1 Attitude Control Functions

The attitude control functional requirements for the AAP-2 mission are listed in the second column of table 2-1. The third column gives the duration of each control function, and the last column gives the limits on attitude and attitude rate and indicates those control functions performed by the CSM or LM/ATM attitude control systems rather than by the WACS. Table 2-2 lists similar data for S-IVB Stage control system for the AAP-4 vehicle. Data in tables 2-1 and 2-2 are taken from reference 2.

2.3.2 Control Impulse Requirements

The control impulse required for each attitude control function, as specified in reference 2 is listed in the fourth column of table 2-1 for the OWS of the AAP-2 vehicle, and in table 2-2 for the S-IVB Stage of the AAP-4 vehicle.

2.3.3 Control System Performance Requirements

Control system performance requirements, as abstracted from tables 2-1 and 2-2 are listed in table 2-3 together with other requirements from reference 2.

2.3.4 Docking Constraints

The docking constraints, which are specified in reference 2 as the basis for the applicable requirements in tables 2-1 through 2-3, are as follows:

Closing rate	0.5 ft/sec nominal, 1 ft/sec maximum
Lateral rate	0 to 0.5 ft/sec
Lateral displacement	0 to 1 ft
Angular displacement	10 deg in each axis
Angular velocity	0.1 deg/sec in each axis

2.4 Phase II Requirements

The Phase II requirements specified alternate orientations, in lieu of the gravity-gradient orientation of event 7 in table 2-1, for the cluster during the 28 days of Mission A. These orientations are more favorable for solar panel power and thermal input considerations. Requirements for the remainder of the mission were not changed.

TABLE 2-1
S-IVB/OWS CONTROL SYSTEM REQUIREMENTS (AAP-2)*

EVENT	ATTITUDE CONTROL FUNCTION	TIME	IMPULSE REQUIREMENT (N-sec)	COMMENTS
1	Insert S-IVB-OWS (AAP-2) in gravity-gradient stabilized position	Before IU loses power (7.5 hours)	0	Use IU with S-IVB APS
2	Maintain gravity-gradient position during CSM-M&SS** rendezvous	7.5 to 48 hours after AAP-2 launch	7,500	System must maintain pitch and yaw within $\pm 20^\circ$ of the local vertical, and can allow random roll attitude
3	Stabilize OWS after one successful and one missed docking attempt by CSM-M&SS	1/4 orbit allowed to remove docking disturbances	15,000	Rate stabilization only. OWS angular rates must be less than $0.1^\circ/\text{sec}$ about each axis
4	Remove undocking disturbances, and stabilize OWS-M&SS for CSM docking attempt	1 orbit allowed	3,500	Rate stabilization only. $0.1^\circ/\text{sec}$ angular rate
5	Stabilize OWS after one successful and one missed docking attempt by CSM	1/4 orbit allowed	15,000	Rate stabilization only. OWS angular rates must be less than $0.1^\circ/\text{sec}$ about each axis
6	Insert OWS-M&SS-CSM in gravity-gradient stabilized position with MDA down	Start of 23-day mission	0	CSM will perform this maneuver
7	Maintain gravity-gradient stabilized position	During 23-day mission	7,500	System must maintain pitch and yaw within $\pm 20^\circ$ of local vertical, and can allow random roll attitude
8	Position OWS-M&SS-CSM as may be required other than gravity-gradient	During 23-day mission	0	CSM will perform necessary maneuver and return the system to gravity-gradient stabilized position

* From Reference 2

** Mapping and Survey System

TABLE 2-1 (cont'd)

EVENT	ATTITUDE CONTROL FUNCTION	TIME	IMPULSE REQUIREMENT (N -sec)	COMMENTS
9	Reacquire gravity-gradient stabilized position after CSM undocking	End of 28-day mission	5,500	Vehicle must be left in storage so that MDA points toward the earth; vehicle is not allowed to tumble
10	Maintain gravity-gradient stabilized position during CSM & LM/ATM rendezvous	Start of 56-day mission for 3 orbits	7,500	System must maintain pitch and yaw within $\pm 20^\circ$ of the local vertical; random roll is acceptable (attitude)
11	Restabilize OWS after two docking attempts by LM/ATM	1/4 orbit after each attempt	10,000	System must remove docking disturbances from one unsuccessful docking attempt and one successful docking attempt to prepare for CSM docking; rate stabilization is 0.1/sec about each axis
12	Stabilize OWS after one successful and one missed docking attempt by CSM	1/4 orbit	2,000	Rate stabilization 0.1/sec about each axis
13	CMG* spin up and dump	Not defined	0	CSM will provide stabilization
14	Maintain gravity-gradient stabilized position in the event of LM/ATM free-fly or tether operation	Two days	7,500	System must maintain pitch and yaw within $\pm 20^\circ$ of local vertical
15	Stabilize cluster for LM/ATM redock after free-fly mode or tether operation	1/4 orbit	0	CSM will provide stabilization
16	CMG Despin	End of 56-day mission	0	CSM will provide stabilization, if this maneuver is required
17	Reacquire gravity-gradient stabilized position after CSM undocking	End of 56-day mission	1,000	This requirement exists only if CMG is inoperative

* Control Moment Gyro

TABLE 2-1 (cont'd)

EVENT	ATTITUDE CONTROL FUNCTION	TIME	IMPULSE REQUIREMENT (N-sec)	COMMENTS
18	Maintain gravity-gradient stabilized position for CSM revisit rendezvous	Not applicable	7,500	System must maintain pitch and yaw within $\pm 20^\circ$, and can allow random roll. Needed only if CMG's are inoperative
19	Stabilize OWS after one successful and one missed attempt by revisiting CSM	1/4 orbit	10,000	Rate stabilization only. $0.1^\circ/\text{sec}$ about all axes required. Not needed if CMG's are operating
TOTAL			99,500 (N-sec) 22,500 (lb-sec)	

TABLE 2-2

S-IVB CONTROL SYSTEM REQUIREMENTS (AAP-4)

EVENT	MANEUVER	TIME	IMPULSE REQUIREMENT	COMMENTS
1.	Insert S-IVB-LM/ATM (AAP-4) in gravity gradient stabilized position	Before IU loses power (7.5 hours)	0	Use IU with S-IVB APS
2.	Maintain gravity-gradient position during CSM rendezvous	7.5 hours to 48 hours	7,500	System must maintain pitch and yaw within $\pm 20^\circ$ of the local vertical, and can allow random roll
3.	Stabilize AAP-4 after one successful and one missed docking attempt by CSM	1/4 orbit required	2,000	Rate stabilization only. AAP-4 angular rates must be less than 0.1/sec about any axis
TOTAL			9,500 (N-sec) 2,100 (lb-sec)	

TABLE 2-3

CONTROL SYSTEM CAPABILITY REQUIREMENTS⁽¹⁾

Thrust:	50 Newtons (11.3 pounds)	
Attitude:	Pitch & Yaw	$\pm 20^\circ$ with respect to local vertical
	Roll	No control required
Rates:	Pitch	Orbital rate within ± 0.01 deg/sec ⁽²⁾
	Yaw	± 0.01 deg/sec ⁽²⁾
	Roll	± 0.01 deg/sec
Impulse	(Based on 8-meter: moment arm; S-IVB aft skirt)	
	AAP-2	99,500 N-sec (22,500 lb-sec)
	AAP-4	9,500 N-sec (2,100 lb-sec)

(1) Excluding boost and passivation (from reference 2):

(2) These requirements represent capability for minimum rates rather than maximum limitations. The requirements of table 2-1 are assumed to govern, since these requirements are not compatible with gravity-gradient-assisted attitude control and the $\pm 20^\circ$ attitude limit (see section 4.2). These rates may also be exceeded during docking and capture transients.

2.4.1 Solar Orientations for Mission A

The alternate cluster orientations for Mission A which were specified in reference 1 for study during Phase II are listed in table 2-4. The orientations listed fall into two general categories. The first are solar orientations for which the vehicle z-axis, which is normal to the plane of the solar panels when they are held parallel to the vehicle longitudinal axis, is solar oriented to maximize solar panel power and thermal input to the OWS. Included in this category are Cases 1, 2 and 3 in table 2-4. The difference between these cases lies in the type of attitude control that is utilized about this solar oriented axis. The criterion for selecting the method of control about this axis is the minimization of control impulse which is attributable primarily to gravity-gradient torques.

For Case 1, the longitudinal principal axis, which is assumed to be the axis of minimum moment of inertia, is controlled to lie in the orbit plane as shown in figure 2-1. The y-axis will thus be perpendicular to the local vertical at the terminator. Case 1A is identical except that the axis of mid-valued moment of inertia is held in the orbit plane, so that the axis of minimum moment of inertia is controlled to be perpendicular to the local vertical at the terminator. If the x-axis is the axis of minimum inertia, this will cause the y-axis to lie in the orbit plane as shown in figure 2-2.

For Case 2, attitude about the solar-oriented z-axis is uncontrolled; however, in order to limit the yaw rate, which could be induced by unsymmetrical roll engine impulse, yaw rate signals can be added to the roll error signal. This will cause pulsing of the roll thruster which will decrease yaw-rate. This orientation is shown in figure 2-3.

Case 3 is the same as Case 2 except the vehicle is spun about the yaw-axis at an optimum rate which is restricted to values equal to or less than 1.0 degree per second. This can result in a degree of spin stabilization so that impulse usage is reduced. This orientation is also shown in figure 2-3.

The other type of orientation is that of Case 4 in which control about the pitch and yaw axes holds the vehicle in the gravity-gradient orientation with the longitudinal principal axis along the local vertical. Roll attitude about the longitudinal axis is controlled to orient the solar

Table 2-4

SOLAR ORIENTATIONS FOR MISSION A

<u>CASE</u>	<u>DESCRIPTION</u>
1	z-axis toward the sun x-axis (principal) in the orbit plane
1A	z-axis toward the sun y-axis (principal) in the orbit plane x-axis (principal) perpendicular to the local vertical at the terminator
2	z-axis toward the sun x-axis free to rotate about the solar vector
3	z-axis toward the sun commanded to rotate about the solar vector at optimum rate to minimize WACS propellant consumption (Equal to or less than 1 deg/sec)
4	x-axis along the local-vertical Roll control for maximum sun orient- ation.
5	x-axis perpendicular to the orbit plane Roll control for maximum sun orient- ation Solar panels articulated for maximum sun orientation (Specialized version of Case 1A)

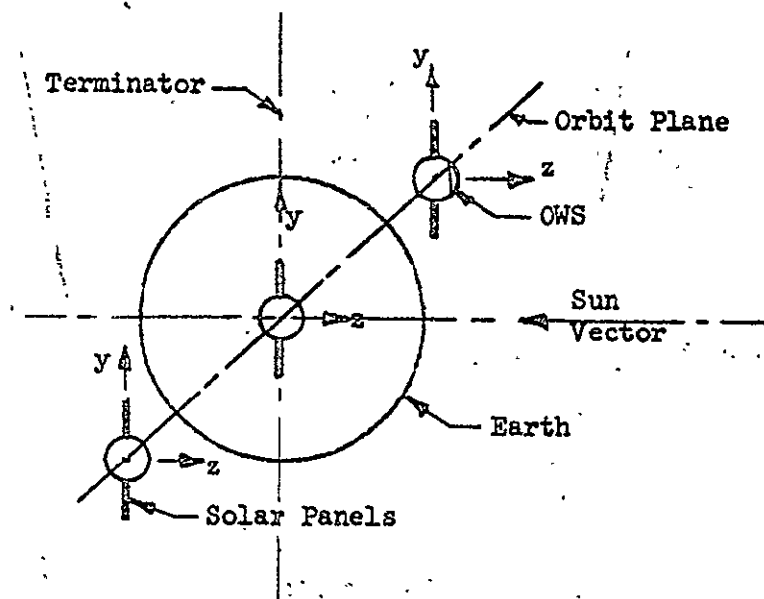


FIGURE 2-1 ORIENTATION CASE 1.

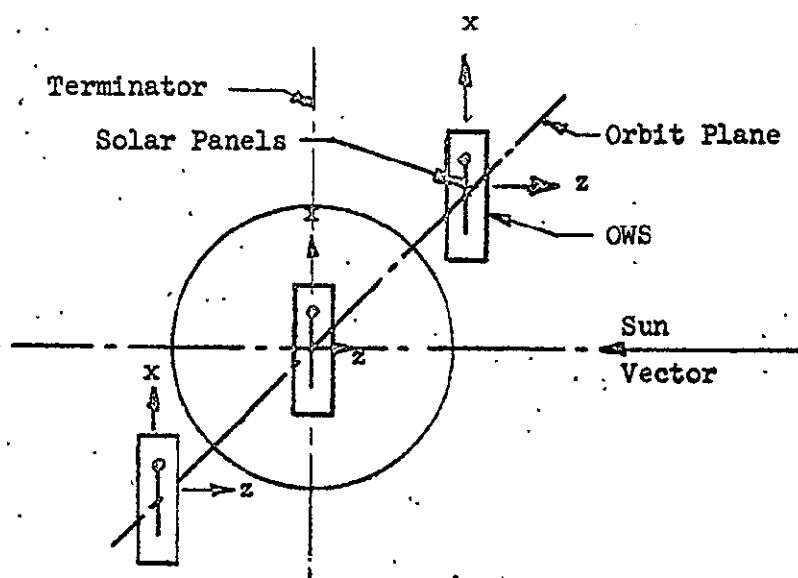


FIGURE 2-2 ORIENTATION CASE 1A,



panels toward the sun to the maximum extent consistent with the pitch and yaw control. Articulation of the solar panels (rotation of the panels with respect to the stage) for this orientation can further increase solar panel power. This orientation is shown in figure 2-4.

Another orientation has been defined for which impulse usage can be determined based on study results for Case 1A. For Case 1A with zero beta angle, (the angle between the orbital plane and the sun line) the longitudinal axis is perpendicular to the orbit plane. The results of this case at zero beta are thus applicable to the case in which the longitudinal axis is held perpendicular to the orbit plane, and vehicle roll combined with solar panel rotation or articulation about the y-axis maintains normal solar incidence on the panels. This case is identified as Case 5 in subsequent paragraphs, and is shown in figure 2-5.

2.4.2 Docking Configurations

Two CSM docking configurations were specified for study for each orientation, these were:

1. The CSM end-docked to docking port number 5.
2. The CSM side-docked to docking port number 4, which is on position plane IV making the CSM longitudinal axis parallel to the plane of the solar panels.

2.4.3 Attitude Deadbands

The effect of attitude deadbands between the limits of ± 5 degrees and ± 30 degrees was to be determined.

WORKSHOP ATTITUDE CONTROL SYSTEM NEAR SOLAR ORIENTATIONS

WACS-7
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CASE 4: X-AXIS ALONG LOCAL VERTICAL, ROLL ABOUT X-AXIS FOR OPTIMUM SOLAR PANEL ORIENTATION.

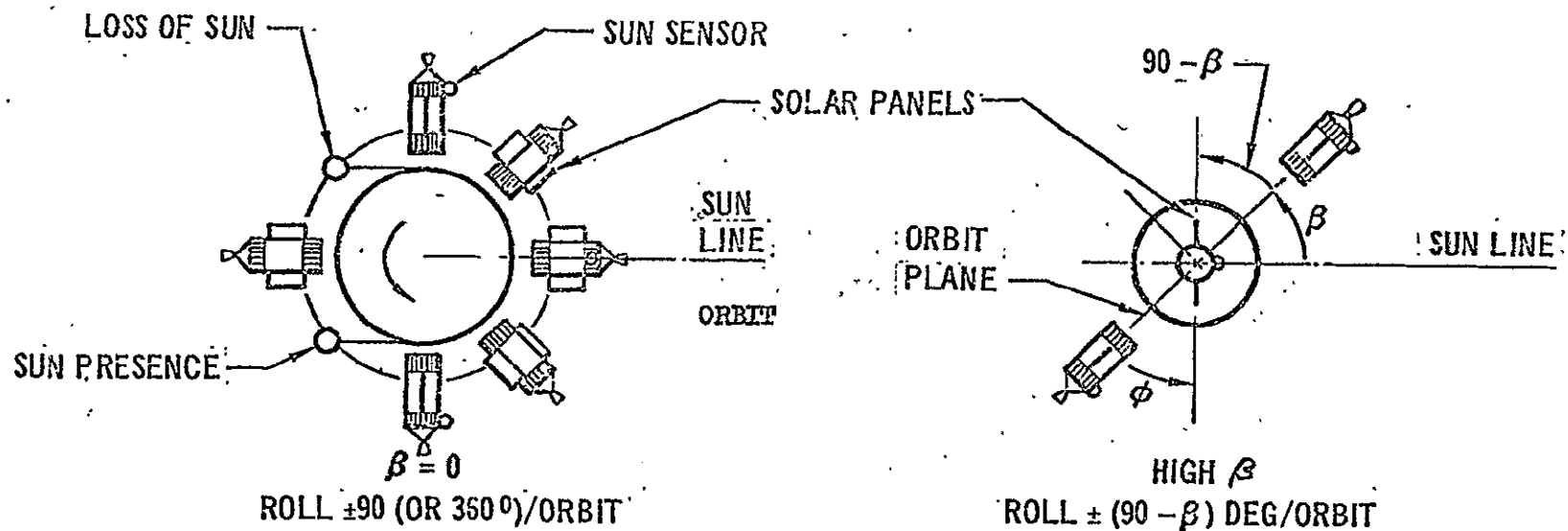


FIGURE 2-4 ORIENTATION CASE 4

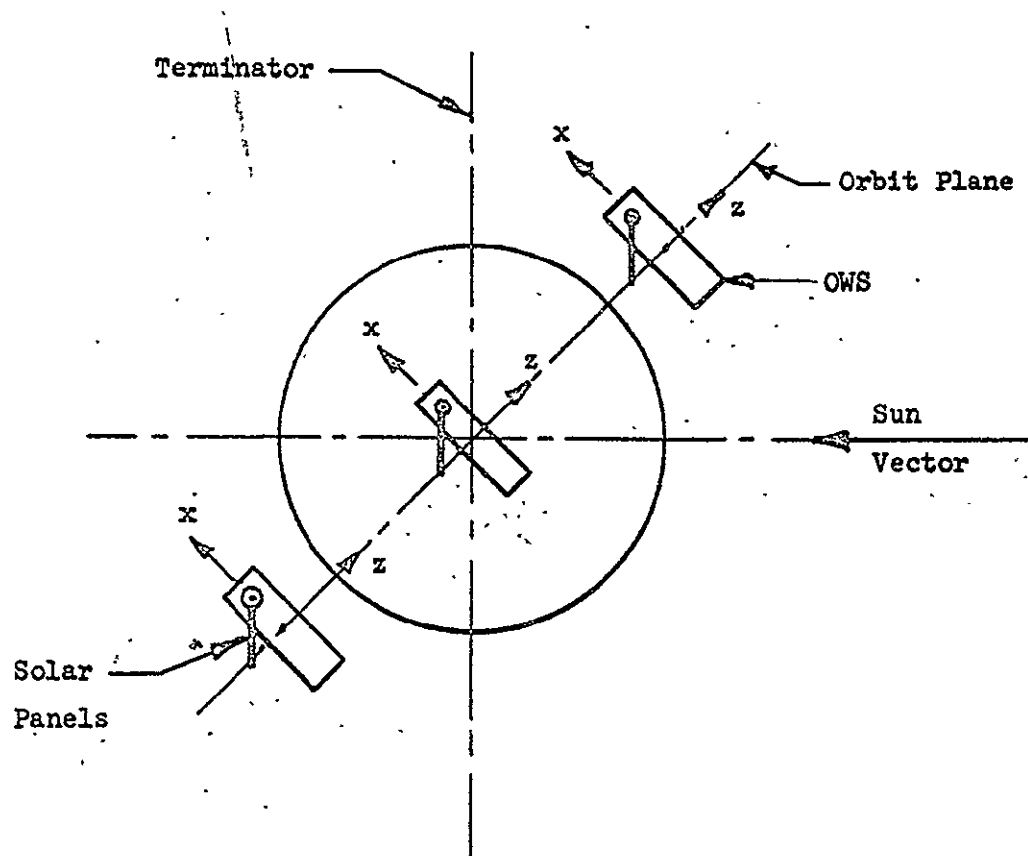


FIGURE 2-5 ORIENTATION CASE 5

3.0 CONTROL SYSTEM CONCEPTS

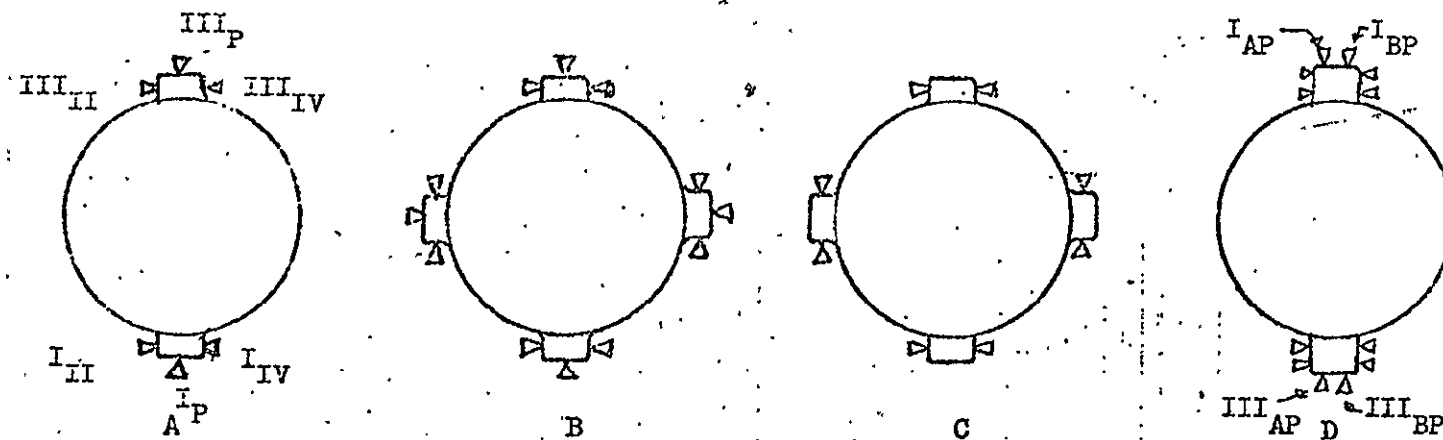
In selecting attitude control system concepts for the Orbital Workshop, three areas must be considered. These include the sensors for obtaining attitude and attitude rate signals, the control logic which commands corrections based on this intelligence, and the reaction-control thrusters and propellant system which provide the corrective moments. The choice of sensors is dependent primarily on the orientation and operational-mode requirements; the choice of reaction-control configuration is dependent primarily on vehicle configuration and disturbances; and the control logic is dependent primarily on operational-mode requirements and sensor selection, but must also be compatible with the reaction-control configuration. The considerations relating to the selection of concepts in each of these three areas are discussed in the succeeding paragraphs.

3.1 Thruster Location and Orientation

The configurations of the thruster cluster on the CSM and on the S-IVB represent two thruster configurations which could be utilized. The four thrust vectors for each CSM cluster lie in a plane which is tangent to the vehicle surface, so that pitch and yaw moments are obtained by thrusting parallel to the vehicle longitudinal centerline. The S-IVB thrust vectors for each module lie in a plane perpendicular to the longitudinal centerline of the stage, so that pitch and yaw moments are obtained by thrusting perpendicular to the vehicle longitudinal centerline. For a long vehicle such as the OWS, the S-IVB configuration gives the advantage of greater efficiency, in terms of propellant required to maneuver and counteract disturbances, because of the longer moment arm.

The candidate thruster configurations which were evaluated during this study are shown in figure 3-1. All have arrangements which are similar to the present S-IVB APS modules used on the Saturn IB or V. These configurations are compatible with existing propulsion hardware, and except for configuration C, require no change in the IU for control during boost and S-IVB Stage passivation. The thrusters have been assumed to be located on the S-IVB aft skirt because this longitudinal location provides the most efficient propellant usage as discussed in section 4.1.1.

VEHICLE VIEWED LOOKING FORWARD . .



- Notes:
1. Configurations indicate thruster arrangement only, internal configuration of modules and interconnections between modules are not specified.
 2. Thruster identification numbering system used for S-IVB is shown for configuration A. Modification of numbering system to indicate redundant thrusters shown for configuration D pitch thrusters.

FIGURE 3-1 CANDIDATE THRUSTER CONFIGURATIONS

Another selection which must be made is between adding additional Auxiliary Propulsion Subsystem (APS) modules for Workshop attitude control and replacing the present S-IVB APS modules with ones having larger propellant capacity, which can satisfy both the boost and coast requirements. The candidate thruster configurations shown in figure 3-1 are applicable to either approach, depending on the number of modules and the size of each module. The relative advantages of these two approaches are discussed in section 8 of this report.

3.2 Modes of Operation

The Phase I attitude control requirements for each mission event listed in tables 2-1 and 2-2 and the additional Phase II requirements can be grouped to arrive at seven modes of operation. The requirements and use of each mode are discussed in the following paragraphs, and summarized in table 3-1.

The LOCAL VERTICAL HOLD mode holds the longitudinal principal axis along the local vertical, which is the gravity-gradient stable orientation, within ± 20 degrees during rendezvous, and was considered for Mission A in the Phase I study.

The RATE mode is used to reduce vehicle body rates induced by unsuccessful docking disturbances (the CSM may control following successful docking) to less than 0.1 degree per second.

The GRAVITY-GRADIENT CAPTURE mode minimizes rate and attitude disturbances relative to the gravity-gradient stable orientation (with the MDA down) to ensure passive control to this orientation after active control is terminated for storage.

In the STORAGE mode, power is applied to all thermal conditioning systems in the APS modules and in the electronic assemblies, but all control loops are open. If tumbling cannot be prevented during storage by passive means, the pitch and yaw horizon sensors and associated electronics may be active in this mode. Alternative approaches include periodic activation of the GRAVITY-GRADIENT CAPTURE mode during storage, or the use of an additional mode, utilizing only the pitch and yaw horizon sensors, which would be activated periodically during storage to reduce oscillations to a small value.

TABLE 3-1

MODES OF OPERATION

WACS-5
8-22-67

<u>MODE</u>	<u>REQUIREMENTS</u>	<u>USE</u>
I U	NORMAL CONTROL BY I.U.	1. DURING BOOST 2. FIRST 7 1/2 HOURS ORBITAL COAST
LOCAL VERTICAL	PITCH } ± 20 DEG. OF GRAVITY-GRADIENT AND YAW } STABLE ORIENTATION, MDA DOWN ROLL } 0.01 DEG/SEC MAXIMUM	1. RENDEZVOUS
RATE	PITCH, YAW, & ROLL RATES ± 0.1 DEG/SEC MAXIMUM	DOCKING
GRAVITY-GRADIENT CAPTURE	PITCH } ATTITUDE DEVIATION FROM GRAVITY- GRADIENT STABILIZED ORIENTATION AND YAW } ATTITUDE RATES LOW ENOUGH TO ROLL } PRECLUDE TUMBLING WHEN DEACTIVATED. (UNDISTURBED), MDA DOWN	1. FOLLOWING CSM SEPARATION FOR REENTRY PRIOR TO ORBITAL STORAGE PERIODS 2. PRIOR TO RENDEZVOUS
STAND BY	POWER TO ALL ATTITUDE CONTROL COMPONENTS, CONTROL LOOPS OPEN.	1. DURING DOCKING ATTEMPTS 2. TELEMETRY DURING STORAGE 3. ACTIVATION
STORAGE	ALL POWER OFF EXCEPT THERMAL CONDITIONING	STORAGE
INTERROGATE (OPTIONAL)	POWER TO GYROS AND HORIZON SENSORS	TELEMETRY DURING STORAGE
SOLAR	ORIENT SOLAR PANELS TO SUN	MISSION A

In the STAND-BY mode power is applied to all attitude control components but all control loops are open. This mode will be used during control by the CSM or LM/ATM immediately prior to transfer of control to the WACS, and may be used intermittently during storage for telemetry.

The Phase II requirements add an additional SOLAR mode for control to align the solar panels normal to the sun vector during Mission A. The specific requirements for this mode depend on the orientation selected.

The IU mode permits control of the thrusters by the IU during boost and initial orbital coast, as for the normal S-IVB Stage.

3.3 Control System Concepts; Phase I Modes

In establishing control system concepts which can meet the requirements for the various modes of operation listed in table 3-1 which are applicable to Phase I, pitch and yaw requirements can best be considered separately from roll. Concepts for meeting the requirements for each mode of operation are discussed in the following sections, starting with the most critical. Although pitch and yaw requirements and mechanization are discussed together, in depicting system mechanization yaw and roll are shown together because common thrusters are used for obtaining yaw and roll control moments.

In the following figures, dual rate gyros, horizon sensors, spatial amplifiers, and thrusters are shown, however, approaches to redundancy for improving reliability by counteracting failures are discussed in section 3.5.

3.3.1 Gravity-Gradient Capture; Pitch and Yaw

The gravity-gradient capture mode will be used primarily to capture the passive gravity-gradient stable orientation, with the MDA down, after separation of the CSM, and prior to deactivating active attitude control for storage. It may also be used to obtain the correct orientation with the MDA down prior to activating the local vertical hold mode if tumbling has occurred. To capture the gravity-gradient orientation with the MDA down requires that the desired orientation be obtained within an allowable tolerance, which is a function of the maximum vehicle rates that are obtained

at the switching limits in this mode of control. These rates are functions of the minimum pulse width derived by the control logic.

The horizon is the best reference for sensing attitudes with respect to the local vertical. It can readily be sensed with sufficient accuracy to meet the attitude-control requirements in section 2, and solution of navigation equations is not required as for an inertial reference. If altitude is known, one horizon sensor in each of the pitch and yaw axes is adequate to determine the local vertical. The use of two sensors in each axis to sense the horizon on opposite sides of the vehicle can eliminate errors caused by altitude uncertainty, and provides redundancy which permits operation with reduced accuracy in the event of a horizon-sensor failure.

Damping signals can be provided from any of several sources, or a combination of these sources. These include; lead shaping of the horizon sensor attitude error signals, derived rate feedback around the switching amplifier, and rate gyro signals. Several factors influence the choice of damping signals. Both attitude error shaping and derived rate feedback provide damping signals which are proportional to the angular velocity relative to the gravity-gradient orientation, however, lead shaping of the horizon sensor output will provide no damping signals for attitude errors greater than the horizon sensor saturation limit. This will cause the switching lines in the phase-plane to become vertical at this attitude, and effectively eliminate damping for large attitude excursions.

Rate gyro damping signals about the axis perpendicular to the orbit plane include a bias equal to orbital rate when the vehicle is in the gravity-gradient orientation. This bias must either be nulled out, which is practical only if capture of a specific roll orientation can be assured, and roll attitude excursions are small, or rate gains must be low enough that orbital rate is smaller than the rate-axis intercept of the switching line in the phase plane. One approach to the solution of this problem is to use a non-linear rate gain so that the rate gyro gain is zero for rates which are less than a value somewhat greater than orbital rate. This, in conjunction with

lead shaping of the attitude error signal, will give switching lines which are symmetrical about the gravity-gradient null in the region of the phase plane near this null point, and provide an effective rate ledge for high attitude errors to limit peak rates, thus improving impulse usage during capture from large amplitudes. The switching lines in the phase plane which result from this system of control in the orbit plane are shown in figure 3-2 for typical gains. The figure shows that the switching lines are symmetrical about the gravity-gradient orientation in the small region around null, which is applicable to steady-state limit cycles, but is unsymmetrical for larger excursions from null as would be encountered during a large capture transient. Further simulation studies are required to establish acceptable gains and network time constants for this system concept.

A block diagram of the pitch axis attitude control system in the gravity-gradient mode, which incorporates the techniques discussed, is shown in figure 3-3. The offset command is used to account for the misalignment of the principal axes with respect to the body axes which is especially significant with side docked modules.

In determining the field of view required for the horizon sensors, several factors must be considered. These include:

1. The view angle required for the ± 20 degree switching limits required for local vertical hold.
2. The view angle required for the lower switching limits of approximately ± 5 degrees required for gravity-gradient capture.
3. Variations in view angles required to accommodate the variations in principal axes orientation caused by side docked vehicles.
4. The view angles required for both opposing horizon sensors to be active at each switching limit.
5. The upward field of view required to ensure a right-side-up capture following large attitude excursions.
6. The desirability of a continuous wide-angle field of view to obtain a continuous attitude-error signal which is amenable to shaping to obtain damping signals.

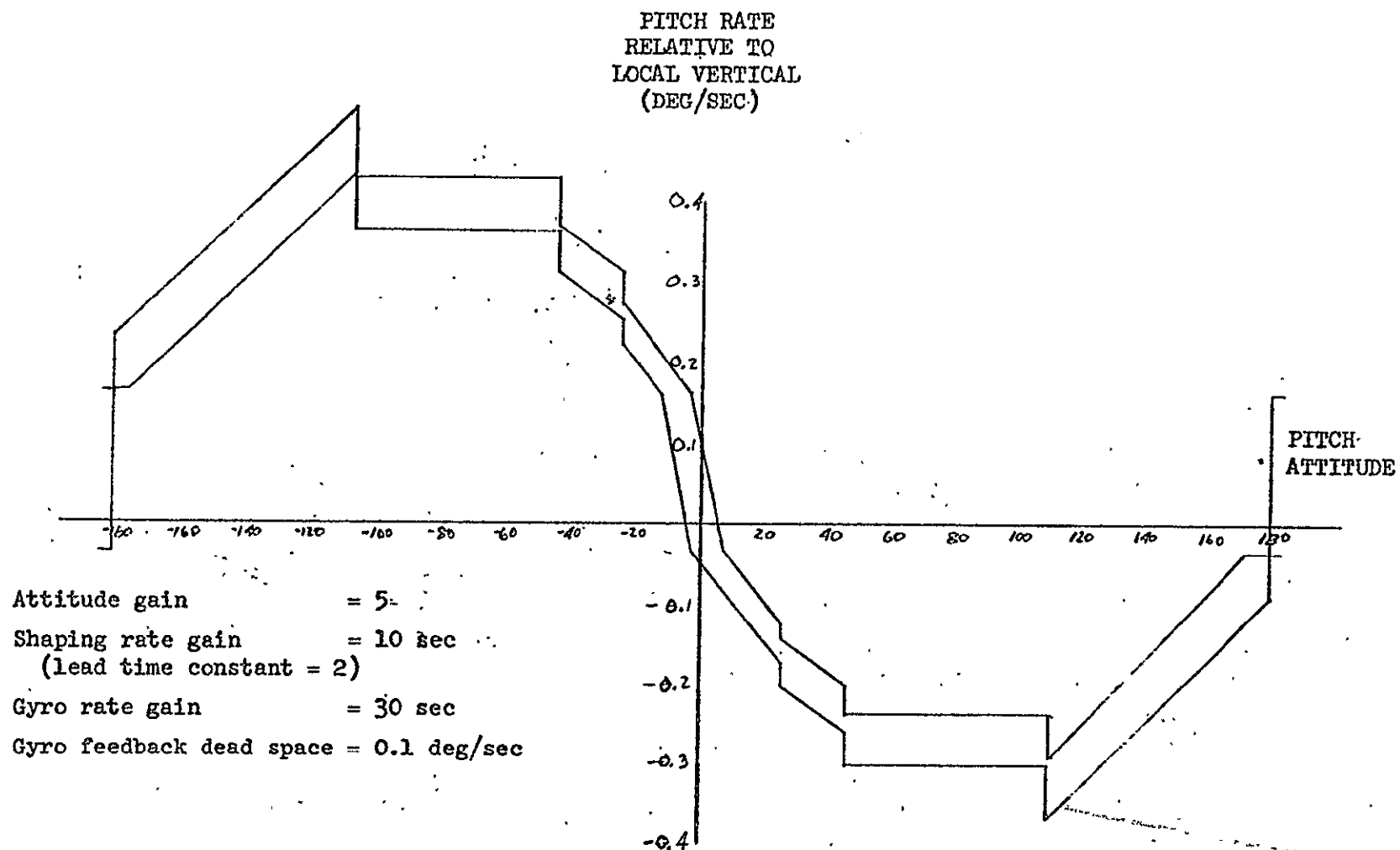


FIGURE 3-2. PITCH PHASE PLANE SWITCHING LINES GRAVITY-GRADIENT CAPTURE MODE

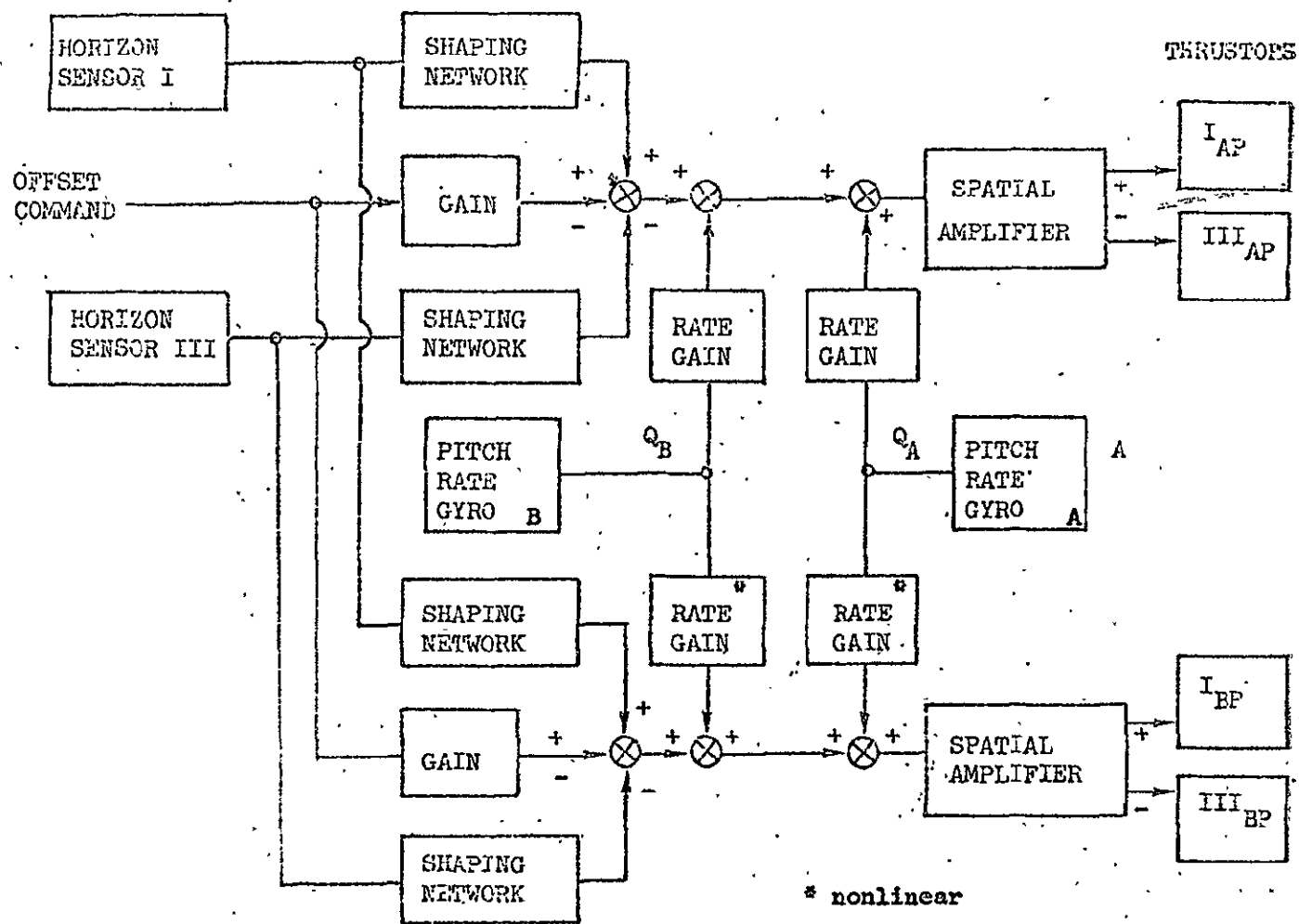


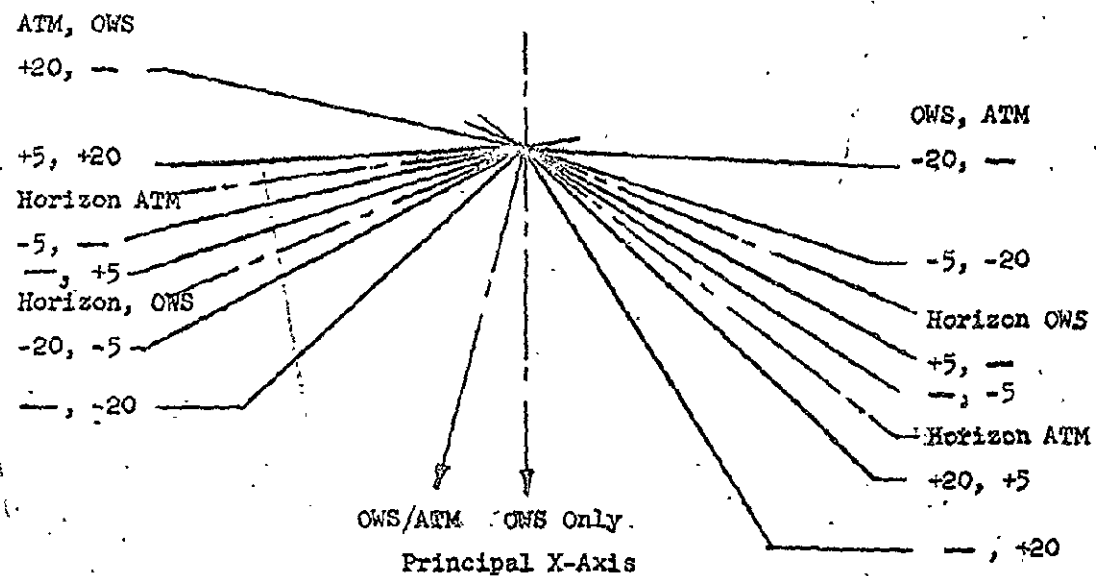
Figure 3-3 PITCH ATTITUDE CONTROL BLOCK DIAGRAM
GRAVITY-GRADIENT CAPTURE MODE

7. The field of view required beyond the desired switching point to assure that switching will occur before saturation is reached for adverse tolerances and cross-axis coupling.

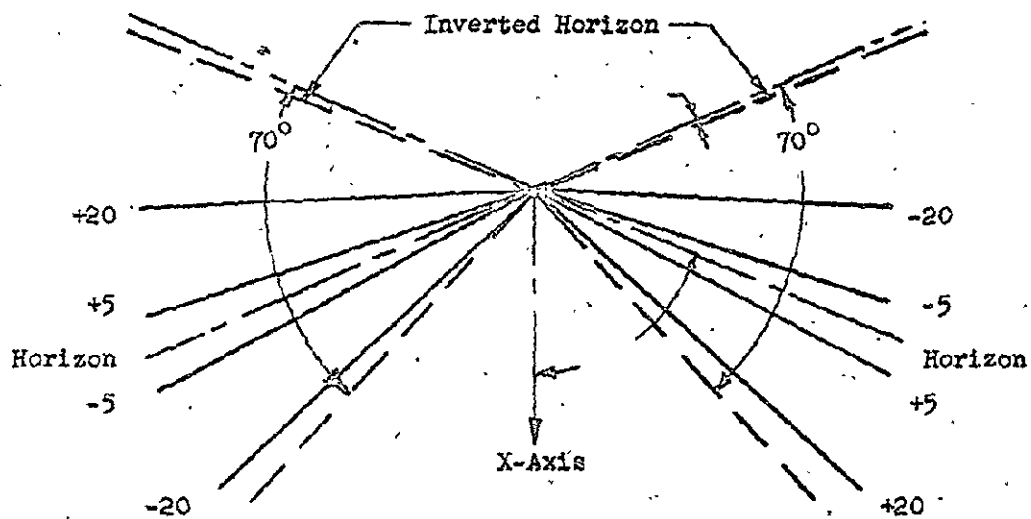
The field of view required in the pitch and yaw axes to meet each of these considerations is shown in figure 3-4. The location shown for the horizon accounts for the offset of the horizon as measured by infrared sensors from the target to the earth's surface. The four switching points required to accommodate the ± 5 degree and ± 20 degree deadbands for each of two cluster configurations are shown in figure 3-4a and 3-4b for the pitch and yaw axes, respectively. These points are very nearly symmetrical about the longitudinal axis for the OWS alone configuration and for the end-docked CSM configuration of Mission A, for which the longitudinal principal axis very nearly coincides with the longitudinal body axis. For the OWS/ATM configuration, which exists during storage following Mission B, the axis of symmetry of these switching points in pitch are displaced about 15 degrees from the body axes. For these two configurations a field-of-view of 70 degrees is adequate as it will provide a margin of 7.5 degrees beyond each of the extreme switching points for both horizon sensors in the pitch axis. Although a 55-degree field of view provides the same margins in the yaw plane, a 70-degree field-of-view will provide the same margins and give an additional upward field-of-view for both sensors to assist in capture from an inverted position.

Smaller fields-of-view could be utilized if a compromise is made so that both horizon sensors are not active at all switching points. This would decrease accuracy because of errors caused by variations in altitude from the design value, and in the event of a horizon sensor failure, would cause loss of corrective control pulses on one side of the deadband.

Figure 3-5 shows the output error signal of each yaw horizon sensor, and the error signal derived by subtracting the signals from the two opposing sensors. The figure shows the higher slope in the error signal output of the yaw axis about the upright (zero degree) position, and for small deflections, about the inverted position. The figure also shows the lower slope for larger deflections which is one-half of the value about the normal null orientation. These are



a) Pitch Axis



b) Yaw Axis

FIGURE 3-4 HORIZON SENSOR FIELD OF VIEW REQUIREMENTS

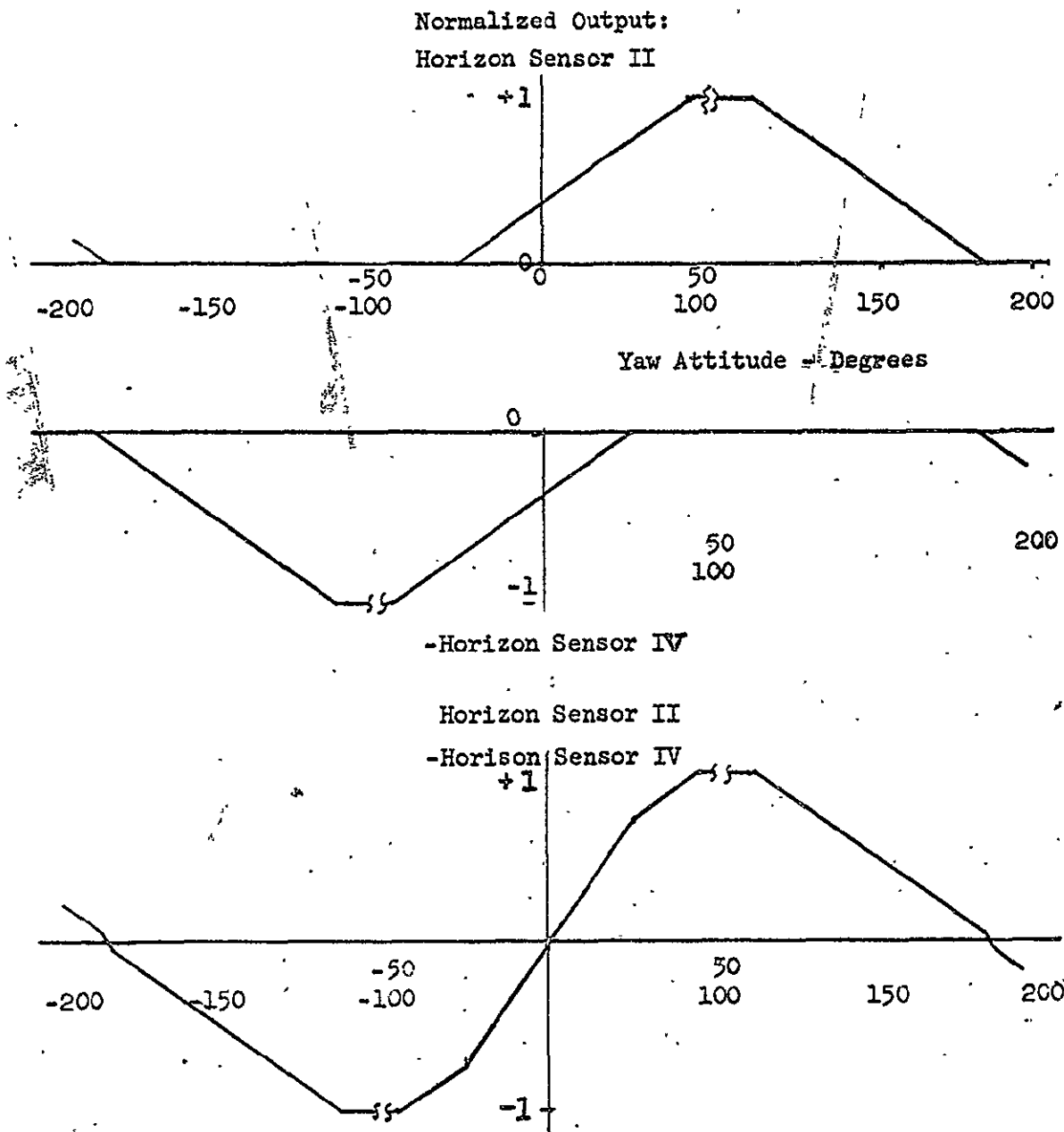


FIGURE 3-5 HORIZON SENSOR OUTPUT SIGNAL, YAW-AXIS

caused by the effective overlap in the fields-of-view of the two horizon sensors and the fact that only one horizon sensor produces an output for deviations about either the normal or inverted orientations which are greater than the overlap. Tolerances can decrease the effective overlap, and may give an underlap about the inverted position.

To correct for an inverted capture, the horizon sensors in at least one axis must have an upward field-of-view which is sufficient to ensure an adequate error signal to command the spatial amplifier in the event of an inverted orientation. The upward field-of-view, the expected libration amplitude, and the deadband limits (attitude gain) are factors which determine whether an inverted orientation will be sensed and an inversion command derived. As shown in figure 3-4, the upward field-of-view in the yaw plane is two degrees more than required to sense the horizon when inverted, but with tolerances could conceivably be less than required. Because of this small overlap and the reduced gain in the inverted position, an attitude libration of greater than the underlap plus twice the deadband limits is required to develop an error signal of sufficient magnitude to command an inversion. For this reason, the deadband limits for the capture mode should be as low as possible, considering tolerances and propellant consumption. For deadband limits of ± 5 degrees, a libration amplitude of 8 degrees plus an amount determined by tolerances is required to obtain an inversion command, neglecting rate effects.

In the event the OWS is inverted with libration amplitudes which are too small to develop an inversion command, an attitude bias command similar to those which are required to account for the variations in longitudinal principal-axis orientation (caused by variations in the modules docked to the OWS, and the port to which they are docked) can be used to obtain an inversion command. The attitude command would be removed when sufficient attitude change has been achieved to maintain the inversion command.

Some of the Phase II solar orientations (Cases 1A and 5), for which the mechanizations are discussed in section 3.4, require additional yaw horizon

sensors which have a more upward, or aft, field-of-view than those shown in figure 3-4. These can generate an inversion signal in the event of an up-side-down capture for zero libration amplitude for all configurations. Inversion commands can also be achieved by use of the pitch command normally used to compensate for the beta angle for the Case 5 orientation as discussed in section 3.4.5.

3.3.2 Gravity-Gradient Capture; Roll

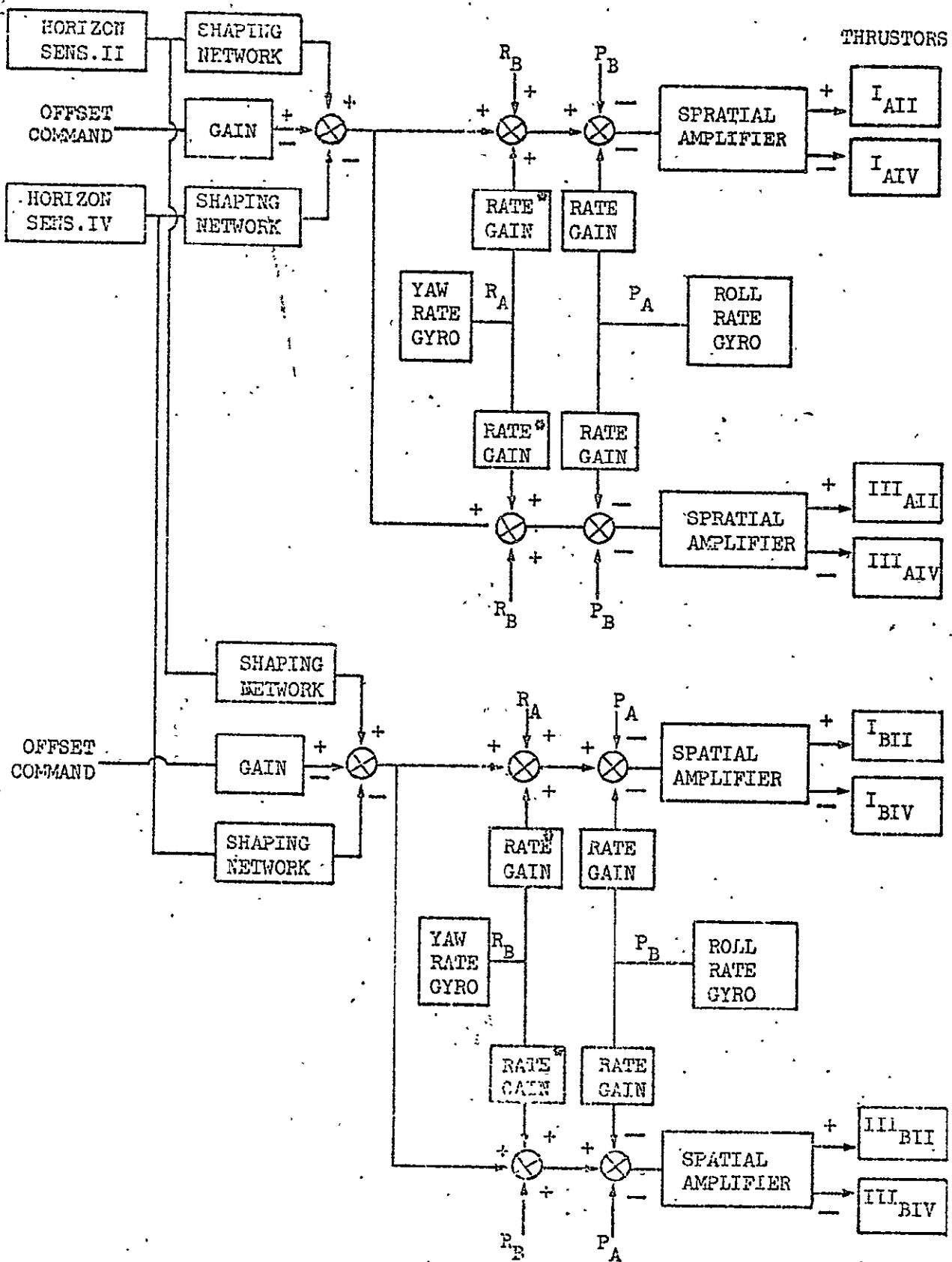
The MSFC requirements listed in section 2 require only that the longitudinal axis be vertical within the specified tolerances. Capture of the vertical gravity-gradient orientation for the OWS alone, which has nearly equal pitch and yaw inertias, requires that the body roll rates be limited, in addition to the pitch and yaw restrictions discussed in the previous section.

Figure 3-6 is a block diagram of the yaw-roll control system for the gravity-gradient capture mode which includes roll rate gyros for limiting body roll rates.

With the CSM docked to a side port, the difference between pitch and yaw inertias is increased, thus providing greater roll-restoring torques for the gravity-gradient stable orientations which are 180 degrees apart. Either stable orientation can be acquired by providing damping signals from roll-rate gyros. However, the relative stability of these two orientations is affected by aerodynamic torques which are stabilizing only about the CSM trailing position.

Either of two approaches can be used to capture a particular roll orientation for solar panel orientation should it be required. Roll attitude could be sensed, or a roll inversion maneuver of 180 degrees could be commanded if an inverted roll orientation were captured. Possible methods of sensing roll attitude include a magnetometer, ion angle-of-attack sensor, or gyro-compass. All of these approaches increase system complexity.

The yaw and roll axes are coupled in the same manner as the S-IVB APS because of the thruster arrangement which uses the same engines for yaw



* Nonlinear

Figure 3-6 YAW-ROLL ATTITUDE CONTROL BLOCK
 DIAGRAM SPATIAL AMPLIFIER (NONLINEAR MODE)

and roll. Consequently, the yaw axis is shown with the roll, although yaw requirements are identical to those for pitch.

3.3.3 Local Vertical Mode; Pitch and Yaw

The pitch and yaw requirements for the local vertical mode are identical to those for the gravity-gradient capture mode, except that greater attitude deviations are permissible and capture from any orientation is not required. Because initial attitude errors and rates will be very small on activating this mode, if the capture mode is used first, much less damping will be required. Thus, the rate gyros shown in the pitch and yaw axes for the capture mode will not be required. Elimination of pitch and yaw gyro operation during the 23 day mission will increase reliability.

Figure 3-7 is a block diagram of the pitch axis in the local vertical mode of operation.

3.3.4 Local Vertical Mode, Roll

The roll-axis requirements for the local vertical mode are similar to, but less stringent than, those for the gravity-gradient capture mode. The requirement to limit roll rates during this mode of operation can be met by using body-mounted rate gyros to derive roll control signals. Figure 3-8 is a block diagram of the yaw-roll axes for this mode of operation.

Roll attitude control can be achieved, if required for solar panel orientation, by one of the methods discussed in section 3.3.2.

3.3.5 Rate Mode

Control of body rates, as required after an unsuccessful docking attempt, can be achieved by deriving control signals from body-mounted rate gyros. Figure 3-9 is a block diagram of the rate mode for the yaw-roll axes. The gain will be set so that rates are limited to 0.1 degrees per second or less. Figure 3-10 is the pitch-axis block diagram.

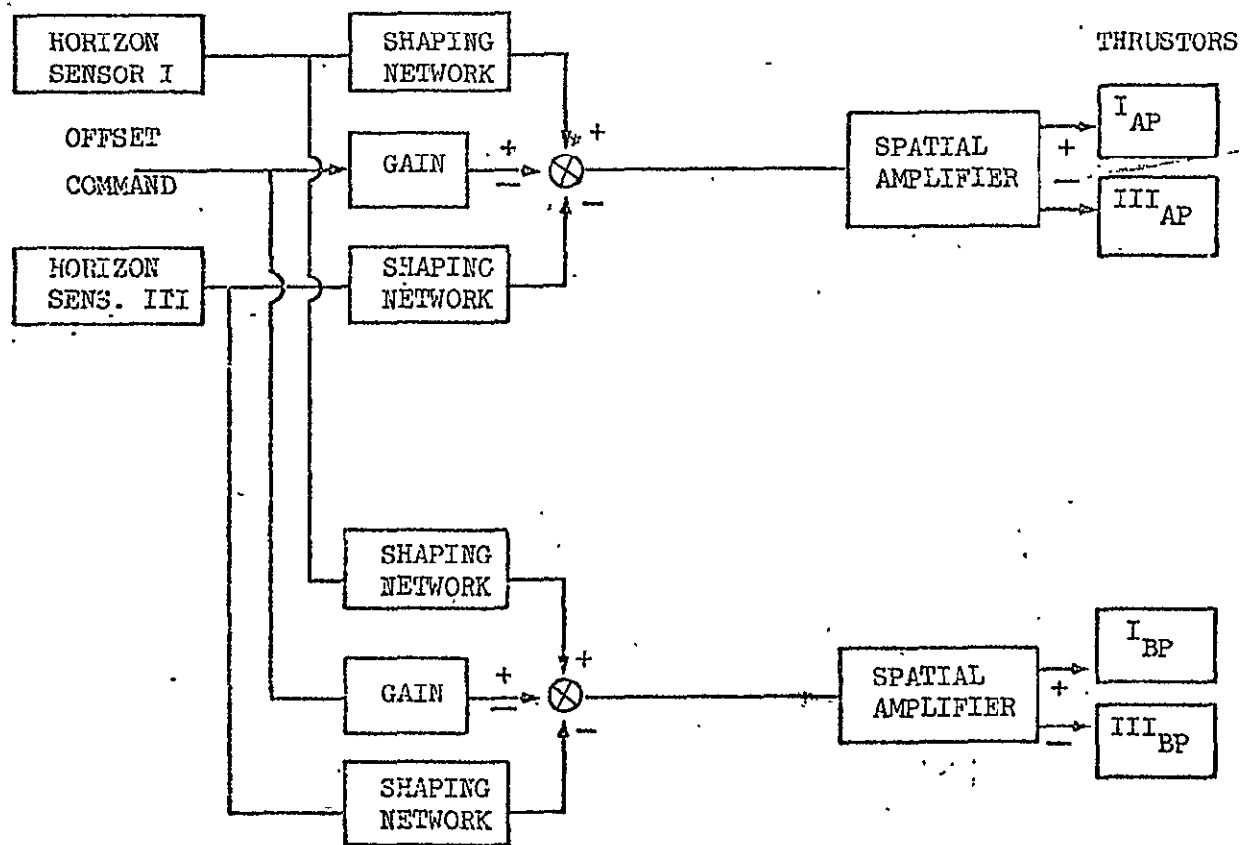


Figure 3-7 PITCH ATTITUDE CONTROL BLOCK DIAGRAM LOCAL VERTICAL MODE

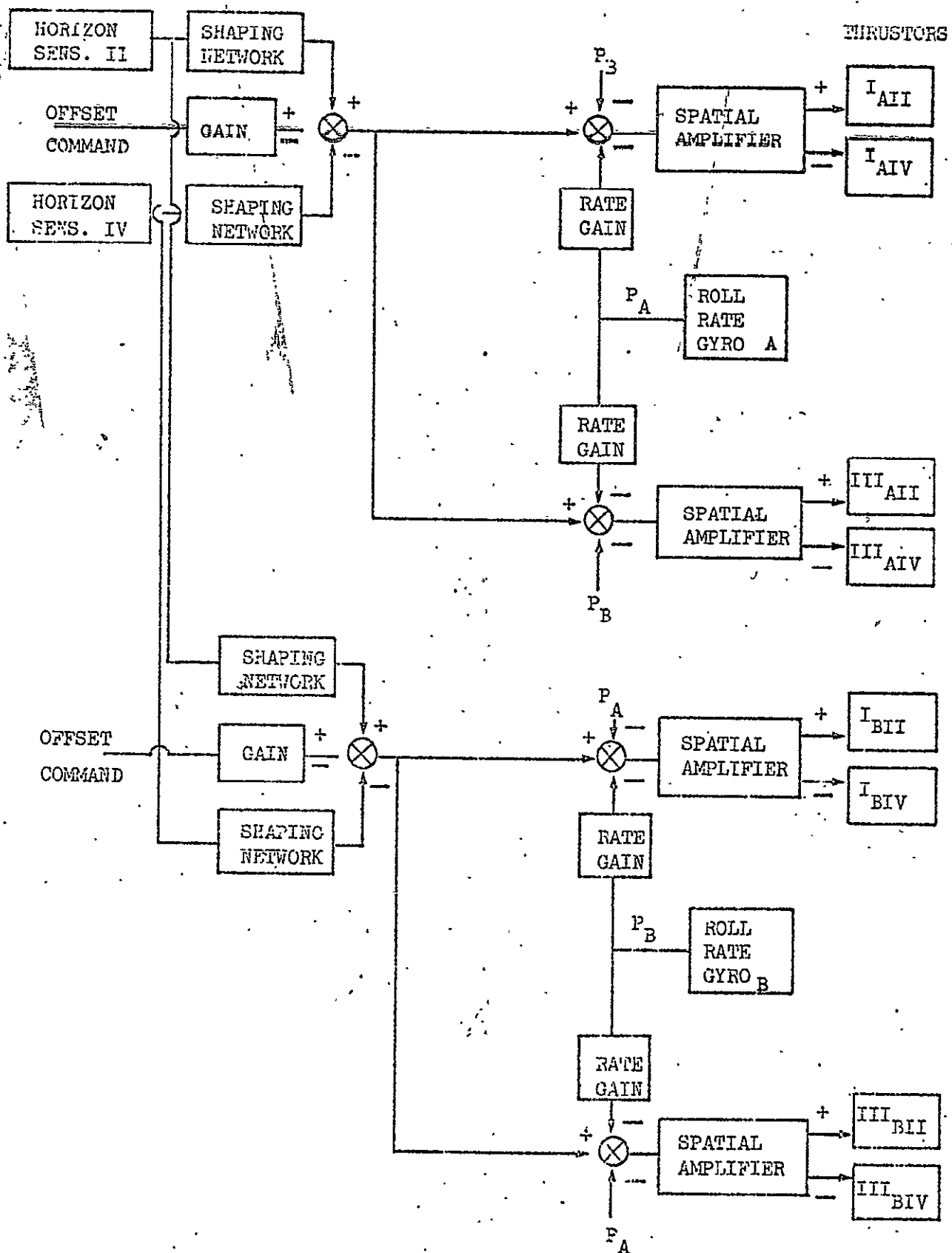


Figure 3-8 YAW-ROLL ATTITUDE CONTROL BLOCK
DIAGRAM LOCAL VERTICAL MODE

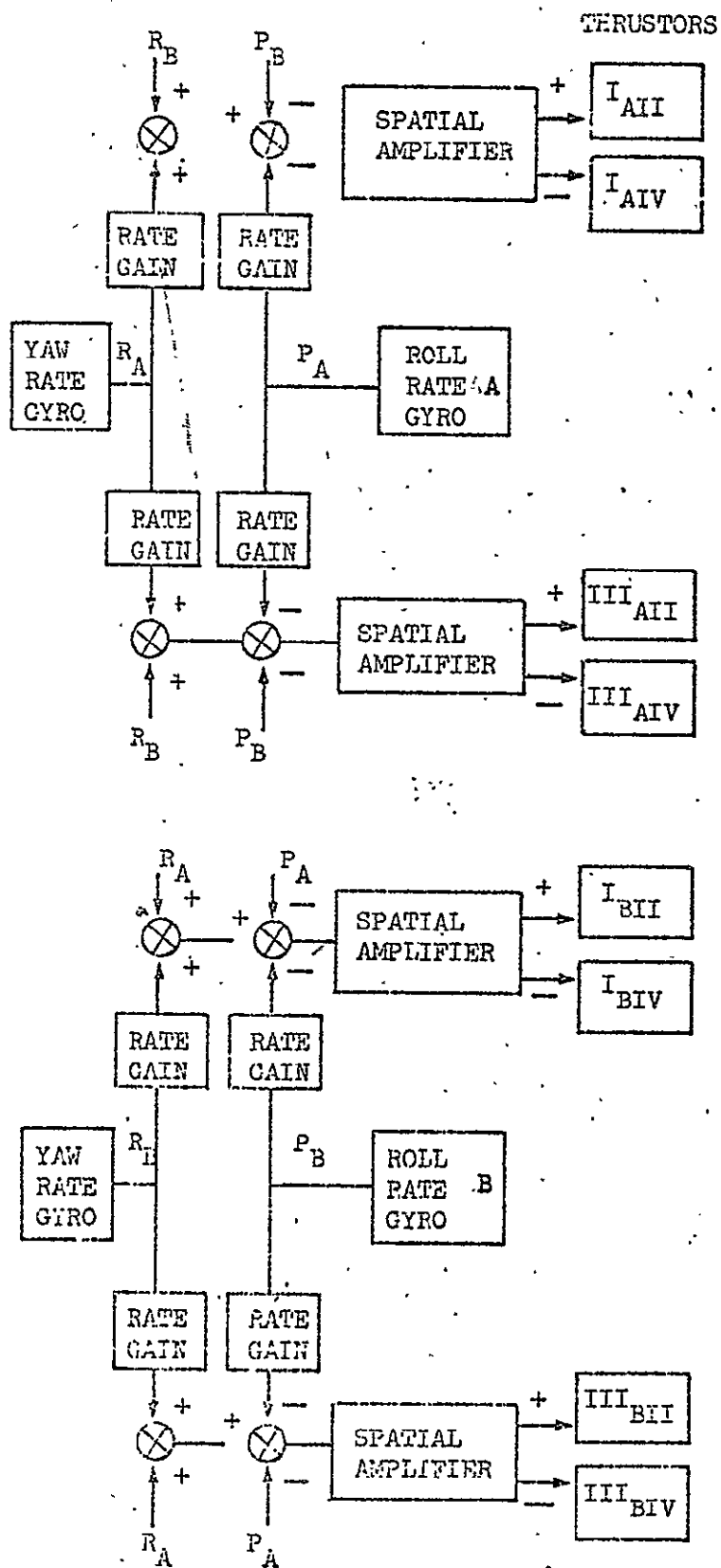


Figure 3-9 YAW-ROLL ATTITUDE CONTROL
BLOCK DIAGRAM RATE MODE

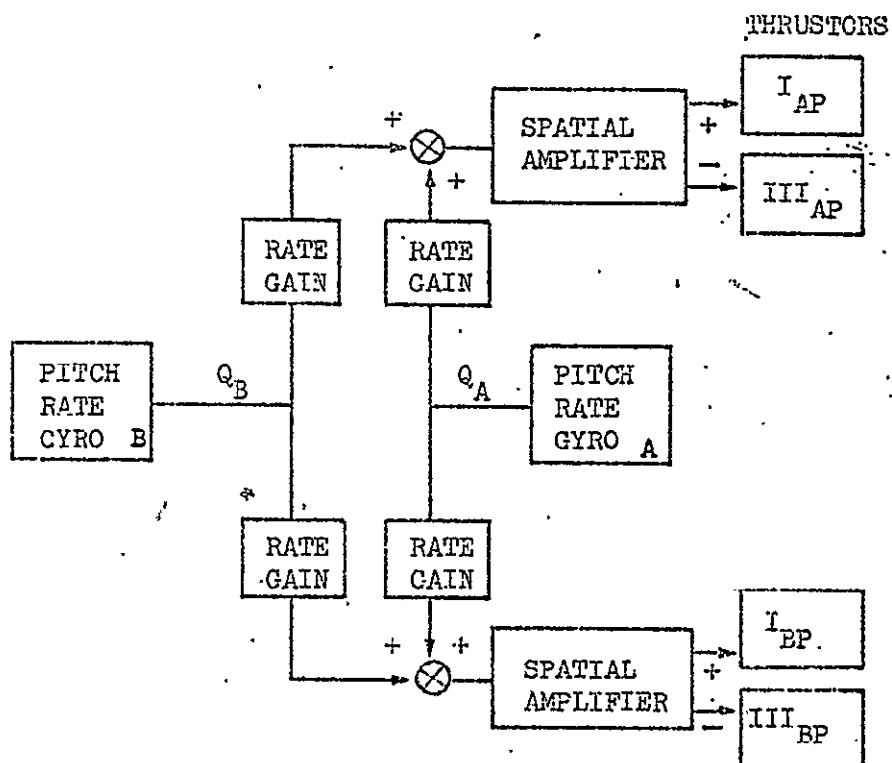


Figure 3-10 PITCH ATTITUDE CONTROL BLOCK DIAGRAM RATE MODE

3.4 Phase II System Mechanization Concepts

Control system mechanization concepts for each of the orientations evaluated during Phase II are discussed in the following sections and one or more control system mechanizations are recommended for each orientation. Some of the considerations in choosing system mechanization are common to several orientations and to more than one axis. For brevity these considerations are discussed only for the first case to which they are applicable, and this discussion referred to in subsequent cases.

In the following sections, only the methods of deriving pitch, yaw, and roll error signals for input to the spatial amplifiers are discussed. The pitch and yaw-roll spatial amplifier commands are derived from these error signals in the same manner as is done for the S-IVB Stage, which is shown in figures 3-3 and 3-6 in the Phase I section of this report.

The axis system, solar panel location, and CSM docking configuration used for the Phase II study are shown in figure 1-1. The S-IVB axis system, for which the positive x-axis (roll) is in the direction of flight, the positive y-axis (pitch) is along position plane IV, and the positive z-axis (yaw) is along position plane I is used. The solar panels are mounted off the II-IV plane, but are pivoted about an axis which is parallel to the II-IV plane when deployed. Two CSM docking locations were considered: the CSM end-docked to port 5, and side-docked to port 4 which coincides with position plane IV as shown in figure 1-1.

3.4.1 Case I Mechanization

3.4.1.1 Pitch and Roll Axes, Case I

Orientations for which the yaw-axis is solar-oriented have common problems related to control about the pitch and roll axes. These orientations include Cases 1, 1A, 2, 3, 3A, 5 with minor distinctions, and the roll-axis for 4. For these systems, control problems can be subdivided into two orbit phases: when the sun is visible, and when the sun is occulted by the earth. Figure 3-11 shows the block diagram of the recommended system.

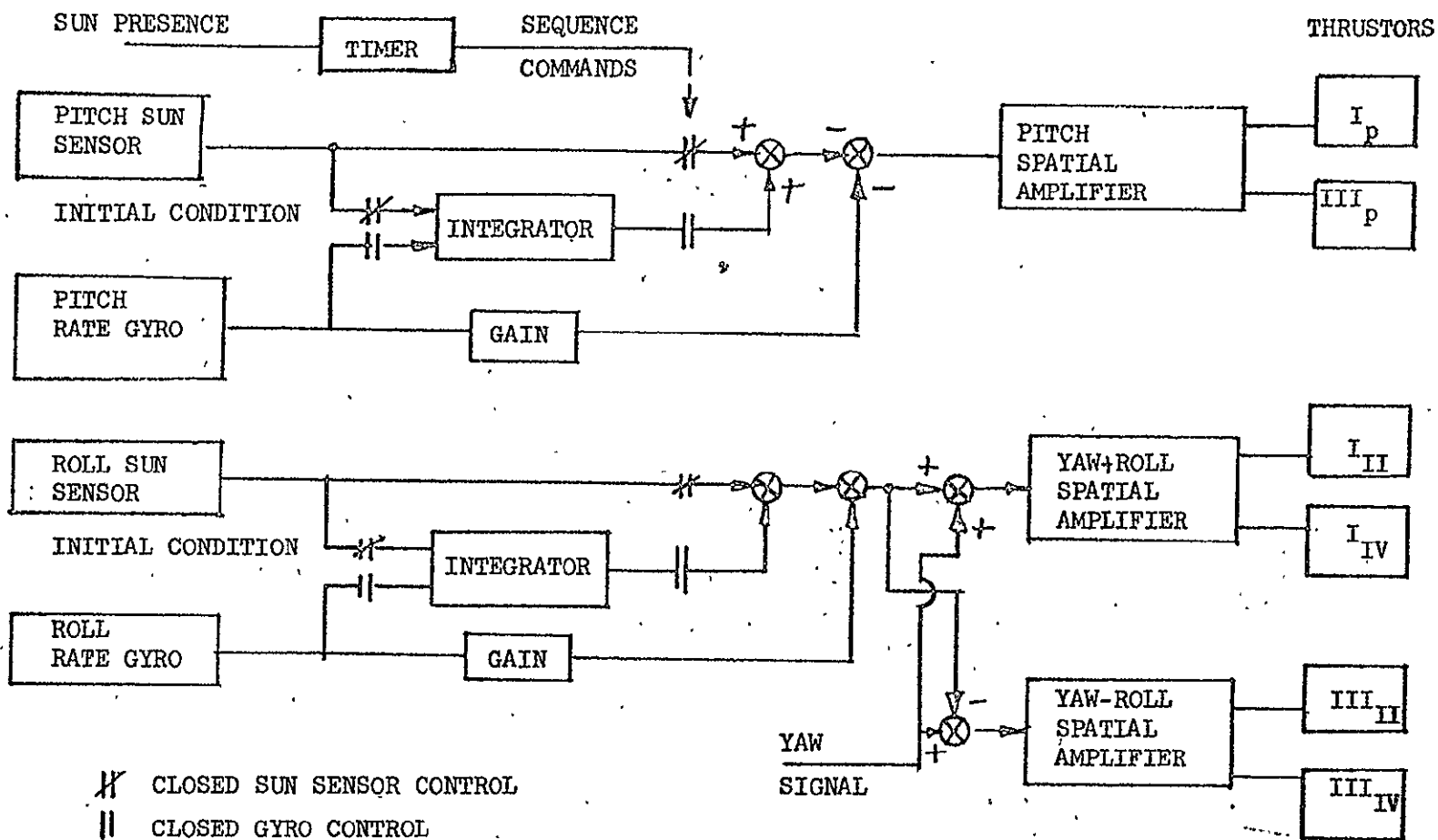


FIGURE 3-11 BLOCK DIAGRAM - SOLAR ORIENTED PITCH AND ROLL AXES

With the sun visible from the OWS, sun sensors provide attitude error signals. Damping signals are provided by rate gyros and additional damping is provided by derived rate feedback around the switching amplifiers and by shaping.

During sun occultation, a gyro reference is required to provide attitude error signals. Two problems must be accounted for in switching from sun sensor to gyro control. 1) Attitude errors due to deadband operation must be preserved during the transfer of control to preclude shifting of the null to the attitude error which exists at the time of switching. This is especially important for systems with large deadbands in order to prevent large transients and the resulting control propellant usage when the sun is reacquired. 2) Switching of control must be accomplished at a time in the orbit (between terminator passage and sun occultation) when atmospheric effects do not degrade the sun sensor signals, either by gain reduction or refraction. Switching times in the orbit are shown in figure 3-12.

Either a rate-integrating-gyro (RIG), or a rate gyro and electronic integrator could be used. The latter, which is the system used on the IM/ATM, is shown. With the RIG system, a sample and hold circuit is required to hold attitude errors during switching to a gyro reference.

For the recommended rate gyro and integrator system shown on the block diagram, attitude errors are preserved during switching by having the integrator initial condition (IC) follow the sun sensor output, or attitude error signal, during sun sensor control, so that the attitude error from the integrator after switching equals that from the sun-sensor before switching.

On switching from gyro to sun sensor control, similar signal preservation is not required or even desired as the sun sensor provides the desired reference attitude which will automatically correct errors which have accumulated during gyro control because of gyro drift and orbital regression.

Time of switching to gyro or sun sensor reference is controlled as shown in the block diagram of figure 3-11 by a timing sequence activated by the sun presence

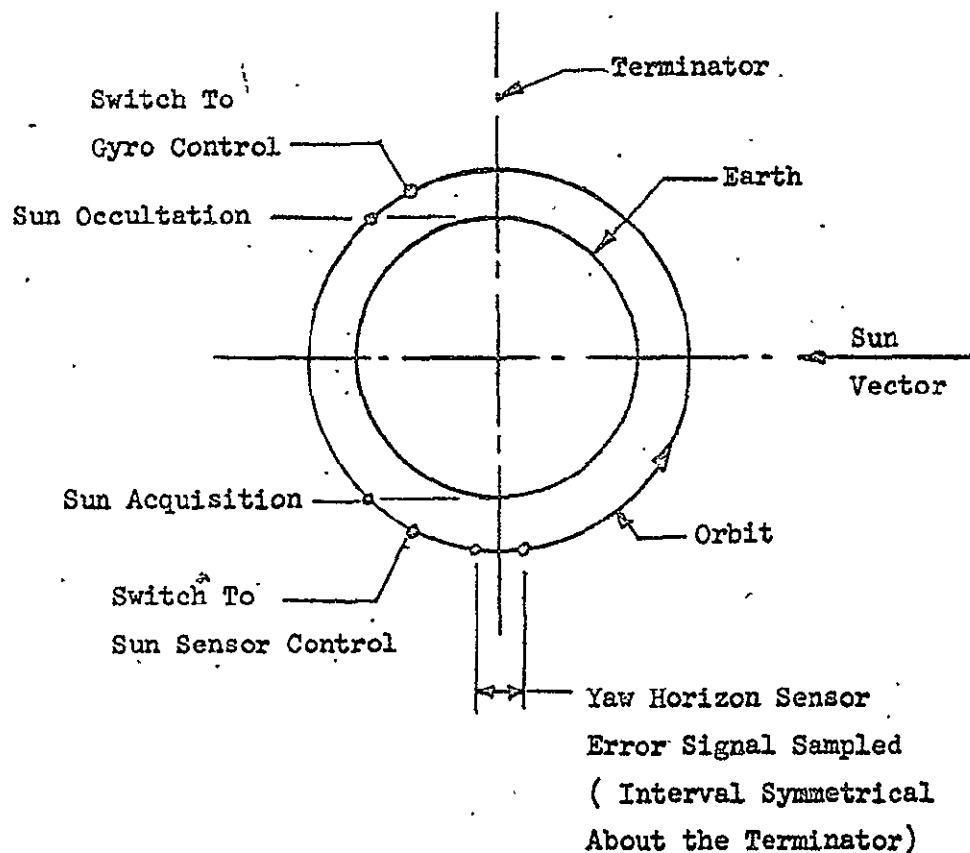


FIGURE 3-12 SENSOR SWITCHING TIMES CASES 1 AND 1A

signal. A switching sequence based on an orbital event times for a zero beta angle, (i.e., the sun vector in the orbit plane) can be used throughout the mission. Although sun sensor control could be used for a greater percentage of the orbit for high beta angles, the fixed sequence of the recommended system is adequate and is more easily mechanized.

3.4.1.2 Yaw-Axis, Case 1

To keep the x-axis in the orbit plane within allowable tolerances requires one of the following:

- 1) A continuous orbit-reference attitude-sensing system
- 2) An inertial reference, a means of commanding a time variable attitude to account for orbital regressions, and a means of periodically realigning the inertial reference.
- 3) An inertial reference and a means of periodically realigning the reference to the orbit plane, or
- 4) An orbit centered attitude reference which is usable part of each orbit and an inertial reference for use during that part of the orbit for which the orbit centered reference is not usable.

Method Number 3 is the recommended system.

Figure 3-13 shows a block diagram of the recommended system. The system uses the integrated output of a rate gyro as a source of attitude error from an inertial reference with damping provided by the rate gyro output signal. The yaw horizon sensors are used to periodically correct for attitude errors caused by gyro drift and other sources. This is accomplished as shown in the figure by using the difference between the yaw attitude errors measured by the horizon sensors and by the integrated rate gyro output to develop a drift correction signal which is added to the rate gyro signal at the input to the rate integrator. The error measurement is made at the terminator because at this point in the orbit the yaw horizon sensor output signal will null when the vehicle x-axis is in the orbit plane for all inclinations of the orbit relative to the sun vector. The correction can be made at only one terminator crossing to be compatible with the field-of-view of the horizon sensors used for local vertical control. The switch is closed to sample the error for a fixed period of time which is symmetrical about the terminator as shown in figure 3-12.

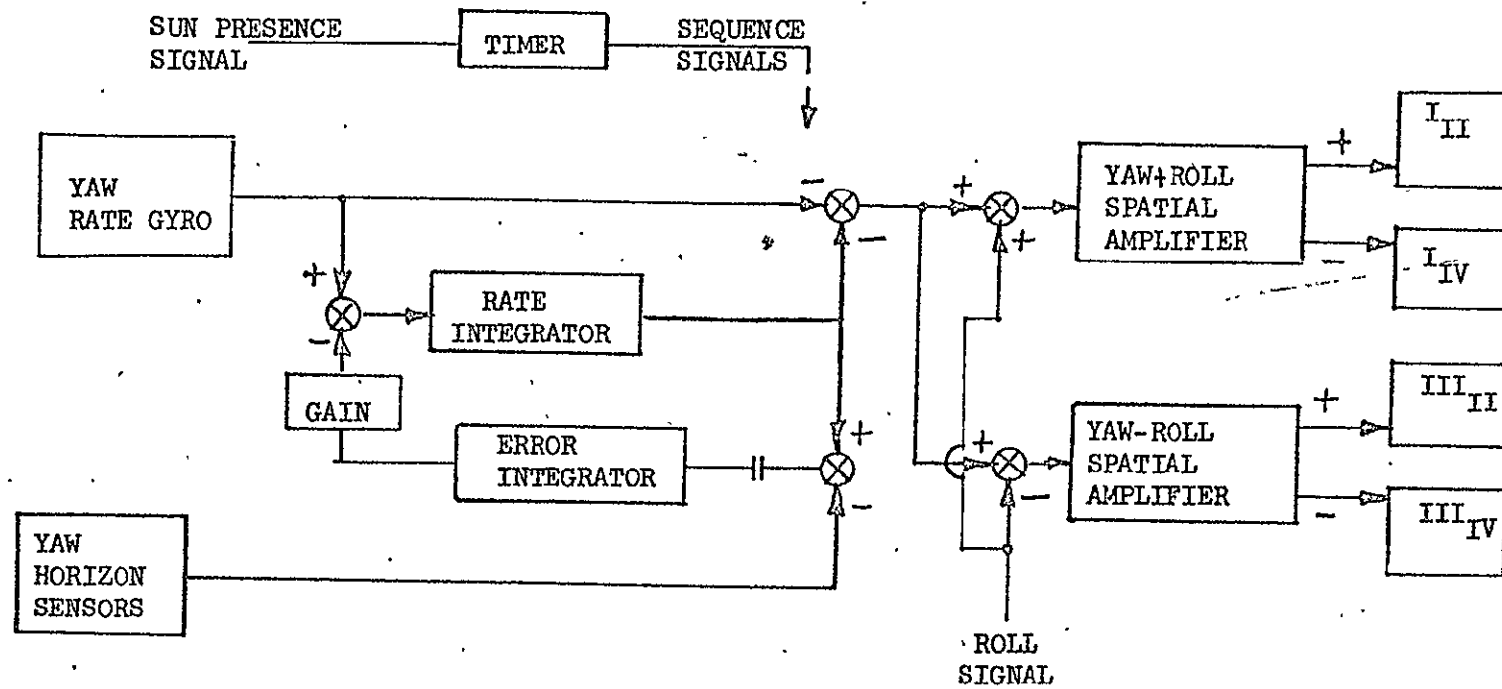


FIGURE 3-13 . BLOCK DIAGRAM YAW ATTITUDE CONTROL CASE 1
x-AXIS IN THE ORBIT PLANE

The error integrator serves two purposes: 1) it averages the error measurement over a finite time interval to minimize noise effects, and 2) it holds previously determined average drift rate corrections. The attitude drift error as measured at the error integrator output is subtracted from the rate gyro signal at the rate integrator input. The gain can be established such that the measured drift will be partially corrected in one orbit, which is the time between drift corrections. This will effectively provide filtering of the drift correction process for improved system stability.

Sequencing signals to activate the sampling switch at terminator passage can be derived from a timing sequence initiated by the sun presence signal in a manner similar to that shown in figure 3-11 to switch from sun sensor to gyro control. Sequence times can be preprogrammed with periodic update capability from the ground, or timing of sun occultation (T_{so}) and sun presence (T_{sp}) can be used for on-board derivation of the time from sun presence to the terminator as

$$T_{pt} = 1/2 (T_{sp} - T_{so}).$$

3.4.2 Case 1A Mechanization

Because of the similarity between yaw-control for Case 1 for which the x-axis is in the orbit plane, and Case 1A for which the x-axis is perpendicular to the local vertical at the terminator and the y-axis is in the orbit plane, the control mechanization described in the previous section for Case 1 can be used for Case 1A with one revision. Additional yaw horizon sensors must be installed since the fields-of-view of the yaw horizon sensors used for gravity-gradient orientation control are not suitable for this orientation. The additional horizon sensors must be installed so that their field-of-view is parallel to the plane of the solar panels and the centerline of the field-of-view is coincident with either the positive or negative y-axis as shown in figure 3-14. The field-of-view required is dependent on the principal axis variation, deadband size, and the maximum drift expected during one orbit.

The mechanizations of the pitch and roll axes for this case are identical to those for Case 1 shown in figure 3-11 and described in section 3.4.1.1.

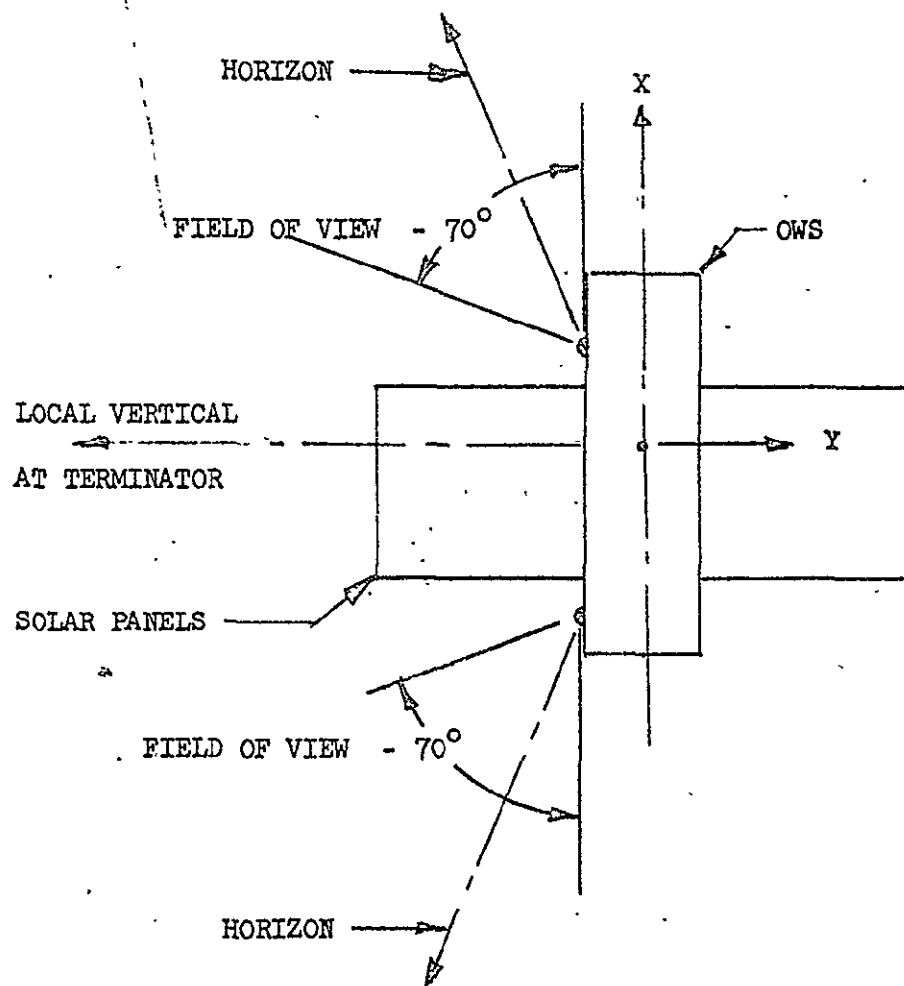


FIGURE 3-14 YAW HORIZON SENSOR FIELD OF VIEW
FOR SOLAR MODE

3.4.3 Case 2 Mechanization

The orientation of the vehicle about the yaw axis is uncontrolled for this case. However, a yaw-rate gyro can be used to limit yaw-rate buildup caused by yaw torques attributable to unsymmetrical yaw-roll engine characteristics or disturbances. The mechanization for this system is shown in figure 3-15. The commanded yaw-rate shown in the figure is zero for this case.

The control mechanization for the pitch and roll axes are identical to that for Case 1 shown previously in figure 3-11 and described in section 3.4.1.1.

3.4.4 Case 3 Mechanization

For this case the yaw axis is oriented toward the sun and yaw-rate is controlled to the optimum value which minimizes control-impulse usage. Mechanization of pitch and roll axes is identical to that of Case 1 described in section 3.4.1.1 and shown in figure 3-11.

The control mechanization for the yaw axis is the same for Case 2 which is discussed in section 3.4.3 and shown in figure 3-15. The yaw-rate gyro output is subtracted from the commanded optimum-yaw-rate to obtain the yaw error signal for the spatial amplifiers.

3.4.5 Case 4 Mechanization

The mechanizations of the pitch and yaw attitude control systems for this case, which holds the longitudinal principal axis of the vehicle along the local vertical, are the same as used for the local vertical hold mode of Phase I described in section 3.3.3 and shown in figures 3-7 and 3-8.

Roll-axis mechanization involves all of the same considerations as for the roll-axis of Case 1 which are discussed in section 3.4.1.1; however, several other factors must be considered. These can best be discussed by considering the roll angle histories for several beta angles as shown in figure 3-16.

Comparison of the roll angle histories for betas of 20 degrees and 45 degrees, shows that during sun occultation roll attitude must be changed, and that the magnitude of this attitude change is a function of the angle beta. It is approximately 80 degrees for a beta of 52 degrees, and approximately

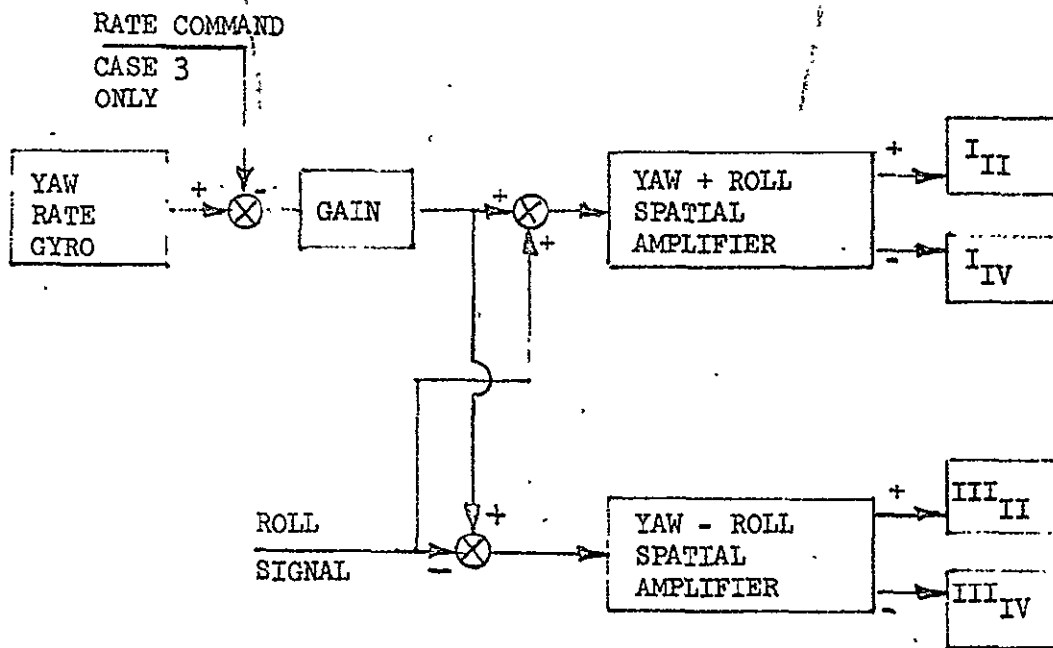


FIGURE 3-15 YAW ATTITUDE CONTROL BLOCK DIAGRAM
CASES 2 AND 3

WORKSHOP ATTITUDE CONTROL SYSTEM GRAVITY GRADIENT ROLL HISTORY - CASE 4

WACS-13
8-22-67

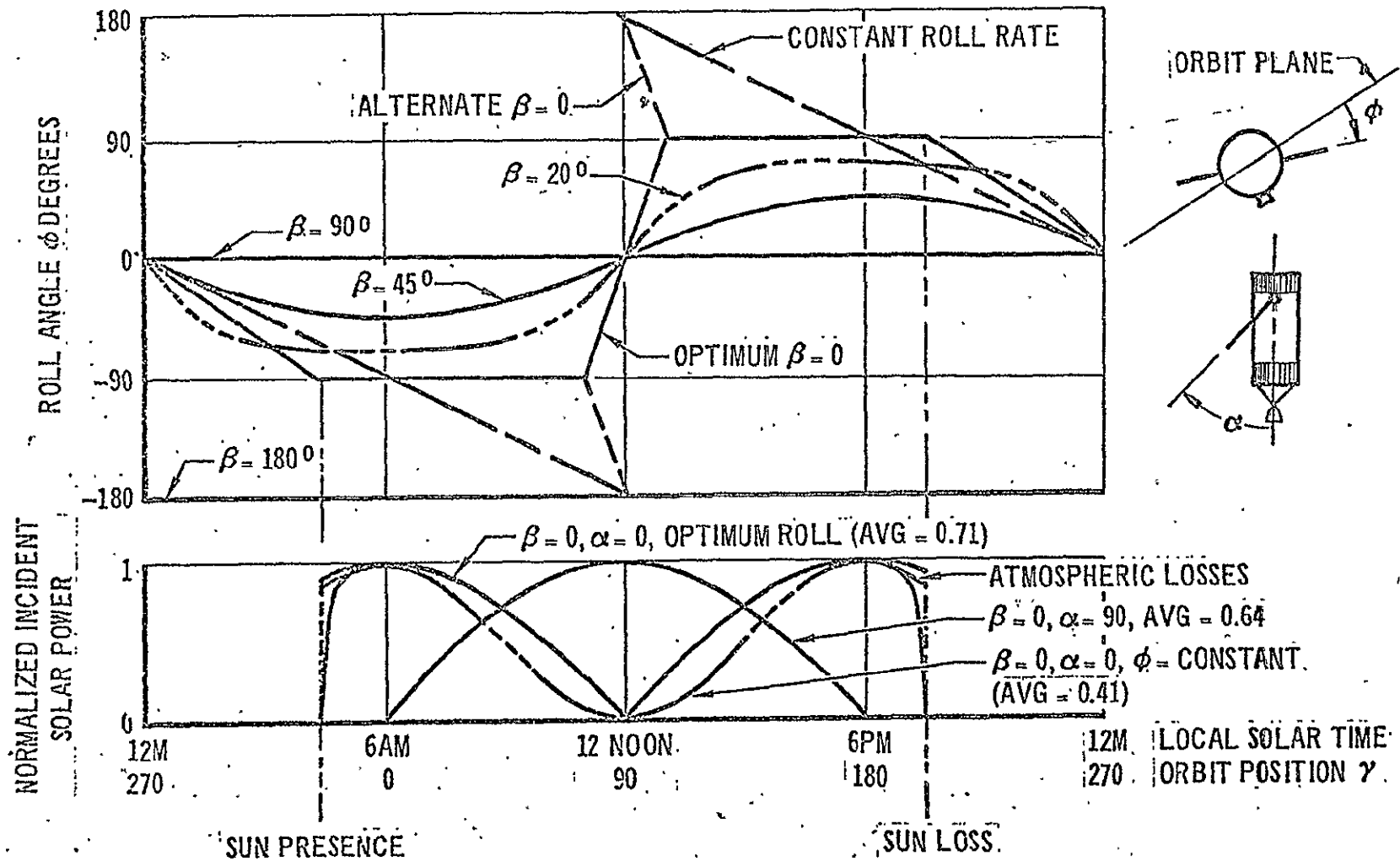


Figure 3-16. Roll Attitude History for Various Beta Angles, Case 4

140 degrees for a beta of 20 degrees. At lower beta angles, extremely high roll rates, which result in high impulse usage, are required when the sun is at the zenith to accomplish an attitude change of up to 180 degrees. These rates can become infinite for beta angles less than the pitch and yaw deadbands. The high roll rates can be prevented by commanding the required change in roll attitude at a reasonable rate near the meridian as is done during sun occultation. The rate used is a compromise between control impulse usage and solar panel power output.

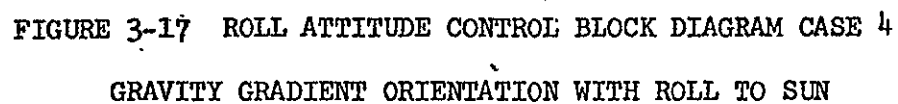
A block diagram of the roll axis mechanization of this case is shown in figure 3-17. This mechanization differs from that for the roll-axis for Case 1 only in the addition of a commanded change of roll attitude, at a predetermined rate, twice per orbit; during sun occultation, and when the sun is near the zenith. The magnitude of the command can be changed periodically, as beta changes, by ground command, by an on-board program, or by on-board computation of the command as a function of beta. Beta can be determined on-board from measurements of sun occultation time for a circular orbit of known altitude. The rate-error integrator develops a roll attitude error by integration of the roll rate error and thus provides control of the roll attitude within the established deadbands, and errors caused by the gyro and integrator.

An alternate method of roll control at low beta angles which reduces impulse usage is the use of the constant rate of change of roll attitude, of a magnitude nearly equal to orbital rate, so that the solar panels are nominally perpendicular to the sun vector at each terminator. Mechanization of this scheme requires updating of the roll rate reference by means of a roll sun sensor.

Solar power can be increased for any of these mechanizations by articulating the solar panels to provide more optimum panel orientation relative to the sun.

3.4.6 Case 5 Mechanization

For this case, the orbit orientation, the vehicle longitudinal axis is held perpendicular to the orbit plane. The orientation of the vehicle relative to the orbit for this case is thus identical to that for Case 1A for a beta angle of zero. Consequently, the control system mechanization is nearly



identical to that for Case 1A. Several minor modifications are necessary because the vehicle z-axis is displaced from the sun vector by rotating about the pitch axis by an angle equal to minus beta which holds the z-axis in the orbit plane.

The recommended mechanization for this case is shown in figure 3-18. The only change from Case 1A for the pitch axis is the addition of the pitch attitude command equal to minus beta. This could be accomplished electrically, as shown, or by mounting the pitch sun sensor on the solar panels and commanding the correct solar panel angle. The command can be obtained by preprogramming with periodic updating capability, or by periodically commanding a new angle from the ground, or from the airlock control panel. For any of these approaches, override capability from the ground or airlock can be provided. Because the maximum rate of change of beta is less than four degrees per day, a change of panel angle once per orbit or once every several orbits would insure that deviations from the preferred vehicle attitude from this source are much less than from other sources, such as deadband.

The other considerations affecting the mechanization of the pitch axis are the same as those for Case 1 and 1A which are discussed in section 3.4.1.1.

The mechanization of the yaw and roll axes is also shown in figure 3-18. One additional feature is required in the yaw and roll axes for this case compared to Case 1A. Because of the sun vector can be offset from the vehicle z-axis by an angle beta, a vehicle yaw error will induce an error signal in the roll sun sensor given by

$$\phi = (\theta + \beta) \sin \psi - \phi \cos \psi$$

where θ , ψ , and ϕ are the Euler angles describing vehicle attitude.

Because β can be much larger than θ , which is limited to slightly larger than the deadband, and for the small yaw deadbands used, ψ is small, the major yaw-to-roll coupling induced at high values of β can be removed by a coupling signal given by

$$\phi = -\beta \psi$$

This is accomplished by the cross-feeding yaw attitude error (ψ) into the roll axis through a gain equal to β as shown in figure 3-18. Periodic changes in gain will be made simultaneously with changes in pitch attitude command; however, since gain is not critical, gain changes will not be required at each

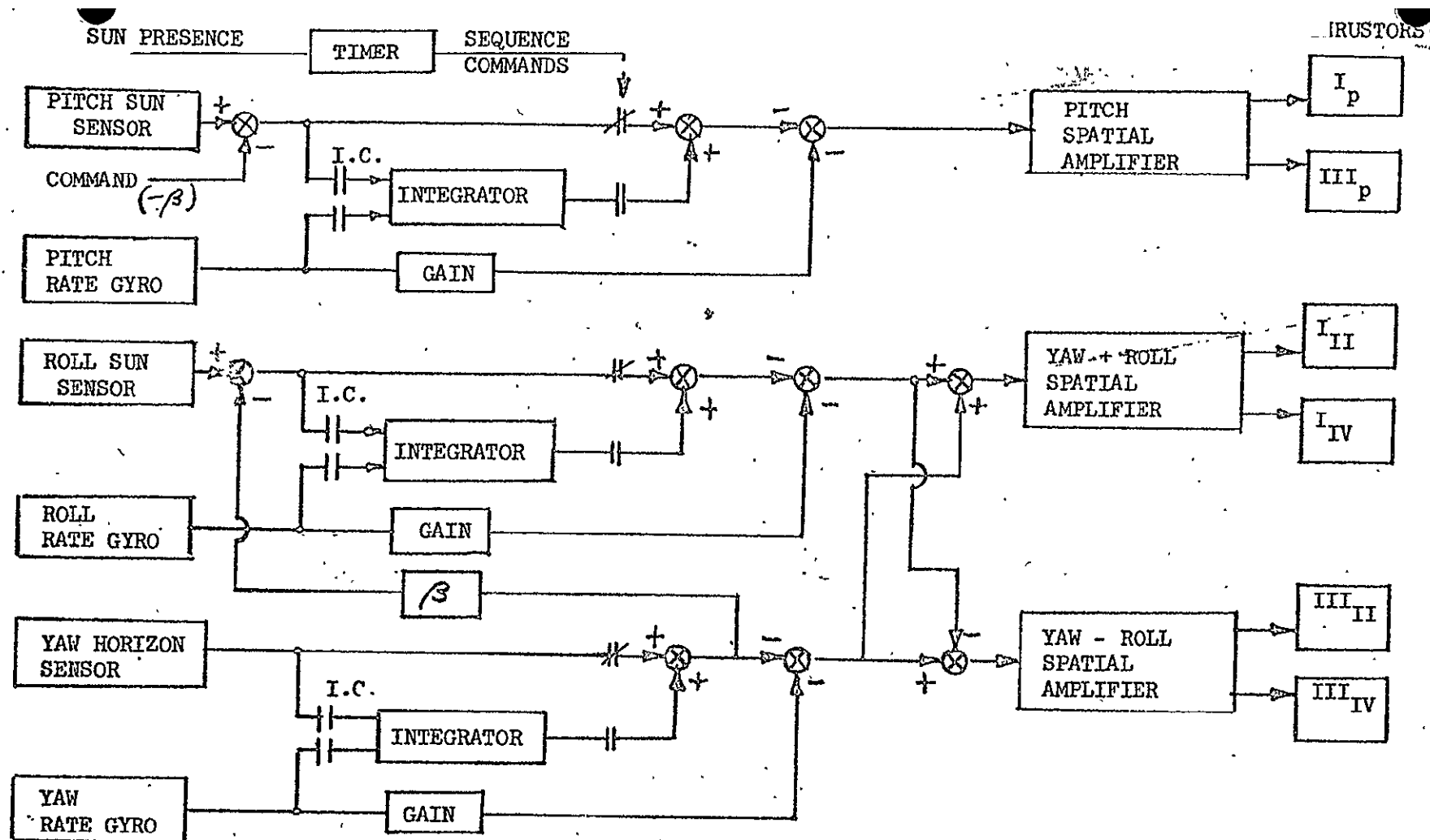


FIGURE 3-18 BLOCK DIAGRAM, ATTITUDE CONTROL CASE 5

change in attitude command.

The yaw mechanization shown in figure 3-18 uses techniques similar to those used for the pitch and roll control systems except that horizon sensors are used in place of sun sensors. Because the vehicle longitudinal axis is perpendicular to the local vertical, within tolerances caused by deadband and drift, the side-looking yaw horizon sensors will provide usable yaw-attitude-error signals for a significant distance along the orbit before and after the terminator crossing. Thus, yaw attitude control can be switched from the gyro to the horizon sensor during this period. During the time that the attitude error signals are obtained from the horizon sensors, the rate integrator will be in an initial condition mode with the yaw-attitude error applied to the initial condition input on the integrator; so that, when control is switched to the gyro, the initial yaw-attitude error at the output of the rate integrator will equal the attitude error from the horizon sensor.

The maximum duration of satisfactory horizon sensor control is dependent on the orbit altitude, the maximum expected deviations of pitch and roll attitude from nominal, and the size of the deadspace in the field-of-view of the horizon sensors about the nominal local vertical. For the OWS Mission, horizon sensor may be possible for up to one fourth of the orbit.

The mechanization shown in figure 3-13 for Case 1A could also be used for Case 5. The considerations affecting this yaw-axis mechanization are identical to those for Case 1A discussed in sections 3.4.1.2 and 3.4.2. However, the yaw drift correction with the horizon sensors can be made a fixed time interval from the sun presence. This is because the vehicle longitudinal axis is perpendicular to the orbit, so that the horizon sensor null attitude is not affected by deviations from the terminator.

3.5 Redundancy and Failure Correction

An important consideration in selecting control system concepts for the WACS, is the selection of methods of improving mission reliability by providing redundancy to negate the adverse affects of failures in subsystem components. The type of redundancy for a particular component, including the method of transferring to a redundant channel, must be based on the effect of the primary failure modes on the mission, the allowable failure detection and correction time, and the probability of the failure occurring.

The types of redundancy considered for the WACS include those used in the Saturn guidance and control system (reference 4). However, during orbital operations of the OWS, failure response times are much greater than during boost and failures can not impair safety to an extent comparable with those during boost. Consequently, different redundancy approaches may be appropriate. Different approaches are applicable to the electronic and propulsion subsystems because of the problems in mechanizing some approaches for the different subsystems. For both systems the failures which must be considered include failure to operate, causing a null output, and failure to turn off either mechanically or electronically, causing a full or partial output.

Possible methods of providing redundancy include:

- a. Dual channels with capability of commanding switching from one to the other by an automatic failure detection system, by a crew-operated control, or by command link from the ground or CSM.
- b. Triple redundant components with a comparator which switches to a back-up channel in the event of a difference in output signal between the active channel and a reference channel. This is the system used in the IU portion of the Saturn flight control system. The automatic switching feature permits rapid detection and correction of failures, as is required during boost. This is a special case of item a above in that a third channel and a comparator serves as the automatic failure detection system.

- c. Multiple channels permanently connected so that failure of a single channel to either null or saturation causes no more than a gain change and bias signal. This usually requires three channels.
- d. Dual channels with output signals summed, so that loss of signal from either channel would result only in a reduction in gain. Provisions can be included to disconnect either channel in the event of a full-output failure.
- e. Two complete active systems, with slightly different system gains, so that one system pulses at lower error signals with the other acting as an automatic backup. Provisions can be included for changing relative gains in the control logic for the two systems, so that primary and backup functions can be periodically interchanged between redundant systems so that propellant consumption can be equalized.

Possible methods of implementing propulsion system redundancy include:

- a. Use of two independent propulsion systems, including tankage and thrusters, to negate effects of propellant leakage or thruster failure to operate.
- b. Use of series-redundant control valves, with redundant thrusters, to minimize the probability of a thruster-on failure.
- c. Use of quad valves on each thruster to minimize the probability of either type of thrust failure.
- d. Use of shutoff valves for each engine or cluster which are automatically closed in the event of a thruster-on failure.
- e. Provisions for interconnecting propellant supplies in the event of a thrust-off failure in the system with the maximum propellant remaining.

As noted in reference 4, the duplex redundancy technique is very desirable in terms of simplicity and reliability improvement but is limited by the problem of determining the functioning channel after a failure has occurred.

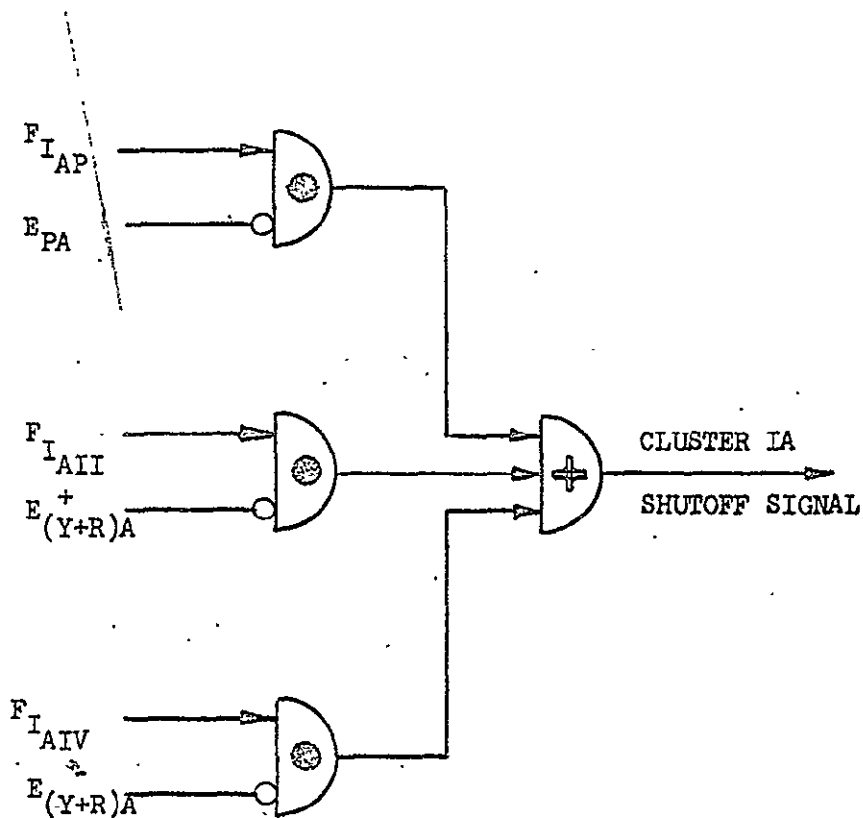
For a switching system, such as the WACS, the approach of duplex systems with slightly different gains, as listed in item e above, provides automatic transfer of control to the operative channel in the event of an inoperative failure. A method of detecting a full-on failure is also required, particularly for the thruster which can fail with a valve or valves open.

One approach to detect and correct a full-on thruster failure is shown in figure 3-19. This system consists of logic circuitry which compares a signal that indicates propellant flow to the thruster with the error signal to the spatial amplifier for that thruster. If there is flow to the thruster and the spatial amplifier input signal is not of the sign and magnitude which would command this thruster on, a signal is developed which can shut-off propellant to that particular thruster or to the entire three-engine cluster. The propellant-flow signal can be obtained from thruster chamber pressure, or propellant flow-meters. The error signal can be obtained at the input to the spatial amplifier or by appropriate summing of signals directly from the sensors.

Another method of correcting for a thrust-on failure is to derive an engine or cluster shut-off signal from the flow indication signal for each thruster if the on-time for that thruster exceeds the expected value, based on predicted disturbances for that mode of operation, by an established margin.

Gyro failures causing null or near null output are not critical for the dual-redundant switching system. A gyro failure which causes a non-null output can be detected by the output exceeding the maximum rate which is expected during the particular mission phase. A logic circuit, similar to that shown for thruster shut-off, can be used to shut-off and disconnect the fault gyro.

For any of these automatic systems, the addition of an override capability by ground command or crew control can further enhance reliability.



KEY:

F_{IAP} = FLOW TO PITCH THRUSTOR, CLUSTER "A" IN MODULE I
(II, IV INDICATES YAW-ROLL THRUSTORS)

E_{PA}^{+} = ERROR SIGNAL TO PITCH SPATIAL AMPLIFIER "A" POSITIVE

CLUSTER IA SHOWN, IB, IIIA, AND IIIB SIMILAR

FIGURE 3-19 CLUSTER SHUTOFF LOGIC

Further reliability analyses during detail design of the system are required to firmly establish the approach used.

3.6 Commands

Commands must be furnished to the WACS to accomplish the following three functions:

1. Change the mode of operation and apply power to the gyros prior to switching to a mode which utilizes gyros.
2. Command attitude offsets from the sensor null.
3. Control switching of electronic signals or actuation of propellant valves to correct for component failures.

These commands can be originated remotely on the ground (or in the CSM), in the Airlock Module Astronaut Manual Control Panel, or be derived from on-board based on system parameters. The method used for each command must be selected after study of mission control procedures and timing requirements for each mode. The required commands, based on preliminary studies, are listed in table 3-2.

The total number of commands listed is based on sequential transmission of several commands to obtain the desired system state (i.e., turn on gyro power, select mode of operation, set-up offset commands, select operating gyro, etc.). However, the switching desired for initial operation could be established prior to launch if desired.

TABLE 3-2 REQUIRED COMMANDS

<u>Function</u>	<u>Number</u>
Mode of Operation	8
Gyro Power (Two packages, either or both on and off)	4
Offset Commands	
Pitch (beta) magnitude	16
Sign	2
Scale (beta, high; gravity-gradient, low)	2
Yaw (gravity-gradient: ± 4 deg in $1/2$ deg increments)	9
Primary Channel (A, or B; gain change)	2
Spatial Amplifier Gyro Signal	
(Six amplifiers: Gyro A, Gyro B, off)	18
(All A to A and all B to B; all to A; all to B, all open)	4
Horizon Sensor Heads	13
(Six; open or closed; all closed)	
Sun Sensors	
(All amplifiers to A, all to B, A to A and B to B)	3
Cluster Shutoff	8
(Four clusters; on and off)	
Manifold Interconnect	4
(Fuel and oxidizer, open and closed)	
Timers (A or B, on or off)	4
Static Inverter	3
(All gyros to inv. A, all to B, A to A and B to B)	
	103

4. ATTITUDE CONTROL DYNAMICS

Three-axis simulations and single-axis analysis were performed to evaluate the dynamic response characteristics of the candidate systems, and to investigate potential solutions to problems associated with the WACS. The simulation was used to evaluate the effects of control system gains, compensation, thrust, and minimum-impulse bit on system performance for each mode of system operation. The analysis techniques were used primarily to define the minimum engine impulse bit, and to determine impulse usage during orbital-coast control.

4.1 Phase I Analysis

4.1.1 System Dynamic Model Description

The dynamic equations used in the simulation included the effects of gravity-gradient torques, gyroscopic coupling, skewed principal axes, and the control engine moments. The coordinate system is shown in figure 4-1. Vehicle angular position is referenced to a coordinate system which is fixed at the vehicle CG, and rotates with the orbital velocity vector. The X-axis is aligned along the velocity vector, the Z-axis along the local vertical, positive towards the center of the Earth, and the Y-axis completes a right-handed coordinate system.

The angular position of the vehicle principal axes, with respect to the local vertical coordinate system, are described by the Euler angle rotations Θ_p (rotation in the orbital plane), Ψ_p (rotation in the plane perpendicular to the orbital plane) and Φ_p (roll position). The order of rotation used to derive the direction cosine matrix is Θ_p, Ψ_p, Φ_p .

A block diagram of the vehicle and control system dynamics is presented in figure 4-2. The control moments about the body axes were transformed to moments about the principal axes, using the direction cosine matrix which describes the position of the vehicle body axes with respect to the principal axes. The control moments were then combined with the gravity-gradient and gyroscopic moments, to define the vehicle rotation about its principal axes. Vehicle angular position and rates, as sensed by ~~horizon~~

$\dot{Q}_p, \dot{R}_p, \dot{P}_p$ PITCH, YAW, ROLL ANGULAR ACCELERATION ABOUT BODY AXIS
 $\dot{Q}_p, \dot{R}_p, \dot{P}_p$ PITCH, YAW, ROLL ANGULAR ACCELERATION ABOUT PRINCIPAL AXIS
 Z_{cg}, Y_{cg}, X_{cg} PITCH, YAW, AND ROLL CG LOCATIONS

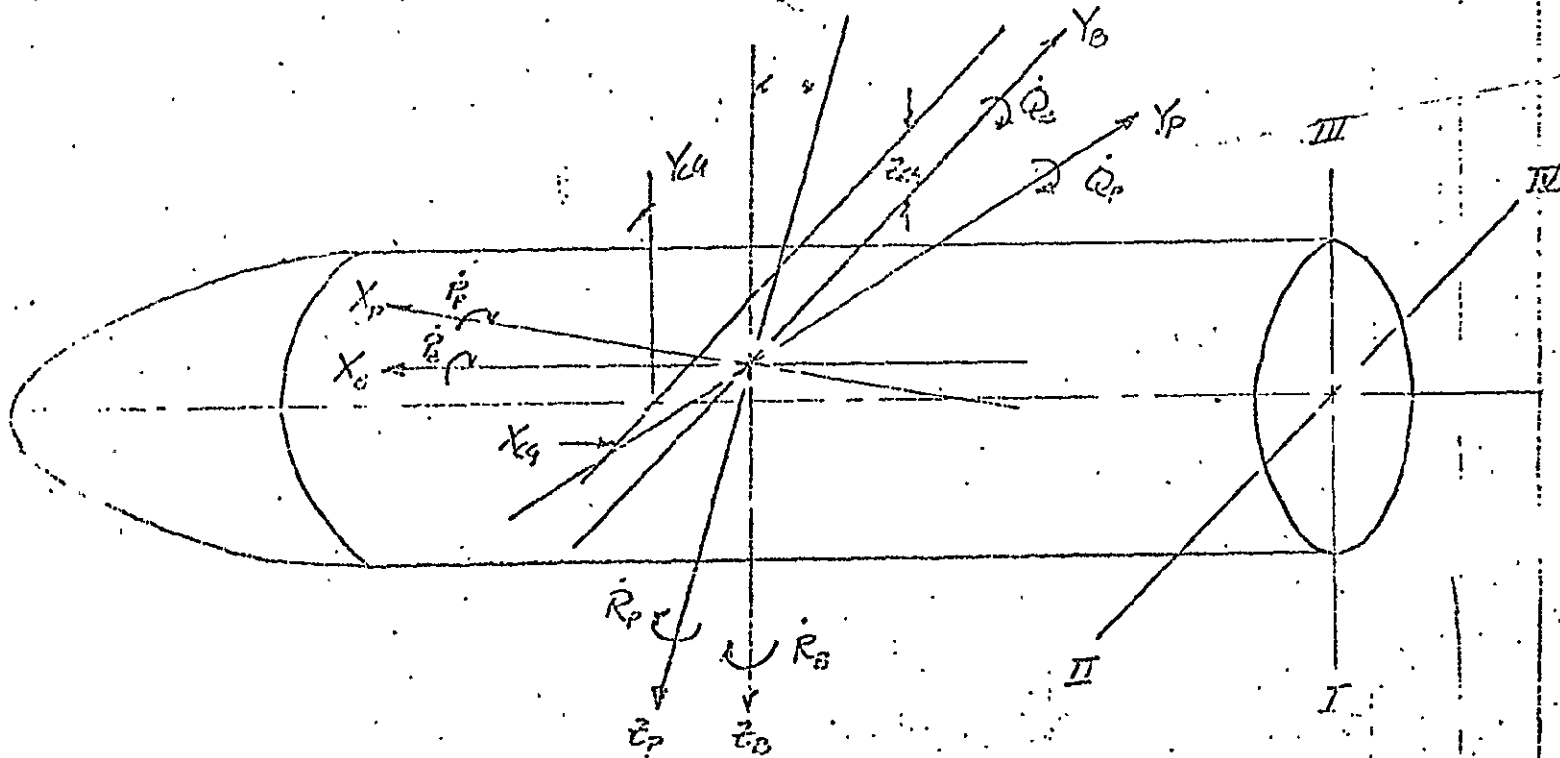
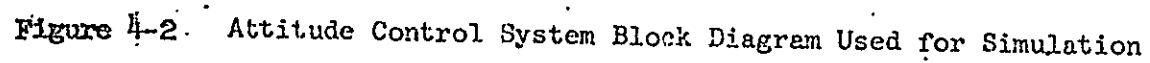


Figure 4-1. S-IVB/OWS Coordinate System in Orbit



sensors and rate gyros which are mounted on the vehicle body axes, were defined, based on the angular position and rates of the principal axes and the relationship between principal and body axes, as shown in figure 4-2.

4.1.2 Results of Dynamic Analysis

Transient response characteristics, impulse usage, duty cycle, and overall WACS compliance with system requirements were analyzed for all system configurations in each of its modes of operation. Vehicle mass data used in these analyses are listed in table 4-1.

4.1.2.1 Docking

One of the most critical flight modes during the OWS mission will be to dock the CSM to the S-IVB prior to the 23-day mission. Following a docking attempt, the WACS must reduce the workshop angular rates to 0.1 degree per second or less within approximately one-quarter orbit. The WACS will be in a rate mode in all three axes to remove these disturbances. There are no attitude-position requirements during docking or after a docking attempt.

A preliminary evaluation of the disturbances due to a missed docking attempt showed that the workshop body rates could be as high as 1.5 degrees per second in pitch and yaw and up to 0.5 degree per second in roll. However, more data on docking procedures and docking mechanism characteristics are required to accurately determine these values. Simulations of the WACS response to initial body rates were made to determine settling time, and impulse usage as a function of engine thrust level. Thrust levels of 12, 50, and 150 pounds were evaluated. In each case, a minimum pulse width of 0.050 second was used. The settling time (defined as the time to reach zero rate) and peak attitude excursion following a missed docking are shown in figure 4-3 as a function of thrust level. It is noted that a thrust level of 12 pounds provides adequate control of the vehicle following the docking disturbances, as disturbance rates are removed in less than 100 seconds which is much less than the one-quarter orbit (1420 seconds) allowed.

PARAMETER	VEHICLE CONFIGURATION		
	SOLAR PANELS ONLY	SIDE-DOCKED CSM	END-DOCKED CSM
l_r (ft)	11.51	11.51	11.51
l_p (ft)	27.15	31.992	47.217
l_{yr} (ft)	28.112	32.953	47.217
I_{xx} (slug-ft ²)	0.12006×10^6	0.250×10^6	0.15085×10^6
I_{yy} (slug-ft ²)	0.93477×10^6	1.888×10^6	2.8492×10^6
I_{zz} (slug-ft ²)	0.95476×10^6	1.950×10^6	2.8212×10^6
y_{cg} (ft)	0.425	6.7	-0.2770
z_{cg} (ft)	-0.208	0.0	-0.03452
a_{xx}	0.99984	0.9655	0.99996
a_{xy}	0.1236×10^{-1}	0.2602	-0.5847×10^{-3}
a_{xz}	0.7994×10^{-2}	0.0	0.82069×10^{-2}
a_{yx}	0.14676×10^{-1}	-0.2602	0.36117×10^{-2}
a_{yy}	-0.79601	0.9655	0.92743
a_{yz}	-0.60544	0.0	-0.37398
a_{zx}	-0.112415	0.0	-0.73926×10^{-2}
a_{zy}	0.60515	0.0	0.37399
a_{zz}	-0.79585	1.000	0.92740

TABLE 4-1 Orbital Workshop Mass Data

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM PITCH TRANSIENT RESPONSE

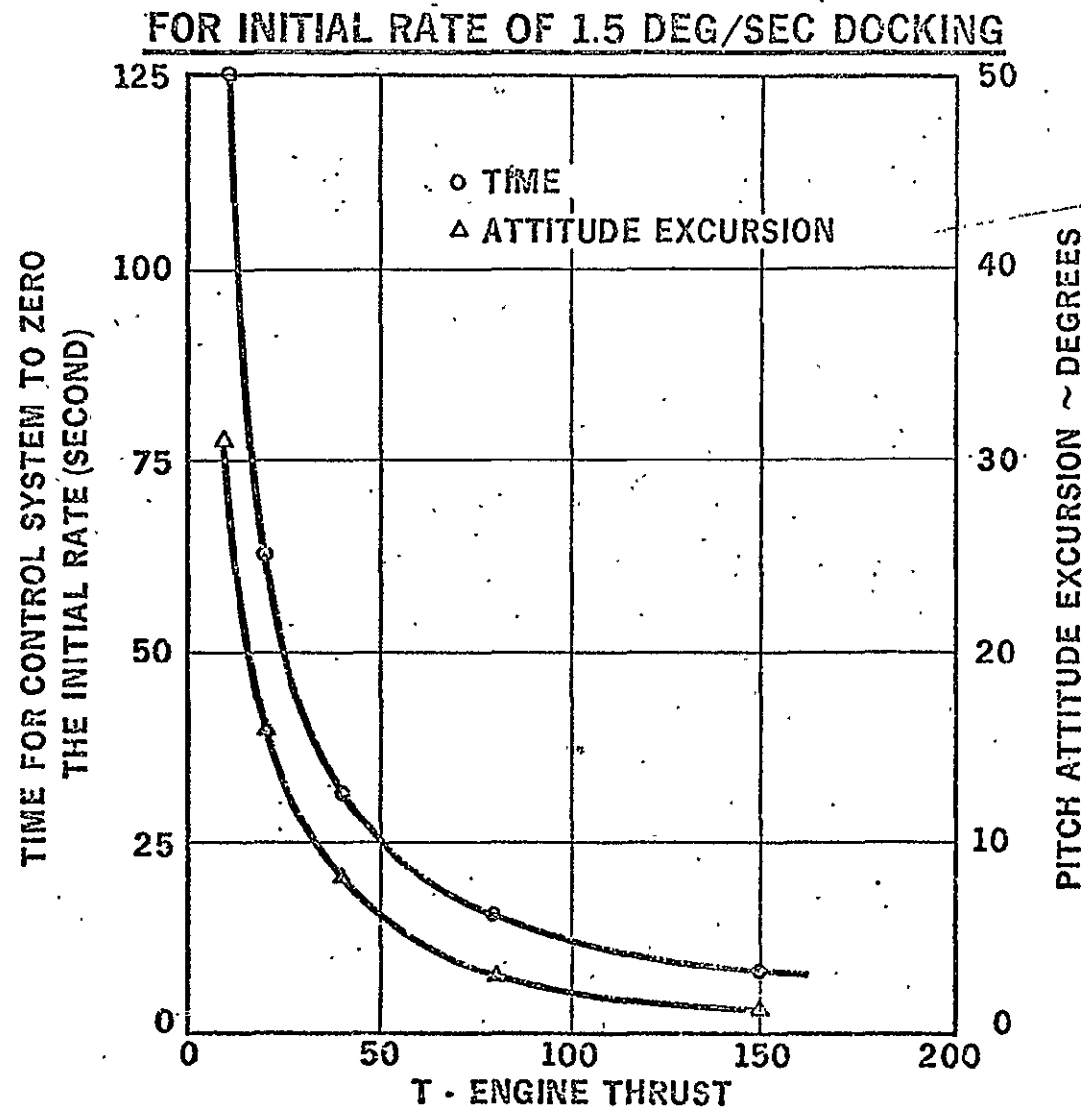


Figure 4-3. Time for WACS to Control Initial Docking Disturbances for Given Thrust Levels

4.1.2.2 Local Vertical Hold

During the period when the CSM is performing the rendezvous maneuvers, and during the 23-day mission, the WACS will be in a local-vertical-hold mode. The workshop will be required to maintain a gravity-gradient stable orientation within ± 20 degrees in the pitch and yaw planes, with the MDA directed toward the earth. In this mode the roll rate must be maintained less than 0.01 degree per second during rendezvous and during the 23 day mission.

The roll rate restriction imposes an upper limitation on the minimum-impulse-bit size. Configuration 1, the OWS alone, is most critical in establishing the minimum-impulse-bit because it has the lowest roll inertia. Figure 4-4 shows the change in roll rate for this configuration as a function of impulse bit. This figure shows that a change in roll rate of 0.01 degree per second is obtained with an impulse of 1.8 pound-seconds. The control logic, which mixes yaw and roll, will ensure that in the limit cycle mode only one module can pulse with yaw errors and roll errors; but two modules can pulse with roll errors only. Thus, with an impulse bit of 1.8 pound-seconds, or 0.9 pound-second with two redundant thrusters per module, a single pulse will cause roll rate to change from one switching limit to the other in response to roll errors or from zero to the switching limit with yaw errors. Although this value of minimum impulse bit will meet rate requirements, a lower value must be used to assure that tolerances on pulse width, thrust, switching limits, etc. will not cause continuous thrusting. A minimum impulse bit less than that required to reverse the roll rate direction from the switching lines is desirable to improve roll gravity-gradient capture time. Thus, assuming control logic such that only two roll engines, in opposite modules, can pulse simultaneously 0.9 pound-second is the upper limit on minimum impulse bit. A value of 0.6 pound-second has been selected to allow for tolerances. Roll phase plane plots are shown in figure 4-5.

Performance in the pitch and yaw axes is also affected by minimum-impulse-bit size. Based on a single-axis analysis the maximum pitch rates resulting from a minimum engine impulse bit for a given pitch attitude limit

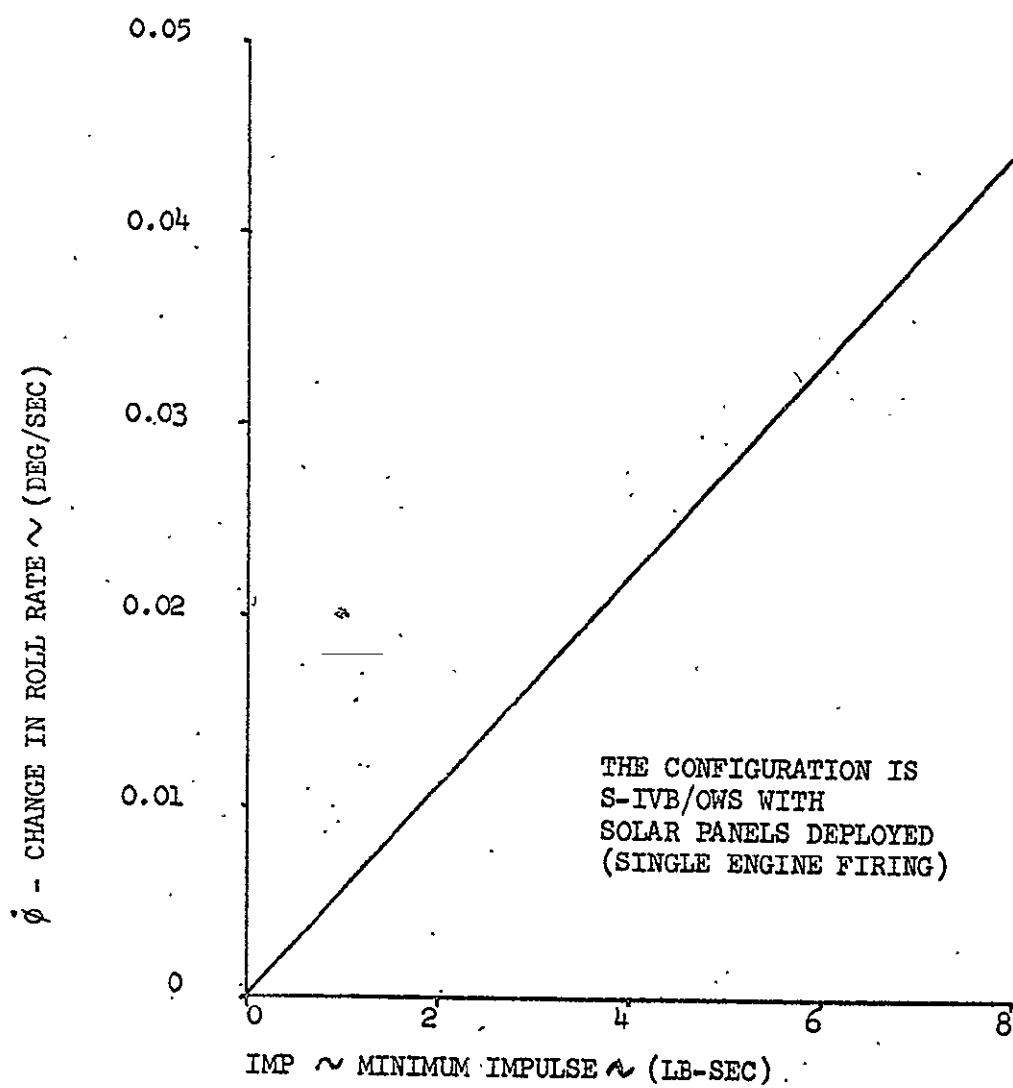


FIGURE 4-4. CHANGE IN BODY ROLL RATE DURE TO A GIVEN MINIMUM ENGINE IMPULSE BIT

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM UNCOUPLED ROLL PHASE PLANES

S-IVB/OWS ALONE

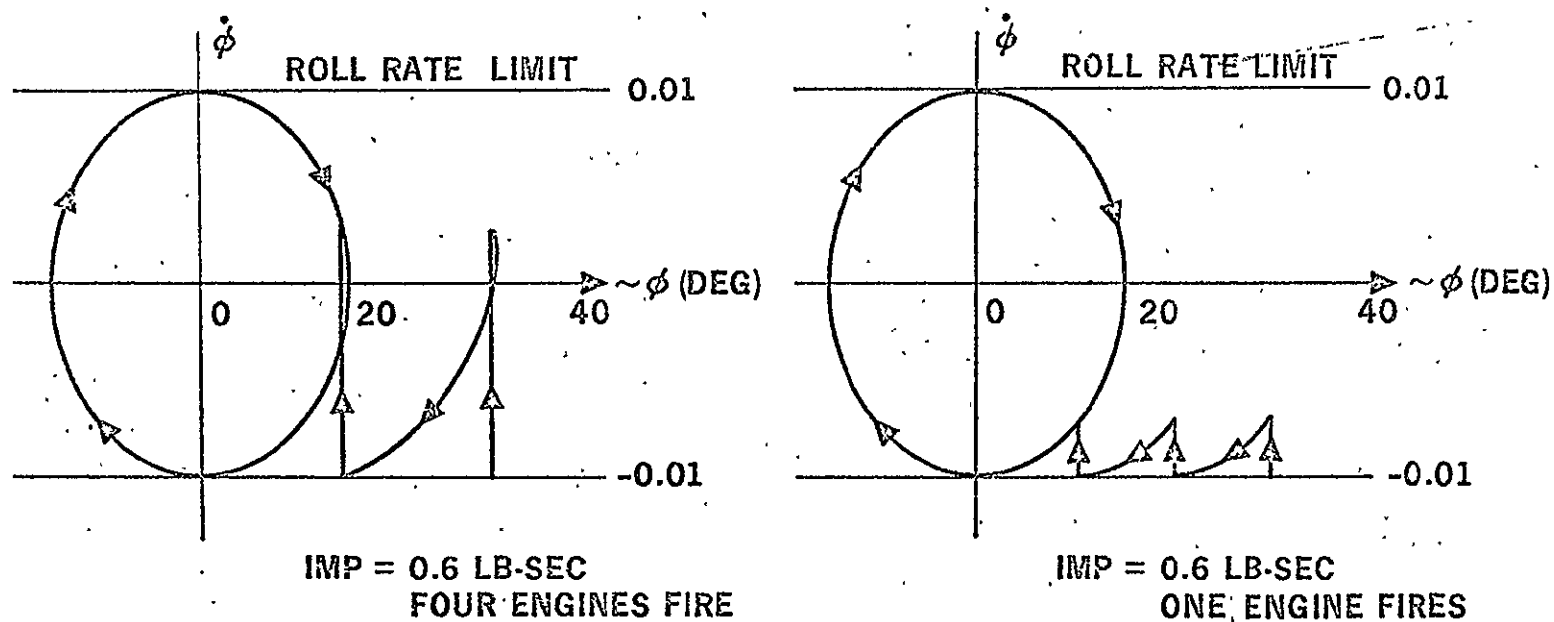


Figure 4-5. Roll Phase Planes for Uncoupled Analysis for the S-IVB/OWS with Solar Panels Deployed

are shown in figure 4-6. For the 20-degree deadband, rates of approximately 0.035 degree per second are developed by gravity-gradient torques alone. It is noted that the effect of the minimum engine impulse bit on maximum pitch rate increases as the attitude limits are decreased.

The sensitivity of the uncoupled WACS duty cycle to varying pitch attitude deadbands for several values of engine minimum impulse bit is contained in figure 4-7. Based on the single axis analysis, the duty cycle significantly increases as the pitch-attitude limit is decreased, or as the minimum impulse bit size is increased. For example, using a pitch attitude limit of 20 degrees, the duty cycle time is reduced by 23 percent by increasing the minimum engine impulse bit from 0.6 to 7.5 pound-seconds. This is equivalent to increasing impulse usage by a factor of 15.3.

The pitch impulse usage per day for a given, minimum engine-impulse bit is given in figure 4-8. Increasing the pitch-attitude limit reduces the required impulse significantly, but reducing the minimum-impulse bit reduces the required impulse by a greater amount. Figure 4-8, which is for the more critical S-IVB/OWS configuration, indicates that the 23-day mission would require less than 700 pound-seconds of impulse for the pitch axis to control gravity-gradient, providing the minimum engine-impulse bit is 0.6 pound-second or less. Impulse due to disturbances, coupling, and aerodynamics must be added to this value, and impulse for the other axes included also. Figure 4-9 shows a coupled, pitch-axis, phase-plane plot for the local-vertical mode for the S-IVB/OWS with side-docked CSM configuration.

Because aerodynamic torques were not available initially, they were not included in either the simulation or single-axis analysis. Consequently, a comparison was made between gravity-gradient torques and aerodynamic torques for a range of angles around the gravity-gradient local vertical stable orientation, to ensure that destabilizing aerodynamic moments would not cause roll-attitude divergence. The comparison was made for the configuration used for the 23-day mission with the CSM and M&SS docked to the S-IVB. The data used to compute the aerodynamic torque were obtained from reference 5. It is expected that removing the M&SS will not change the aerodynamic characteristics drastically. Figure 4-10 shows the roll

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM MAXIMUM PITCH ATTITUDE RATE

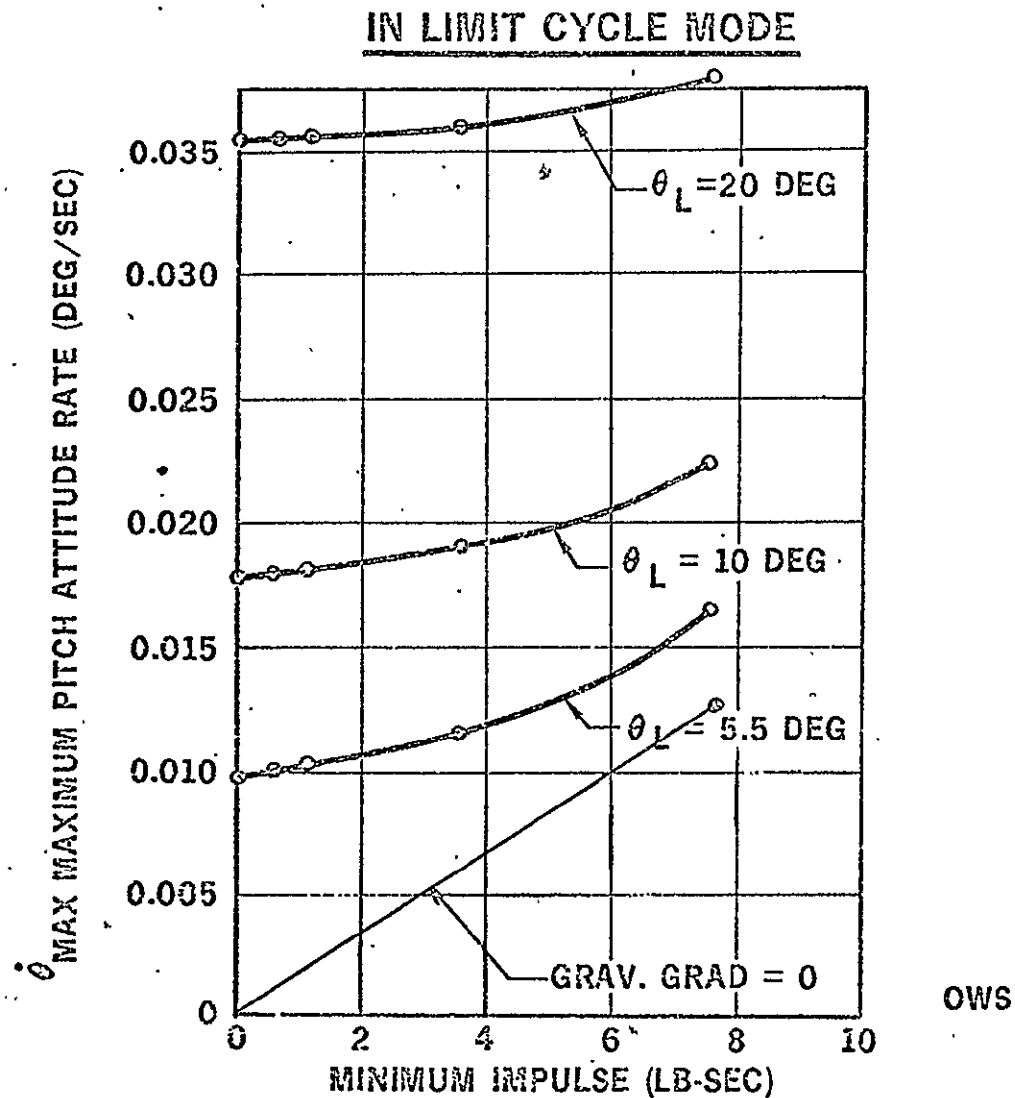


Figure 4-6. Maximum Pitch Attitude Rate Due to a Minimum Engine Impulse for Given Attitude Limits

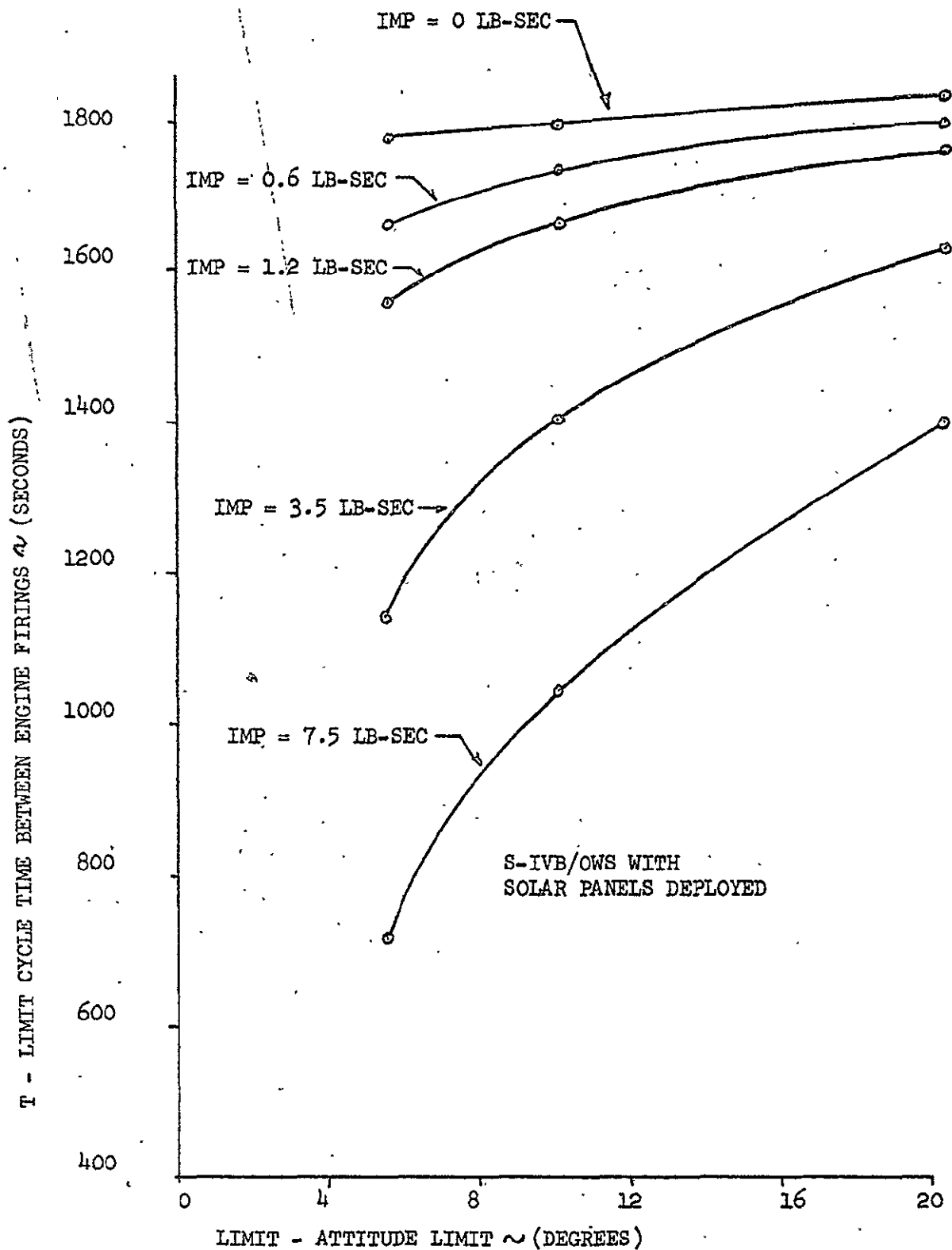


FIGURE 4-7. DUTY CYCLE AS A FUNCTION OF PITCH ATTITUDE LIMITS FOR GIVEN ENGINE IMPULSES

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM REQUIRED IMPULSE PER DAY

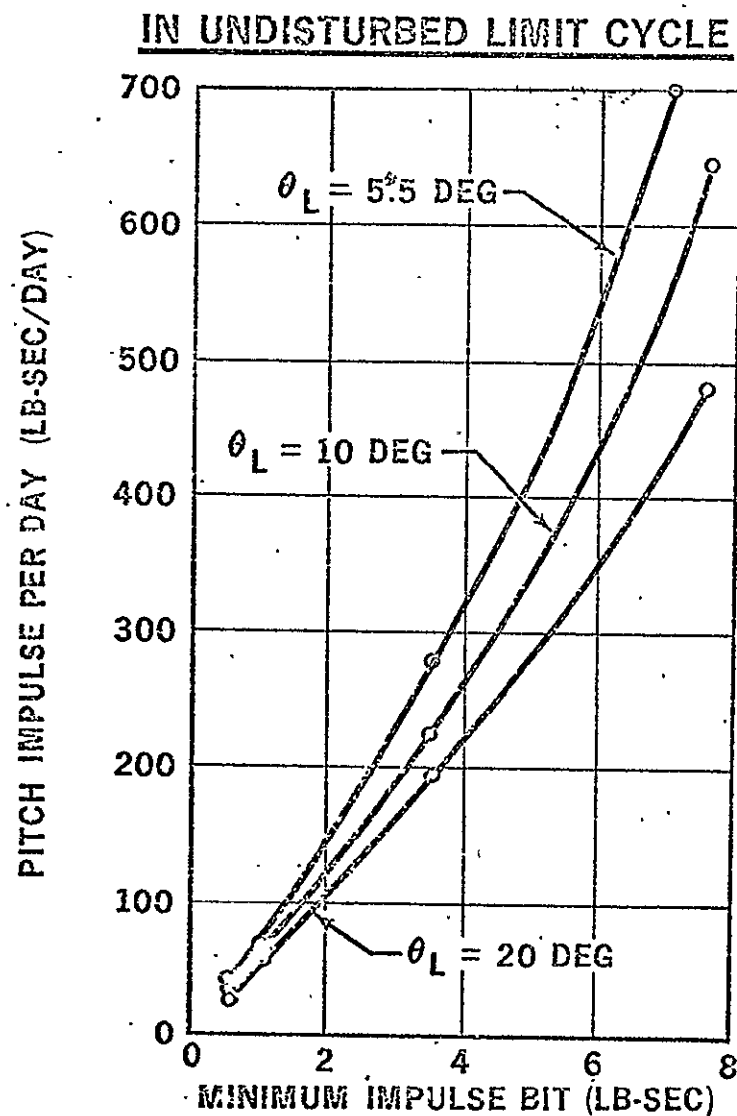


Figure 4-8. Required Impulse Per Day for a Minimum Engine Impulse and Attitude Limit

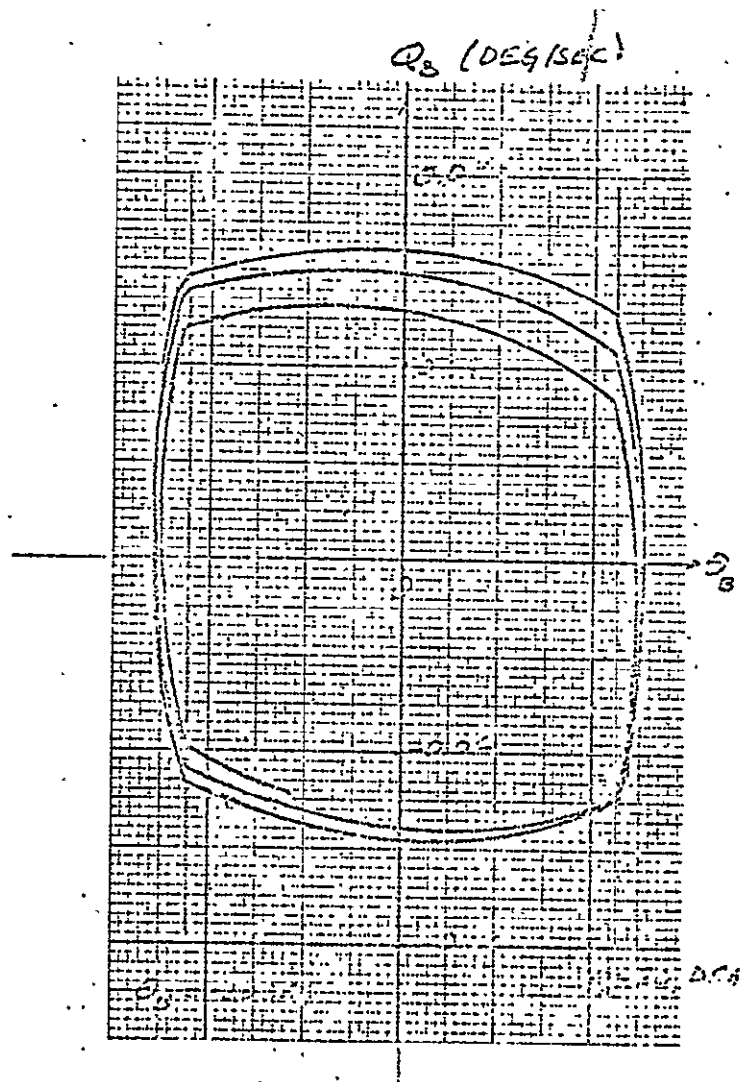


Figure 4-9. Pitch Axis Phase Plane From Simulation for the S-IVB/QWS with CSM.

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM TORQUE

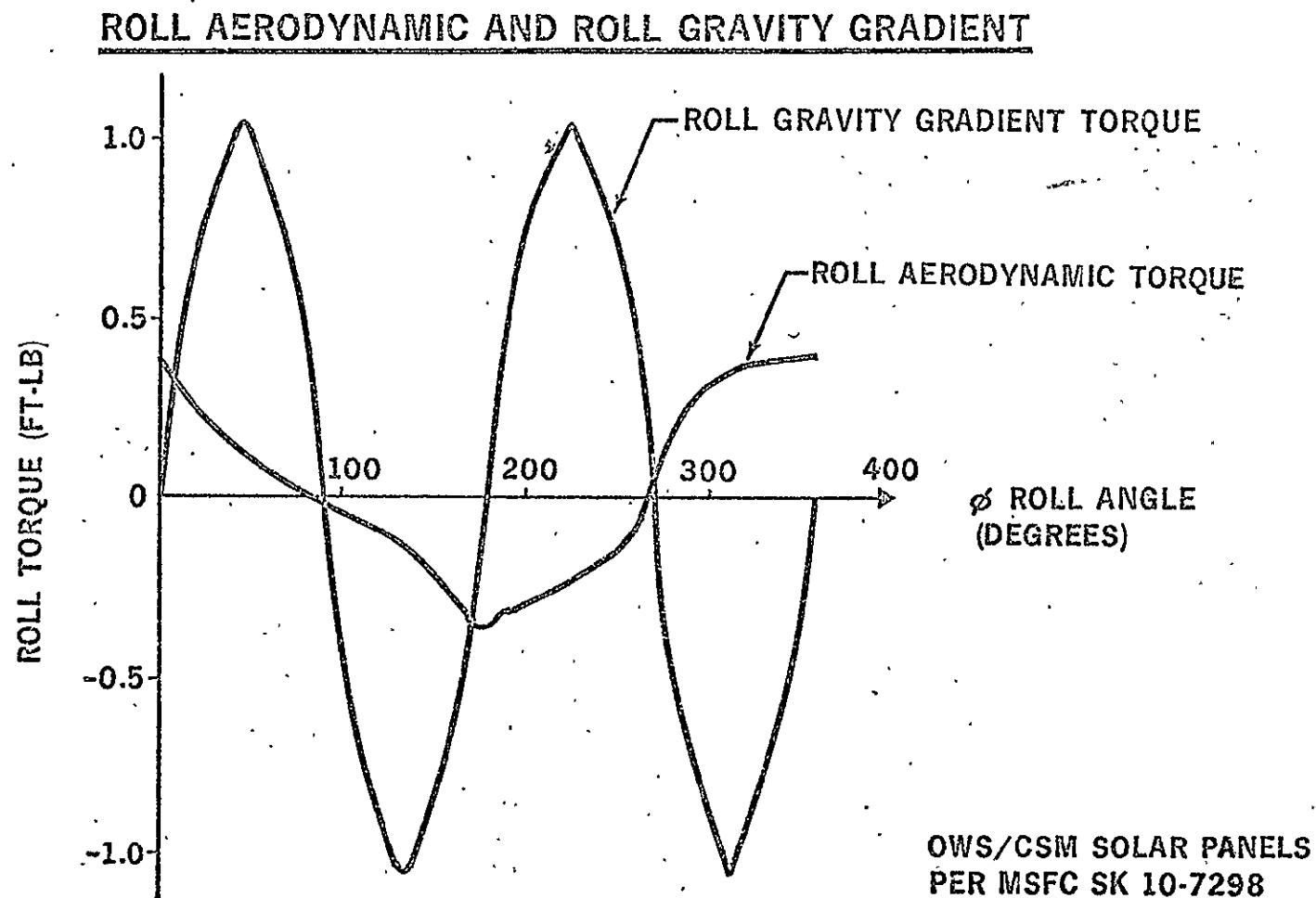


Figure 4-10. Roll Aerodynamic Torque and Roll Gravity Gradient Torque as a Function of Roll Attitude

gravity-gradient and roll aerodynamic torques as a function of roll attitude for a pitch attitude of 90 degrees.

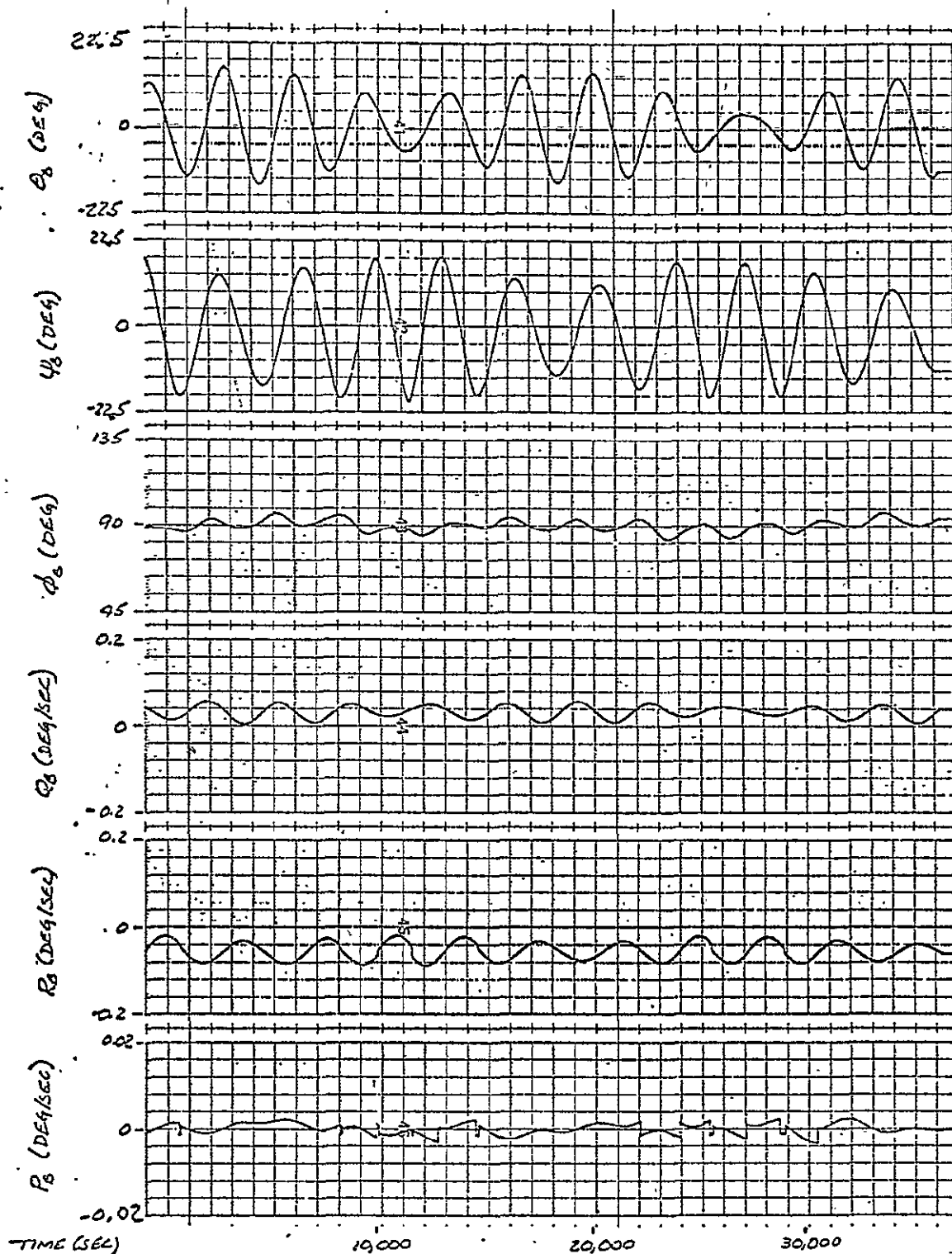
Based on figure 4-10, it appears that the roll aerodynamic torque for roll angles between 0 and 180 degrees will act with gravity-gradient to stabilize the CSM/OWS combined vehicle. Therefore, the CSM/OWS configuration for the 23-day mission will be stable in gravity-gradient with the CSM trailing. With the CSM leading ($\phi = 270$ degrees), aerodynamic moments are destabilizing. These aerodynamic moments will change for other solar-panel arrangements.

Similar comparisons of gravity-gradient and aerodynamic torques were made for pitch and yaw. It was found that the aerodynamic torques were negligible in yaw, while they were sufficiently large in pitch to cause the pitch-stable gravity-gradient position to be shifted up to four degrees.

Transient histories of orbital coast for several configurations, using various initial conditions, were obtained by simulation techniques. It was found that rate feedback was not required for control. Figure 4-11 shows transient histories of body attitudes and body rates for the S-IVB/OWS in orbit with its solar panels deployed. The pitch and yaw attitudes were controlled to 20 degrees, and the roll rate was maintained below 0.01 degree per second. It is shown in figure 4-11 that deviations in roll-attitude (ϕ_R) are limited by the combination of a roll-rate switching limit and restoring-torques from gravity-gradient and centrifugal torques.

4.1.2.3 Gravity Gradient Capture

Analysis of the gravity-gradient capture mode was required to 1) establish system rate and position gains and attitude error limits which will assure capture from large initial attitude errors with optimum transient response and impulse usage, and 2) to determine an acceptable impulse bit and attitude gain to preclude tumbling from the passive gravity-gradient position. The capture mode performance and requirements are different for each of the following configurations, because of the different inertia parameters.



INITIAL CONDITIONS: $\phi_p = -69 \text{ DEG}$ $Q_p = 0.1 \text{ DEG/SEC}$ $A = 0.05 \text{ P Y}$
 $\psi_p = 95 \text{ DEG}$ $R_p = -0.0632 \text{ DEG/SEC}$ $A_{\text{roll}} = 0.0 \text{ P Y}$
 $\theta_p = 5 \text{ DEG}$ $P_p = 0$ $A_{\text{pitch}} = 100 \text{ Roll}$

FIGURE 4-11. TRANSIENT HISTORY OF BODY ATTITUDE & RATES

- a. OWS alone with solar panels deployed
- b. OWS with CSM docked to a side port
- c. OWS with LM/ATM docked to a side port

The yaw-axis phase-plane plot, which includes a simplified version of the switching lines from figure 3-2, and shows a capture trajectory for the gravity-gradient capture mode, is shown in figure 4-12. An 0.05 degree per second rate ledge will limit propellant consumption while giving maximum capture times, from an inverted position, of approximately one-half orbit. The 10-degree attitude-error limit will be within the horizon-sensor linear range for expected principal axis variations. The rate gain, attitude gains, and attitude-error limits can be adjusted to optimize propellant consumption and capture time for the final configuration.

The maximum permissible body rates, as a function of pitch and yaw angle deviations from the local vertical for passive capture of the gravity gradient orientation, are defined by the capture envelopes shown in figure 4-13. These curves are derived based on energy considerations. Gravity-gradient capture is assured, if the combined pitch and yaw body attitudes and rates lie below the curves when active control is terminated. The envelopes for both configurations that require gravity-gradient capture are given in the figure. The allowable pitch and yaw rates for the ATM docked to the S-IVB/OWS are considerably lower than for the S-IVB/OWS alone configuration with solar panels deployed (0.52 degree per second compared to 0.66 degree per second). The maximum allowable attitudes are approximately the same (37 degrees). The roll attitude was assumed to be 90 degrees from its stable gravity-gradient position. However, the effect of roll attitude on all envelopes determined was negligible, because the roll gravity-gradient terms were small.

The capture envelopes given in figure 4-13 were determined, based on the assumption that all three rates (pitch, yaw and roll) were equal. It was found by limiting the roll rate to 0.01 degree per second, which is the specified roll-rate limit, that the maximum permissible pitch and yaw rates could be increased by only a negligible amount (0.001 degree per second) for capture.

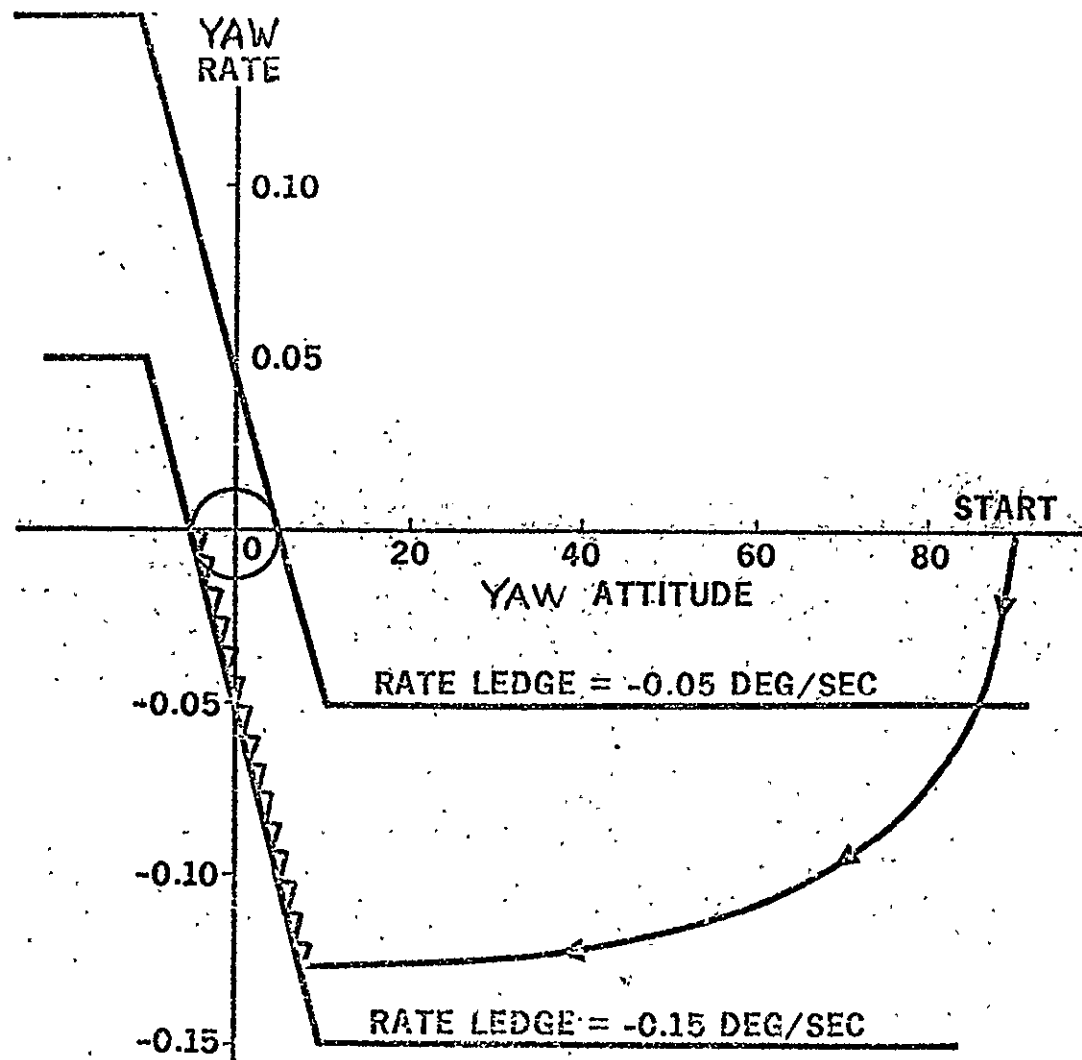


Figure 4-12. Yaw Phase Plane for Gravity Gradient Capture Mode (Uncoupled)

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM GRAVITY GRADIENT CAPTURE LIMITS PRIOR TO ORBITAL STORAGE

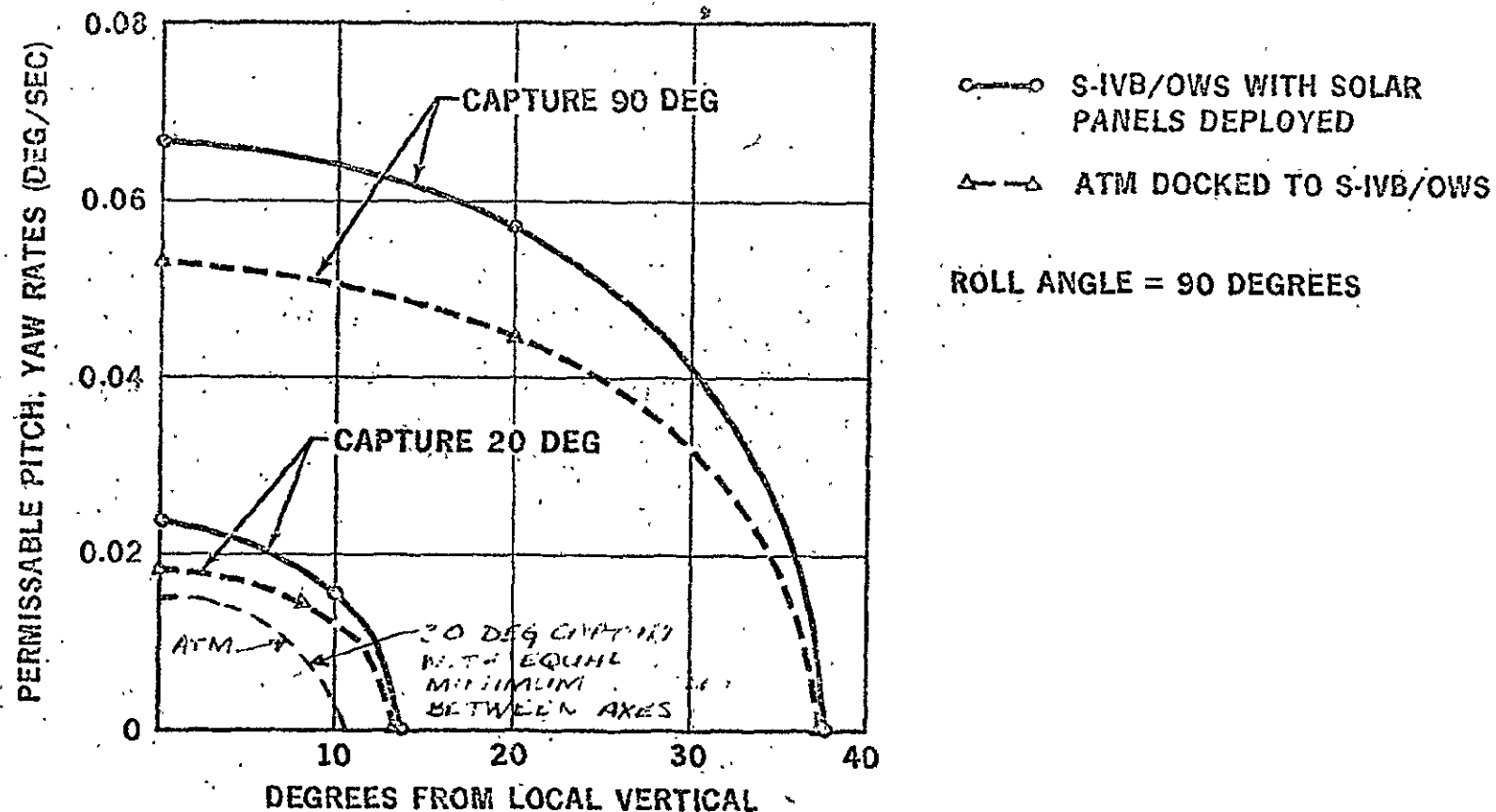


Figure 4-13. Envelopes of Allowable Pitch and Yaw Rates Versus Angle Deviation From Vertical to Capture in Gravity Gradient

Figure 4-13 also shows boundaries which provide a margin on capture such that pitch or yaw attitude excursions do not exceed 20 degrees. This requires much lower rates and attitude errors than required to ensure capture. Rates of approximately 0.016 degree per second are required for attitude errors of 5 degrees. A roll-rate restriction of 0.016 degree per second restricts the minimum engine-impulse bit to 0.8 pound-second. However, if the body rates in all three axes are assumed to have values corresponding to equal minimum-impulse bits, the minimum engine-impulse bit can be approximately 3.0 pound-seconds. The capture limits for this case for the ATM configuration are also shown in figure 4-13.

Figure 4-14 shows the response of the OWS to an initial roll displacement of 90 degrees without attitude control. It can be seen that this roll displacement does not cause tumbling about the pitch or yaw axes.

4.1.3 RCS Thrustor Location Study

The effect on the WACS impulse usage of locating the APS modules at the present S-IVB APS location, the IU, and the Air Lock was investigated. Figure 4-15, which contains the results of this analysis for the various S-IVB/OWS configurations, shows the relationship between impulse usage and APS module location for the impulse usage of reference 4, which is used primarily in the pitch and yaw axes. The impulse was normalized to unity for the present S-IVB APS location.

The results of figure 4-15 indicate that the present location of the APS engines is superior to all other locations for the configurations used during Mission A, Mission B, and orbital storage after Mission B. The location at the forward end of the Airlock is superior only for the configuration of the OWS alone. Considering the total mission, the present location of the APS on the S-IVB is superior. Location in the IU is inferior in all cases.

Several other factors favor the S-IVB location over the Airlock. It is necessary to locate engines on the S-IVB aft skirt for control during powered roll, because thrustors on the Airlock would be covered during at least a part of the time that roll control of the S-IVB is required,

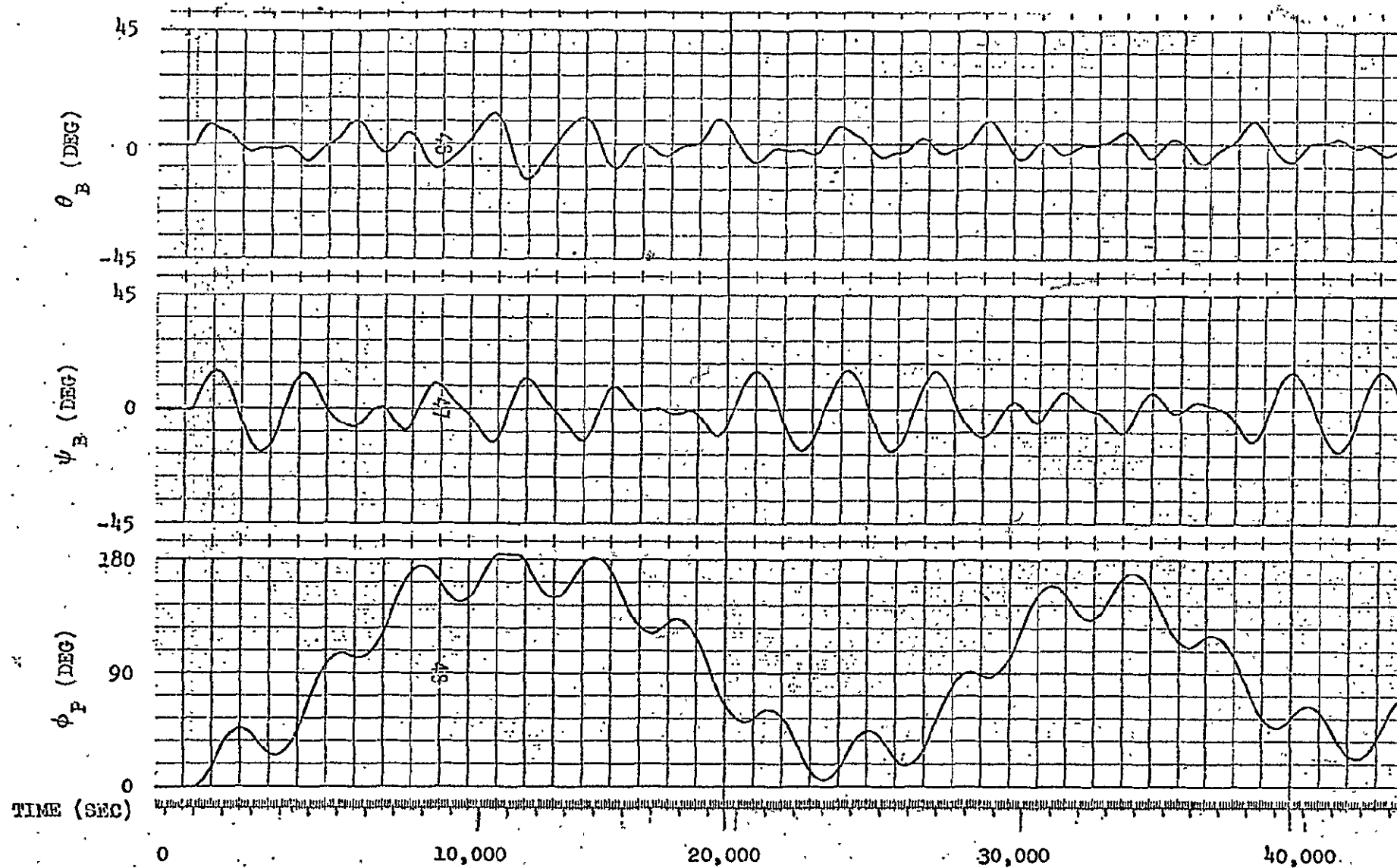


Figure 4-14. Transient History of Body Attitudes for an Initial Roll Attitude of Zero Degrees and the WACS Inactive

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM NORMALIZED IMPULSE FOR ALTERNATE LOCATIONS

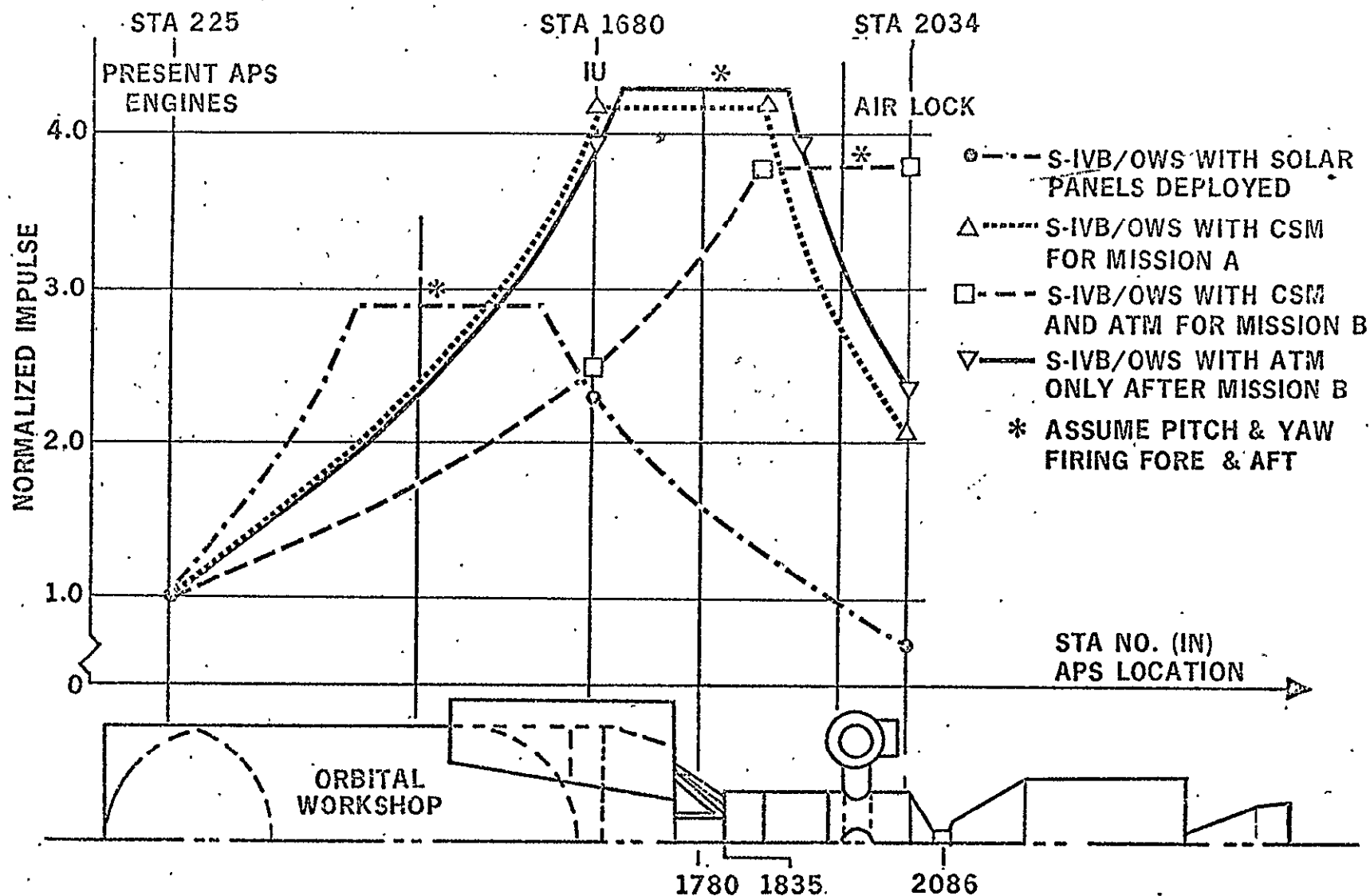


Figure 4-15. Normalized Impulse for a Given Control Lever Arm for Several Configurations

(during J-2 engine burn). This would require multiple APS locations. Also, impingement of reaction-control thruster plumes on the CSM or LM/ATM would be much more severe for Airlock-mounted thrusters.

4.1.4 Summary of Dynamics Derived System Requirements

The additional system requirements obtained from the simulation and analysis are summarized in the following sections and tables 4-2 through 4-5.

4.1.4.1 Impulse per Module

The impulse per module and total impulse for four possible propulsion configurations is given in table 4-2. This table includes the boost and passivation requirements determined by Douglas, in addition to the AAP requirements furnished by MSFC in reference (2). Several assumptions were made in determining the minimum propellant requirements for each module. It was assumed that 60 percent of the total required impulse, to control limit cycle, could come from one module because of variations in impulse-bit size between thrusters, one-sided disturbances such as aerodynamic torques in the pitch axis, bias voltages, and cross-axis coupling effects. These effects could cause one module to fire more frequently than the other during limit-cycle operation. It was assumed that 90 percent of the impulse for AAP requirements could be derived from either module, based on the considerations above and the unidirectional nature of docking disturbances, which, as indicated in section 4.1.4.2, account for most of the impulse requirements.

4.1.4.2 Pulse Width Distribution and Duty Cycle

For purposes of evaluating the specific impulse of the engines during each mission phase, the total required impulse was divided into four categories of duty cycles as given in table 4-3. Docking and separation require large continuous pulses, and are covered in the first category. Category two includes duty cycles between five and ten percent, while category three includes duty cycles between one and five percent. Unperturbed limit cycle is covered in the fourth category as having a duty

TABLE 4-2
ESTIMATED IMPULSE REQUIREMENTS FOR
FOUR PROPOSED PROPULSION CONFIGURATIONS

CONFIGURATION	IMPULSE REQUIREMENTS (pound-second)					
	PASSIVATION & BOOST		COAST & DOCKING		ALL PHASES OF CONTROL	
	Total	Per Module	Total	Per Module	Total	Per Module
A Nonmanifolded	13,500	9,000	22,500	20,500	36,000	29,500
A Manifolded	13,500	6,750	22,500	11,250	36,000	18,000
B Nonmanifolded	13,500	4,500	22,500	10,250	36,000	14,750
B Manifolded	13,500	3,375	22,500	5,625	36,000	9,000
C Nonmanifolded	13,500	9,000	22,500	20,500	36,000	29,500
C Manifolded	13,500	3,375	22,500	5,625	36,000	9,000
D Nonmanifolded	13,500	4,500*	22,500	10,250*	36,000	14,750*
D Manifolded	13,500	3,375*	22,500	5,675*	36,000	9,000*

* These impulse requirements for configuration D, which has two separate systems in each module, are per system or half module.

TABLE 4-3
PRELIMINARY ESTIMATED PULSE WIDTH DISTRIBUTION
AND DUTY CYCLE FOR S-IVB WORKSHOP ATTITUDE CONTROL SYSTEM

Mission Sequence (Times from S-IVB Cutoff)	TOTAL IMPULSE PER MODULE* (S-IVB CONFIGURATION)			
	Continuous Pulses (>0.1 sec)	Pulsed		
		10% > Duty -Cycle > 5%	5% > Duty -Cycle > 1%	1% > Duty -Cycle
1. S-IVB Separation to 7 1/2 Hours After S-IVB Cutoff	2,350	2,870	1,680	2,100
2. 7 1/2 Hours to 48 Hours	0	0	0	1,550
3. 1 to 2 Hours Following (2) (Start of 23-Day Mission A)	5,500	500	500	650
4. During 23-Day Mission A Assume Min. Imp = 0.6 lb-sec)	0	0	0	1,550
5. End of 23-Day Mission A	600	0	0	500
6. 3 to 6 Months Following End of 23rd Day	No Usage	No Usage	No Usage	No Usage
7. Start of 56-Day Mission B	3,400	200	200	300
8. Any Time During 56 Days (2 Day Duration)	1,000	0	0	550
9. End of 56-Day Mission B	150	0	0	50
10. 6-12 Months Following End of 56-Day Mission B	No Usage	No Usage	No Usage	No Usage
11. Start of 2nd Day Mission C	1,400	0	0	1,900
TOTALS	14,400	3,570	2,380	9,150

GRAND TOTAL 29,500

* For configuration on A Non manifolded of Table 4-I.

cycle less than one percent. Powered roll and LH_2 venting are two typical examples of category two, while LOX venting is typical of category three.

The impulse for duty cycles less than one percent during undisturbed limit-cycle operation was sized, based on an engine minimum impulse of 0.6 pound-second. A decrease in impulse per pulse, because of lower specific impulse at low duty cycles, will reduce the total impulse.

4.1.4.3 Minimum Pulse Bit

The maximum value of the minimum-impulse-bit was established in section 4.1.2.2 as 0.6 pound-second based on the roll rate requirements. Lower values are desirable for minimizing impulse usage during limit-cycle operation, and for more accurate gravity-gradient capture. During IU operation (boost and passivation), the present 0.065-second pulse-width can be used.

4.1.4.4 Thrust During Boost and Passivation

A roll thrust of 25 pounds is required during boost and passivation for the present S-IVB APS thruster arrangements, based on previous analyses. If dual thrusters (6 per module) are used, the thrust per engine can be reduced by a factor of 2.

4.1.5 Results of Failure on Impulse Requirements

Table 4-4 contains the estimated changes in impulse requirements due to possible failures in the WACS for both manifolded and nonmanifolded propellant tanks. The impulse indicated in table 4-4 refers to table 4-2. No estimates were made for cases which result in "loss of control" of the CWS. For purposes of simplicity, the following assumptions were made with respect to possible failure:

- a. The loss of a propellant tank resulted in a total loss of a module for the nonmanifolded case.

TABLE 4-4

ESTIMATED CHANGES IN IMPULSE REQUIREMENTS TO TOLERATE POSSIBLE
FAILURES WITHOUT LOSS OF CAPABILITY

TYPE OF FAILURE	PROPULSION CONFIGURATION			
	A	B	C	D
Loss of Tank Non-Manifolded	No Control	1.33 x Imp Per Module	No Change	2 x Imp Per Module
Loss of Tank Manifolded	2 x Imp Per Module	1.4 x Imp Per Module	1.33 x Imp Per Module	1.33 x Imp Per Module
Loss of Engine Non-Manifolded	No Control	1.33 x Imp Per Module	No Change	No Change
Loss of Engine Manifolded	No Control	1.05 x Imp Per Module	No Change	No Change
Loss of Line Nonmanifolded	No Control	1.33 x Imp Per Module	No Change	2 x Imp Per Module
Loss of Line Manifolded	No Control	No Control	No Control	No Control
Loss of Power to System Non-Manifolded	No Control	1.33 x Imp Per Module	No Change	2 x Imp Per Module
Loss of Power to System Manifolded	No Control	No Change	No Change	No Change
Engine Failure "ON" Nonmanifolded	No Control	2 x Imp Per Module	No Control	2 x Imp Per Module
Engine Failure "ON" Manifolded	No Control	No Control	No Control	No Control

NOTE: The factors per module refer to Table 4-1
Assuming separate tank for each engine cluster

- b. The loss of a single fuel line resulted in single module loss for the nonmanifolded case and complete WACS loss for the manifolded case.
- c. The loss of power to a module resulted in total loss of the module.
- d. Failure due to an "engine on" situation resulted in total consumption of the propellant in the module that contains the "on" engine and the opposing module for the nonmanifolded case (a roll-yaw engine failure may not be so severe). Total loss of the propellant resulted for the manifolded case.

ORBITAL WORKSHOP ATTITUDE CONTROL SYSTEM SUBSYSTEM REQUIREMENTS SUMMARY

MODE	AXIS	REQUIREMENTS/BASIS OF REQUIREMENTS***			
		ATTITUDE DEG	RATE DEG/SEC	THRUST LB	MINIMUM IMPULSE BIT LB/SEC
I.U. (BOOST)	ROLL	$\pm 1.$		$> 12.5^{**}$	0.5-7.5 ACCEPTABLE
LOCAL VERTICAL	P, Y	$\pm 20^{\circ}$ SPEC	$(W_0 \pm 0.04)$	*	≈ 0.6 IMPULSE USAGE (≈ 150 LB)
	ROLL	$(\pm 20^{\circ})$ / SOLAR PANELS	< 0.01 / $\pm 20^{\circ}$ HOLD	*	≈ 0.6 / RATE LIMIT OF .01
GRAVITY-GRADIENT CAPTURE (20 DEGREE CAPTURE)	P, Y	$\approx \pm 5^{\circ}$	≈ 0.01		≈ 3.0 2 YAW ENGINES
	ROLL		≈ 0.033		≈ 3.0 (2 ENG)
RATE (DOCKING)	P			(12)	7.5 OR LESS ACCEPTABLE
	Y		0.1	(12)	
	R			(12)	

*DISTURBANCES TBD

MSFC REQUIREMENT OR INHERENT IN MODE REQUIREMENTS, IF NOT SPECIFIED.

** FOR 6 ENGINE/MODULE CONFIGURATION

4.2 Phase II Analysis of Solar Orientations

4.2.1 Dynamic Model Description

The dynamic model used to simulate the vehicle in a solar orientation is similar to the model used for the gravity gradient orientation which is discussed in section 4.1.1. Figure 4-16 defines the OWS coordinate system with respect to its earth orbit and the sun. The solar panels were assumed to lie along the y-axis for all configurations simulated.

The sign convention, engine module locations, and vehicle configuration are shown in figure 4-17. The order of the Euler angle rotation used for Phase II solar orientations was changed to pitch, roll, and yaw to minimize changes to the analog simulation used in Phase I. A block diagram of the simulation similar to figure 4-2 in section 4.1.1 can be drawn for the solar orientation. However, changes are necessary to include the angle between the earth orbit and the sun vector (β) and to convert the horizon-sensor attitude-error equations to sun sensor error equations. The same small angle approximations were made in developing the sun sensor error equations as for the horizon sensor error equations. These variations are contained in the block diagram in figure 4-18.

The same technique of writing equations of motion was used for Phase II as for Phase I. Moments about the body axes due to the body mounted engines were derived and transferred into the principal axes using the transform matrix from section 4.1.1. Rotational equations of motion which included gravity-gradient and gyroscopic torques and the transformed engine moments were written in terms of rotations about the principal axes. This simplified the simulation by eliminating the quadratic terms involving products of inertia. Aerodynamic torques were not included in the simulation because their effect on impulse usage was considered small at the 260 nautical mile altitude, and further, because of their cyclic nature they will tend to add to gravity-gradient torques throughout half of an orbit and subtract during the other half of the orbit.

4.2.2 Impulse Requirements

The following sections summarize impulse requirements to counteract gravity-gradient torques for each of the five orientation cases. The important

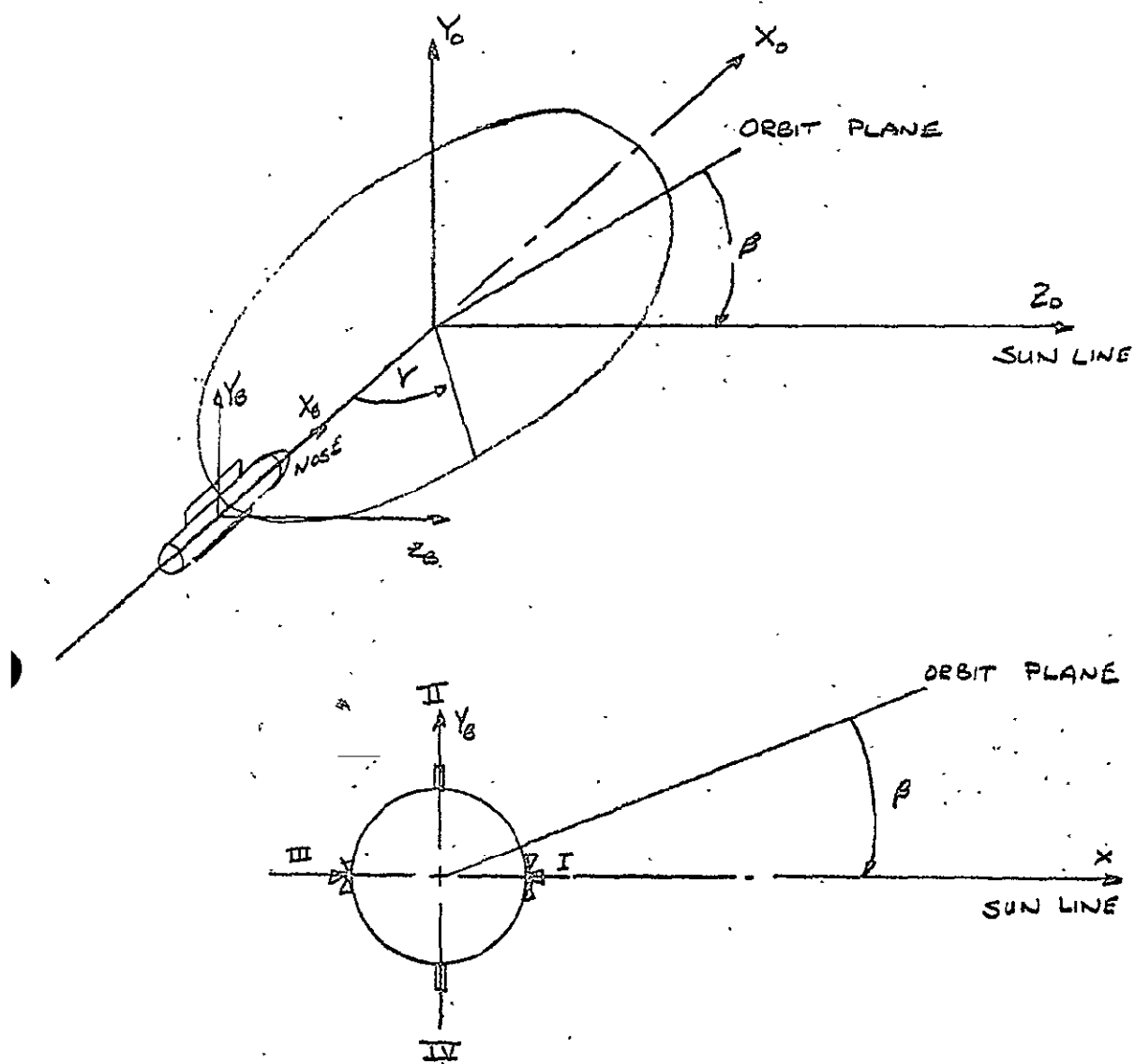
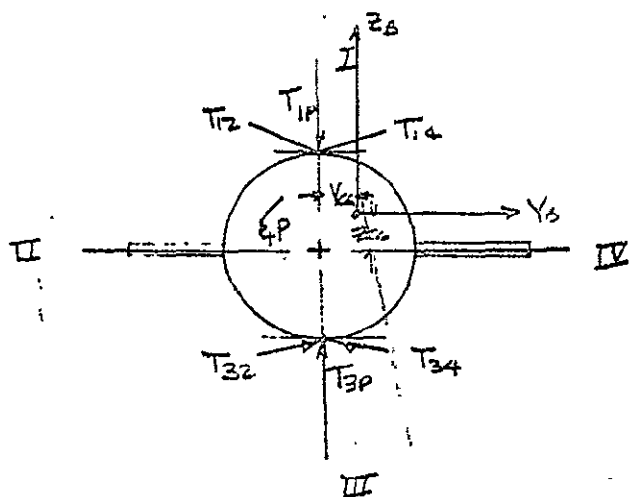
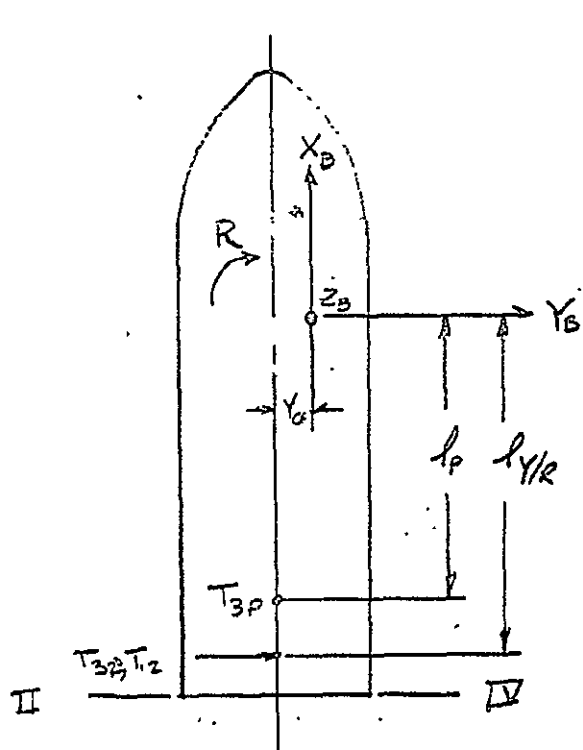


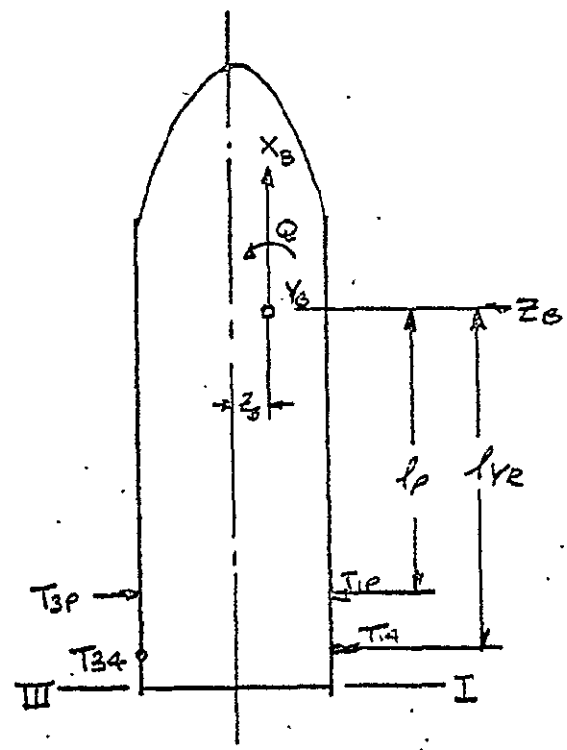
FIGURE 4-16. WORKSHOP VEHICLE COORDINATE SYSTEM



ROLL PLANE
(LOOKING AFT)



YAW PLANE
(LOOKING AFT)



PITCH PLANE

FIGURE 4-17 SIGN CONVENTION FOR WORKSHOP
SIMULATION MODEL

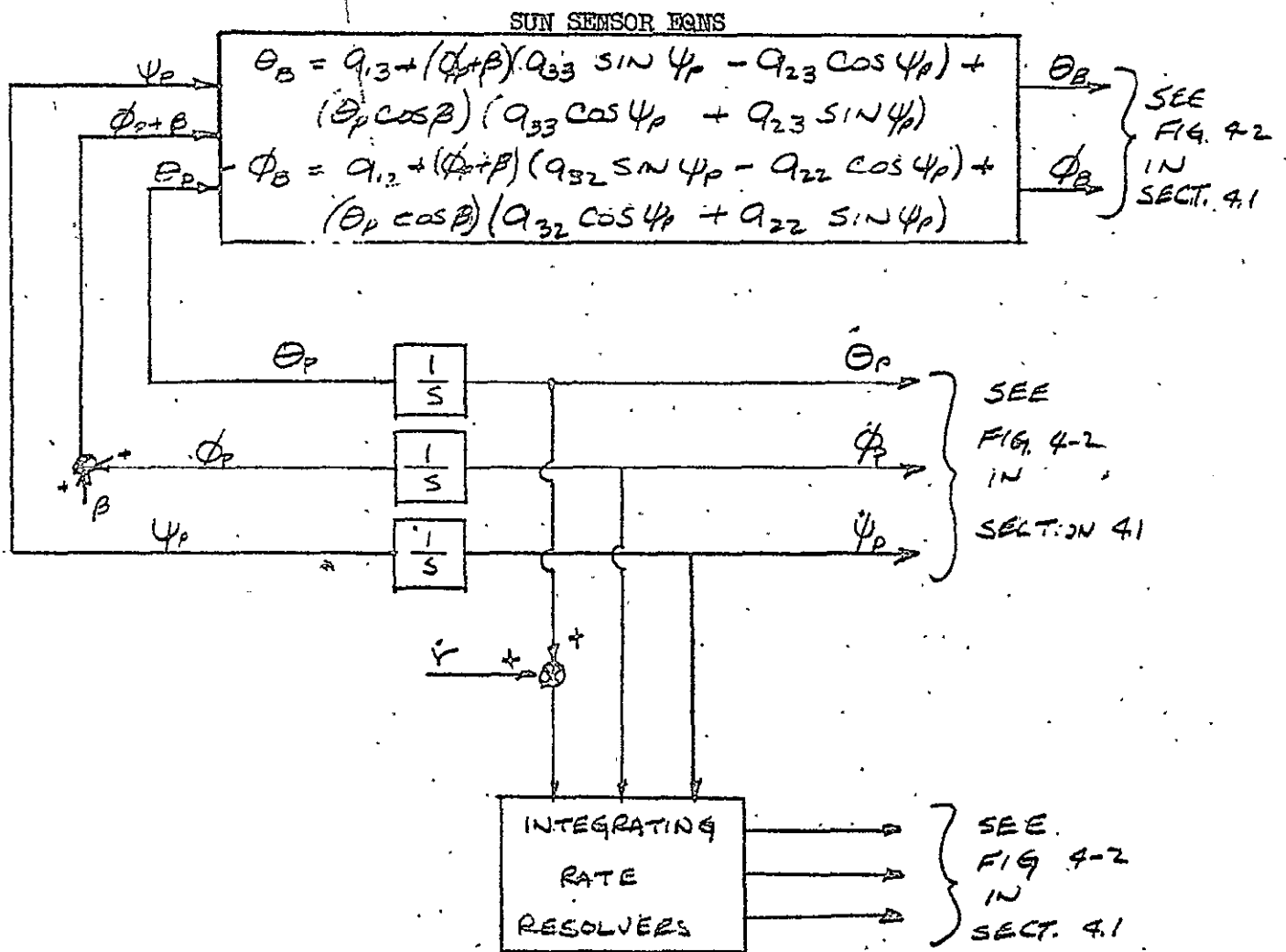


FIGURE 4-18 BLOCK DIAGRAM OF SIMULATION MODEL FOR SOLAR ORIENTATION

WACS parameters have been parameterized to permit optimization of the control system such that the required impulse per orbit is minimized. Both side-docked and end-docked CSM configurations have been evaluated and compared for each case. Mass data used to obtain the simulation results for each configuration is listed in table 4-1.

4.2.2.1 Case 1 Impulse

The workshop orientation for Case 1 is shown in figure 2-1 of section 3.4.6. The x-axis, which is the longitudinal axis of the vehicle, is controlled such that it is in the plane of the orbit while the z-body axis is controlled to be parallel to sun vector, thus orienting the solar panels in a plane normal to the sun vector.

The effect of attitude deadbands of 5, 10, and 30 degrees on impulse per orbit as a function of the angle between the orbit plane and the sun vector (beta) is shown in figures 4-19 and 4-20 for both configurations. It was observed from the simulation results that, for the mass data used, impulse usage was symmetrical about zero degree of beta for both configurations, with the exception of one set of parameters which were used only for Case 3. Therefore, impulse-usage results are shown only for positive betas. Pitch and yaw-roll impulse per orbit are indicated separately in figure 4-19, while total impulse per orbit is given in figure 4-20. Yaw-roll impulse requirements for the side-docked configuration are considerably greater than for the end-docked configuration as shown in figure 4-19. This is attributed to the larger gravity-gradient torques in roll which are caused by the greater difference in pitch and yaw inertias and the shorter yaw control arm, caused by the more aft center of gravity, for the side-docked configuration compared to the end-docked configuration. As a result of the high yaw-roll impulse requirements for the side-docked configuration, the total impulse requirements for the end-docked configuration are less. Reducing the attitude deadbands results in an increase in propellant consumption for both configurations.

A rate deadband of 0.01 degree per second was used in the simulation to obtain the data for figures 4-19 and 4-20. Figure 4-21 shows that for an

WORKSHOP ATTITUDE CONTROL SYSTEM IMPULSE BY AXIS - CASE 1

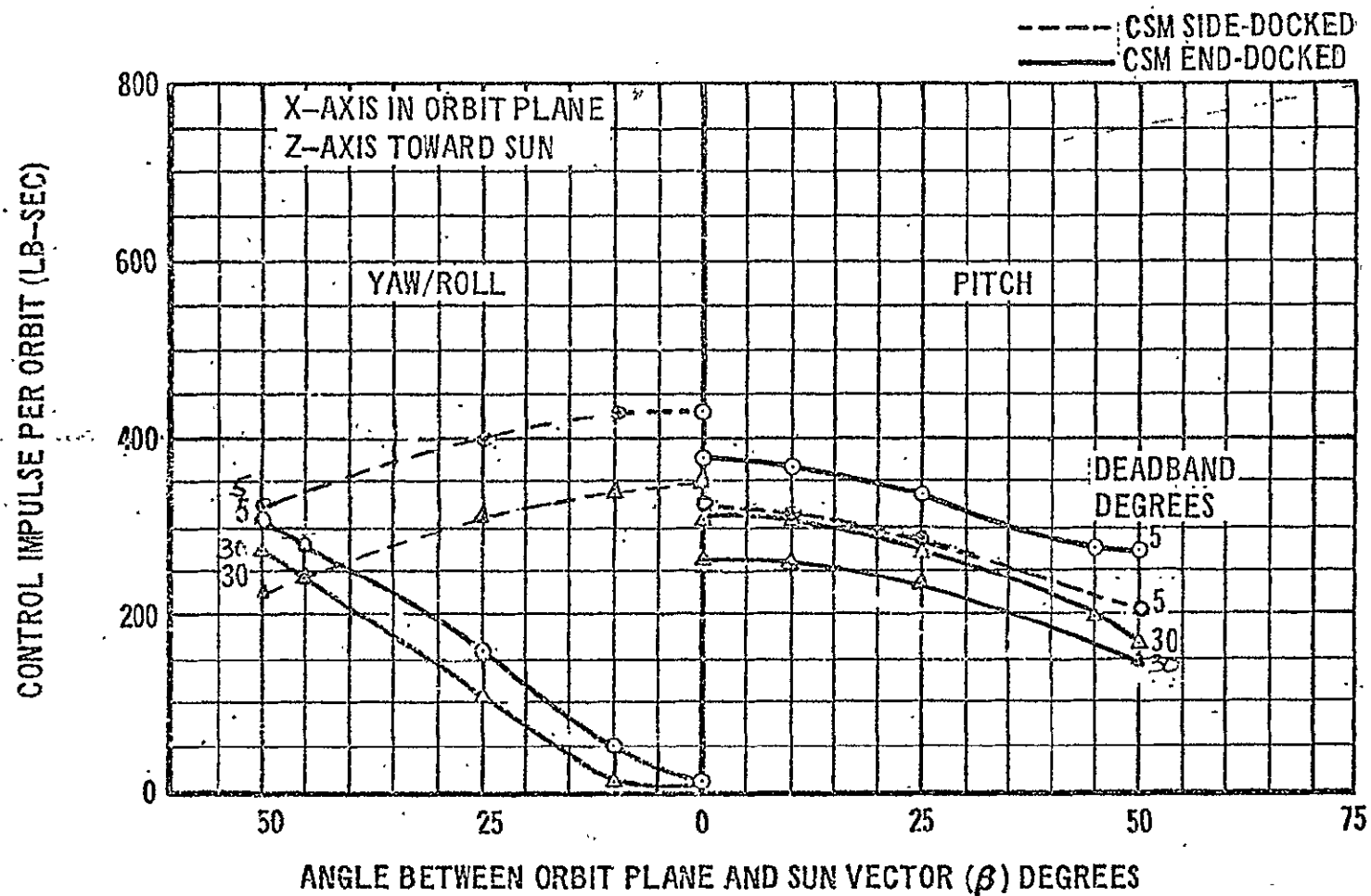


FIGURE 4-19 Impulse by Axis - Case 1

WORKSHOP ATTITUDE CONTROL SYSTEM

TOTAL IMPULSE - CASE 1

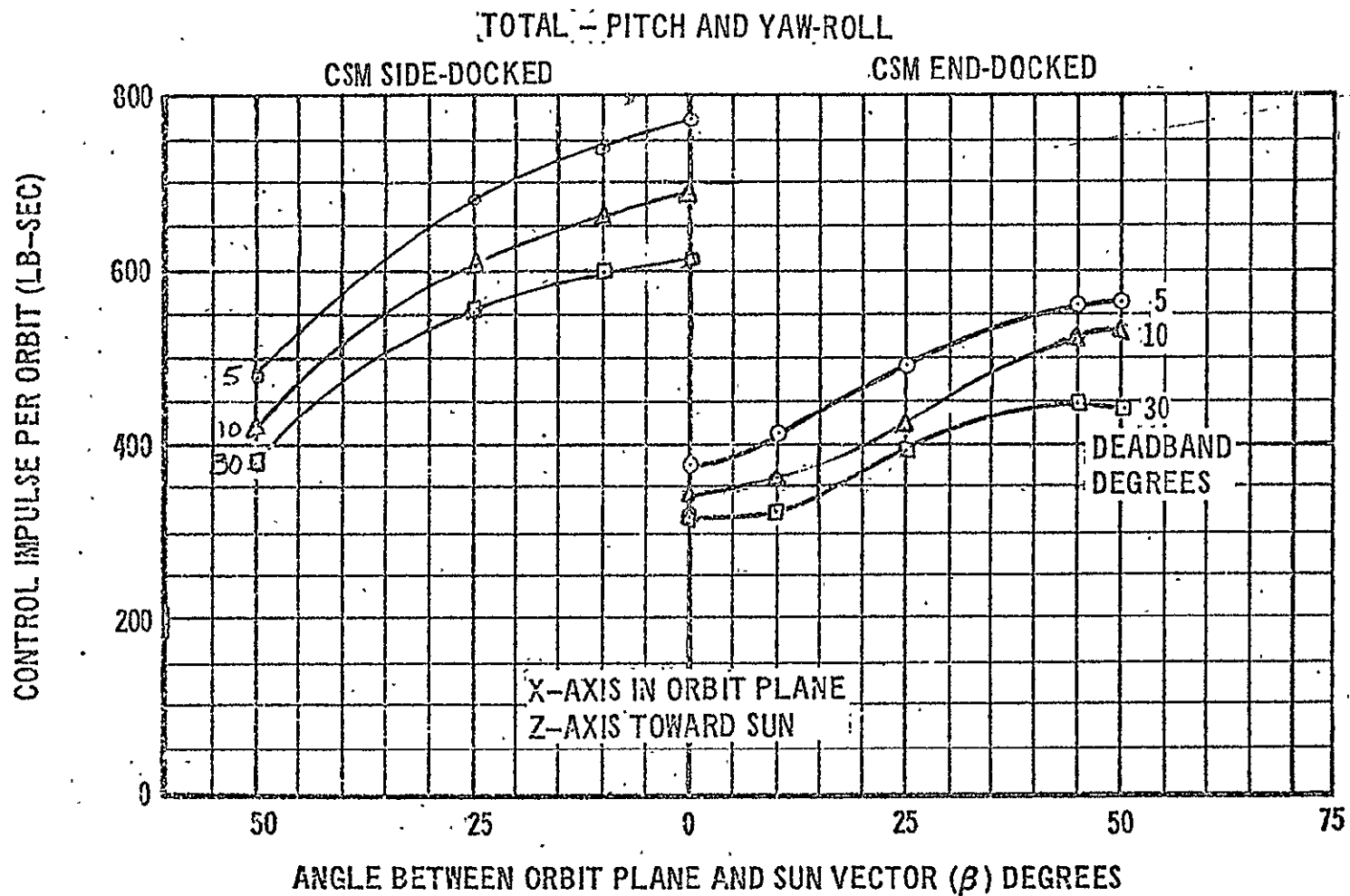


FIGURE 4-20 Total Impulse - Case 1

WORKSHOP ATTITUDE CONTROL SYSTEM RATE GAIN EFFECT ON IMPULSE USAGE - CASE 1

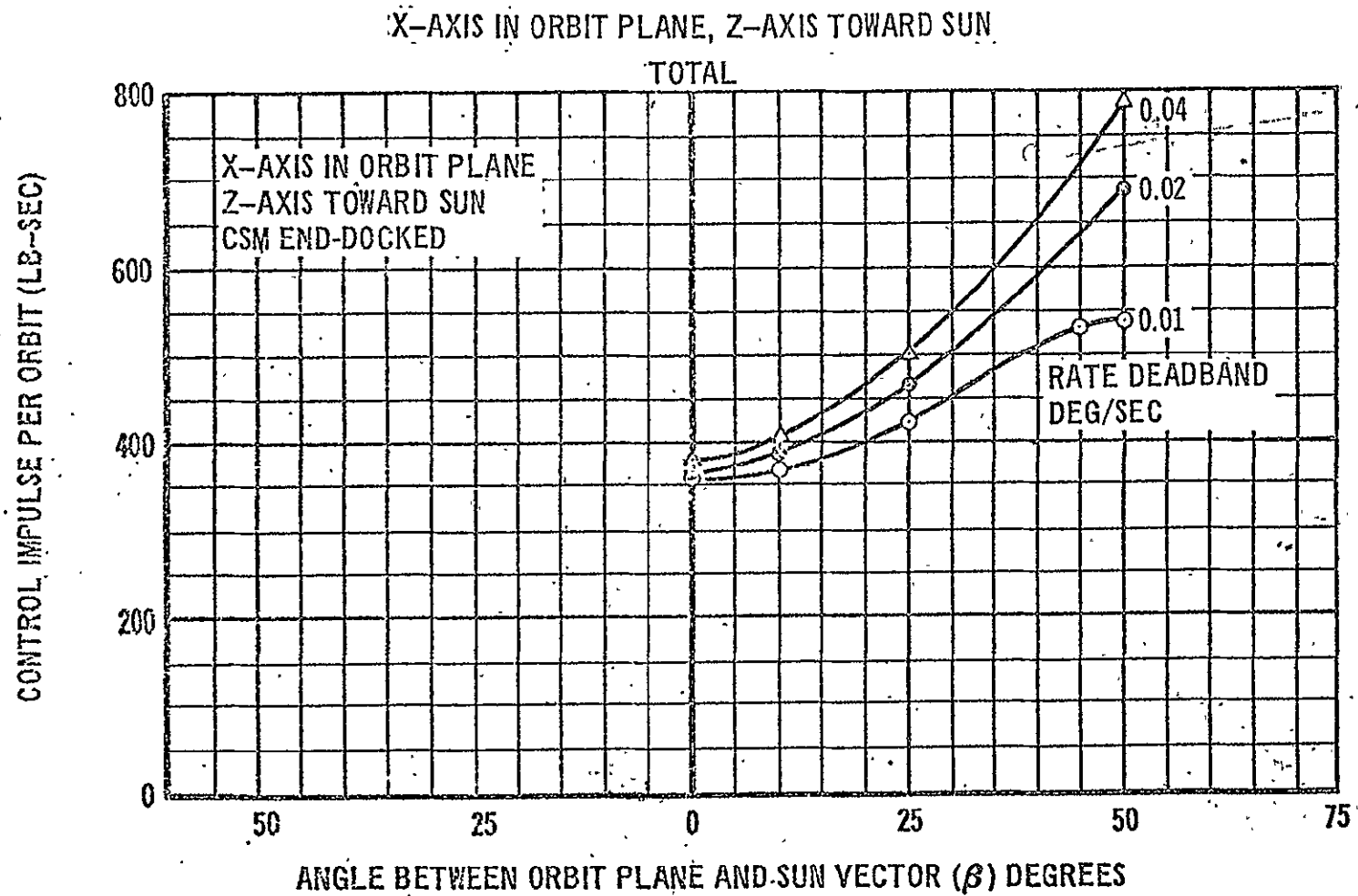


FIGURE 4-21 Rate Gain Effect on Impulse Usage - Case 1

attitude deadband of ± 10 degrees, a 0.01 degree per second rate deadband decreased impulse usage compared to higher rate deadbands for Case 1 with the end-docked configuration. Other simulation results indicated that this low rate-deadband was also preferred for attitude deadbands greater than 10 degrees. Similar results were found for the side-docked configuration.

4.2.2.2 Case 1A Impulse

Case 1A is similar to Case 1 with the exception that the principal y-axis is controlled to lie in the orbit plane rather than the x-axis. This positions the workshop vehicle so that the gravity-gradient torques that are imposed on the OWS when beta is zero are minimized; therefore, the propellant consumption per orbit is considerably reduced from Case 1 for both configurations as shown in figure 4-22. However, the gravity-gradient torques seen by the OWS increase as the vehicle longitudinal axis is rotated away from the local horizontal through the angle beta. The impulse increases for both configurations as the angle between the orbit plane and the sun vector increases.

Total impulse per orbit for several attitude deadbands for Case 1A is shown in figure 4-22. The impulse is reduced at large beta angles for increased attitude deadband. However, at small beta angles impulse increases with increasing deadband. Thus, smaller deadbands for low beta angles and larger deadbands at high beta angles are optimum. Propellant consumption may be further reduced for small beta angles by reducing the pitch and yaw deadbands beyond the 5 degrees shown in the figure. Simulation results not included in this report indicated that for Case 1A rate deadbands of 0.01 degree per second in each axis are better than larger rate deadbands. Based on figure 4-22 the end-docked configuration requires considerably less propellant than the side-docked configuration for Case 1A. The primary reason for the additional impulse is evidenced in figure 4-23 which contains the yaw-roll and pitch impulse individually as a function of beta. This figure indicates that the side-docked configuration requires considerably more impulse in yaw-roll than the end-docked configuration while requiring approximately the same for pitch control. This indicates that the longer control lever arms for the end-

WORKSHOP ATTITUDE CONTROL SYSTEM

TOTAL IMPULSE CASE -1A

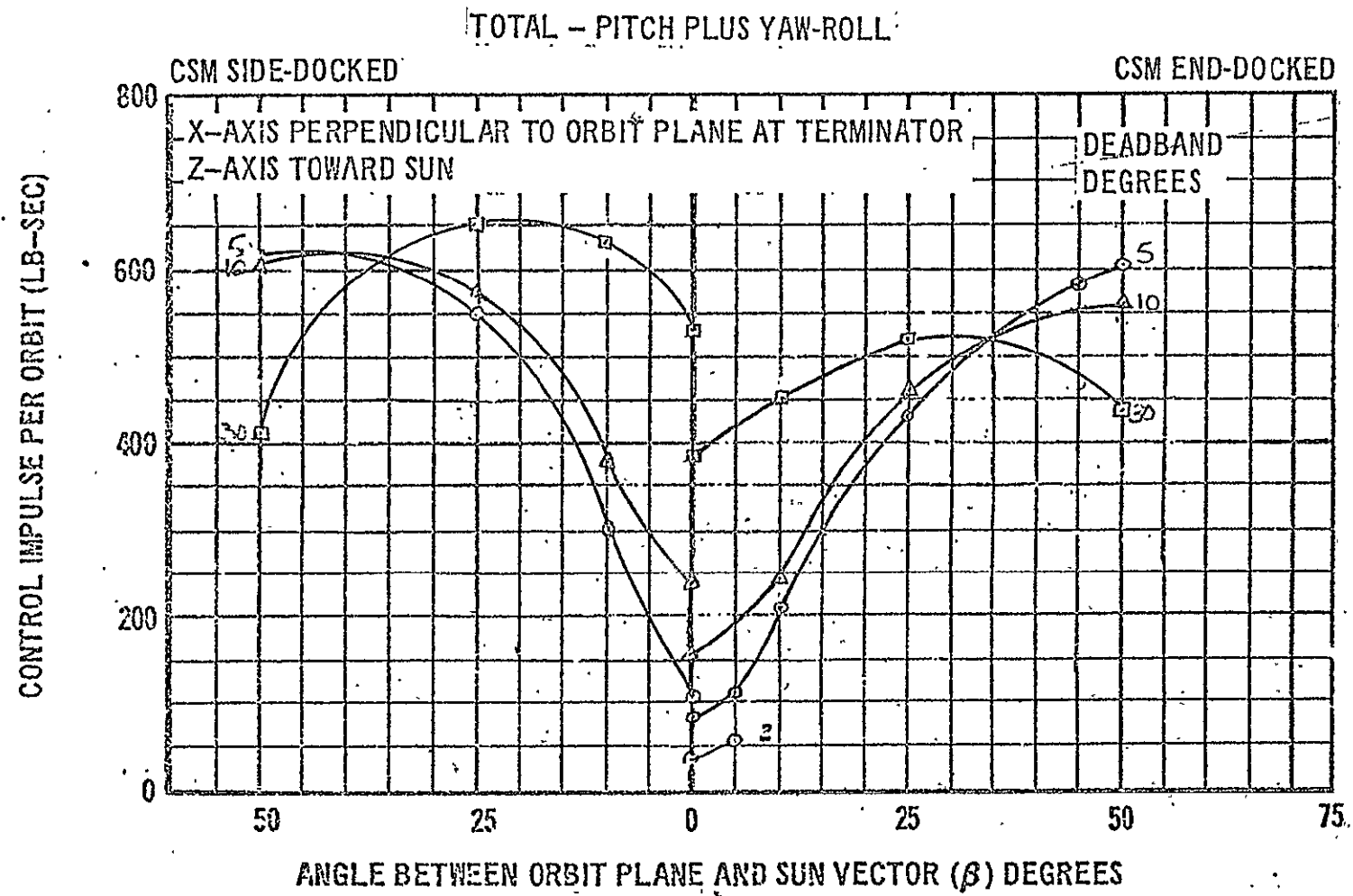


FIGURE 4-22 Total Impulse - Case 1A

WORKSHOP ATTITUDE CONTROL SYSTEM IMPULSE BY AXIS - CASE 1A

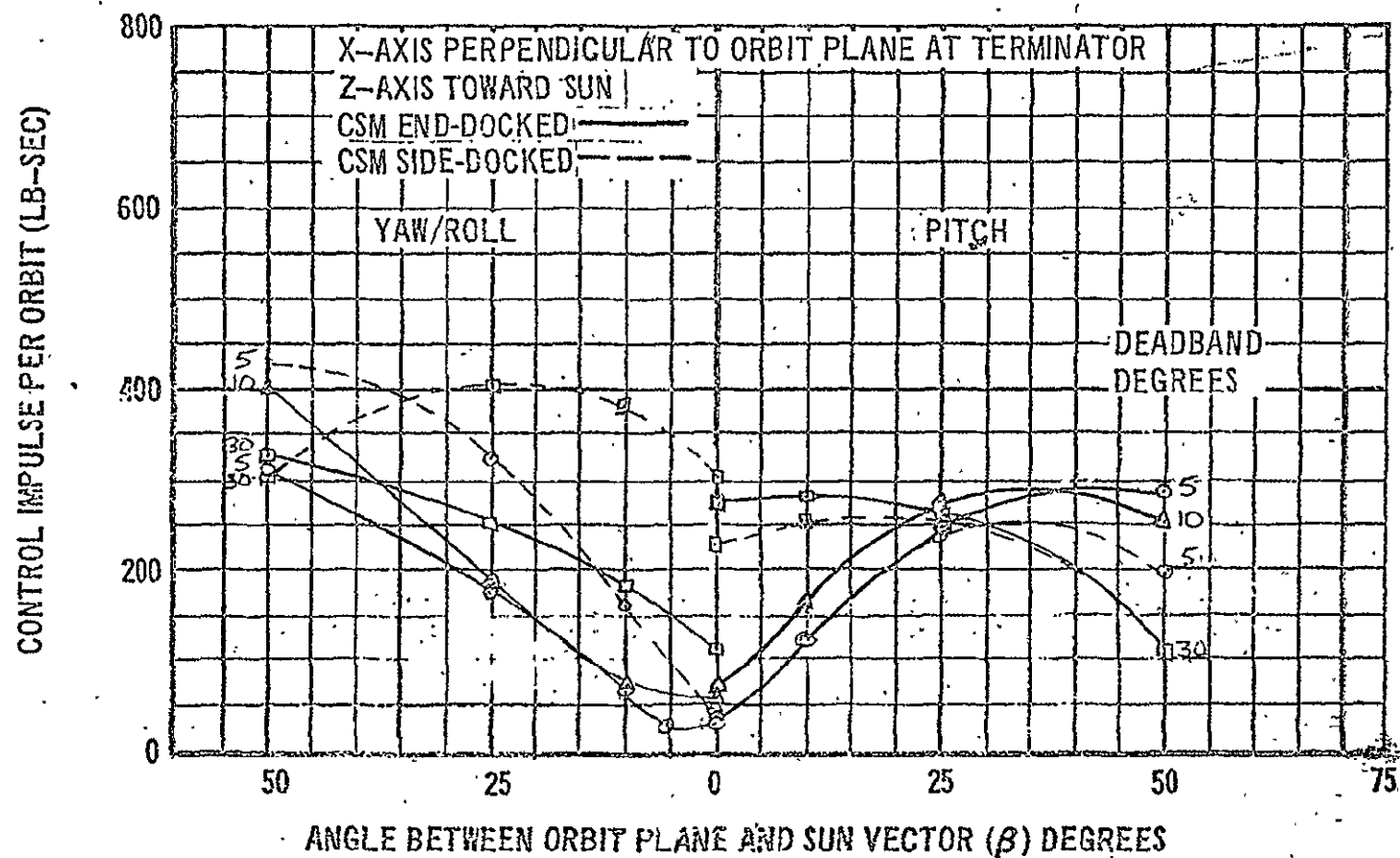


FIGURE 4-23 Impulse by Axis - Case 1A

docked configuration in pitch and yaw and the increased roll gravity gradient torques are significant as discussed in section 4.2.2.1. Results of theoretical impulse computations for the end-docked configurations for Case 1 and 1A are shown in figure 4-24. These computations were made for an uncoupled system, using separate engines for each axis, and included only the gravity-gradient torques. Comparing the Case 1 curve of figure 4-24 with figure 4-20, indicates that the simulation curve for the five-degree deadband closely matches the theoretical curve.

4.2.2.3 Case 2 Results

The OWS is uncontrolled in yaw attitude for Case 2. This enables the vehicle to rotate about the z-axis which is controlled, as in the initial two cases, to be parallel to the sun vector. The results from the simulation indicate that there is no advantage in this method of control over the previous cases except for the end-docked configuration for beta angles greater than 25 degrees. On the other hand, as evidenced by comparing figure 4-25 with figure 4-22, the Case 1A mechanization requires considerably less propellant than the Case 2 mechanization to control the end-docked configuration for low beta angles. A similar observation for low beta angles can be made for the side-docked configuration. The impulse requirements for Case 2 are nearly identical to those of Case 1 for the side-docked configuration (compare figure 4-25 with figure 4-20). Thus, the lower Case 1 impulse requirements at high beta angles for the side-docked configuration are also achieved by the Case 2 mechanization. Figure 4-26 contains the impulse by axis as a function of beta angles of Case 2. The initial conditions for all simulation runs for Case 2 were zero for all body attitudes and rates.

4.2.2.4 Case 3 Results

In Case 3, the vehicle is rotated about the yaw-axis (z-axis) while controlling the z-axis to be parallel to the sun vector. Both end-docked and side-docked configurations were evaluated over a range of values for the WACS parameters to determine their effect on system impulse usage. Yaw spin rate and attitude and rate deadbands were each varied independently to determine the optimum values.

WORKSHOP ATTITUDE CONTROL SYSTEM GRAVITY GRADIENT CONTROL IMPULSE THEORETICAL UNCOUPLED

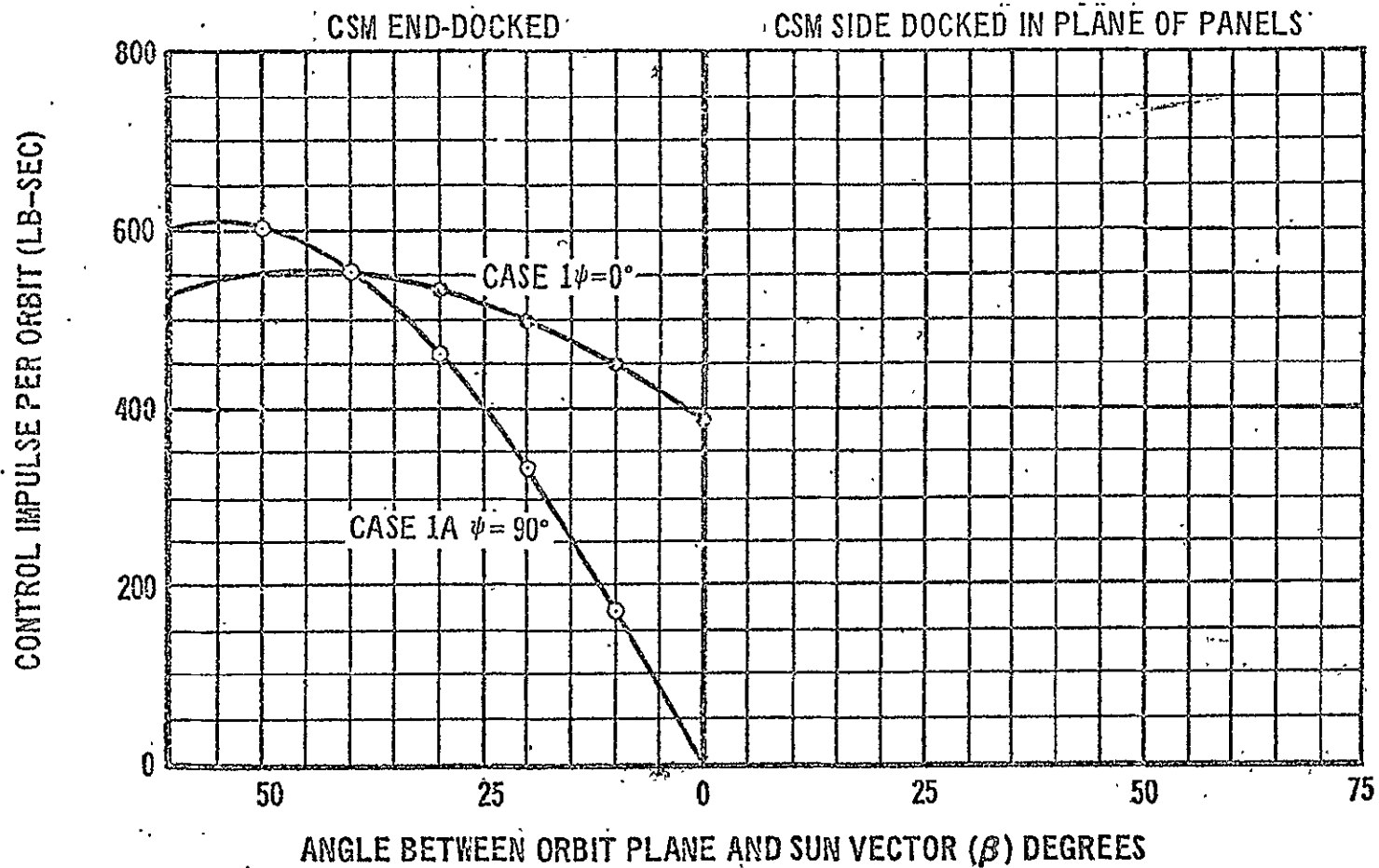


FIGURE 4-24 Theoretical Uncoupled Gravity-Gradient Control Impulse

WORKSHOP ATTITUDE CONTROL SYSTEM IMPULSE BY AXIS - CASE 2

WACS-20
8-22-67

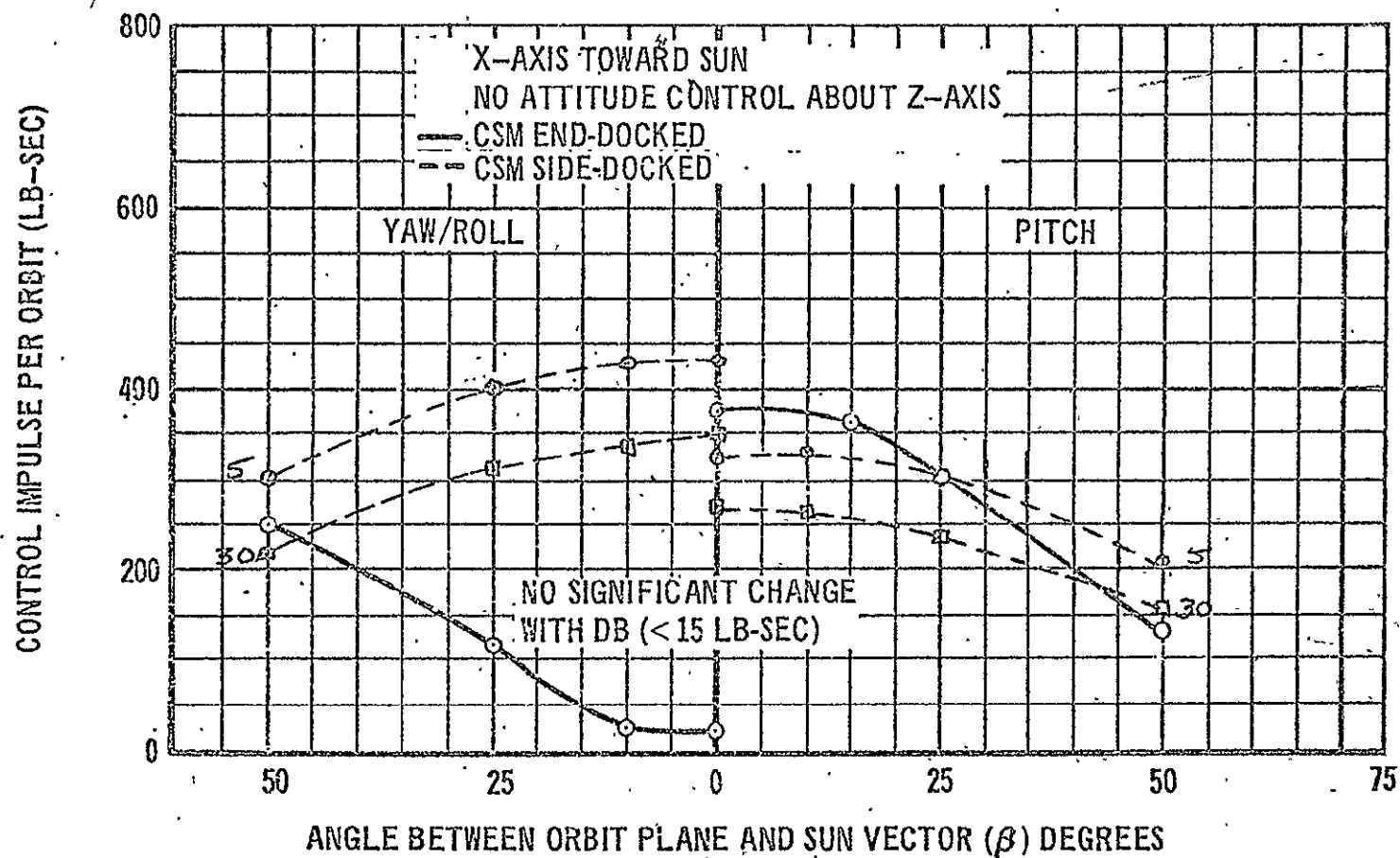


FIGURE 4-25 Impulse by Axis - Case 2

WORKSHOP ATTITUDE CONTROL SYSTEM TOTAL IMPULSE - CASE 2

WACS-21
8-21-67

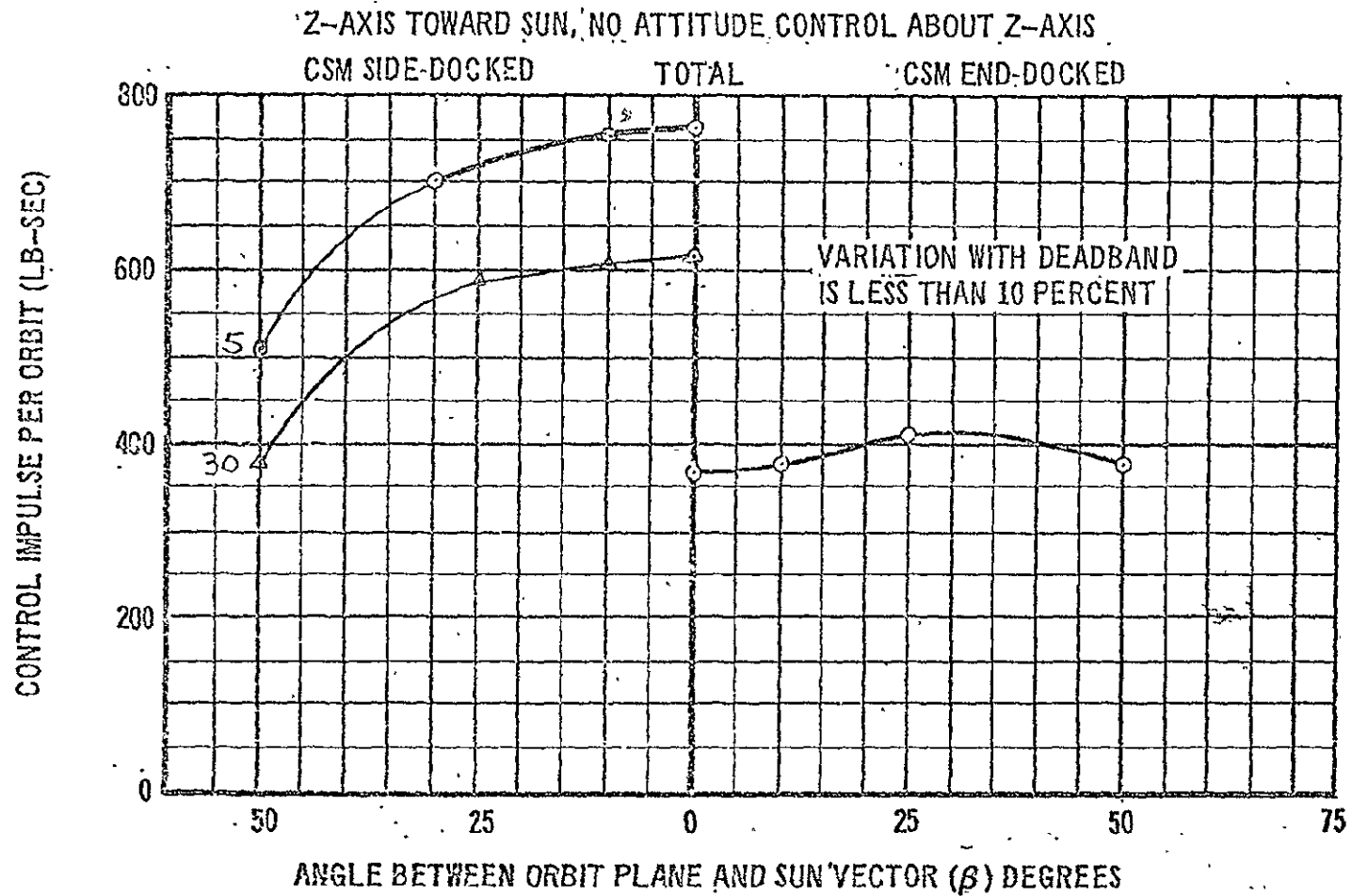


FIGURE 4-26 Total Impulse - Case 2

For the end-docked configuration, the yaw spin rate was found to be most critical. The z-axis for the end-docked configuration was the axis of mid-valued moment of inertia, unlike the side-docked configuration for which the z-axis was the axis of maximum moment of inertia, making the end-docked configuration unstable in pitch for Case 3. The effect of this instability on impulse as a function of yaw spin rate is seen in figure 4-27. A 10-degree attitude deadband in pitch and roll and a 0.01-degrees-per-second rate deadband in each axis with a beta of 50 degrees were the parameters used in the simulation to obtain the data for figure 4-27. Impulse requirements increased sharply for the end-docked configuration when the yaw-rate exceeded 0.7 degree per second. This increase is attributed to the gyroscopic torques which increase as the spin rate increases because of the instability. The instability occurs only in the pitch plane as indicated in figure 4-28 which shows that the pitch impulse increases for increasing yaw spin rate while the yaw-roll impulse remains relatively constant for the end-docked configuration. Revised mass data received from MSFC after completing the simulation indicates that the moment of inertia about the z-axis is expected to be the maximum inertia. Thus, it is anticipated that the impulse will decrease as the yaw spin rate increases as it did for the side-docked configuration.

Figure 4-29 contains the results obtained from the simulation by varying the pitch and roll attitude deadbands for both configurations. The figure contains the relationship between total impulse per orbit and the angle beta. The figure indicates that the optimum attitude deadband for the end-docked configuration is a function of beta, while the maximum deadband is optimum for the side-docked configuration.

Simulation results indicated that the most critical parameter for Case 3 was the rate deadband. Figures 4-30 and 4-31 show the relationship between impulse per orbit and rate deadband for the side-docked and end-docked configurations, respectively.

Because of the stable nature of the side-docked configuration, the propellant usage decreases as the rate limit for each axis increases. The

WORKSHOP ATTITUDE CONTROL SYSTEM YAW RATE EFFECTS ON IMPULSE - CASE 3

WACS-23
8-22-67

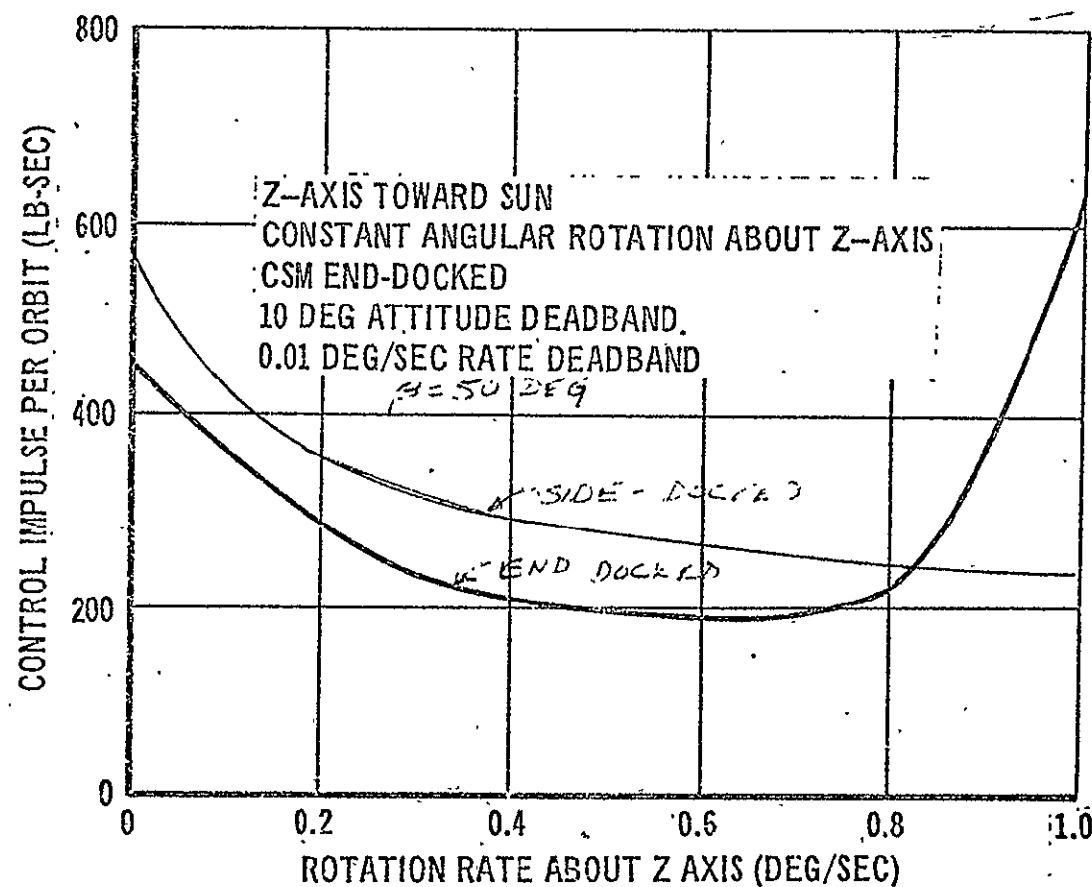


FIGURE 4-27 Yaw Spin Rate Effects on Impulse - Case 3

WORKSHOP ATTITUDE CONTROL SYSTEM YAW RATE EFFECT ON IMPULSE BY AXIS

WACS-22
8-22-67

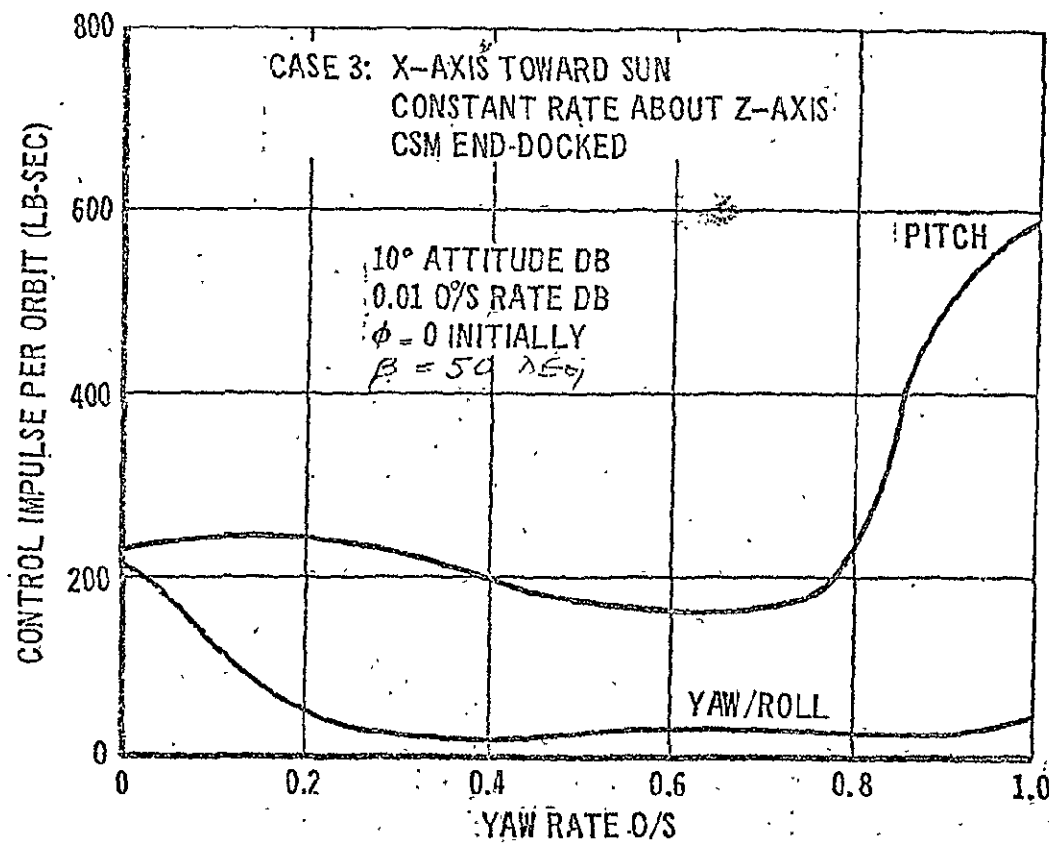


FIGURE 4-28 Yaw Spin Rate Effects on Impulse - Case 3

WORKSHOP ATTITUDE CONTROL SYSTEM TOTAL IMPULSE - CASE 3

WACS-25
8-22-67

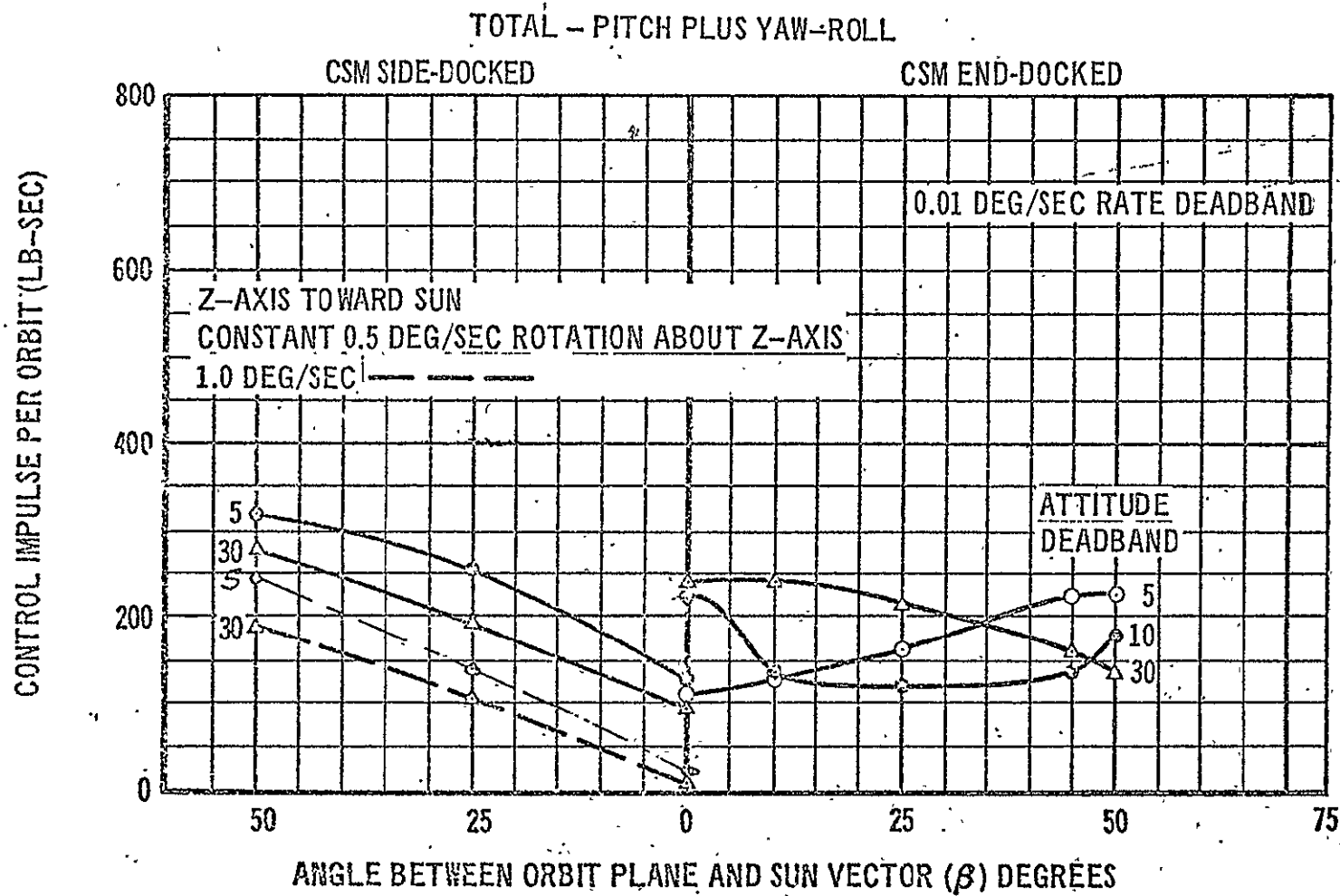


FIGURE 4-29 Total Impulse - Case 3

SPIN RATE = 1.0 DEG/SEC
 PITCH, YAW DEADBAND = ± 10 DEG

- - SPIN AXIS ALIGNED WITH PRINCIPAL AXIS
- △ - SPIN AXES 2 DEG OFF PRINCIPAL AXIS

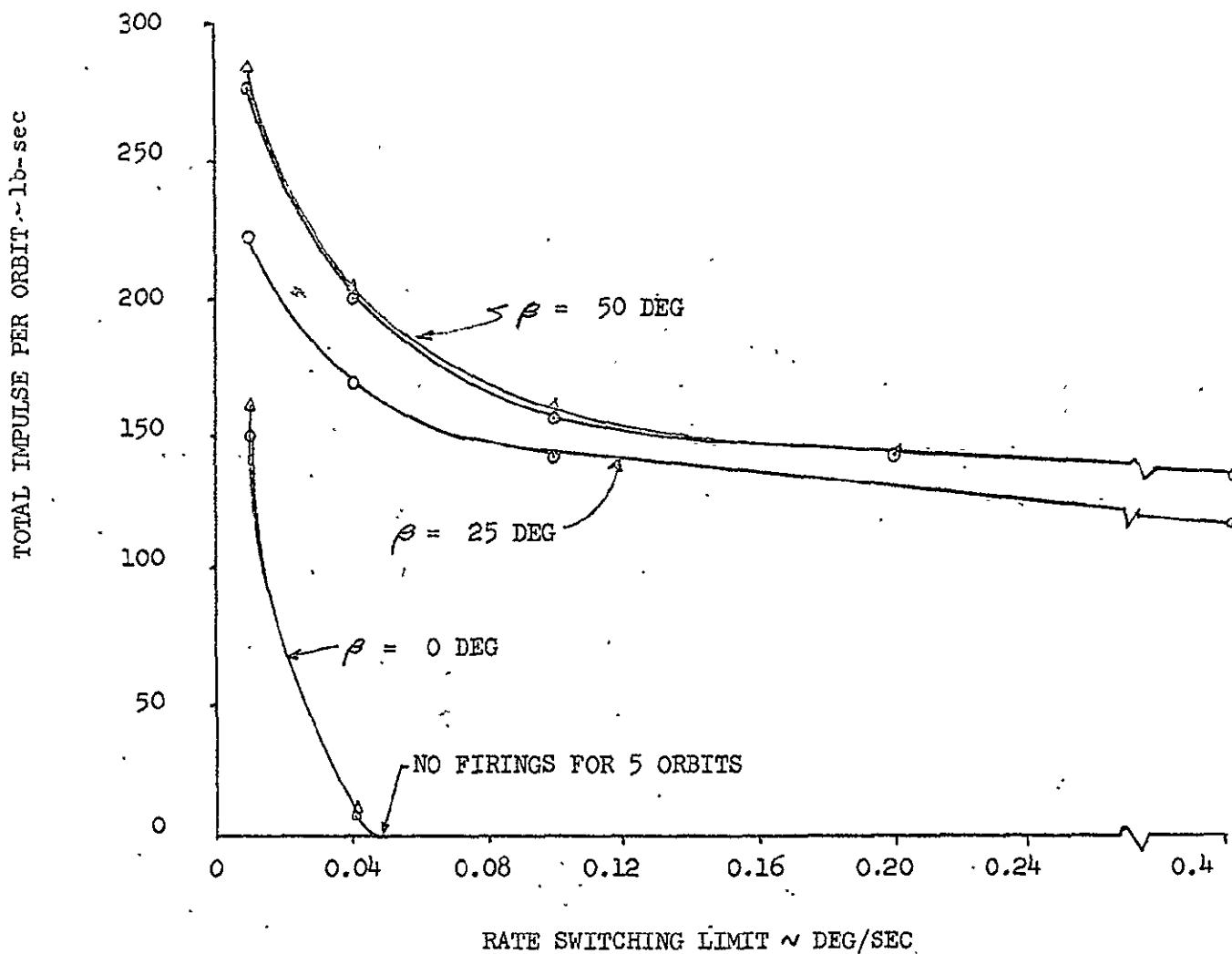


FIGURE 4-30 IMPULSE VERSUS RATE DEADBAND FOR CASE 3
 WITH SPIN AXES MISALIGNED FROM PRINCIPAL AXES

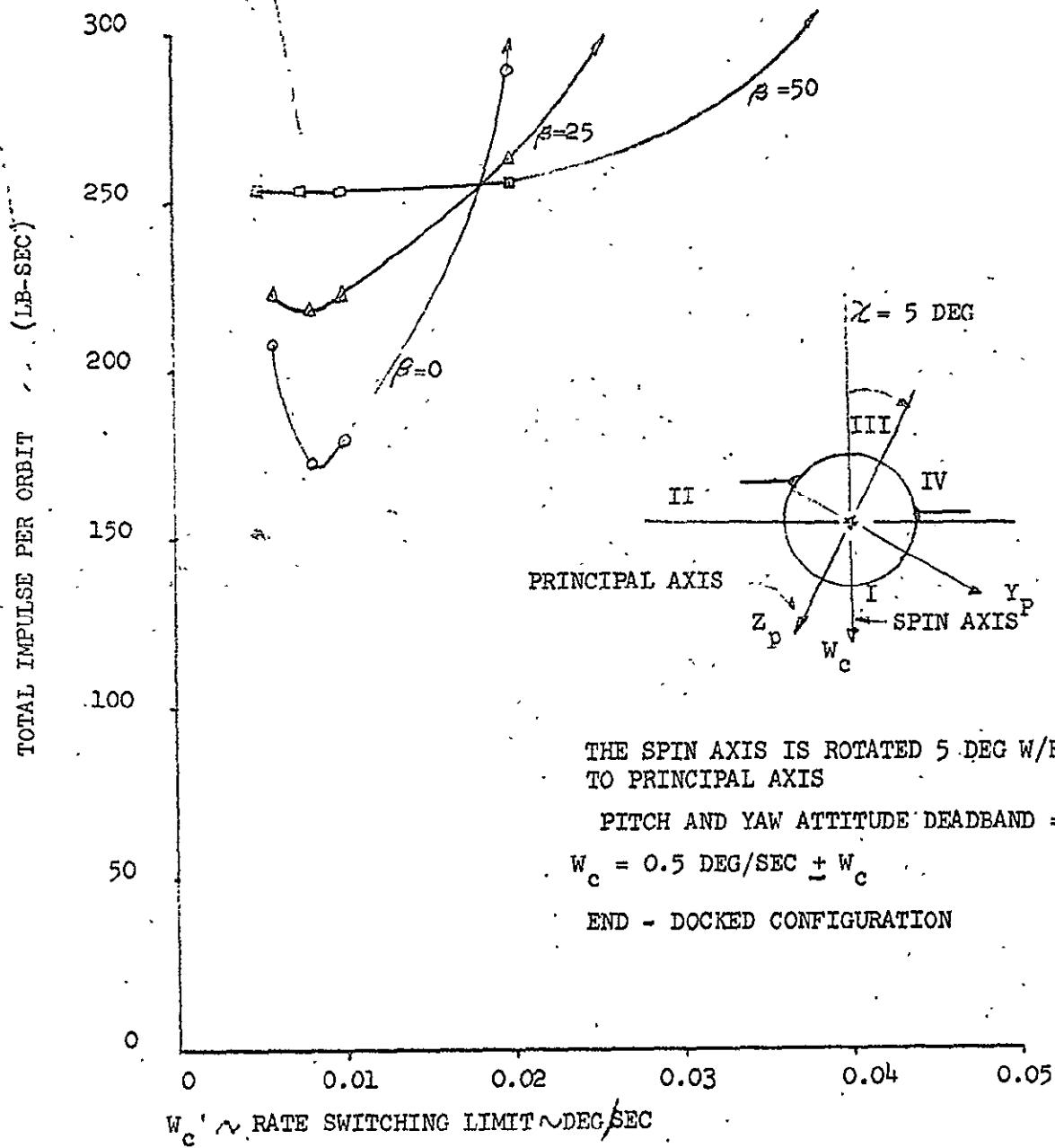
impulse usage increases for rate deadbands greater than 1 degree per second because the rate gain is too low to provide adequate system damping. The system is spin stabilized such that there are no engine firings when beta is zero and the rate deadbands are greater than 0.06 degree per second.

The optimum rate deadband in all three axes for the end-docked configuration is approximately 0.008 degree per second for the deadbands evaluated as shown in figure 4-31, which also indicates that the impulse requirements per orbit increase as beta increases for optimum rate deadbands. The WACS parameters that were used in the simulation model were a 10-degree attitude deadband in pitch and roll and a yaw spin rate of 0.5 degree per second. The impulse per orbit increases as the rate deadband increases beyond the optimum point. This is attributed to the instability in the pitch axis previously discussed and is further compounded by principal axes being rotated five degrees from the spin axis.

The effect of the principal axes being offset from the spin axis for two betas of zero and 25 degrees is contained in figure 4-32. The effect is shown to be relatively small for small offset angles.

The impulse usage doubles for a five-degree rotation while it is 14 times greater for a 15-degree rotation. The figure illustrates that small rotations of two degrees or less, which is anticipated for the side-docked configuration, results in a small increase in propellant consumption. Although these results are not directly applicable to the side-docked configuration because of the different inertia properties, the low sensitivity to offsets of the principal axes from the spin axis is expected to apply.

The effect of rotating the engine modules 20.5 degrees from the spin and principal axes is compared to the non-rotated case in figure 4-33. Both systems were simulated for the entire range of beta (-50 to 50) using both positive and negative yaw spin rates. The variation in impulse requirements caused by changing the yaw spin rate from positive to negative results from the unstable pitch gyroscopic torques which vary as a function of the yaw spin direction and beta angle. It is noted from the figure that the maximum impulse occurs at a beta of 30 degrees instead of 50.



TOTAL IMPULSE VERSUS RATE SWITCHING LIMITS FOR GIVEN β ANGLES.

FIGURE 4 - 31

PITCH & YAW ATTITUDE DEADBAND = ± 5 DEG
 PITCH & YAW RATE DEADBAND = ± 0.01 DEG/SEC

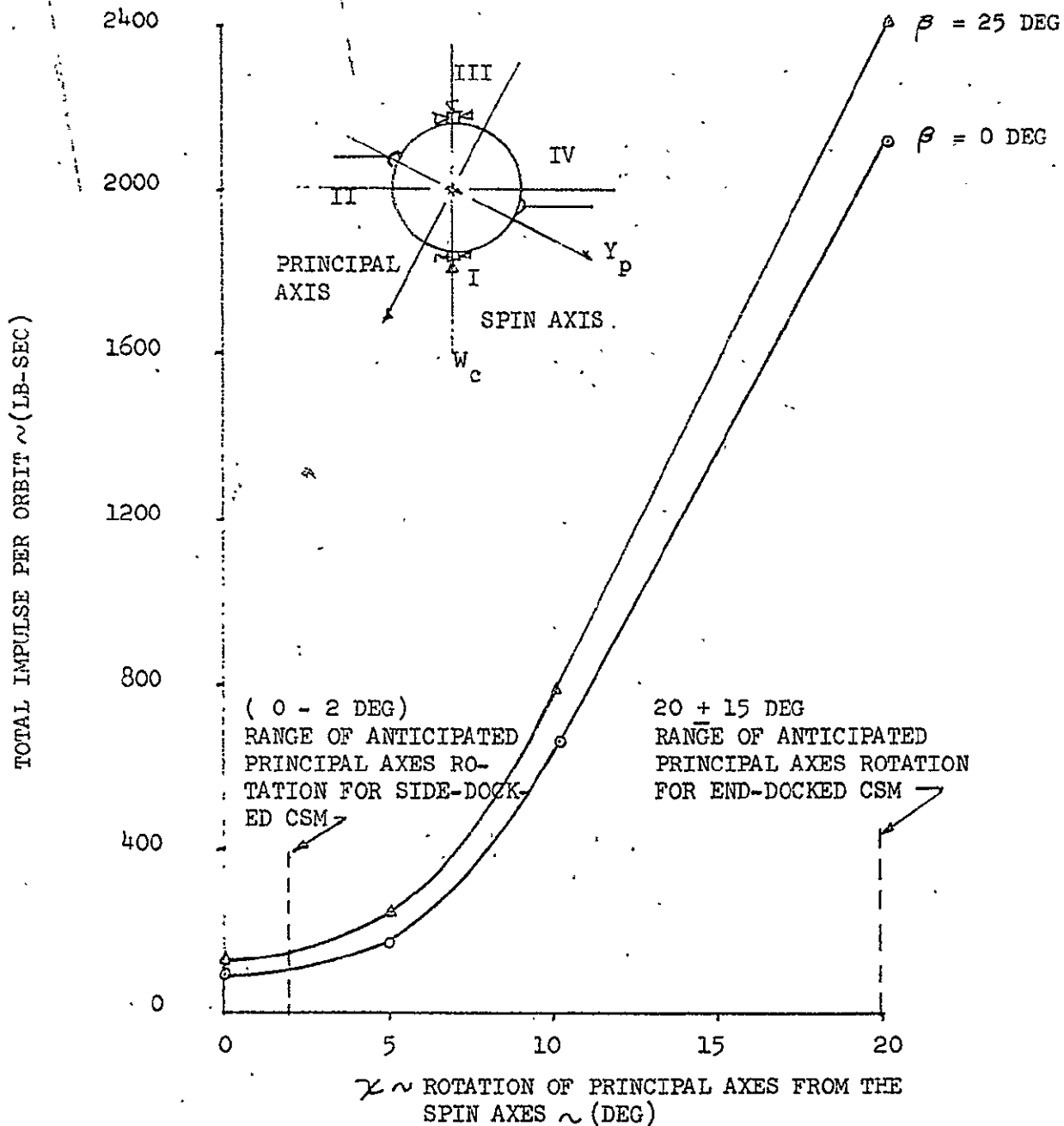
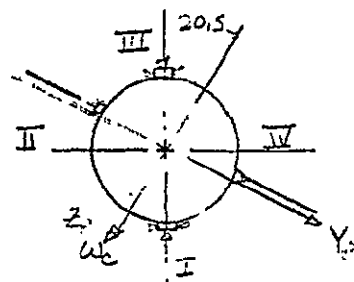


FIGURE 4-32 IMPULSE VERSUS ANGLE OF ROTATION OF PRINCIPAL AXES FROM SPIN AXES FOR END-DOCKED CONFIGURATION

Effects of Rotating Engines off the Principal Axes
For A Workshop Vehicle



Pitch and Yaw Attitude Deadband = ± 5 deg

Pitch, Yaw & Roll Rate Deadband = ± 0.01 deg/sec

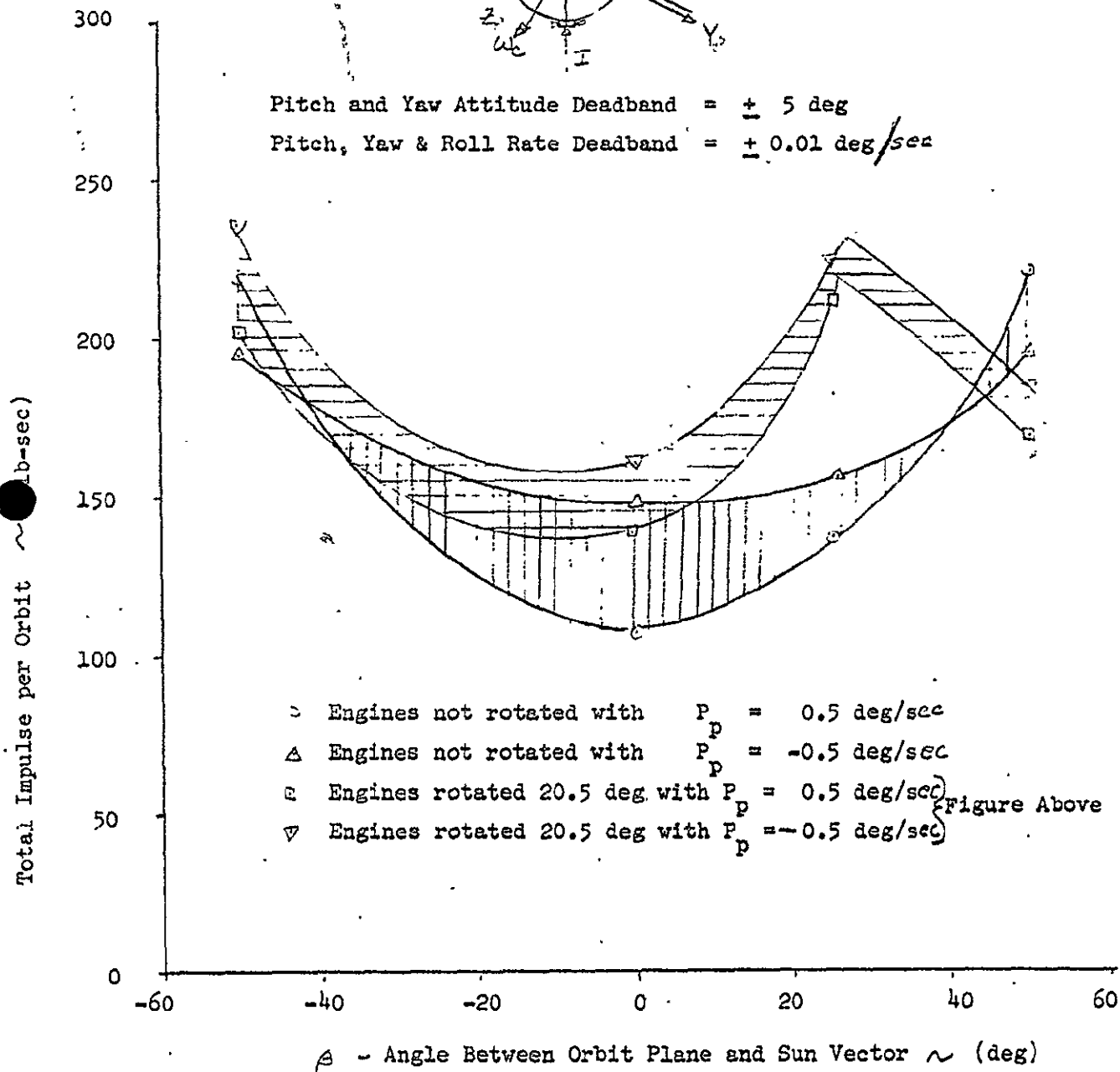


Figure 4-33. Impulse versus Beta for Workshop Vehicle Whose Engines are Rotated off the Principal Axes

degrees due to the 20.5-degree engine shift from the principal axes. It is estimated that the shift results in approximately a 10 percent increase in impulse for this case.

4.2.2.5 Case 4 Results

Case 4 utilizes the gravity-gradient stable orientation for which the x-axis is controlled to lie along the local vertical and the vehicle is rotated about the x-axis such that an optimum solar panel orientation is achieved. Unlike the previous cases, where the impulse requirements increased as beta increases, the impulse for this case is minimum when beta is 90 degrees because with a fixed roll attitude the panels are positioned optimally for the entire orbit, thereby eliminating the roll commands to the vehicle. The impulse usage goes up as beta decreases because roll rate commands increase in order to orient the panels to the sun. As discussed in section 3.4.7, the optimum roll attitude history varies considerably as a function of beta and for a low values of beta several alternate roll control schemes can be used to minimize impulse without significantly affecting solar panel power. Figure 4-34 shows the impulse per orbit versus beta for both end and side-docked configurations for several roll attitude control mechanizations. The mechanizations for which data are shown are Cases 4, 4A, 4B and 4C as described in the following paragraphs. (See also section 5)

The impulse per orbit for Case 4A, which uses a fixed roll attitude, is independent of beta, and requires only a small amount of impulse (5 pound-seconds per orbit) to control the vehicle in its gravity-gradient stable position. This impulse is estimated from figure 4-8, assuming pitch and yaw impulse requirements are equal.

Impulse data by axis obtained from the simulation for Case 4C, which uses a constant roll rate equal to orbital rate and 20-degree pitch and yaw attitude deadbands, is also contained in figure 4-34. Although not shown in figure 4-34, simulation results indicated that the impulse requirements increased by a factor of 10 when the attitude deadbands in pitch and yaw were reduced to ± 5 degrees. The impulse in pitch and yaw-roll are approxi-

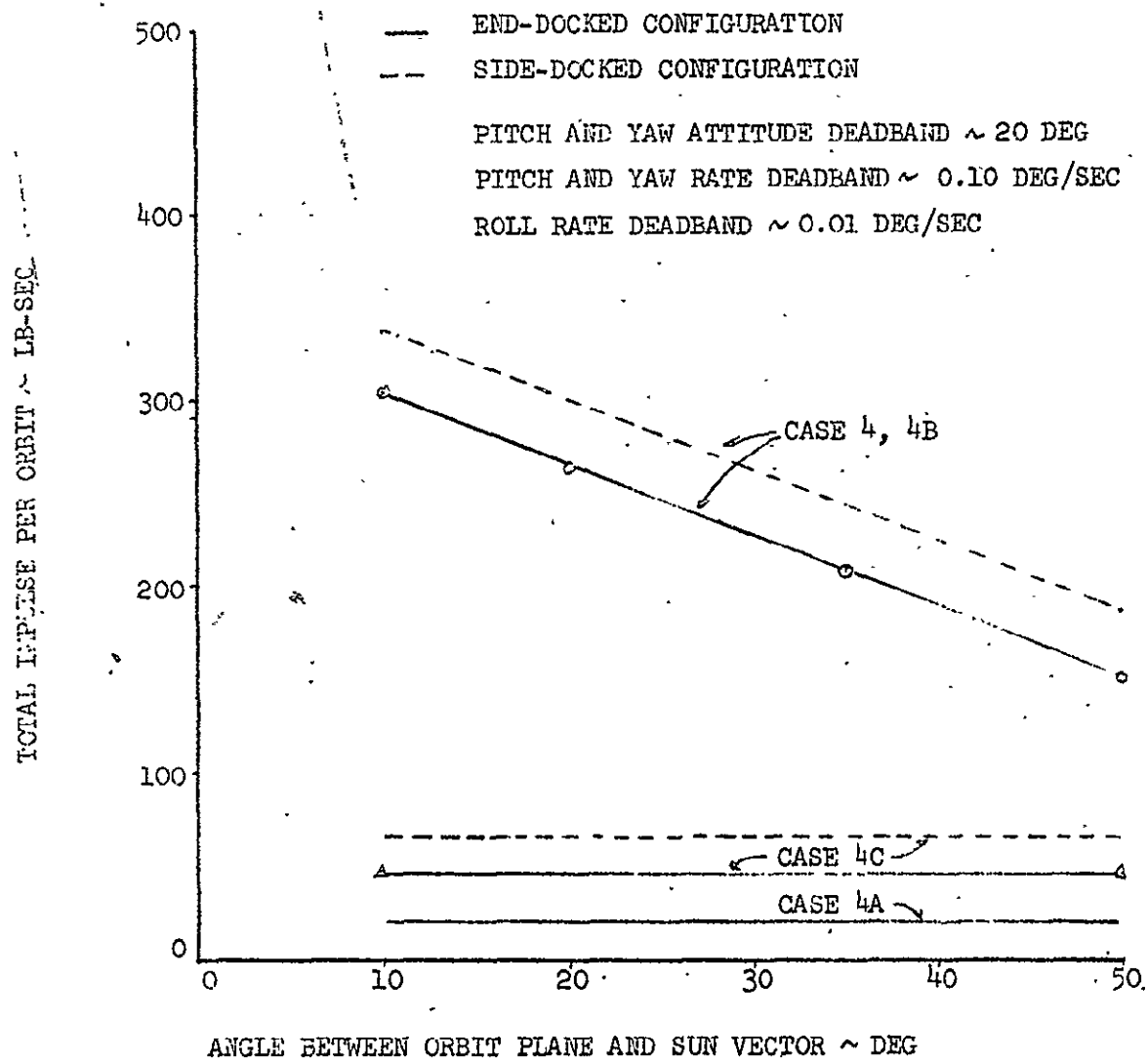


FIGURE 4-34 TOTAL IMPULSE PER ORBIT VERSUS
 BETA ANGLE FOR CASE 4

mately equal for Case 4C as indicated in figure 4-35.

Also shown in figure 4-35 is the impulse distribution between pitch and yaw-roll engines for the end-docked configuration for Cases 4, and 4B. The rate deadband in pitch and yaw used in the simulation was 0.1 degree per second while it was 0.01 degree per second in roll. The larger rate deadbands in pitch and yaw were used because simulation results indicated that the smaller deadbands increased the propellant consumption. The required impulse per orbit for Cases 4 and 4B resulted in more propellant usage than Case 5 while requiring less than Case 1. However, of significance is the reduced amount of power obtained from the panels because of the nonoptimum panel orientation with respect to the sun unless panels are articulated as discussed in section 5.

4.2.2.6 Case 5 Results

Case 5 is similar to Case 1A for a beta of zero in that the x-axis is controlled to be perpendicular to the local vertical at the terminator and thus perpendicular to the orbit plane when beta is zero. The variation between the two cases is that for Case 1A the z-axis of the vehicle is controlled to be parallel to the sun vector, while for Case 5 the z-axis is biased by beta such that it remains in the orbit plane and the x-axis remains perpendicular to the orbit plane for all values of beta. The panels are rotated toward the sun through the angle beta by a separate control system for Case 5.

The impulse data for Case 1A, which is shown in figure 4-22, can be used to determine the impulse per orbit for Case 5 with beta representing the offset of the pitch axis null from the perpendicular to the orbit plane. The figure indicates that impulse is minimized by reducing the attitude deadbands. A curve showing the required impulse for a two degree pitch and yaw deadband for beta angles in the range of zero to five degrees is also shown in figure 4-22. This curve indicates that pitch and yaw deadbands smaller than ± 5 degrees will reduce the impulse requirements for Case 5. This impulse savings results because the pitch and yaw gravity-gradient torques are reduced. Larger roll attitude deadbands are expected to be

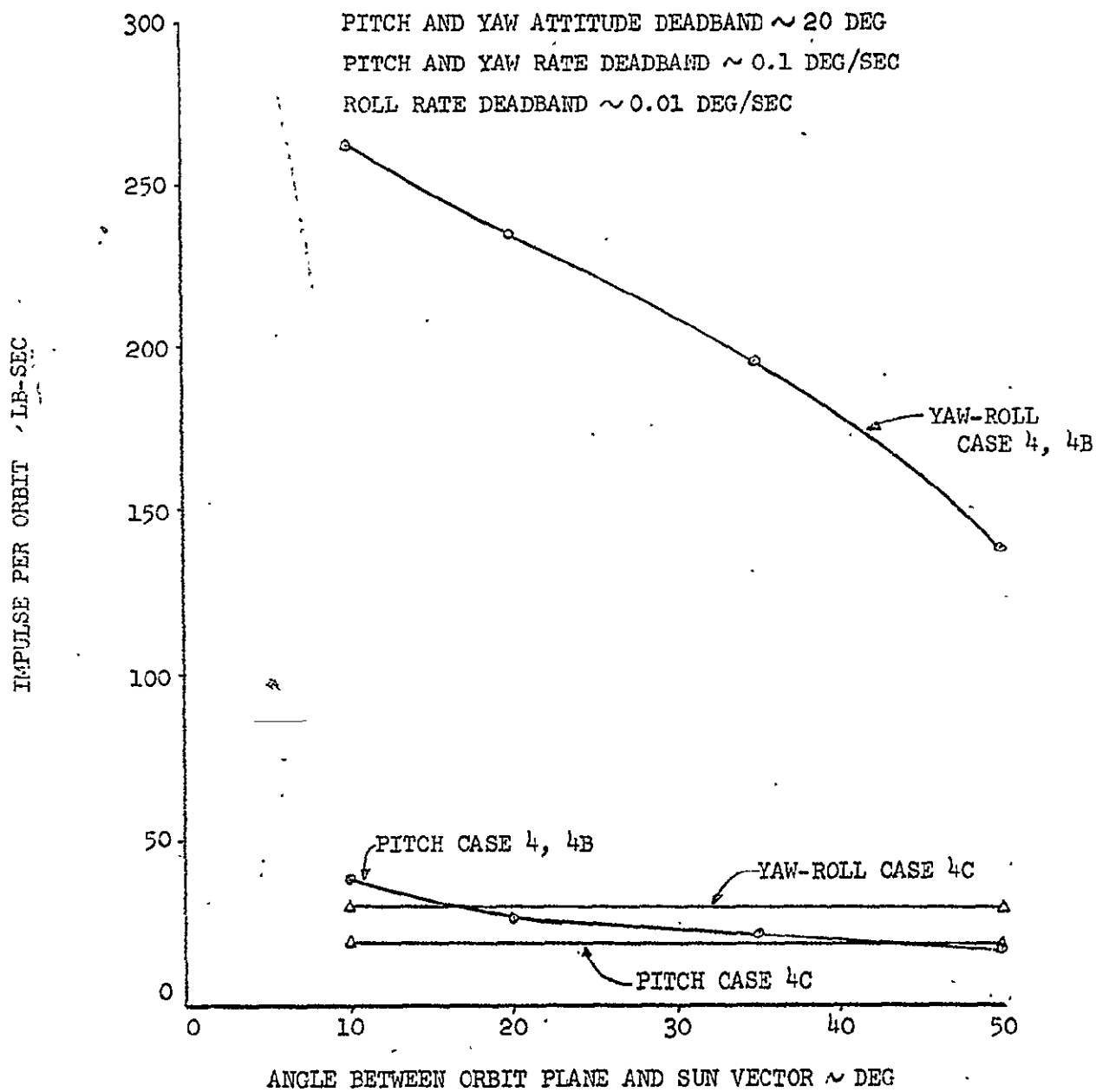


FIGURE 4-35 PITCH AND YAW-ROLL IMPULSE PER ORBIT VERSUS
 BETA FOR END-DOCKED CONFIGURATION

optimum for Case 5, because of the cyclic nature of the roll gravity-gradient torques.

The destabilizing nature of the pitch and yaw gravity-gradient torques cause a one-sided limit cycle in pitch and yaw; thus, the gravity-gradient torques, and the impulse usage, are functions of the sum of the deadband and the null offset caused by various tolerances. Impulse can thus be minimized by a combination of small pitch and yaw deadbands, and the use of accurate sensors and attitude commands in pitch and yaw. The use of integral feedback to minimize these null offsets and further reduce impulse usage should also be considered.

4.2.3 Impulse Requirements due to Aerodynamic Torques

The effects of aerodynamic torques were not included in the simulation because data for the latest solar panel configuration was not available at the time the study was initiated, and the effect on impulse usage was expected to be small at the 260 nautical mile altitude. This conclusion was based on the aerodynamic data available from reference 5 which indicated that aerodynamic torques were less than approximately 20 percent of maximum gravity-gradient torques. The effects on impulse usage are further reduced because aerodynamic torques subtract from the gravity-gradient torques for part of each orbit and add for the remainder of the orbit.

4.2.4 Summary of Impulse Requirements for the 28 Day Mission

The impulse requirements for each of the orientations studied are listed in table 4-6. The attitude deadbands that yielded the minimum impulse values for each configuration are also given in the table, and the optimum configuration for each orientation is indicated by a heavy outline. A rate deadband of 0.01 degree per second was used for all cases except the side-docked configuration of Case 3. For this case a rate deadband of 0.2 degree per second, based on the results presented in figure 4-30, was used to obtain the impulse requirements. The impulse estimates for each case were derived from the simulation results by adding a 20 percent factor to the recorded simulation results to account for aerodynamic torques, and to allow for simulation variations. This assumes that the impulse required

Case No.	END-DOCKED			SIDE-DOCKED		
	Optimum End-Docked	Maximum Impulse 28 Days (lb-sec)		Optimum End-Docked	Maximum Impulse 28 Days (lb-sec)	
		$\beta < 25^\circ$ (13° Avg)	$\beta > 25^\circ$ (35° Avg)		$\beta < 25^\circ$ (13° Avg)	$\beta > 25^\circ$ (35° Avg)
1	Attitude Deadband ~30 deg.	170,000	225,000	Attitude Deadband ~30 deg.	300,000	255,000
1A	5 deg for $\beta < 25^\circ$ 30 deg for $\beta > 25^\circ$	135,000	258,000	5 deg for $\beta < 25^\circ$ 30 deg for $\beta > 25^\circ$	205,000	300,000
2	Attitude Deadband ~30 deg	195,000	205,000	All Attitude Deadband are equal	308,000	255,000
3	Attitude Deadband 10 deg	65,000	60,000	30 deg Attitude 0.2 deg/sec Rate Deadband	30,000	82,000
5	5 deg Deadband	55,000	55,000	5 deg Deadband	103,000	103,000
4, 4B	20 deg Deadband	140,000	110,000	20 deg Deadband	155,000	130,000
4A	20 deg Deadband	6,000	6,000	20 deg Deadband	6,500	6,500
4C	20 deg Deadband	25,600	25,600	20 deg Deadband	30,000	30,000

For total mission impulse, add 22,500 pound-seconds for rendezvous, docking, etc., as shown in table 2-1, and 13,500 pound-seconds for boost and passivation as shown in table 4-2.

TABLE 4-6 Workshop Attitude Control System

Data Summary for 28 Day Mission

for other disturbances is negligible. The summary table indicates that Cases 3 and 5 and certain variations of Case 4 require considerably less propellant than the other cases.

4.2.5 Effect of Impulse per Pulse and Thrust

The minimum engine impulse bit used for control during the 28 day mission is expected to be a significant factor in minimizing the propellant consumption. A minimum impulse bit of 0.6 pound-second was used in the simulation to obtain the impulse data given in the previous section. This impulse-bit was sized to minimize the impulse usage during gravity-gradient control. By increasing the bit size the specific impulse of the engine increases, thereby decreasing the impulse required for disturbances. This is true for all cases for which the total disturbance torques cause operation on one side of the deadband rather than two-sided limit cycle operation. This situation exists for most of the mission for all orientations, except pitch and yaw axes of the gravity-gradient cases. For control about an orientation for which gravity-gradient torques have an unstable null, as is the case for the pitch axis of Case 5, larger minimum impulse bits can decrease impulse usage by driving the axis further toward the null between pulses. This decreases the average torque and thus the impulse usage. Thus, further study is required to optimize the pulse width for each mode of operation.

5.0 SOLAR PANEL POWER

The average value of continuous electrical power at the buses was estimated for all of the attitude control orientations studied based on the solar array configuration recommended by Douglas Aircraft Company in reference 6. The curves of power versus beta angle have been computed for 1,200 square feet of active surface. Active surface is defined for this study to be the total surface of the panels which is used for mounting solar cells; it thus excludes surface lost between the panels that is used for deployment mechanisms and structure and other auxiliaries, but does include space on the panels between the solar cells.

The average power for each of the different attitude control configurations, which are described in section 3, was computed by integrating the incident solar flux on the panels during a complete orbit for a particular beta angle (including shadowing effects). A value of 8.35 watts of electrical power per square foot of active surface at 25°C, was used to arrive at the value of electrical energy supplied to the Airlock power conditioning equipment. The mathematical model of the Airlock batteries and power conditioning equipment for the 260 nautical mile circular orbit was supplied by McDonnell Astronautics Company; it is

$$P_B = \frac{1.597 E_O}{4.47 - T_L}$$

Where P_B is average bus power in watts for a complete orbit,

E_O is input electrical energy in watt-hours during one orbit, and

T_L is the duration of the illuminated portion of the orbit in hours.

In order to make valid comparisons between power curves for different attitude control cases, the panel temperature estimates must be compatible. This was done by assuming the average temperature during the illuminated portion of the orbit to be a linear function of total integrated solar flux during the orbit. Temperature is estimated to vary between 70°C (158°F) for normal incidence at a beta of 52 degrees and 25°C (77°F) for

Case 4 at a beta of zero degree. Further thermal analysis is desirable to predict instantaneous temperatures during an orbit and include this factor in the integration of solar cell output power.

The average temperature of the panels for the orbital storage case was assumed for simplicity to be constant at 25°C for the full range of beta angles - no attempt was made to refine this further. Actually, because of the lower value of integrated solar flux on the panels during storage than for any of the active cases, the panels are expected to be cooler than 25°C and thus supply somewhat more average power than is indicated.

5.1 Normal Solar Incidence Orientations (Cases 1, 1A, 2, 3, 3A, 4B and 5)

The attitude control cases which provide normal solar incidence to the panels include all the solar-inertial orientations, plus the gravity-gradient case in which both vehicle roll and panel angle relative to the vehicle centerline (alpha) are controlled to point the panels directly toward the sun (Case 4B). These orientations provide the maximum possible power under any conditions. The power as a function of beta angle and deadband is shown in figure 5-1. The rise in power as beta increases results from the solar panels being exposed to the sun for a greater fraction of the orbit period.

5.2 Gravity-Gradient with Roll Control (Case 4)

In this orientation mode, the panels are held parallel to the longitudinal axis of the workshop (alpha = 0 degree). This orientation becomes more favorable as beta increases. Figure 5-2 shows the power as a function of beta and deadband. The constant-roll-rate maneuver which is used for small beta angles when the sun is near the zenith, has a negligible effect on solar panel power.

5.3 Gravity-Gradient with Variable Array Angle and Constant Roll Attitude (Case 4A)

In this orientation, the roll attitude is constant with the array rotation (alpha) axis perpendicular to the orbit plane. At a beta angle of zero degrees, the panel can be pointed normal to the sun by controlling

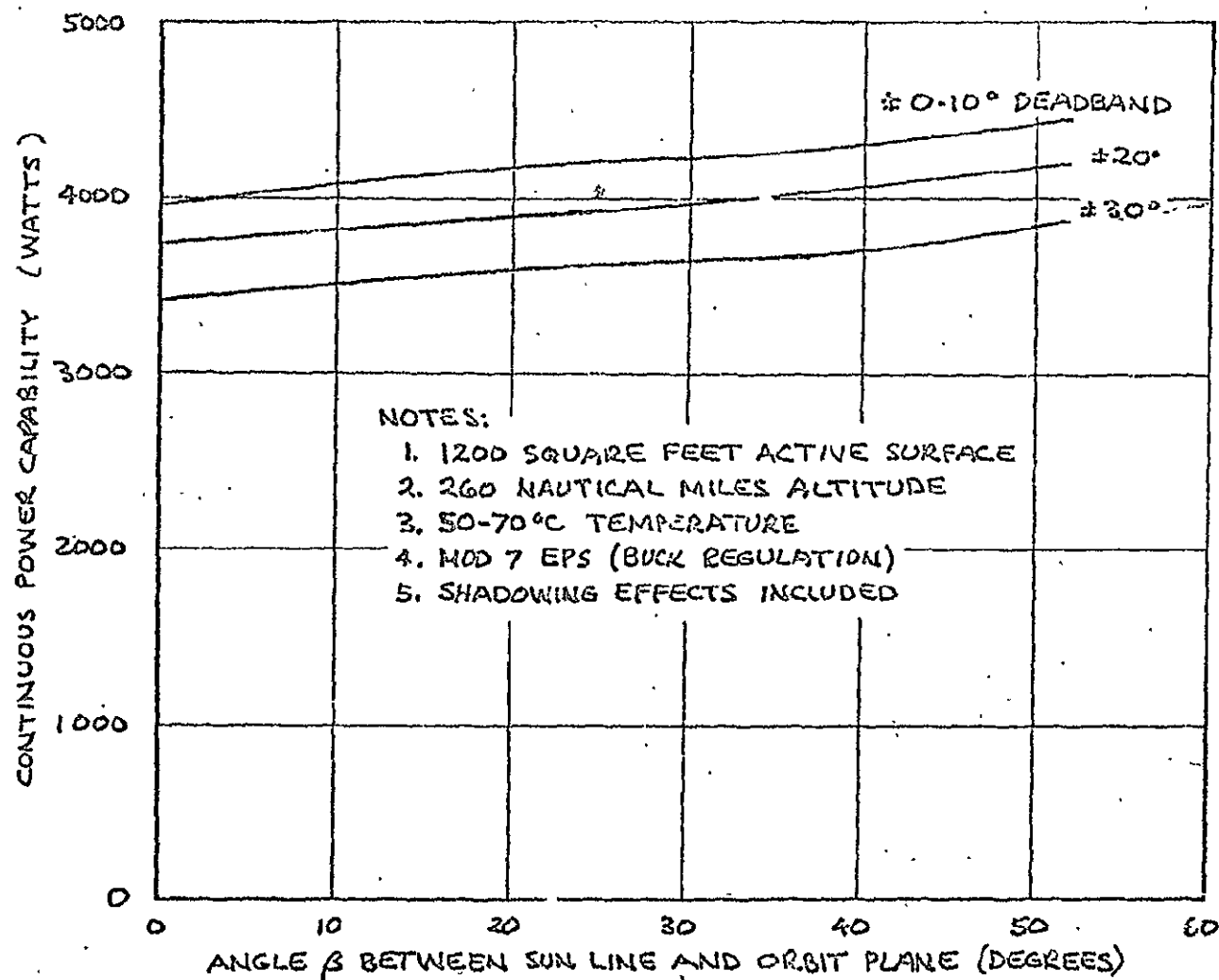


FIGURE 5-1 SOLAR ARRAY/BATTERY BUS POWER CAPABILITY; INERTIAL STABILIZATION WITH ARRAYS
NORMAL TO SUN'S RAYS (CASES 1, 1A, 2, 3, 3A, 4B, & 5)

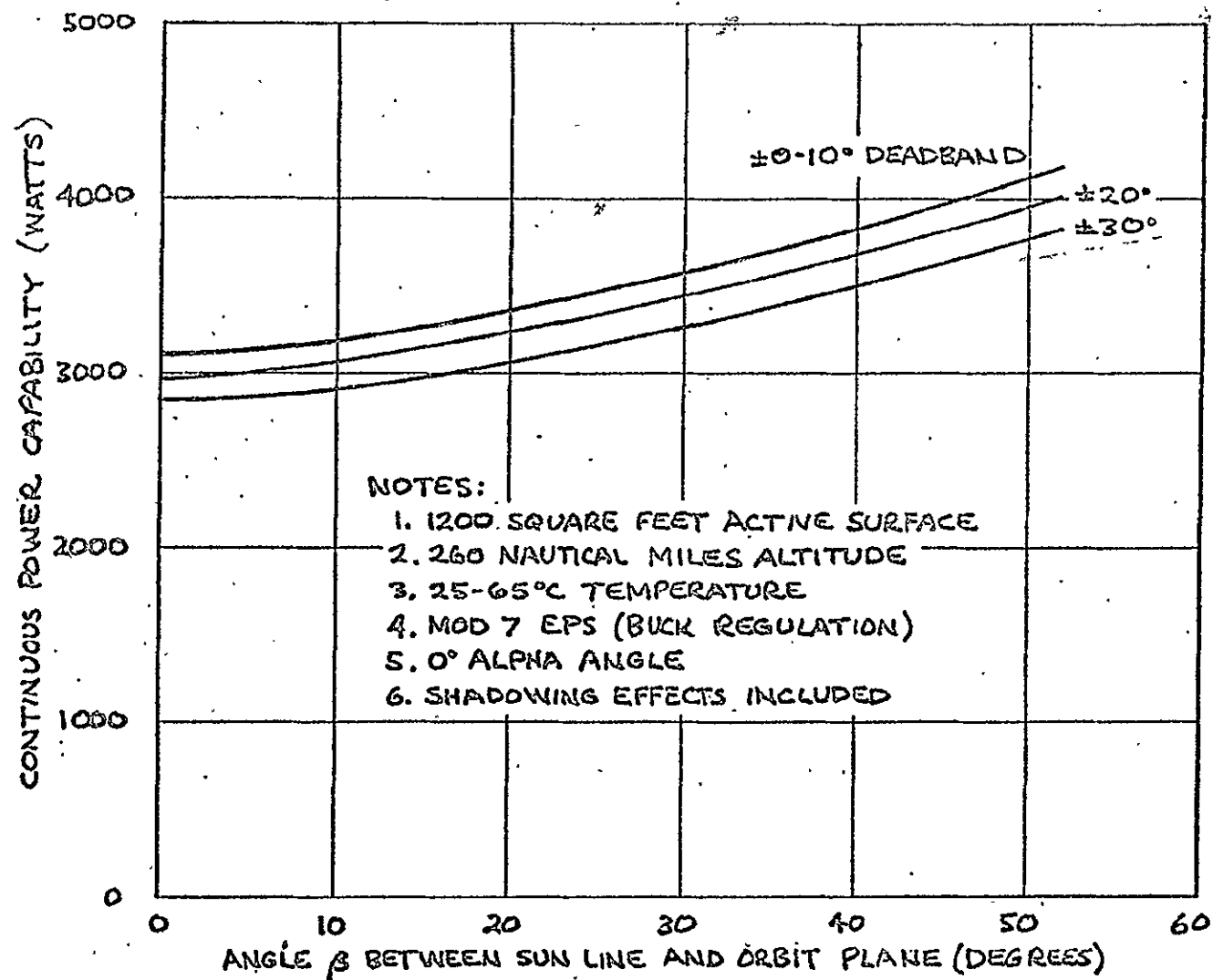


FIGURE 5-2 SOLAR ARRAY/BATTERY BUS POWER CAPABILITY: GRAVITY-GRADIENT
STABILIZATION WITH ROLL CONTROL FOR MAXIMUM SOLAR INCIDENCE (CASE 4)

alpha continuously, to yield the same power as for Cases 1, 1A, 2, 3, 3A, 4B and 5. However, as beta increases, the fall-off from this level is rapid, as shown in figure 5-3.

5.4 Gravity-Gradient with Variable Array Angle and Constant Roll Rate (Case 4C)

For this case, the roll rate is equal to the orbital rate of the Workshop and is synchronized so that, if alpha were to be held at zero degrees, the plane defined by the vehicle centerline and the panel normal would contain the sun vector at the time the OWS crosses the meridian. There is only a slight loss of power at a beta of zero degree compared to Case 4A; it results from the fact that the attitude resulting from the constant roll rate corresponds with the optimal roll attitude required at the terminator for Case 4A. At greater beta angles the constant rate of Case 4C provides a slightly worse match to the optimal attitude. As a result, the power is nearly constant as beta varies, as shown in figure 5-4, and does not increase appreciably with increasing beta.

5.5 Gravity-Gradient with No Roll Control and Array Angle Alpha = 90 degrees Storage

For orbital storage, insufficient stabilizing torques will exist to maintain any particular roll attitude. As a consequence, the solar arrays must be positioned so that their normal is parallel with the workshop longitudinal axis (alpha equal to 90 degrees) in order that they may absorb solar energy independently of the workshop roll attitude. Under worst-case shadowing and beta-angle conditions, the solar power system will supply a minimum of approximately 960 watts, as shown in figure 5-5. As noted in section 5.0, this value is probably conservative, and further, the likelihood of the workshop remaining in the worst shadowing condition for a long duration is very low.

The best-case-shadowing curve shows a dip in the region around beta equal to 10 degrees. For orbit planes with beta away from this region, it is possible to avoid any significant shadowing of the panels. However, at beta angles near 10 degrees, it is impossible to avoid some shadowing.

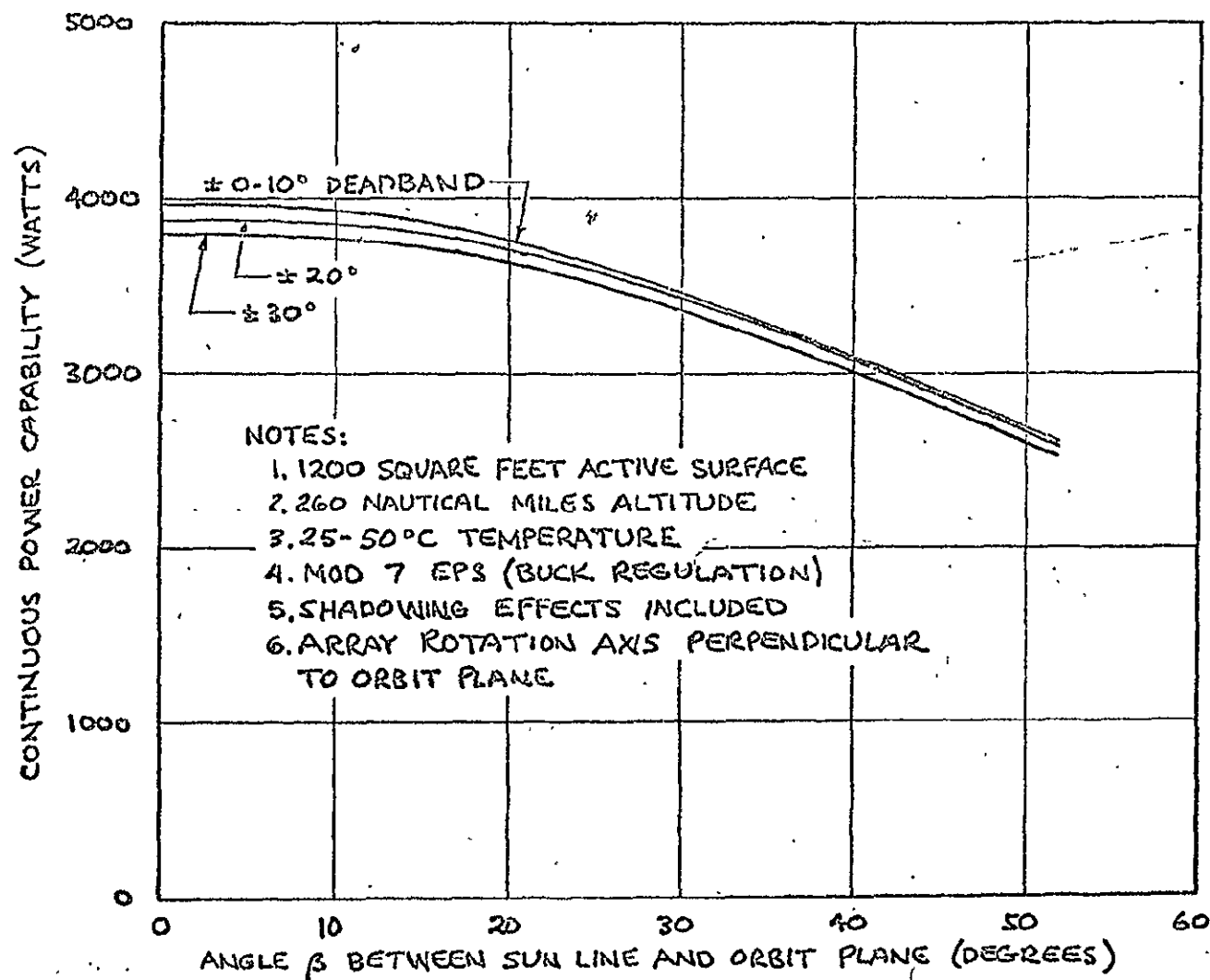


FIGURE 5-3 SOLAR ARRAY/BATTERY BUS POWER CAPABILITY: GRAVITY-GRADIENT
STABILIZATION WITH FIXED ROLL ATTITUDE AND VARIABLE ARRAY ANGLE FOR
MAXIMUM SOLAR INCIDENCE (CASE 4A)

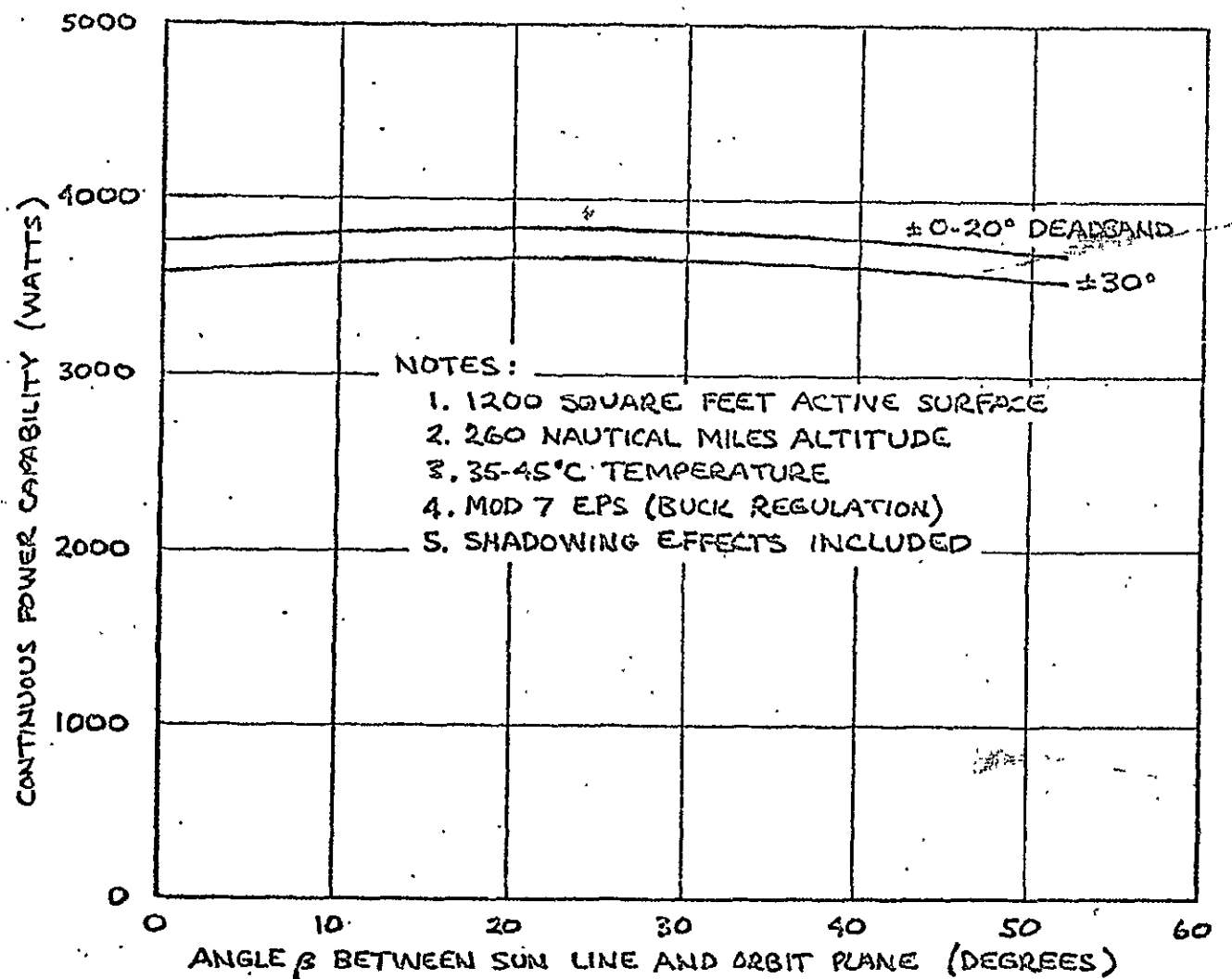


FIGURE 5-4 SOLAR ARRAY/BATTERY BUS POWER CAPABILITY: GRAVITY GRADIENT STABILIZATION,
 CONSTANT ROLL RATE EQUAL TO ORBIT RATE, CONTINUOUS CONTROL OF α ANGLE FOR MAXIMUM SOLAR INCIDENCE
 (CASE 4C)

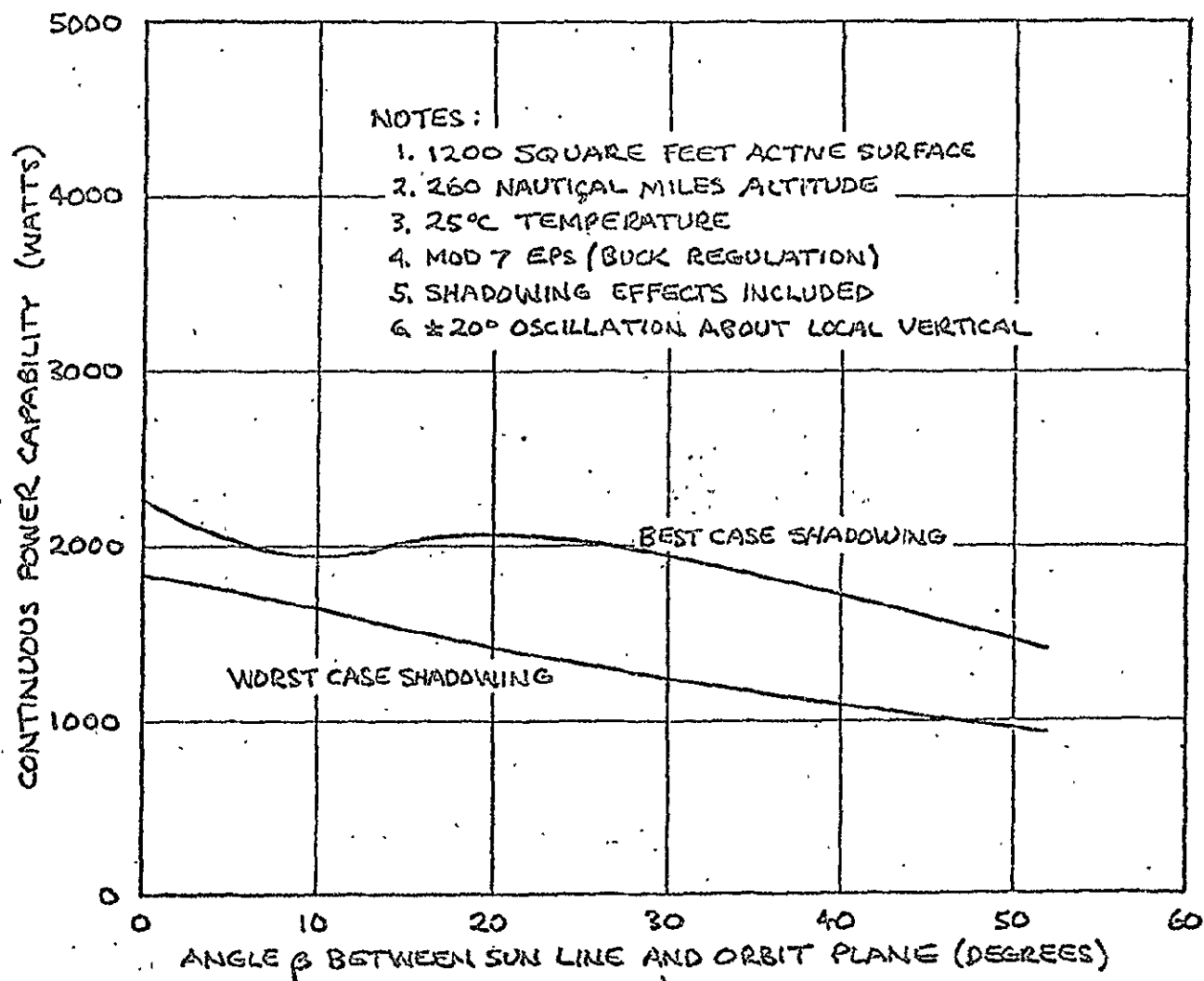


FIGURE 5-5 SOLAR ARRAY/BATTERY BUS POWER CAPABILITY:

ORBITAL STORAGE IN GRAVITY GRADIENT AT $\alpha = 90^\circ$:

5.6 Summary

The available power for the different alternative attitude control modes for Mission A is compared in figure 5-6. It is obvious that the orientations providing normal incidence to the arrays provide the most power. The next best single orientation method appears from a power standpoint to be that with the constant roll rate (Case 4C), because it provides reasonably high power for the entire range of beta orbit angles. It might be desirable to utilize different attitude control modes, for different values of beta to maximize power output from the solar panels.

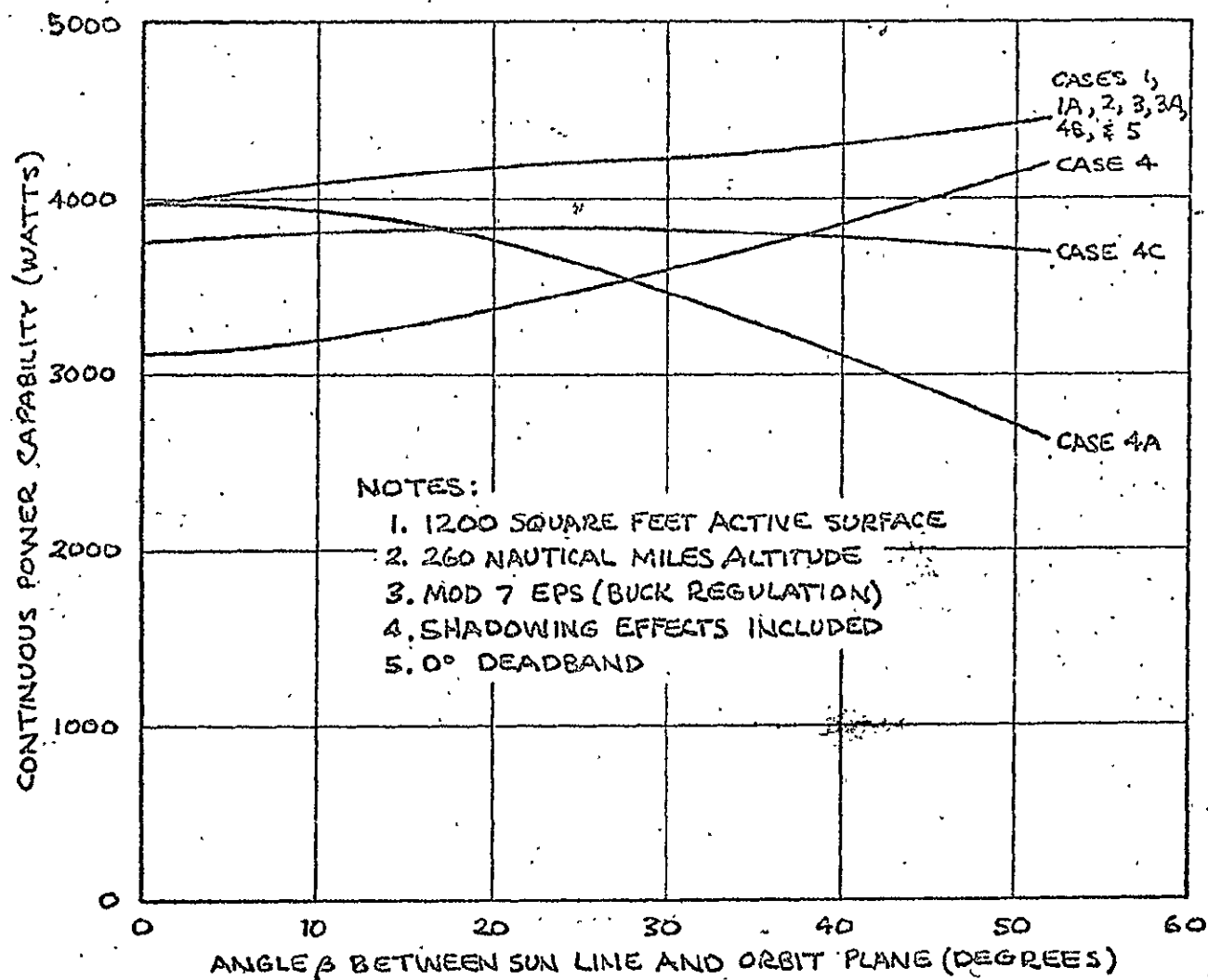


FIGURE 5-6 SOLAR ARRAY/BATTERY BUS POWER CAPABILITY:

COMPARISON OF ATTITUDE CONTROL ALTERNATIVES

1.0 THERMAL CONDITIONING

6.1 Phase I

The Phase I study was a broad approach to the general problems associated with thermal conditioning of the WACS. This general approach was necessary because specific components and systems had not been selected nor had final recommendations been made. The external environmental energy inputs to the vehicle were based exclusively on a gravity-gradient orientation; it was additionally assumed that a portion of this incident energy was blocked by the solar panels and cluster modules. Since extreme "blockage" is not likely to represent a typical mission orientation (especially throughout an entire mission), a certain degree of conservatism is automatically included in the analysis results.

6.1.1 Propulsion Subsystems

This section summarizes the results of the thermal evaluation of several WACS configurations employing existing S-IVB Auxiliary Propulsion System (APS) modules and includes the effects of possible manifolding of the propellants as well as the use of high performance insulation (i.e., multiple layers of radiation shielding; e.g., NRC-2 aluminized mylar). The various configurations discussed consider using two S-IVB/V APS modules with either S-IVB/V or S-IVB/IB propellant tanks or four S-IVB/IB APS modules. The modules are assumed to be symmetrically located around the vehicle.

Also considered as possible variations to any given system are propellant manifolding, substituting an oxidizer with a lower freezing point than the nitrogen tetroxide of the existing system and replacement of both propellants with a monopropellant. In addition to the above considerations, different numbers and kinds of engines for each configuration are included.

The thermal control of these configurations is primarily dependent on the modules' external envelope and the particular temperature required to assure that all system components remain within their operational limits. Once the thermal limitations are established, two basic module envelopes (IB and V) can be analyzed with regard to the vehicle and orbital geometry

as well as their attachment locations relative to the vehicle. Additions to or modifications of the basic data to account for manifolding, engine type and number variations, etc., can be done separately, where applicable, to account for all the possible systems considered.

Table 6-1 is a summary of the power requirements necessary to maintain the propulsion modules within given temperature limits for a vehicle in a gravity-gradient orientation. For configurations using the S-IVB/V APS propellant tanks or monopropellants such as hydrazine, the minimum allowable temperature is approximately 40°F . For systems using the existing hypergolic bi-propellants and the S-IVB/IB APS propellant tanks the minimum allowable temperature is 20°F . If an oxidizer such as red fuming nitric acid is substituted for the nitrogen tetroxide of the existing system a minimum temperature of 0°F (possibly as low as -40°F) may be used. The maximum and minimum values listed in the table give the yearly variation of the required power supply due to the annual variation of the sun vector relative to the orbit plane. If the vehicle employs a slow roll rate while maintaining the gravity-gradient orientation, the power requirements would be evenly distributed over each module. These values are shown in Table 6-1 also.

Table 6-2 lists the modifications necessary to the basic data in Table 6-1 to arrive at the total power requirements for each system considered. For instance, the maximum power required for a system with the following configuration is tabulated below:

Configuration

- a. Two Saturn V APS modules with high performance insulation
- b. Modules oriented in the orbital plane
- c. Six 25 pound thrust engines
- d. Monopropellant - hydrazine (e.g., 40°F allowable temperature limit)
- e. Manifolding with a single line wrapped with aluminized tape (e.g., $\epsilon = 0.05$)
- f. Vehicle fixed in gravity-gradient orientation (i.e., no roll)

TABLE 6-1

POWER REQUIREMENTS FOR OWS AACS IN GRAVITY-GRADIENT ORIENTATION (PHASE I)

MODULE/ ENERGY CONDITION	AVG. TEMP. WITH NO ACTIVE THERMAL CONTROL (°F)				POWER REQUIRED TO MAINTAIN AVG. TEMP. OF															
					40°F (watts)				20°F (watts)				0°F (watts)				-40°F (watts)			
	I	II	III	IV	I	II	III	IV	I	II	III	IV	I	II	III	IV	I	II	III	IV
IB: Max	22	43	22	-42	35	-7	35	135	-5	-47	-5	95	-40	-81	-40	60	-97	-148	-97	3
Min	-22	7	-22	7	108	63	108	63	68	23	68	23	33	-12	33	-12	-24	-69	-24	-69
V: Max	15	64	15	-63	101	-116	101	344	17	-200	17	260	-56	-273	-56	187	-178	-395	-178	65
Min	-46	-17	-46	-17	294	215	294	215	210	131	210	131	137	58	137	58	15	-64	15	-64
Net Energy Required for Vehicle Employing Slow Roll																				
4-IB Modules: Max					198				38				None				None			
Min					342				182				42				None			
2-V Modules: Max					215				47				None				None			
Min					509				341				195				None			

ASSUMPTIONS:

- Optical properties of external surfaces $\alpha = 0.3$; $\epsilon = 0.28$
- Orbital Inputs - Minimum energy conditions: spring, 6 a.m. insertion
Maximum energy conditions: winter, noon insertion
- Values listed in table are for four circumferential locations (I-IV) 90° apart with axis I-III in the orbit plane and axis II-IV normal to the orbit plane
- Negative watts indicate a surplus at that location

TABLE 6-2

MODIFICATIONS TO BASIC DATA TABULATED IN TABLE I FOR COMPLETE SYSTEM POWER REQUIREMENTS
PHASE I

ATTACHMENTS IB Modules V Modules	ADD: 3 watts/module ADD: 5 watts/module	
*MANIFOLDING 4 Module Configuration 2 Module Configuration	Lines polished ($\epsilon = 0.15$)	Lines wrapped with aluminum tape ($\epsilon = 0.05$)
	ADD: 26 watts/line	13 watts/line
	ADD: 13 watts/line	6 watts/line
ENGINE LOSSES Average Temperature ($^{\circ}\text{F}$) 40 20 0 -40	25 lb thrust (watts/engine)	150 lb thrust (watts/engine)
	ADD: 1.5	6
	ADD: 1.2	5
	ADD: 1.1	4.2
	ADD: 0.9	3
HIGH-PERFORMANCE INSULATION Multiply the values in Table 6-1 by 0.5		

*Assumes a 3/4-inch stainless steel line, supported every two feet with a fiberglass attachment 1 inch long and 0.5 square inches cross-sectional area; with an average temperature gradient of 200 $^{\circ}\text{F}$.

Power Requirements

- a. From table 6-1 in the 40°F column, read 294 watts each for Position I and III, yielding:
 $2(294) = 588$ watts for the basic system
- b. For high performance insulation (table 6-2) multiply by 0.5 yielding:
 $(0.5)(588) = 294$ watts
- c. Add 10 watts for the module attachments (table 6-2) yielding:
304 watts
- d. Add 9 watts for losses from six small engines (table 6-2) yielding:
313 watts
- e. Add 6 watts for one manifolding line (table 6-2) yielding:
319 watts

Hence, the total power required for this particular system is approximately 320 watts. Combinations or variations for any system employing the basic elements considered in the two tables may be evaluated in a similar manner.

The factor 0.5 used in table 6-2 to account for the application of high performance insulation is an estimated value. It is believed that an optimized system could achieve a greater reduction in the power requirements than this value if "optimization" is permitted by the other design constraints of the WACS.

6.1.2 Electronic Subsystem

This section discusses various problems associated with the thermal protection of the electronic hardware. The requirements include a gyro package, an electronics package, a horizon sensor and an inverter. In this group of hardware is equipment that is characterized during operation by high power dissipation, requiring thermal provisions that prevent the equipment from overheating (e.g., a heat sink). On the other hand, it is not desirable from an equipment lifetime nor power requirements standpoint to have the equipment "turned on" during those months that the stage will be inactive. The thermal problems associated with a reduced power

condition require thermal isolation of the equipment to prevent the equipment from exceeding the minimum allowable temperatures. These two separate operational conditions produce opposing requirements for the thermal protection system.

Below are several approaches that may be taken for thermal protection of the electronic hardware for the WACS.

1. Leave hardware "on" 100 percent of the time and design a passive thermal protection system around the overheating problem.

This approach has the greatest power consumption and the most stringent requirements on equipment lifetime and reliability.

2. Design an active heating system that maintains the hardware at the lowest allowable operating temperature 100 percent of the time (i.e., includes inactive periods when stage is "powered down"). This heating system is superimposed over the thermal design that is used to solve the overheating problem while the equipment is operating.

This system has the advantage of requiring lower power levels than the first approach to maintain the equipment within its operational range, but is inefficient since it is superimposed over a structural design used to solve an overheating problem (e.g., heat sinks).

Another advantage relates to equipment lifetime and stems from the fact that the active thermal control system (rather than the electrical components) is generating the heat to maintain the desired temperatures.

3. Design the structural attachments to prevent an overheating condition while equipment is operating. Define non-operational temperature limits for each piece of hardware and design an active thermal system with dual capabilities. The first capability would be the ability to maintain the non-operational temperature levels during the inactive periods; the second capability would be to "warm up" each piece of hardware just prior to activation.

This system would require less power than either of the above two systems and has another advantage in as much as the non-operational temperature limits for some of the hardware may be below those temperatures that result solely from the orbital environment.

4. Mount equipment on a "thermal switch" which operates in a manner that might be referred to as a thermal diode. It has the ability to passively dissipate large heating loads through a heat sink during the operational phases of the equipment and, therefore, provide thermal isolation. The "switches" can be used singly or in groups; the surface dimensions of one switch are approximately 6 inches by 6 inches (used on the Surveyor Spacecraft). The energy dissipated per switch is approximately 0.147 watt per degree temperature difference across the device; e.g., with a temperature difference of 100°F, 16 switches on a 2 foot by 2 foot panel would dissipate over 230 watts.

This system has the potential of having much lower power requirements than any of the active systems discussed above. This is primarily because the power used to maintain any temperature level (operational or non-operational) during the inactive periods is not conducted into the heat sink which is necessary to prevent overheating when the equipment is operational.

Without design and installation details for particular pieces of equipment as well as their environmental limitations, it is only possible to approximate power requirements for the thermal protection of the electronic hardware for the WACS. These power requirements may range from a "worst case" of 100 percent of total energy being generated within the equipment (e.g., mounting through a perfect heat sink) to a small fraction of this amount (i.e., systems 3 or 4 above).

In the absence of specific equipment details it is estimated that 100 watts would be sufficient for the thermal protection of the electronic hardware associated with the WACS. The active thermal system may be required to deliver this load to one of several pieces of equipment or at a reduced

level to equipment in several locations. The equipment installation locations can be distributed around the vehicle so as to exclude the possibility of having the "worst case" thermal environment occur for more than one location at a time.

6.1.3 Testing

Thermal vacuum testing should be performed on the WACS to verify the thermal performance of the active and passive thermal conditioning subsystems. It is not possible for analysis alone to completely verify the total thermal response of any module or component under any given set of conditions. Testing with the proper instrumentation can assure that the thermal conditioning system will adequately meet the mission requirements.

The effects of long term exposure to the space environment on the WACS optical properties must be determined in light of specific design requirements. Testing may be required to determine the degree of degradation resulting from surface exposure to plume impingement, ultra-violet radiation, vacuum environment, and cyclic orbital temperatures. These factors must be considered in selecting the materials to meet the thermal design requirements.

The use of high performance insulation (i.e., multiple layer radiation shielding) would require an evaluation of the "heat shorts" associated with the application. Although the techniques used to minimize these heat shorts are well known, an exact quantitative determination is difficult to perform analytically. This occurs because the heat short result from physical and geometrical deviations from the ideal (theoretical) application techniques and are most accurately determined experimentally.

Douglas has ample facilities for performing thermal vacuum verification testing of the thermal conditioning subsystem on the WACS. There are two 5 foot diameter by 6 foot long high vacuum (10^{-9} torr) chambers with helium and LN_2 shrouds and one 10 foot diameter by 12 foot long (BEMCO) chamber with a LN_2 shroud capable of pumping down to approximately 10^{-6} torr. Each chamber is capable of absorbing a heat load of several kilowatts. Any of these chambers would adequately meet the environmental requirements necessary to perform the testing.

A "worst case" test of the thermal conditioning system for the propulsion subsystem could be performed with an "unloaded" module (i.e., no propellants or pressurization gas) or with a module using simulated propellants. The module attachment structure could be simulated and the module allowed to radiate to the LN_2 shroud (representing space) under a given operational power load (or a series of different loads) to the conditioning system until steady state thermal conditions are reached. The magnitude of the power required to maintain the specified temperature is a measure of the total system heat shorts and can be used with the proper instrumentation to verify not only the actual system performance, but the analytical predictions as well.

6.2 Phase II

The Phase II study reflects the results of narrowing the candidate propulsion systems considered in the Phase I study and adding the effects of solar orientation for the vehicle. Also included is an additional tabulation of "nominal" power requirements as well as those tabulated in the Phase I analysis where blockage of the environmental inputs to the modules by the overall vehicle geometry (e.g., cluster modules, solar panels, etc.) was considered. The nominal power tabulation assumes little or no interference from the rest of the vehicle with the environmental energy incident on the propulsion system modules.

6.2.1 Propulsion Subsystem

The power requirements in this section are based on the use of two propulsion modules and up to four tanker modules. Table 6-3 lists the analysis assumptions and ground rules and table 6-4 lists the average power requirements under these conditions.

6.2.2 Electronic Subsystems

The discussion presented above in the Phase I study is applicable to the Phase II study with regard to the electronic subsystem.

6.2.3 Testing

The testing section in the Phase I study is also applicable to the Phase II study.

TABLE 6-3

WORKSHOP ATTITUDE CONTROL SYSTEM STUDY
THERMAL CONDITIONING SUBSYSTEM

ASSUMPTIONS

Solar orientation during mission phases A and B

Gravity-gradient orientation at all other times

Maintain APS temperature limits of 20°F to 125°F

Extreme blockage of environmental inputs to modules by vehicle protuberances (e.g., solar panels, module clusters, etc.) may occur temporarily but "nominal" assumptions in this regard are considered to be more representative of mission averages

GROUND RULES

*Compute power requirements based on maintenance of operational temperatures throughout mission

Manifold propellants

Employ multiple layer radiation shielding to minimize losses

* "Average" power requirements would be reduced considerably by permitting temperatures to be below "operational" values during inactive mission phases.

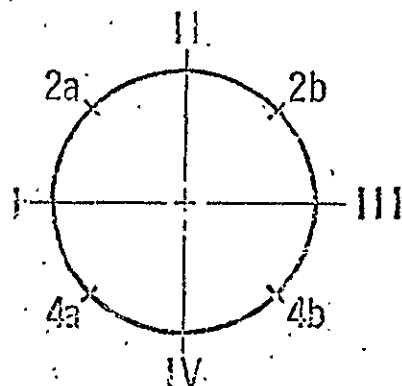
WORKSHOP ATTITUDE CONTROL SYSTEM APS THERMAL POWER REQUIREMENTS

WACS-57
8-22-67

TABLE 6-4

*AVERAGE POWER /MODULE (WATTS)

MODULE AND LOCATION



ENVIRONMENTAL INPUTS COMPUTED ON BASIS OF	EXTREME BLOCKAGE DUE TO VEHICLE GEOMETRY	NOMINAL BLOCKAGE DUE TO VEHICLE GEOMETRY	** ORIENTATION	SOLAR	GRAVITY GRADIENT				
				SOLAR	GRAVITY GRADIENT				
					$\beta=50$	$\beta=0$			
PROPULSION MODULE I	0			0	25	117			
PROPULSION MODULE III	170			140	25	117			
TANKER MODULE 2a	0			0	0	97			
TANKER MODULE 2b	175			150	0	97			
TANKER MODULE 4a	0			0	127	97			
TANKER MODULE 4b	175			150	127	97			

*MANIFOLDING OF PROPELLANTS REQUIRES AN ADDITIONAL 10-25 WATTS PER CONFIGURATION.

** POSITION I TOWARD SUN DURING SOLAR ORIENTATION

POSITION I - III IN ORBITAL PLANE DURING GRAVITY GRADIENT ORIENTATION

7.0 ELECTRONIC SUBSYSTEM

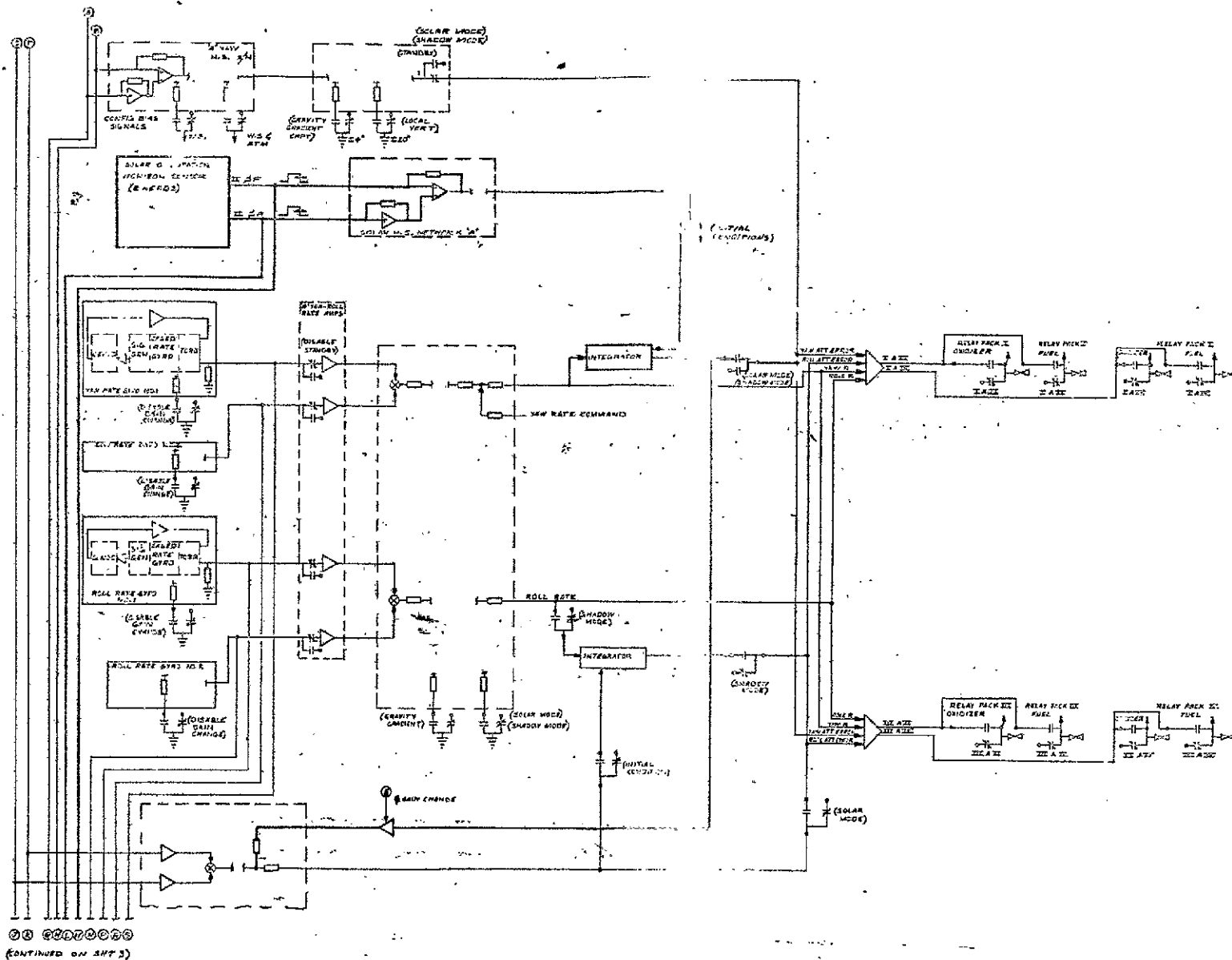
7.1 General System Description

A block diagram of the WACS electrical system is shown in figure 7-1. The basic drawing is for case 5 with deletions for case 3 indicated with heavy lines. The system used for the phase I modes of table 3-1 is the same as that for case 3 except that sun sensors and rate integrators are not required.

The WACS system is made up of two redundant channels (A and B). Each channel feeds a separate spatial amplifier which controls a redundant set of valves on the APS. Each channel has a rate gyro which can also be used to control the other channel if necessary. Attitude information is provided from the Horizon Sensor electronics. The output from each of the sensing heads, which are designated I and III in the pitch plane and II and IV in the yaw plane, are 16 Hertz square waves which are pulse width modulated. The signals from the horizon sensors are distributed to the required pitch and yaw-roll channels. At the input to each channel the pulse width modulated signals are transformed to a proportional analog voltage.

Two methods of conversion are presently under consideration. One system is a filter which imposes some restrictions on the response time of the system. The second method makes the digital-to-analog conversion using a circuit that samples the integrated input signal once each cycle and holds this level until the next update. With either of the conversion methods the signals will be amplified using integrated circuit differential amplifiers. The amplifiers can be used to derive the error signal which is the difference between the output signals of opposing sensors.

Also feeding into the horizon sensor shaping networks are configuration bias signals. These signals are inserted into the shaping network to compensate for changes in the vehicle principal axes when different payloads and vehicles are attached to the workshop. The bias signals into the network can be positive or negative and will be derived from the plus or minus 20 volt power supply and dropped through an appropriate resistance.

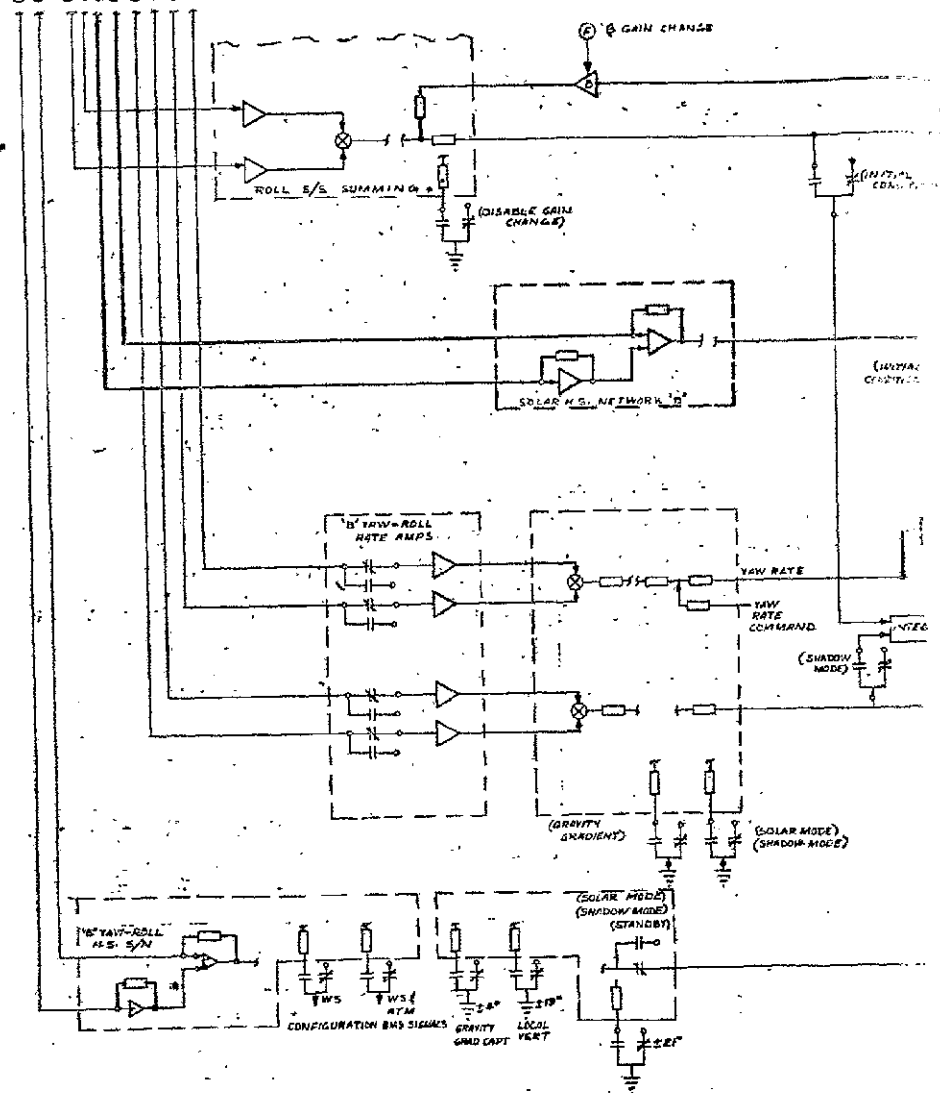


FOLDOUT FRAME

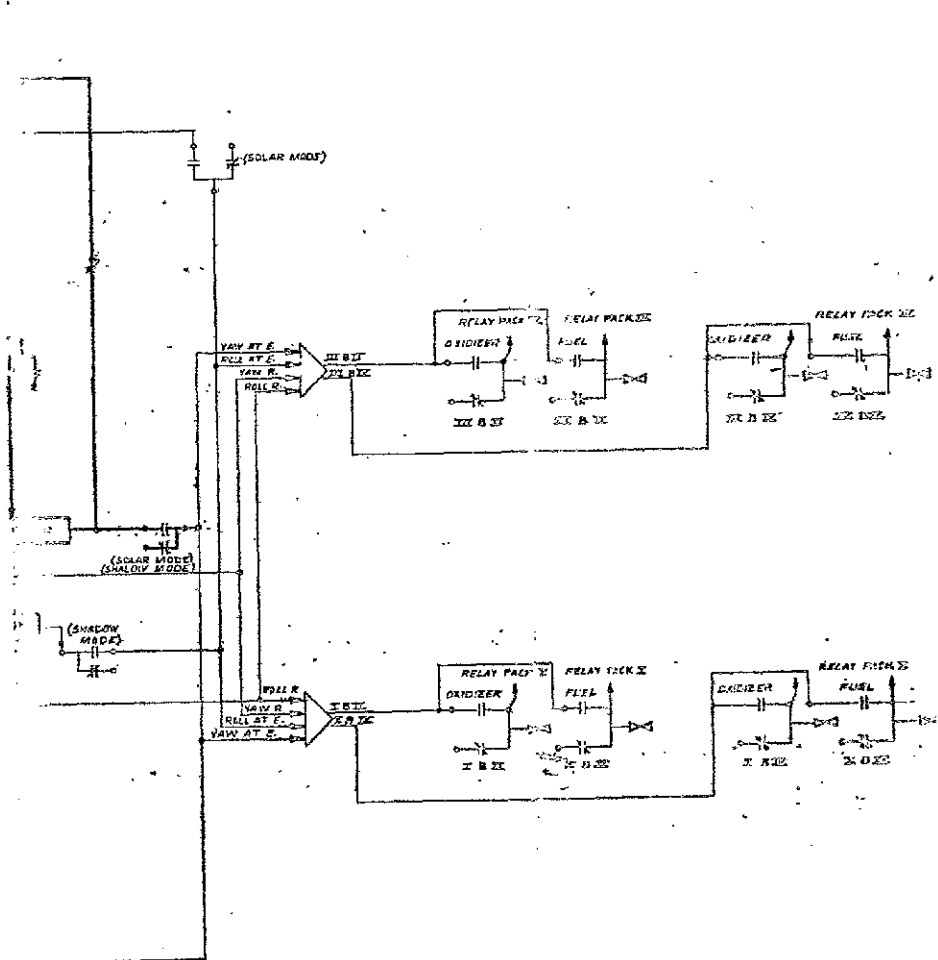
FOLDOUT FRAME 2

11 (CONTINUED FROM FIG. 1)

24 25 26 27 28 29 30



FOLDOUT FRAME 1



FOLDOUT FRAME 2

FIG. 11 CONTINUED FROM FIG. 1

Gain changes in the horizon sensor attitude error loop are provided for the plus and minus 20 degree local vertical mode and plus and minus 4 degrees for gravity gradient capture. Use of slightly different gains in each channel will cause one channel to be operative and the lower gain channel to serve as an automatic backup.

As noted above, one rate gyro is associated with each channel. In the event of a single gyro failure, the output from the remaining gyro can be inserted into both channels. By monitoring gyro output via telemetry, and utilizing a radio command link the defective gyro can be switched completely out of the circuit. During habitation the crew can disconnect the disabled gyro from the Astronaut Manual Control Panel.

The output of the spatial amplifiers are fed through normally open relay contacts to the valves of the APS. These relay contacts, which electrically parallel the relays in the control relay package, will connect the WACS spatial amplifier outputs to the oxidizer and fuel valves of the appropriate engines upon actuation. The actuation of these relays will thus permit control of the APS valves by the WACS. Relays in the S-IVB Stage control relay package will be held at the normally open position when control by the IU is terminated, or when the IU/S-IVB battery system is passivated. This circuitry is devised to minimize valve actuation time and is predicted on: 1) The output transistors of the relay drivers in the spatial amplifiers being capable of providing current to drive the valves, and 2) The circuitry being capable of providing a pulse of 10 milliseconds. In the event that the spatial amplifier cannot provide the current of a pulse of 10 milliseconds, some design may be necessary in the Schmidt trigger circuits and the relay driver circuits of the spatial amplifiers. During operation of the WACS, the relays within the control relay packages would be open thus eliminating feedback into the S-IVB circuitry during workshop operations.

The portion of the system discussed thus far is that used for the phase I modes of operation. The case 3 system requires the addition of the following equipment:

1. Two axis solar sensor (2 units for redundancy)
2. Timer package (2 units for redundancy)
3. Solar switch (2 units for redundancy)
4. Additional summing and integration circuits within the electronics package.

The two-axis solar sensors for which signals can be switched between channels in a manner similar to the gyros will provide pitch and roll attitude error information during the solar mode. During the shadow mode, attitude error in pitch and roll will be obtained by integrating the output of the rate gyros. Switching between these two modes will be controlled by a solar switch and a timer package which has two outputs. As the workshop moves from the earth's shadow into the sun, the solar switch provides an initiate signal to the timer. Output number 1, which occurs approximately three minutes after initiate, switches from shadow mode to solar mode, Output number 2, which occurs approximately 55 minutes after initiate, switches to the shadow mode. The timer then resets awaiting another initiate signal after traversing the earth's shadow. Two solar switches and timer packages will be used for redundancy considerations. Only rate control is provided in the yaw plane during the case 3 solar mode. A continuous command of approximately one degree per second is inserted into the rate channel.

The case 5 system requires the following additional equipment over case 3:

1. One horizon sensor system with two heads
2. Additional outputs from the timer package
3. Interface with command receiver relay panel and astronaut manual control
4. Additional summing and integration circuits within the electronics package
5. Beta angle generator (in power distributor or electronics package)

The pitch channel for case 5 will operate in the same manner as for case 3 except a command will be summed with the pitch sun sensor output to keep the vehicle perpendicular to the orbital plane. The roll channel for case 5 is the same as for case 3 except a voltage equivalent to the product of yaw

attitude error and beta angle is summed with the roll sun sensor output to cancel out the yaw-to-roll cross-coupling effect. Yaw attitude control is provided by inserting the rate and the rate integrated output of the gyro package into the spatial amplifier. However, gyro drift compensation must be provided by utilizing a horizon sensor system consisting of two heads pointing forward and aft along position II. The yaw axis will be switched from gyro to horizon sensor control near the terminator as described in section 3.4.6.

In order to sequence the system as discussed above, it will be necessary to provide additional outputs from the timer package to turn the horizon sensor on and off. Functions previously discussed in case 3 will remain the same.

Beta angle voltages and gain changes may originate from three sources:

- 1) Ground control via the command receiver R.F. link, 2) Astronaut manual control panel, and 3) Preprogrammed data on-board. Commands will be transmitted from the ground at the proper time to change the beta angle voltage to the pitch sun sensors and the gain of the amplifier between the yaw and roll channels. It is estimated that the beta angle command must be changed in increments of not more than 3 degrees to provide a maximum error of 1.5 degrees between nominal vehicle position and the orbit plane. This will require capability for 32 different commands. Amplifier gain need be changed to only eight levels for each sign. The beta angle generator and gain sequence will consist primarily of a resistor panel with a voltage source. Relays on the relay panel will cut resistors in and out of the circuits at the proper time. The R.F. link will be the primary mode of control. The astronaut manual control will be used as a backup in the event of failure of the R.F. link. With this pitch attitude control system, the solar panels can be controlled by a closed loop positioning system using either a sun sensor mounted on the solar panels, or the output from the beta angle generator to derive command signals.

The method of switching gyros, sun sensors and the electronics, as discussed above, is not the only option for increasing reliability as pointed out in

paragraph 3.5. An evaluation of reliability data must be performed before a definite operational mode is defined.

7.2 Gyros

The use of conventional spring restrained rate gyros is not feasible because of threshold and zero offset limitations. The existing IU rate gyros were found to be a typical example. Characteristics of the IU rate gyros are as follows:

Type:	Nortronics GR-H4T .35B
Range:	± 20 deg/sec
Scale Factor:	200 mv/deg/sec $\pm 3\%$
Zero Offset: (Includes Non-G Sensitive Drift)	5 MV (0.02 deg/sec) Factory Adjustment 25 MV (0.1 deg/sec) Max. for all conditions
Threshold:	0.01 deg/sec Max.
Self Test Torquer:	0.125 deg/sec/ma DC
g-Sensitive Drift:	0.03 deg/sec/g
Excitation:	26V, 400 Hz
Size:	0.967 in. Dia X 2.0 in. Long
Weight:	5 oz.
Cost:	Approx. \$2,500 each

Very little, if any, improvement in threshold or zero offset can be achieved with conventional spring restrained rate gyros, thus dictating the use of electronically caged rate integrating gyros.

Of the gyros investigated, only two are available in a flight qualified packaged system which would be suitable for this application. These are the DELTA Gyro Assembly (Douglas P/N 7842941) with three Honeywell GG87 gyros and the Apollo Gyro Assembly (Honeywell P/N GG-362) with three Honeywell GG 248 gyros.

Also, it was learned that the gyro packages being developed for the ATM may be available in time for use in the WACS system.

The Apollo unit is a three-axis system capable of operating in either rate or attitude modes (± 20 degrees). The Delta unit is a three-axis system normally operated open loop as attitude gyros (± 10 degrees) with caging loops for initial alignment. The caging circuits could be modified for rate mode operation.

The Apollo gyro package has an MTBF of 17,400 hours; it is estimated the MTBF for the Delta package would be somewhat less.

Considering the Apollo and Delta packages the Apollo system is preferred because of the lower gyro torquer gain and probable less extensive modifications required.

A comparison of the ATM and Apollo gyro packages may be found in table 7-1.

In general, the relative merits of each of the gyro systems varies with the specific area of interest. Those areas in which the two systems are roughly equivalent are: cost, long term zero offset, input angle freedom, non-g sensitive drift and power consumption. Considering a three-axis system, the Apollo package is smaller in volume and weight, has a higher MTBF and is already qualified for the Apollo program. However, the ATM packages can be obtained in a shorter period of time, can survive lower temperatures and has a lower threshold.

Either gyro will adequately perform the workshop mission. Considering the factors discussed above, the most important are reliability, delivery schedule, production and qualification status.

The Apollo package, except for schedule, has the edge in these categories. The schedule permitting, the Apollo rate gyro package appears to be the best choice based on information gathered to date. If schedule is a problem, it may be possible to obtain several packages from the Apollo program on a loan basis and be replaced when the production line is activated.

Each of the gyros investigated must be heated for proper operation. For the heated gyros, the operating temperature must be maintained to less than $\pm 10^{\circ}\text{F}$ for specified performance. The unheated gyros will provide specified performance over the specified operating temperature range but improved performance may be obtained if the operating temperature range is minimized.

A new system would be desirable. This system could be designed specifically for the required low rate ranges and scale factors. A lower cost gyro would be selected compatible with the requirements. The heaterless gyros are attractive because of price, power consumption and the elimination of a heater control system.

If schedule and development costs preclude the development of a new gyro assembly, the Apollo and ATM units appear to be the only available systems, with the Apollo unit having an edge as discussed above.

TABLE 7-1. APOLLO & ATM GYRO COMPARISON

	APOLLO	ATM
Part Number	Honeywell Part No. GG-362	MSFC SPEC 50M37705
Gyro Type	Honeywell GG-248	Kearfott C702519018 (Similar to C702519015)
Approx. Cost Per Package	\$105,000 2 Required	\$35,000 6 Required
Size (Each)	14.25" X 6.0" X 7.0"	12.0" X 8.75" X 5.25"
Weight (Total) - pounds	54 * (2 Gyro Packages & 2 Static Inverters)	84 ** (6 Inverter Packages with Internal Static Inverters)
Input Angle Freedom	± 20 deg	± 24 deg
Output Gimbal Freedom	± 4.4 deg	± 2 deg
Signal Generator Scale Factor	0.1V/Deg/Sec	Course: 45V/Deg/Sec Fine: 1000V/Deg/Sec
Torque Generator Scale Factor	400 deg/hr/ma	40 deg/hr/ma
Non-g-sensitive drift-package	± 0.15 deg/hr	Package output fine mode .1 deg/hr
Non-g-sensitive drift-gyro	± 0.5 deg/hr	± 0.5 deg/hr
g-sensitive drift	± 1.2 deg/hr/g	± 0.6 deg/hr/g
Threshold	0.01 deg/hr or less	± 0.00001 deg/hr
Zero Offset	0.1 deg/hr Short term 1 deg/hr Long term	0.1 deg/hr Short term 1 deg/hr Long term
Operating temp. of the gyro	170°F	154°F
Internal heater	28V, 14W	28V, 25W
Temperature Range of Package	0°F to 150°F	-67°F to 131°F
Approx. Power Required (In- cluding estimated average in- ternal heater power)	161 watts ***	182 watts ****

TABLE 7-1. APOLLO & ATM GYRO COMPARISON (continued)

	APOLLO	ATM
Production Status	Produced for Apollo	Under Development for ATM
Delivery Schedule	11-12 months	8 months
MTBF	17,400 hr for each package	25,000 hr for each package
MTBF of a 3-axis system	17,400 hr	8,700 hr
Probability of success of one 3-axis system 1000 hrs of operation	0.942	0.885

* Includes two static inverters at 4 pounds each

** Includes approximately 1 pound in each package for a static inverter

*** Includes 46 watts for two static inverters

**** Includes 47 watts for static inverter internal to gyro packages

7.3 Horizon Sensors

It is advisable to have a horizon sensor with a large field of view which will provide continuous attitude information regardless of vehicle position. Also, because of the long duration of the mission, it is considered desirable to exclude devices with moving parts, if possible. This leaves the electronic scanner and radiometric balance systems for consideration. The radiometric balance system compares the radiation received from opposite portions of the earth. Level attitude is indicated when the fields receive equal amount of earth radiant energy.

The radiometric balance sensors that were investigated had a field of view of not much over 20 degrees and radiance errors were proportional to the field of view. To properly fulfill the operational requirements with a narrow field of view device, it would be necessary to: 1) Have moveable tracking heads with more than one position; 2) Have several horizon sensor heads in the same plane to cover the total angle required; 3) Develop a head with a large field of view; 4) Incorporate a search mode capability within the WACS.

An electronic scanner type of system is the Lunar and Planetary Horizon Sensor (LPHS) presently being developed by Barnes Engineering, which utilizes a linear array of sequentially sampled thermopile detectors to provide scanning without moving parts. This system employs four fields of view (FOV) which are orthogonally projected into space as shown in figure 7-2.

Each FOV of each head is segmented into discrete solid angles which are defined by the radiation detectors in a linear array. The location of the thermal horizon within the FOV is determined by sensing the difference in output between those detectors being irradiated by the energy from the earth and those facing space. Each detector output is sampled sequentially by an electronic commutator. This produces four outputs (one from each head) that are 16 Hertz square waves which are pulse width modulated. A brief breakdown of the specifications of the Barnes LPHS is as follows:

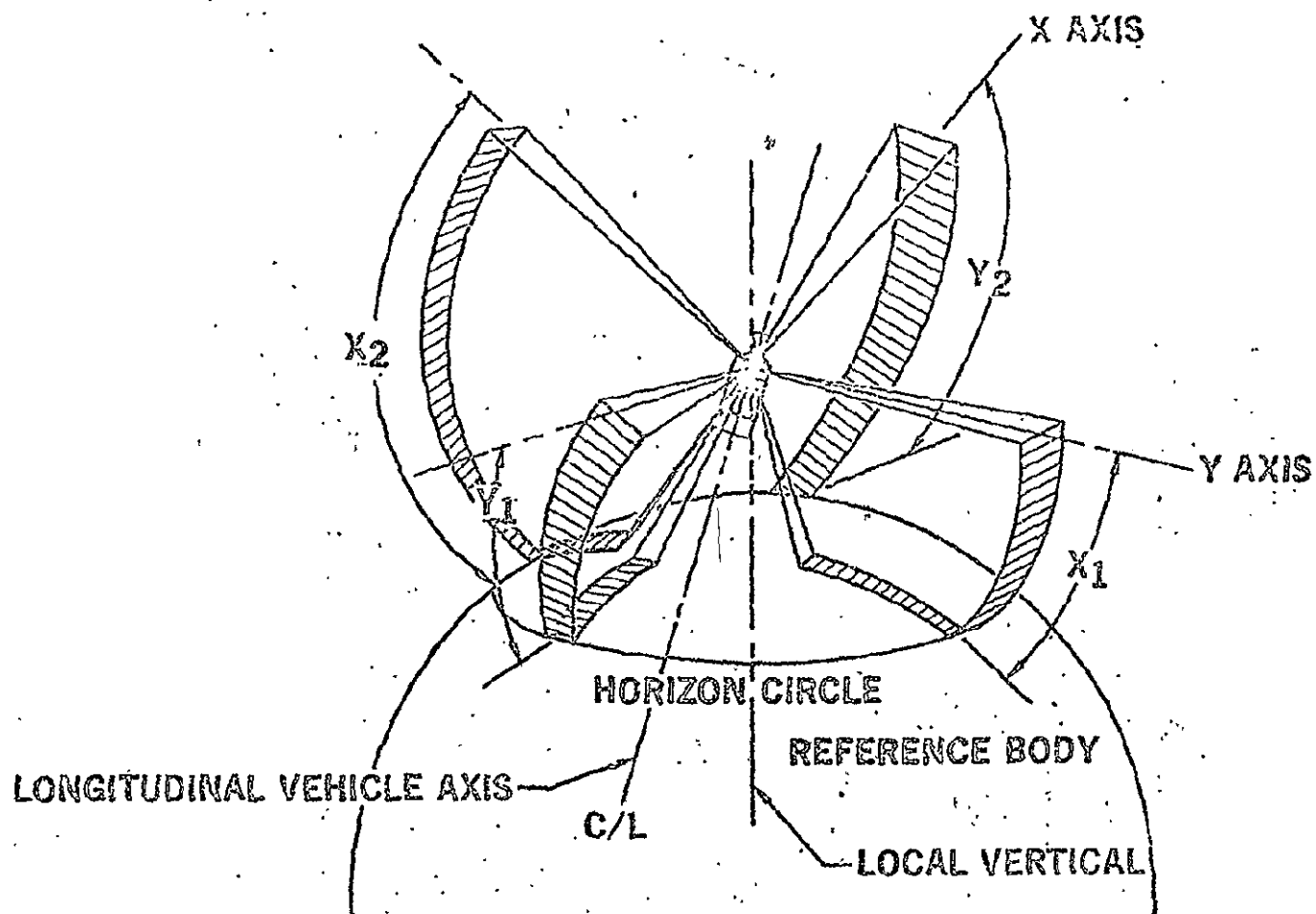


FIGURE 7-2. HORIZON SENSOR FIELDS OF VIEW

Model Number:

13-162

Description:

Arrays of thermopile detector elements are electronically sampled commutator fashion. There are 40 discrete detectors in one array. Four identical scanning heads, each with its optical system and detector array, define the four sectors shown in figure 7-2. Each commutated signal, after filtering, shows a profile equivalent to that which would be generated by a detector mechanically scanned across the space/earth horizon.

Estimated Weight:

15 pound for system consisting of four heads and one electronic box. 2 pound each head. 7 pound electronics box.

Power:

5 watts at 50V, 2400Hz, single phase, square wave*

Estimated Size:

Each head: 6 in. long x 2 7/8. high x 2 3/4 in. wide. Electronics box: 7 1/2 in. x 6 in. x 2 1/2 in.

Optics:

Spectral band: 14 to 40 microns

Field of view (each sector): 70 deg x 10 deg

Mechanical:

No moving parts.

Electronics:

All solid state circuitry

Environmental:

Temperature: -40°C to +80°C

Shock: 50G; 10-12 msec. rise time,
0-2 msec decay time

Vibration: 20G; 28-2000 Hz

Acceleration: 12 g

Instrument Accuracy :

± 1 digital count

Probability of Success

90.6% for 3-year mission

* Power input to be changed to +28 VDC.

Outputs:

Horizon crossover is indicated by transition from 0 volts (space in field) to -20 volts (earth in field). Auxiliary pulses show commutation changeover and start of each scan.

Development History:

A feasibility model has been constructed and environmentally tested. The first flight unit, which will be a single axis system utilizing two heads, was expected to be completed about November 1, 1967.

It is recommended that a modified model 13-162 be utilized for the WACS. This unit is presently under development and can be obtained within the required period of time. Estimated cost is \$125,000 for a four head system. A horizon simulator which will be used for checkout, will cost \$40,000 and additional qualification could cost \$25,000. Development of a low altitude radiometric balance unit would cost approximately \$300,000 and would not be as accurate as the model 13-162. Multiple heads, moveable heads, or the addition of a search mode would result in increased complexity of fabrication and operational control.

7.4 Static Inverter

An AC source will be required to supply gyro spin motor power, demodulator reference voltage, signal generator excitation and other loads which vary with specific equipment chosen for the WACS. A static inverter offers the best solution to this problem. A brief summary of inverters investigated may be found in table 7-2. Most inverters were rejected immediately usually for being too large or being out of specification requirements. Inverters with the 15/26V output were primary candidates for the Delta gyro package, those with the 115/199V output were primary candidates for the Apollo and ATM gyro packages. However, either type of inverter could be converted for use with either gyro through the use of a transformer. Also, the ATM gyro package has the capability of accepting an internally mounted static inverter. It is

TABLE 7
INVERTER CHARACTERISTICS

Inverter:	Input	Output	Weight Pounds	Environment	Remarks
		(Nominal + High Power)			
1. Inverter from EOL Program	+28V D.C.	a) 3 ϕ 400 Hz Wye 15/26V 50 V-A b) 3 ϕ 400 Hz Delta 15V 36 V-A c) 20V D.C. 6 watts d) 8V Output #1 15.5 watts e) 8V Output #2 29.6 watts f) Gyro heater output with overload pro- tection 36.7 watts Sync of 4.8 KC $\pm$.01% for frequency $\pm$.01%	Overload Maximum 60 V-A 20.8 V-A 6 watts 17.5 watts 29.6 watts 36.7 watts (for 45 min.)	--	The VACS system requirements at this time would allow deletion of the $\pm$.01% sync input to the inverter with a resultant fre- quency output tolerance of $\pm$ 1%, which is also true for Delta inverter listed below.
2. Delta Inverter	$28 \pm \frac{4}{2}$ volts D-C, capable of withstanding without damage input transients of 80 volts for 100 sec.	a) 3 ϕ 400 Hz 15/26 volts $\pm$ 2 volts 68 V-A starting 50 V-A continuous b) 2 auxiliary windings of 135 $\pm$ 4% A.C. output. Each capable of supplying 70 V-A. c) Frequency 400 Hz $\pm$ 1.0%. A sync 5 $\pm$ 10% volts P/P ut 4 watts to more accurate frequency control.	9	Operating temperature may vary between -55°C and +71°C. During qualification testing program the inverter was operated for approx. 2000 hrs.	
3. Agena Type Inverter	22 to 29 volts D.C.	115 volt rms $\pm$ 1% 400 Hz $\pm$.001% 3 ϕ Delta For continuous operation total load exceed 120 V-A.	18	+30°F to +165°F	Design life 10,000 hrs
4. Apollo	25 to 30 volts D.C.	115VAC 3 ϕ 400 Hz $\pm$ 7 Hz 115 $\pm$ 2.5 VAC (no load) -1 115 $\pm$ 1 VAC (100% load) -1.5 1250 V-A	47	--	
5. Gemini	28 volts D.C.	26 VAC 400 Hz Single phase	--	--	
6. S-IVB P.U.	+28 volts D.C.	115 VAC 400 Hz Single phase	--	--	
7. S-IVB Chilldown Inverter	+56 Volts D.C.	115 VAC 400 Hz $\pm$ 1% Delta output 3 ϕ Quasi-square wave 1500 V-A	--	--	

FOLDOUT FRAME 1

FOLDOUT FRAME 2

assumed that this capability is utilized in the power and weight tables which are given in sections 7.9, 7.10, and 7.11.

The Inverter from the MOL Program would probably be satisfactory for use with the Delta gyro package, however, its maximum overload capability for the 3 phase wye connection is 65 V-A which would be marginal for the Delta package which requires 68 V-A for starting. In any case, it offers more in capability than would actually be required for the WACS. The Gemini inverter package is single phase and therefore unusable. The Delta inverter would be a better choice for use with the Delta gyro package, however, more qualification testing would be necessary.

The Agena inverter with the output circuit changed appears to be the best off-the-shelf solution at this time. However, a smaller inverter would be used if found or designed. In the weight and power tables which follow the lower entries are for an optimum static inverter and the higher entries are for the Agena inverter. Of the other inverters, the Apollo inverter is too large in volt-ampere capacity and weight. Neither of the S-IVB inverters are acceptable because of phase, input voltage, and waveform considerations. The search for a qualified static inverter which more closely fulfills the system requirements should be extended.

7.5 Control Electronics Assembly

The electronics package is a specialized piece of equipment which will be designed specifically for the WACS. The operation of the package is discussed in paragraph 7.1. The spatial amplifiers presently used in the flight control computer could be repackaged with the following equipment to comprise the WACS electronics package:

	<u>Number</u>
Power Supply	2
Rate Amplifier Assembly Channel A	1
Rate Amplifier Assembly Channel B	1
G. G. Horizon Sensor Channel A	1
G. G. Horizon Sensor Channel B	1
Shaping Networks (Rate)	2

	<u>Number</u>
Relay Assembly	1
Solar Orientation Horizon Sensor Channel A	1
Solar Orientation Horizon Sensor Channel B	1
Sun Sensor Networks	2
Integrator Networks A	1
Integrator Networks B	1

The equipment would be assembled in a package that would have the following estimated characteristics:

Size:	22 inches x 18 inches x 6 inches
Weight:	20 - 25 pounds
Power Dissipation:	75 - 95 watts

7.6 Power Distributor

The power distributor will direct power and sequencing signals to the various pieces of equipment of the WACS, as commanded by the R. F. Command System and the Astronaut Manual Control Panel. If the number of commands dictates a decoder could be included in the design of the power distributor. It will be constructed with the same design approach as the power distributors presently used on the S-IVR. The estimated size and weight would be 20 inches x 16 inches x 6 inches and 20 pounds.

7.7 Sun Sensors & Switches

A market survey was performed to determine if sun sensors are available off-the-shelf that could provide pitch and roll attitude error for the WACS during the solar orientation phase of the mission. The results of this survey may be found in table 7-3. A sun sensor which has a linear field of view of ± 10 degrees to ± 15 degrees and an acquisition field of view of ± 50 degrees or greater would be satisfactory to meet the WACS requirements for case 3 or for case 5 if the pitch sensors are mounted on the solar panels. Either of two Bendix wide angle sun sensors, P/Ns 1771858 or 1818787, would be satisfactory with the 1771858 being the better component for the WACS application. For case 5 with the pitch sensors mounted on the stage, a linear field of view in excess of 104 degrees is required.

TABLE

SUN SURVEY - SUN SENSORS & SWITCHES

	HONEYWELL FINE SUN SENSOR SEE DOCUMENT BNC 67-51 DATED APRIL 1967	BENDIX WIDE ANGLE SUN SENSOR 1771858	BENDIX WIDE ANGLE SUN SENSOR 1818787	RESEARCH SUN SENSOR 1771857	RESEARCH FINE SUN SENSOR 1771857	RESEARCH FINE SUN SENSOR 1771857 (SUN SWITCH)	BAILEY BROS. FINE SUN SENSOR ASS'Y. SS-100	BAILEY BROS. FINE SUN SENSOR FE-5 FE-5A	BAILEY BROS. SWITCHING SENS. TE-4, TE-4A TE-4B	BAILEY BROS. COSMOS CS-2
POINTING ACCURACY	± 1.7 arc sec max.				± 1 sec of arc		± 1 arc min. If FE-5 are used. ± 2 arc min. If FE-5A are used	± 1 arc min. for FE-5 ± 2 arc min. for FE-5A		$\pm 5^\circ$ over range of 90° to 180° look- out
TEMPERATURE STABILITY	± 0.05 arc sec/ $^\circ$ F nominal			$\pm 1.5^\circ$ / $^\circ$ F to $+70^\circ$ F	$\pm 0.5^\circ$ / $^\circ$ F to $+70^\circ$ F at 10 minute calibration cycle					
WEIGHT	2 1/2 pounds	3 oz.	3 oz.		12 oz.		6 oz. (170 gms)	0.5 oz.	0.5 oz.	0.5 oz.
POWER, PEAK	18.3 watts	0	0		0	0	0	0	0	0
POWER, AVERAGE	4 watts	0	0		0	0	0	0	0	0
RELIABILITY										
LIFE	27,500 hrs									
DIMENSIONS	2 packages Op-Mech 7 1/4" H X 8 1/2" W X 6 1/2" D. Electronics 4"x6"x8"	2" dia x 2 1/4" long	2" dia x 2 1/4" long		2 1/2" dia x 3 3/4" long		2 3/4" x 1 3/4" x 2"	15/16" x 1" x 1"	15/16" dia x 1"	15/16" dia x 1"
RESPONSE	Yes	No	No	No		No	No	No	No	No
FIELD OF VIEW	Fine field (each axis): ± 2.5 deg $90^\circ \times 90^\circ$ arc min. Medium field (each axis): ± 20 deg.	Total field: 180° Hemisphere Linear field (each axis): $\pm 10^\circ$. Angles greater than 10° , output increases approx. as a cosine function.	Total field: 180° hemisphere Linear field (each axis): $\pm 10^\circ$. Angles greater than 10° , output increases approx. as a cosine function.	Total field: $\pm 10^\circ$	9 degrees	Wide to order $\pm 15^\circ$	$\pm 15^\circ$	$\pm 15^\circ$	Wide to order $\pm 15^\circ$ For a proportion- al device, switches at level shown above	Wide to order $\pm 15^\circ$
SPECTRAL RESPONSE	.8 to 1.0						Nominal peak at 0.8 microns	Nominal peak at 0.8 microns	Nominal peak at 0.8 microns	
FOLDOUT FRAME										

FOLDOUT FRAME 2

TABLE 7-3 MARKET NO. BY SUN SENSORS & SWITCHES (continued)

	HONEYWELL FINE SUN SENSOR SEE DOCU- MENT HRC 67-51 DATED APRIL 1967	BENDIX WIDE ANGLE SUN SENSOR 1771858	BENDIX WIDE ANGLE SUN SENSOR 1818787	BENDIX SUN SENSOR 1771870	BENDIX FINE ANGLE SUN SENSOR 1818737	BENDIX SUN SENSOR 1771872 (SUN SWITCH)	BALL BROS. FINE SUN SENSOR ASS'Y, SS-100	BALL BROS. FINE EYE SUN SENSOR FE-5 FE-5A	BALL BROS. SWITCHING EYE TE-4, TE-4A TE-4E	BALL BROS. COURSE EYE CE-3
SUN PRESENCE SENSING	Yes	No	No	No	No	Primary Function	No	No	Primary Function	No
DEVELOPED FOR:	AOSQ & ATM		Apollo Program				Aerobee rockets & satellites	Aerobee rockets & satellites		
OUTPUT:	Redundant two axis	Two axis continu- ous from -90° to +90° in each axis	Two axis Output -10° to +10° Output -90° to -10° +10° to +90°		Two axis	Indicates which half of a predetermi- ed field the sun may be in	Two axis	Single sensor only		Single sensor only
TEMP. RANGE	30°F - 75°F (opto- mechanical unit) 0°F - 95°F (Electronics)	-70°C to +50°C.					-20°C to +85°C	-20°C to +85°C	-20°C to +85°C	-20°C to +85°C

FOLDDOUT FRAME 1

FOLDDOUT FRAME 2

The sun presence signal which is used to start the timer unit will be obtained by using a solar switch. Of the switches surveyed, the Ball Bros. Switching Eye TE-4A series looks the most promising, however, the field of view of ± 9 degrees would be marginal for large dead bands. For large dead bands it would be desirable to extend the field of view to ± 15 degrees or ± 20 degrees.

7.8 Installation

An investigation was made to evaluate possible positions of gravity gradient horizon sensor heads on the vehicle. In general, tracking head positions forward of the forward skirt of the S-IVB were considered undesirable because the field of view of the horizon sensor was obscured by hardware attached to the orbital workshop. Limited capability is provided by mounting the horizon sensors on the forward skirt of the S-IVB. In some positions, clearance is provided after docking has been accomplished, but the field of view is obscured by vehicles moving in close proximity to the workshop during the terminal phase of rendezvous. The aft skirt of the S-IVB provides the most desirable position from a look angle standpoint as shown in figure 7-3. Further investigation must be performed to verify that exhaust plumes from the APS modules will not obscure the field of view of the horizon sensors.

The yaw solar mode attitude horizon sensors are mounted along fin plane II with one head in the aft skirt pointed aft, and one head in the forward skirt pointed forward. The primary reason for the separation between the two heads is to clear the solar panels when they are extended.

During phase I the alternate locations for the electronic boxes, besides the tracking heads, were investigated. The primary alternatives were to locate the equipment in the aft skirt to minimize interconnection between the packages or to move a major portion of the equipment to airlock or MDA in order to benefit from a more favorable environmental atmosphere during the period of habitation. Two configurations were investigated:

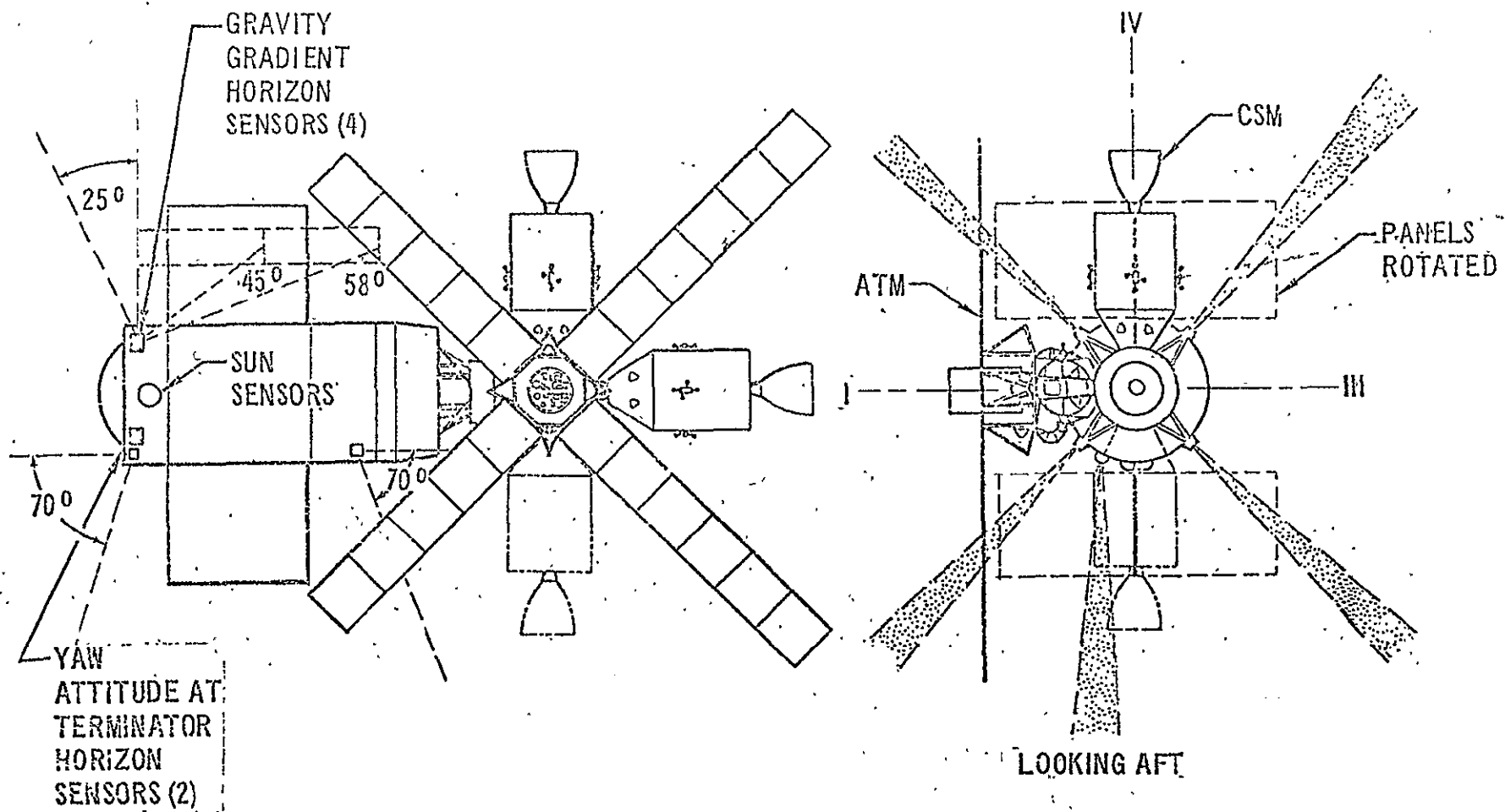


FIGURE 7-3. SENSOR HEAD LOCATIONS

1) All of the phase I equipment shown in figure 7-4, on the workshop side will be located in the S-IVB aft skirt; and, 2) the horizon sensor heads and electronics, the heater control unit and the APS will be located in the S-IVB aft skirt. All other equipment including an additional heater control unit will be located in the airlock. The following is a comparison of the two installations:

- 1) The following connections are considered to be equivalent for either configuration.
 - a) Power distribution power and control signals to:
 - Rate gyro No. 1 and No. 2
 - Static Inverter No. 1 and No. 2
 - Electronics package
 - b) Static inverter power to rate gyros No. 1 and No. 2
 - c) Rate gyro No. 1 to electronics package
 - d) Rate gyro No. 2 to electronics package
 - e) Heater control unit aft to APS and horizon sensor heaters
 - f) Heater control unit forward to WACS electronics unit forward
- 2) A comparison of differences between configurations No. 1 and No. 2 may be found in table 7-4.

The difference in weight between the two locations provides a small advantage for mounting all equipment in the S-IVB aft skirt. Other considerations would be testing, environmental control, and electrical interfaces. If the system were mounted in the S-IVB aft skirt, it would be checked out as an integral part of the vehicle electrical system, and would eliminate the need for a WACS simulator for vehicle checkout, thus simplifying interface problems. For case 2, a power distributor simulator could be used for the airlock checkout. Thermal control in the airlock is only provided during the periods of habitation, which still leaves a heater requirement for the WACS equipment during the storage period. It is therefore recommended the WACS electronics be installed in the aft skirt of the S-IVB vehicle.

A similar study was not performed for phase II since the results were expected to give the same conclusion as for phase I.

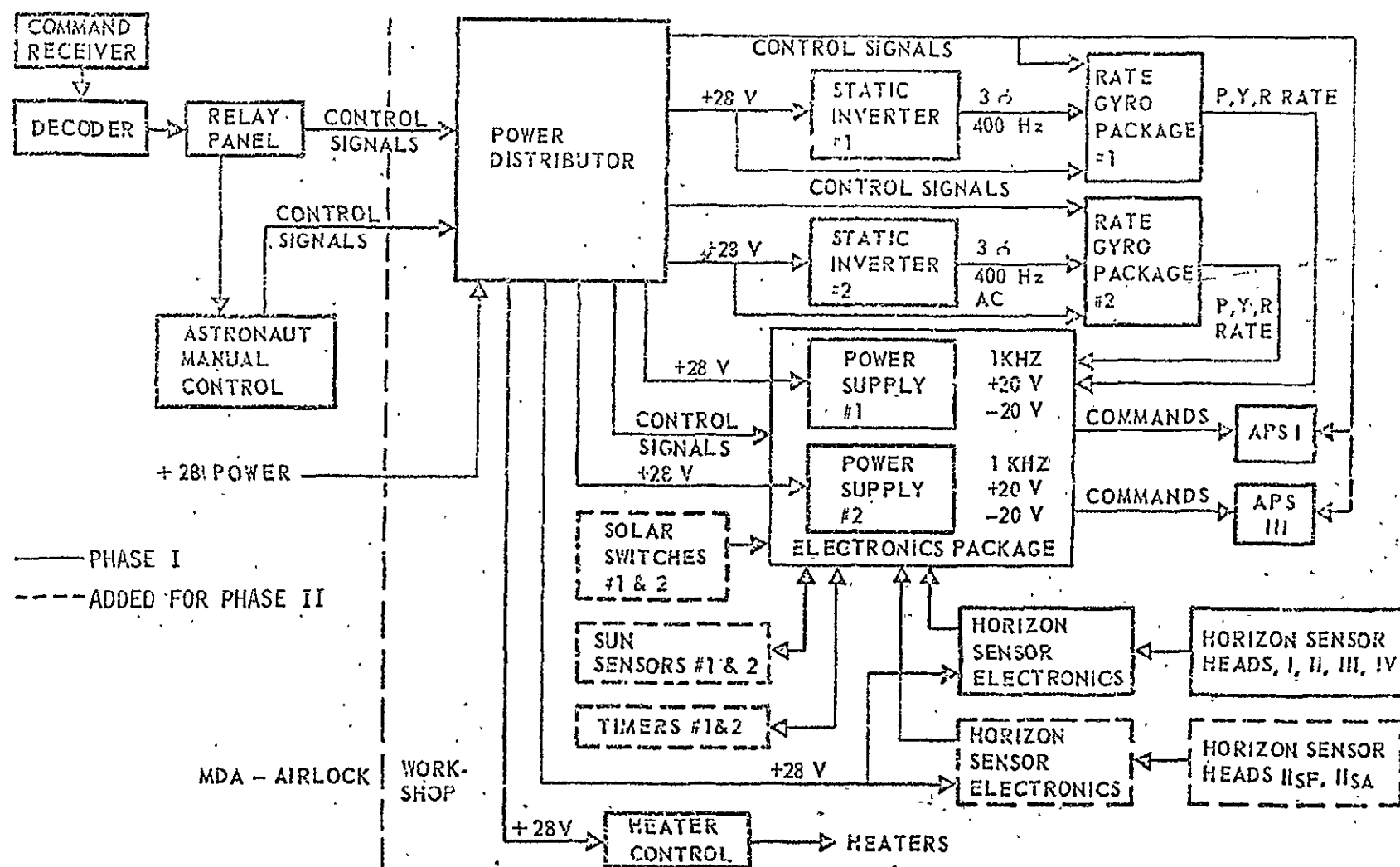


FIGURE 7-4. WACS POWER DISTRIBUTION & CONTROL

CONFIGURATION #1
(most equipment in the aft skirt)

CONFIGURATION #2
(least equipment in the aft skirt)

	Length Ft.	Gauge	Number of Wires	Weight lbs.	Length Ft.	Gauge	Number of Wires	Weight Lbs.
Relay panel to power distributor	65	20	10	4.6	15	20	10	1.1
Astronaut control to power Distributor	65	20	7	3.2	5	20	7	.2
Power distributor to horizon sensor electronics	5	20	2	.1	65	20	2	.9
Horizon sensor electronics to electronic package	5	20	8	.3	65	18	8	2.6
Electronics package to APS #1	15	20	10	1.1	65	18	10	6.5
Electronics package to APS #2	15	20	10	1.1	65	18	10	6.5
+28V to the power distributor	65	8	1	4.5	10	14	1	.2
+28V to aft heater control	65	6	1	7.5	65	8	1	4.5
+28V to fwd heater control	0		0		15	18	1	.1
Fwd heater control unit				2.0*				7.0

TOTAL:

24.4

29.6

Equipment in Transit: Power Distributor, Heater Control Unit, Two Static Inverters, Two Rate Gyro Packages and the Electronics Package

* Weight increase for Aft Heater Control Unit

TABLE 7-4
LOCATION WEIGHT COMPARISON

7.9 Power & Control Signal Distribution

The airlock contains a command system which was used on the Gemini program and Apollo Block I. This system consists of a single box which contains two redundant receivers, a decoder and a relay panel which contains 160 relays of which 90 are in use. It is recommended that all control signals which originate outside the workshop be channeled through this command link. Manual controls, which will allow the astronauts to change mode if necessary, will be provided on a panel mounted in the airlock as a back-up for the command system. This panel will also allow the astronaut to change attitude and initiate malfunction correction action. In general most of these commands will be inserted into the system through the use of toggle switches. Power (+28V) from the airlock power distribution system will be supplied to the power distributor which will switch 28V power to the static inverters, rate gyro packages, heater control package, electronics through redundant relay circuits. Control signals will be fed to the rate gyro packages and the electronics package. Control signals sent to the gyro will allow proper sequencing of pitch, yaw, and rate outputs depending on the appropriate mode utilized in the control system. The control signals sent to the electronics package will energize relays which control configuration bias signals, system gains for various active modes of operation, switching of out-of-tolerance gyros, switching to standby and storage modes. Control signals to the APS will actuate the cartridge valves. The 28V power sent to the electronics package will be sent to two redundant power supplies; one of which supplies power to System A and the other to System B.

Table 7-5 lists the anticipated power usage of the attitude control system during various modes of operation. The operational mode considers the WACS completely activated and performing in one of the various active modes. Storage option number 1 is based on the assumption that all equipment in the system except the gyro package will be kept operating at all times in order to keep the equipment warm. It is anticipated in this mode of operation the gyro will require an external heater to maintain an elevated temperature and to conserve the heater circuits internal to the gyro package which maintains

AVERAGE POWER (WATTS)

	OPERATIONAL MODE		#1 STORAGE - EQUIPMENT OPERATING		#2 STORAGE - EXTERNAL	
	APOLLO ORIENTED SYSTEM	ATM ORIENTED SYSTEM	APOLLO ORIENTED SYSTEM	ATM ORIENTED SYSTEM	APOLLO ORIENTED SYSTEM	ATM ORIENTED SYSTEM
A. SAME AS PHASE I EQUIPMENT						
1. GYRO PACKAGES (2 SYSTEMS)						
ELECTRONICS	39	77 **	0	0	0	0
HEATERS (INTERNAL)	76	105	0	0	0	0
HEATERS (EXTERNAL)	0	0	38	52	38	52
2. STATIC INVERTERS (TO DRIVE SIX GYROS) -						
ELECTRONICS	46-65	(IN GYRO PACKAGE)	28-33	0	0	0
EXTERNAL HEATER					A *	0
3. HORIZON SENSOR - ELECT.	6-7	6-7	6-7	6-7	0	0
EXTERNAL HEATER	0	0	0	0	B *	G *
4. APS VALVES	3-4	3-4	NIL	NIL	NIL	NIL
B. CHANGED FROM PHASE I TO PHASE II						
5. POWER DISTRIBUTOR - EXTERNAL HEATER	5-10	5-10	5-10	5-10	C *	H *
6. ELECTRONICS PACKAGE	75-95	75-95	75-95	75-95	0	0
EXTERNAL HEATER					D *	I *
7. APS HEATERS (2 MODULES)	150-195	150-195	154-259	154-259	154-259	154-259
C. ADDED FOR PHASE II						
8. SOLAR SENSOR (2 UNITS)	0	0	0	0	0	0
EXTERNAL HEATERS			NIL	NIL	NIL	NIL
9. HORIZON SENSOR - ELECT.	6-7	6-7	6-7	6-7	0	0
EXTERNAL HEATER	0	0	0	0	E *	J *
10. TIMER (2 UNITS)	10	10	10	10	0	0
EXTERNAL HEATER	0	0	0	0	F *	K *
TOTAL (CASE 5)	416-508	437-510	322-459	308-440	262-367	276-381

* Estimated Wattage for A + B + C + D + E + F and G + H + I + J + K is 60 watts.

** Includes 47 Watts for Static Inverters in Gyro Packages.

TABLE 7-5 WACS AVERAGE POWER REQUIREMENTS

gyro temperature of approximately 180°F which is far in excess of temperature requirements during storage. Storage option number 2 is based on the premise that it is not desirable to keep the equipment operating at all times and that an external heater should be applied to all equipment to provide a temperature to maintain the equipment at a temperature which would be on the lower limit of the specification values for that equipment.

The difference in average power resulting from the additional phase II equipment was small. The most significant changes in power consumption are those for thermal conditioning which resulted from the more detailed analyses described in section 6. These are a 65-110 watt reduction for APS module heaters during the solar oriented operational phase, and the possible reduction of 106 watts during the gravity gradient storage phase.

A load profile for the complete mission is shown in figure 7-5. This profile shows that power consumption for worst case APS operation rises above the average power dissipation level by 50 to 70 percent. These pulses are of short duration and if averaged over one orbit represent an average power dissipation of approximately 20 watts. The primary design consideration is to have a power source that will supply the required current without excessive voltage drop. This can be supplied by the source in the airlock without difficulty.

A system with two APS modules was used as a basis for table 7-5. The effect of added modules on power requirements is as follows:

Active WACS with two APS PODS	416 - 508 watts
Additional power for tanker modules 2A and 4B*	150 - 175 watts
TOTAL:	566 - 683 watts
Additional power for tanker modules 4A and 2B*	150 - 175 watts
	716 - 858 watts

* Tanker numbers from table 6-4.

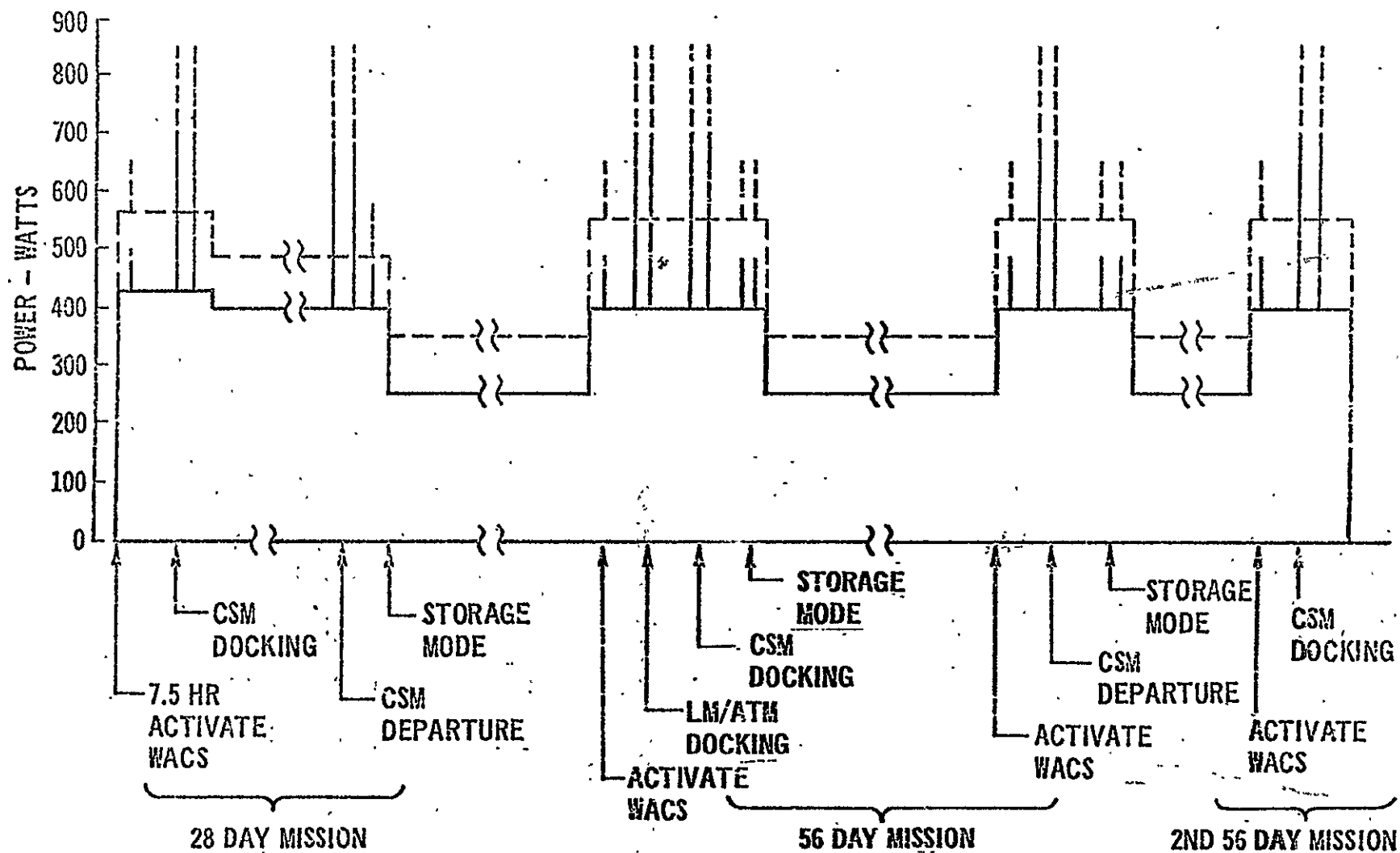


FIGURE 7-5. WACS LOAD PROFILE

	APOLLO ORIENTED SYSTEM		ATM ORIENTED SYSTEM	
	WEIGHT (TOTAL) POUNDS	SIZE (EACH UNIT) INCHES	WEIGHT (TOTAL) POUNDS	SIZE (EACH) INCHES
A. SAME AS PHASE I EQUIPMENT				
1. RATE GYRO PACKAGES (2 APOLLO TYPE) (6 ATM TYPE)	46	14 X 6 X 7	84 *	12 X 8 3/4 X 5 1/4
2. STATIC INVERTERS	8-36	3 X 3 X 2 TO 11 X 10 X 3 1/2	* IN GYRO PACKAGE	
3. HORIZON SENSOR				
(A) ELECTRONICS BOX	7	8 X 6 X 3	7	8 X 6 X 3
(B) 4 TRACKING HEADS	8	6 X 3 X 3	8	6 X 3 X 3
4. WIRING	55-65		60-70	
B. REVISION FROM PHASE I TO PHASE II				
5. POWER DISTRIBUTOR	15-20	20 X 16 X 6	15-20	20 X 16 X 6
6. ELECTRONICS PACKAGE	20-25	22 X 18 X 6	20-25	22 X 18 X 6
C. NEW EQUIPMENT FOR PHASE II				
7. SOLAR SENSORS (2 UNITS)	0.4	2 DIA. X 2 LONG	0.4	2 DIA. X 2 LONG
8. HORIZON SENSOR				
(A) ELECTRONICS BOX	7	8 X 6 X 3	7	8 X 6 X 3
(B) 2 TRACKING HEADS	4	6 X 3 X 3	4	6 X 3 X 3
9. SOLAR SWITCHES (2 UNITS)				
10. WIRING	5-10		5-10	
11. TIMER (2 UNITS)	14	6 X 4 X 3	14	6 X 4 X 3
TOTAL	189-242		224-249	

TABLE 7-6. WACS WEIGHT AND SIZE ESTIMATES, PHASE II

TABLE 7
RECOMMENDED ELECTRONICS SYSTEM

Component	Type	Approximate Weight (Pounds Each)	Estimated Cost Not Considering Qualification Testing (each)	Estimated Time of Procurement or Fabrication from ATP	Estimated Qualification Testing Cost
Horizon Sensor	Modified Lunar & Planetary Horizon Sensor (similar to Barnes Engr. Model #13-162)	15	\$125,000 (4 heads & electronics) \$40,000 (Horizon Simulator)	10 - 11 mo.	* \$25,000
Rate Gyro	Apollo Type (Similar to Minn. Honeywell Part #GG-362)	(2 ²³ req'd)	\$105,000	10 mo.	* \$35,000
Electronics Package	New Development	20	An ECP will be prepared	10 mo.	See ECP
Static Inverter	Modified Agena Inverter or smaller if can be found (similar to Gulton FM417D)	18 (2 req'd)	\$12,000	10 mo.	* \$20,000
Power Distributor	New Development	12	An ECP will be prepared	10 mo.	See ECP
TOTAL (Black Boxes)		129			
Wiring		60			

TOTAL WEIGHT: 189

* These are ROM quotes pending further definition of testing program for extended life testing, etc.

7.10 Weight and Size

Changes in weight resulting from the phase II additions are small. Table 7-6 includes a weight decrease of 28 pounds for an optimized static inverter and small increases in weight for the electronics box, power distributor, horizon sensors, solar sensors, timers and wiring.

7.11 Recommended System

Based on the information gathered during phase I, the recommended WACS is shown in table 7-7. The WACS System will be designed in kit form which is separate from the flight system, and will allow installation into the vehicle with a minimum impact on the boost flight equipment.

The following equipment would be necessary, in addition to that listed in table 7-7, to implement the case 5 configuration of phase II:

- 1) Two two-axis solar sensors
- 2) Two timer packages
- 3) Two solar switches
- 4) One horizon sensor system with two heads

7.12 Testing

Testing of the electronics for the WACS falls into the following categories:

- 1) Augmentation of black box qualification testing previously performed as part of another program testing.
- 2) Design development testing of a black box newly designed for the WACS.
- 3) Qualification testing of a black box newly designed for the WACS.
- 4) Thermal-Vacuum testing of the complete electronics system for 28 days to verify in-orbit operation of the system.

- 5) Complete engineering evaluation test of the total electronics system to verify compliance with engineering design requirements. The system will be activated and put through a test profile consisting of all in-orbit maneuvers and all anticipated malfunction modes with the resulting response evaluated with regard to over-all mission objectives.

The tests to be performed relative to specific prices of hardware are shown in table 7-8.

	Number Required For Flight System	Tests Required	Comments
1. Rate gyro package	2	1, 4, 5	GFE Part
2. Horizon Sensor System #1 (4 heads)	1	1, 2, 4, 5	GFE Part
Horizon Sensor System #2 (2 heads)	1		
3. Sun Sensors	2	1, 4, 5	GFE Part
4. Static Inverter	2	2, 3, 4, 5	
5. Power Distributor	1	2, 3, 4, 5	
6. Electronics Package	1	2, 3, 4, 5	
7. Timer Package	1	2, 3, 4, 5	
8. Heater Control Package	1	2, 3, 4, 5	
9. Solar Switch	1	1, 2, 4, 5	Specification Control Drawing

TABLE 7-8

TESTING REQUIREMENTS FOR WACS ELECTRONIC EQUIPMENT

8.0 PROPULSION SUBSYSTEM

The selection of an orbital workshop attitude-control propulsion system has been significantly constrained by the early mission flight date. The development of a 23-month space lifetime system within the time span allowed (see section 1.1) clearly indicates that complete life testing is impossible and the use of proven hardware is highly desirable. In several instances, more efficient, lighter weight, or simplified system designs have been sacrificed for the increased confidence that proven components give in meeting the flight date. Component development has been eliminated or reduced to that in progress for other programs consistent with the mission schedule. These developments are primarily required for suitable thrusters and tankage.

In preparation for the propulsion system selection, cursory parameteric analyses identified the bipropellant, monopropellant, compressed gas or combinations thereof as possible system contenders. Although the compressed gas system carries an unacceptable weight penalty if used alone, the mission impulse requirements profile may permit combining this system with a liquid-propellant system to reduce thermal control power requirements, avoid materials corrosion problems, or permit use of qualified hardware.

As an initial analysis, the existing S-IVB auxiliary propulsion system (APS) was investigated for possible mission use. Both Saturn IB and Saturn V configurations were reviewed from the aspect of propellant capacity, component lifetime, corrosion problems, growth potential, new modifications, and overall confidence of mission success. Monopropellant systems were designed within existing module envelopes and new tankage was considered only if required.

Because electrical power availability was considered critical throughout the mission, methods of reducing active thermal control needs were sought. Propellants with lower freezing points than that of N_2O_4 were sought to depress the system-operating-temperature lower limit for bipropellants. Monopropellant systems, in themselves, necessitated propellant storage

investigations, and freezing point depression methods were also considered.

Attitude control analysis results, which are presented in section 4.1.4 established minimum impulse bit (MIB) requirements of 0.6 pound-second or less. This eliminated the existing APS thrusters which have a MIB of 7.5 pound-seconds. A thruster survey was conducted to identify those thrusters having the required operating characteristics which could be available within the mission schedule. Both bipropellant and monopropellant thrusters were reviewed.

System pressurization over a long duration was a major concern. Pressurant leakage and component reliability were considered in system design and selection. Proposed systems were analyzed for helium leakage rates and the advantages of nitrogen established. Blowdown system simplicity and reliability merited consideration, even in view of a potentially heavier system than one with pressure regulation.

The detachable module concept for the attitude control systems was selected for the orbital workshop to maintain an independent propulsion system that facilitates development, qualification, checkout, manufacturing, and replacement, if required. The extended orbital operation of the workshop requires active and passive thermal protection of the propellant systems to prevent freezing. Compact system modules, with short propellant lines and buried engine valve installations, greatly simplify the temperature control problem. Detachable modules permit complete system testing in a facility independent from the orbital workshop main stage. Each system produced can be checked for proper operation before installation on the orbital workshop stage.

The following sections present the results of efforts made to permit selection of a realistic auxiliary propulsion subsystem for the OWS attitude control system. System tradeoffs for optimization of weights, reliability, and cost were made for systems which were compatible with the mission schedule. Phase I studies and Phase II studies are described separately.

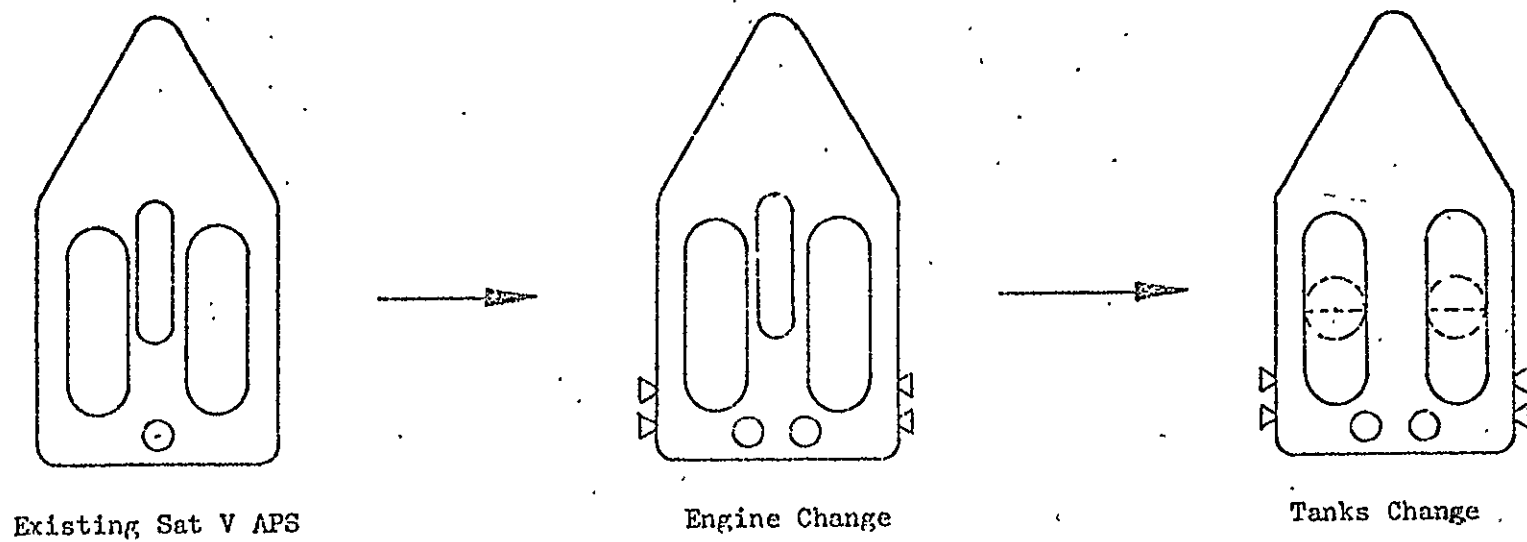
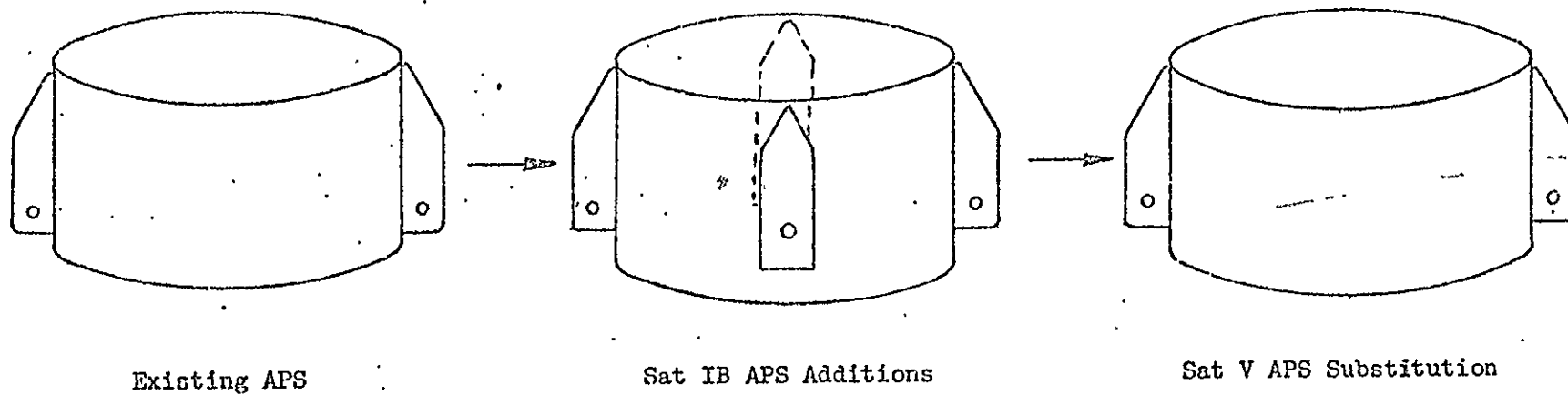


Figure 8-1 S-IVB/OWS APS EVOLUTION

8.1 Phase I Bipropellant Systems

Figure 8-1 depicts the evolution of Phase I system concepts. Initial efforts evaluated the adequacy of the existing S-IVB/IB APS installation. Propellant capacity considerations led to installation of manifolds between the modules, thereby utilizing propellant allocated for unsymmetrical usage. When this method did not provide adequate total impulse, four S-IVB/IB APS modules, mounted on the aft skirt, were considered.

It became evident that there was no advantage to utilizing existing APS modules for boost and passivation, and a separate system for docking and coast. New engines were required for coast; therefore, thruster redundancy was not available, even with four three-engine modules. Analysis indicated that the mission could be accomplished using only low-thrust engines, even with a single-engine failure. After establishing the desirability of a single configuration, thermal and structural investigations revealed that the S-IVB/IB APS is difficult to insulate, and attachment of two additional modules involves extensive aft skirt modifications.

A preferable method of obtaining increased total impulse capability is to exchange S-IVB/V for S-IVB/IB modules. Saturn V APS evaluations, in light of a 23-month mission requirement, considered three major areas; thrusters, tankage, and pressurization subsystems. For control-dynamics reasons, replacement of existing engines was made with low-impulse-bit engines. Six were used per module for reliability. With the performance of Teflon bladders over long-term storage still not thoroughly proven, metal-bellows tankage permitted greater confidence in meeting the launch date. Modifications to the existing APS systems within existing module envelopes resulted in a realistic, feasible system. Beyond this point, little new system design or evaluation was pursued, since this would involve greater development testing, and an increased risk of not meeting the flight date.

Evaluations of bipropellant tankage, thrusters, and pressurization systems are discussed in the following sections, followed by discussion of the evaluation of several system concepts utilizing these components.

8.1.1 Bipropellant Tankage

The requirement for bipropellant tankage is propellant storage for 23 months, during which period the system duty cycle ranges from steady-state engine operations to months of inactivity. The design criteria are as follows:

1. Operational lifetime of 23 months
2. Compatible with S-IVB APS system envelope and operating pressures
3. Man-rated hardware at time of flight
4. 36,000 pound-second impulse capability for 2 modules

Satisfactory storage of bipropellants for attitude-control operation during one to two years of space flight has not been conclusively demonstrated to date. Missions such as Lunar Orbiter have retained residual bipropellants for several months after the system has accomplished its functional requirement. Successful engine steady-state operation has been demonstrated at Lunar Orbiter mission completion; however, only limited data are available and questions of bladder permeation, pressurant gas entrainment, and degree of nitrogen solubility have not been fully resolved. Some form of tank-life testing is required for any tankage qualification, be it bladder or bellows.

To minimize structural development costs, the selected tankage should fit within the envelope of the existing S-IVB/V APS module with a minimum of structural modifications. Operation compatible with present APS pressure limits permits the retention of significant number of existing components.

Table 8-1 presents the DAC review of the available off-the-shelf metal bellows and bladder tanks considered for the orbital workshop APS. The term "off-the-shelf" denotes that the tank is in development or qualification and/or in use for space applications. The tanks considered will meet the envelope requirements of the S-IVB/V APS fairing assembly. However, the majority of tanks listed will require various degrees of redesign and additional qualification testing to meet the WACS requirements. The following tanks are qualified for Saturn IB and Saturn V environments, and are currently available hardware. Each tank meets the WACS propellant capacity requirement and other criteria, except for the service life of 23 months.

THIS CHART SHOWS A SUMMARY OF THE
WHICH HAVE BEEN DEVELOPED OR

[illegible]

* CONFIG: 1= CYLINDER / DIE END 2= CYLINDER MIDDLE END 3= JAW END

THE PROPELLANT TANKS
CURRENTLY IN DEVELOPMENT STAGES

[illegible]

U.S. AIR FORCE

C. 10. ENDS 6 P6 = LONG FLARE/ANS THE = TYPICAL / ANS 10 P6 / LIMITED TO 70% 100% 200%

Table 1

SUMMARY OF EXISTING AND PROPOSED DAMS

FOLDOUT FRAME

FOLDOUT FRAME

The S-IVB/V oxidizer and fuel tank is similar to the LEM oxidizer positive-expulsion tank and the Apollo tanks. The tank is made from 6Al-4V titanium forgings with a design burst pressure of 550 pounds per square inch. The bladder is attached at both ends with flanges, propellant being expelled by collapsing the bladder around a perforated stand pipe mounted on the tank axis. The tank volume of 4089 cubic inches meets the capacity requirements established for the WACS.

The S-IVB/IB metal-bellows tank is an integral assembly consisting of a cylindrical shell, 10.5 inches in diameter, separated into two compartments by a nitrogen storage sphere. Each compartment contains a positive-expulsion metal bellows consisting of 0.006 inch thick stainless steel convolutions, one bellows being used for fuel and the other for oxidizer. A metal liner between bellows and tank shell serves as a guide for axial movement and as a dampener during boost phase vibrations.

After reviewing available tank data for the survey, and additional information obtained from Rocketdyne and Bell Aerospace, it is concluded that the metal bellows offers more reliability in meeting the targeted flight date. The welded, stainless-steel bellows in the positive-expulsion system are superior to bladders of flexible materials, such as Teflon, because they are not permeable, have a longer cycle life, and are not temperature-limited or sensitive over the operating range of the propellants. Thus, S-IVB/IB tanks were selected for the WACS Phase I system.

8.1.2 Bipropellant Thrusters

With the requirement to provide adequate attitude-control thrust for the orbital workshop, from booster separation through 23 months of OWS lifetime, the following criteria developed and are discussed below:

Thruster Design Criteria

1. 25 lb thrust minimum - booster separation (assuming three engines/module)
2. 0.6 lb-sec or less minimum-impulse bit
3. 800 seconds steady-state operation capability

4. 700 seconds pulsed-operation capability
5. 2 years space-lifetime
6. Man-rated hardware at time of flight

Roll-control requirements during booster separation establish the engine minimum-thrust level. With a one-engine-out capability, the roll torque requirement at booster separation can be met with the remaining three roll engines operating at 17 pounds thrust each. An upper thrust-level limit is not in itself a requirement but is indirectly established as a result of minimum-impulse bit. To meet rate requirements during coast, the minimum-impulse bit must not exceed 0.6 pound-second. The larger the engine thrust, the more severe the requirements for system minimum pulse-width. As pulse-width decreases, I_{sp} decreases due to inefficient propellant combustion. Valve dribble volume, cold chamber operation, and off-mixture-ratio operation all adversely affect performance at short pulse widths in the neighborhood of 10 milliseconds. The preferable method of attaining a minimum-impulse bit, from thruster considerations, is to reduce engine nominal thrust to or near the minimum thrust requirement. This approach was taken in reviewing thrusters in the 17 to 30 pound thrust range. Normal engine operation is expected to require a maximum of approximately 700 seconds of pulsed engine operation and 800 seconds steady-state.

Of the wide variety of thrusters in various phases of development throughout the industry, the criteria of flight qualified hardware within 18 months eliminated numerous candidate engines.

Table 8-2 presents a survey of several thrusters which were considered for WACS usage. Each thruster is discussed in detail below.

The Bell 18-25 pound thruster is the latest generation of radiation-cooled engines, and is scheduled for production in quantity under an existing contract for a classified program. However, the vendor firmly stated that qualified production engines could not be made available within less than 24 months.

TABLE 8-2 BI-PROPELLANT THRUSTOR SURVEY

Manufacturer	Engine	Thrust	MIB (lb-sec)	Operating Life (sec)	Current Status-	Delivery Time From ATP (months)
Bell		18-25	.25	>1500	Classified	24+
Marquardt	R-1E	22	.23	13,200	Classified	12
	R-4D	100	.51 ± .18	198,000	LEM, SM, LO	12
Rocketspace	Beryllium	25	.20	10,000	In House	18+
	SE-6	23	.10	130	Gemini	13
	SE-7	23	.10	540	Gemini	13
	OAMS	72	.25	740	S-IVB	12
Thiokol	C-1	100	.4 ± .2	5000	NASA 7-57 Qualification	18+
TRW	URSA	25R	.2	10,000	In House	24+
		150	7.5	1000	S-IVB APS	12

Rocketdyne's many attitude-control engines were reviewed and eliminated for the reasons noted below:

The beryllium thruster is an in-house developed, radiation cooled, latest generation engine. Although there is a high level of confidence for use on the OWS, qualified engines would not be available for more than 18 months.

The SE-6 ablative-cooled Gemini engine is the same as the SE-7, except for operation at higher mixture ratio. The Gemini lifetime requirement of 136 seconds is insufficient for the OWS.

The SE-7 has been qualified for the Gemini attitude-control system. Although the SE-7 operational requirement is 557 seconds, Rocketdyne is confident that the SE-7 could meet the preliminary OWS operational requirement of 1500 seconds. However, for an engine life of 1500 seconds, the engine mixture ratio is 0.7, requiring loading of unequal volume of the propellants thereby compounding growth considerations.

The Gemini OAMS 100-pound thrust engine is ablatively cooled and also used on the S-IVB/V, but derated to 72-pounds thrust. However, the minimum-impulse bit is greater than permitted by the OWS requirements.

The Thiokol C-1 regenerative, radiation-cooled thruster is not available from production tooling for a minimum of 18 months from ATP. The minimum-impulse bit exceeds the OWS requirement.

TRW's 25-pound thrust, radiation-cooled engine is an in-house developed engine which can be available 24 months from ATP. The 150-pound thrust engine currently used on the S-IVB has a limited lifetime, and the MIB is too great for OWS application.

The Marquardt R-4D 100 pound thrust, radiation cooled engine is in production (LEM, LO, SM), available and meets the WACS requirements. However, the minimum-impulse bit is marginally high (0.51 ± 0.18 pound-second) which, therefore, would require minimum pulse-width operation at the lowest propellant utilization efficiency.

The Marquardt R-IE radiation-film-cooled engine is scheduled, under a classified contract, for completion of formal qualification testing in April 1968. The qualification test program requirements equal or exceed the OWS requirements, except for mission lifetime. The addition of a life-cycle test to the present qualification test program is necessary. However, this requirement should not delay the availability of the R-IE thruster for use on the orbital workshop APS.

The modifications necessary to meet the OWS requirements are minor in nature. These modifications include:

1. The addition of a welded connection in each propellant inlet line adjacent to each engine valve, to accomplish propellant recirculation during propellant loading.
2. The inclusion of a port in the chamber for monitoring chamber pressure.
3. The addition of heaters on the mounting flange and/or the oxidizer solenoid valve to maintain the oxidizer in a liquid state.

Tabulated data and a dimensioned outline of the Marquardt R-IE engine is shown in figure 8-2. This is the engine recommended for the workshop attitude control system.

8.1.3 Bipropellant Pressurization Subsystem

The selection of a pressurization subsystem was limited to the consideration of a regulated system, a blowdown system, and a gas-generator system. The gas-generator system would consist of a gas generator, a plenum, and a pressure switch with associated electronics. Due to the complexity of this system, a rather extensive development task would be required. This system is not considered applicable for the schedule prescribed.

The blowdown system is very reliable due to its simplicity. Due to the low pressures within the system (150 to 300 pounds per square inch absolute), the safety of such a system is very good. Also, since no regulator or other devices separate the pressurant tank from the ullage, overpressurization due to internal leakage is not a problem. The main disadvantages of the blowdown system are its size and weight. The

WORKSHOP ATTITUDE CONTROL SYSTEM MODEL R-1E ROCKET ENGINE 22 lb THRUST

WACS-49
8-22-67

THRUST

22 LBS @ 180 PSIA PROPELLANT SUPPLY PRESSURE
OPERATIONAL RANGE - 15 TO 27 LBS THRUST

COOLING

RADIATION

IGNITION

HYPERGOLIC

PERFORMANCE

STEADY STATE SPECIFIC IMPULSE - 276 SEC NOMINAL @ O/F = 1.6
PULSE PERFORMANCE - SEE CURVE
IMPULSE BIT @ 10 MS PULSE WIDTH - 0.12 LB-SEC

VALVES

ELECTRICALLY LINKED SOLENOIDS
POWER REQUIREMENTS - 35 WATTS @ 28 VOLTS DC
OPENING RESPONSE - 9 MS @ 28 VOLTS DC
VALVES CAN BE FURNISHED WITH PIGTAILS
OR MECHANICAL CONNECTOR AS REQUIRED.

LIFE

BURN TIME - 220 MINUTES OR MORE
IGNITIONS - 100,000 +

WEIGHT

2.88 LBS

CHAMBER PRESSURE NOZZLE AREA RATIO

93 PSIA

40:1

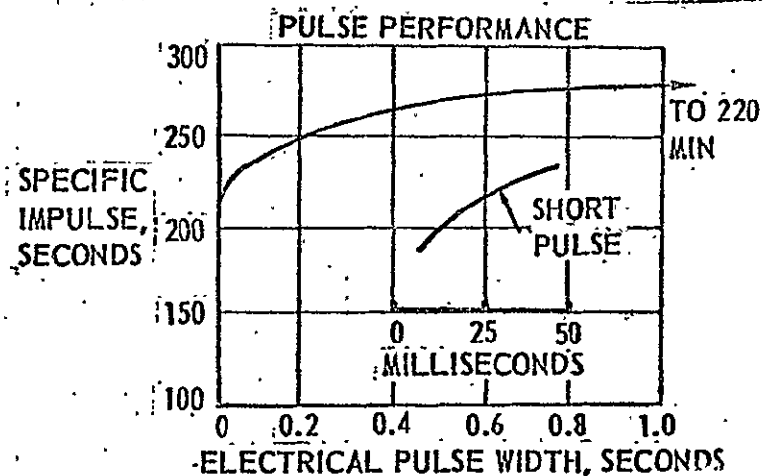
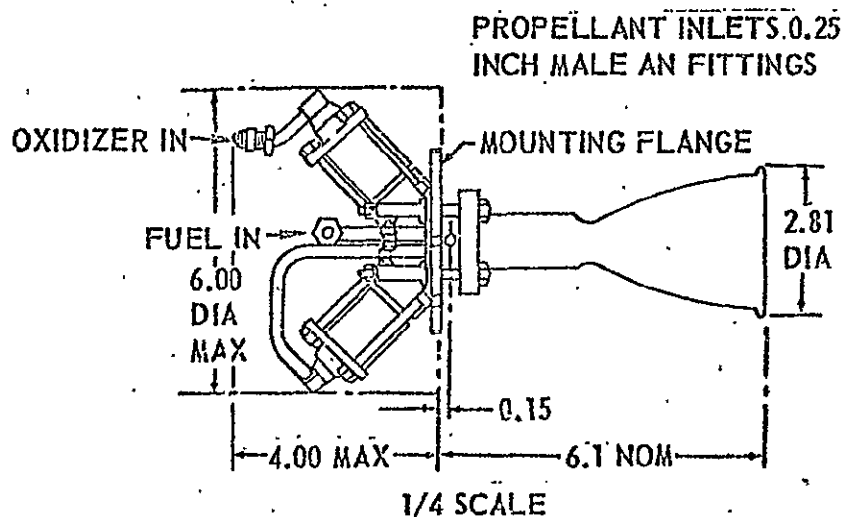


Figure 8-2 Model R-1E Rocket Engine Data

blowdown system requires an initial pressurant volume equal to the initial propellant volume. Because of these requirements, the weight penalty becomes larger as the impulse requirement increases. The use of a blowdown system is desirable, if space is available and the weight penalty can be tolerated.

The regulated system utilizes high-pressure gas and, therefore, requires less space than the blowdown system. Although the pressurant tank is of heavier construction, the overall system weight is less. The use of a regulator reduces the reliability of the system, and introduces the problem of internal leakage. By selecting regulators with low internal leakage, the regulated system is feasible for a 23-month mission. This system also has the advantage of present use in the Saturn S-IVB Stage APS.

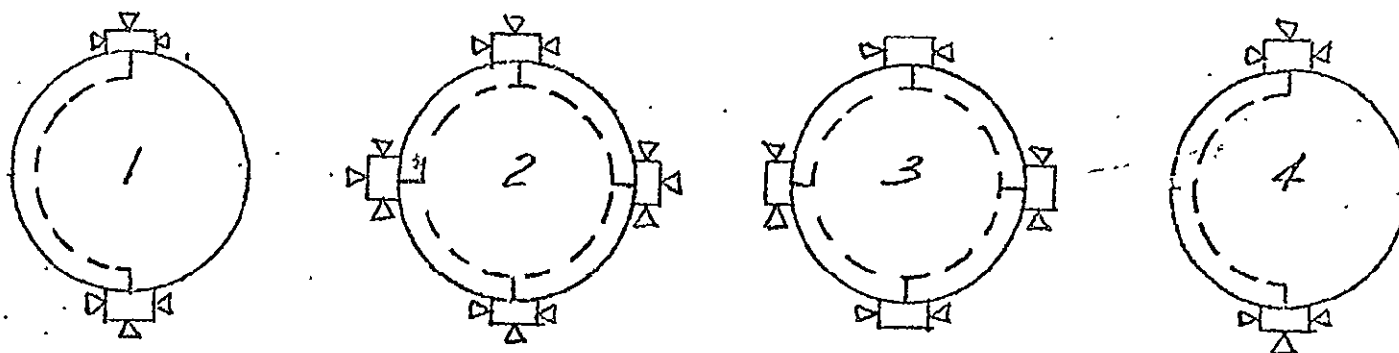
Another consideration related to the pressurization subsystem, is the selection of the pressurant gas. This selection was limited to two candidates, helium and nitrogen. At the present time, helium is utilized for the S-IVB APS modules.

The most significant criteria for selection of the pressurant is the leakage rate. For a given system, the use of nitrogen reduces the system leakage to approximately 0.4 times the leakage using helium. Since the pressurization subsystems are closed systems, the size of the subsystem is independent of the gas used. Use of nitrogen would increase the weight of pressurant, but for the systems considered the additional weight is less than 10 pounds. Therefore, nitrogen was selected as the pressurant.

8.1.4 Bipropellant Systems Installation

Figure 8-3 presents several methods of increasing the attitude-control total-impulse capability of the S-IVB/IB APS.

Configuration 1 uses the existing S-IVB/IB APS modules manifolded together. For a relatively small weight increase, manifolding increases the total usable propellant available for a given mission. Unsymmetrical propellant usage is automatically compensated via the manifold, so that excess propellant is not left in one module after depletion of the other.



	1	2	3	4
Configuration	2 Sat IB Modules	4 Sat IB Modules	4 Sat IB Modules Less Pitch Engines	2 Sat V Modules
Location	Present	90° Apart	90° Apart	Present
Dry Weight	700	1,400	1,292	826
Propellant	124	248	248	664
Wet Weight	824	1,648	1,540	1,490
Manifolding (Approx)	20	40	40	20
M. R.	1.6	1.6	1.6	1.6

Figure 8-3 Methods of Increasing Propellant Capacity

Configuration 2 utilizes four S-IVB/IB APS modules which could be manifolded together. Modules are spaced 90 degrees apart around the aft skirt.

Configuration 3 is the same as 2 except that all pitch engines have been removed. The remaining eight thrusters can accomplish attitude control about all three axes.

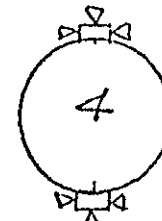
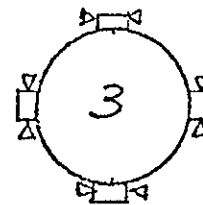
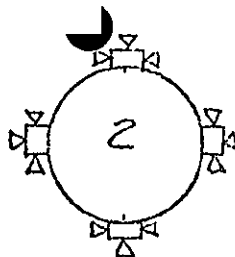
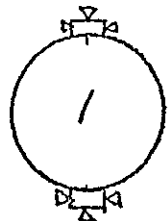
Configuration 4 uses two S-IVB/V APS modules which could be manifolded together. The ullage engine has been removed from each module. The superior propellant mass fraction of the S-IVB/V APS over the S-IVB/IB APS is evident. For similar system weights at liftoff, configuration 4 provides more than 2.5 times the propellant capability of configuration 2 or 3.

Figure 8-4 shows the total impulse and equivalent propellant loading required per module to complete the 23-month mission for both independent and manifolded cases. The advantage of module manifolding is evident. The effect of pitch-engine deletion in the independent-module case of configuration 3 is to double system weight.

In computing equivalent propellant loads per module, changes in specific impulse were accounted for during various mission phases. During boost, the present pulse-width of 0.065 second was used for pulsed operation. This corresponds to a specific impulse of 230 seconds. During the 0.6 pound-second minimum-impulse-bit operation, a specific impulse of 210 seconds was assumed. The specific impulse used for steady-state operation was 280 seconds. These engine performance figures were obtained from Marquardt R-IE engine performance curves.

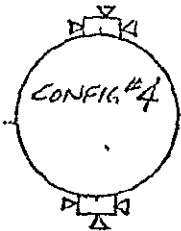
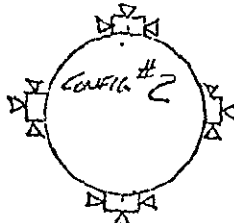
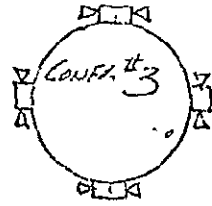
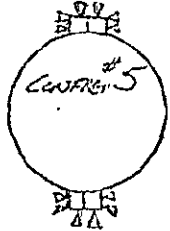
With a propellant capacity of 62 pounds per S-IVB/IB module, configuration 1 is totally inadequate. Independent module usage is not recommended for configuration 2 and is inadequate for configuration 3. The S-IVB/V module is adequate whether manifolded or not.

Figure 8-5 presents the effects of gross component failures on the various configurations. For each configuration, various failure modes were



	REQUIRED IMPULSE PER MODULE							
	Indep	Manif	Indep	Manif	Indep	Manif	Indep	Manif
o Boost & Passivation								
Steady State	2,400	1,800	1,200	900	2,400	900	2,400	1,800
Pulse	6,670	5,000	3,335	2,500	6,670	2,500	6,670	5,000
Subtotal	9,070	6,800	4,535	3,400	9,070	3,400	9,070	6,800
o Docking & Coast								
Steady State	12,500	6,850	6,250	3,430	12,500	3,430	12,500	6,850
Pulse	8,000	4,400	4,000	2,195	8,000	2,195	8,000	4,400
Subtotal	20,500	11,250	10,250	5,625	20,500	5,625	20,500	11,250
TOTAL IMPULSE	29,570	18,050	14,785	9,025	29,570	9,025	29,570	18,050
	REQUIRED PROPELLANT PER MODULE							
	Indep	Manif	Indep	Manif	Indep	Manif	Indep	Manif
o Boost & Passivation								
Steady State (280)	8.6	6.5	4.3	3.3	8.6	3.3	8.6	6.5
Pulse (230)	29.0	21.8	14.5	10.9	29.0	10.9	29.0	21.8
Subtotal	37.6	28.3	18.8	14.2	37.6	14.2	37.6	28.3
o Docking & Coast								
Steady State (280)	44.6	24.5	22.4	12.3	44.6	12.3	44.6	24.5
Pulse (210)	38.1	21.0	19.0	10.4	38.1	10.4	38.1	21.0
Subtotal	82.7	45.4	41.4	22.7	82.7	22.7	82.7	45.5
TOTAL	120.3	73.8	60.2	36.9	120.3	36.9	120.3	73.8
Propellant Available Per Module	62		62		62		332	

Figure 8-4 Impulse and Propellant Requirements by Configuration

FAILURE MODE				
INDEPENDENT:				
Tank Loss	No Control	1.33 RIPM**	RIPM	2 RIPM
Engine Loss (Off)	No Control	1.33 RIPM	RIPM	RIPM
Engine Loss (On)	No Control	1.33 RIPM	No Control	No Control
MANIFOLDED:				
Tank Loss	2 RIPM	1.4 RIPM	1.33 RIPM	1.33 RIPM
Engine Loss (Off)	No Control	1.05 RIPM	RIPM	RIPM
Engine Loss (On)	No Control	No Control	No Control	No Control
Manifold Loss	No Control	No Control	No Control	2 RIPM

* Assuming Separate Tanks for Each Engine Cluster, two Clusters/Module.

** RIPM = Required Impulse per Module Based on No Failure Mode. The Factor Provides for Sufficient Propellant to Complete the Mission in the Event of the Failure Mode Listed.

Figure 8-5. FAILURE EFFECTS ON VARIOUS CONFIGURATIONS

evaluated to determine if attitude control were lost or, if not, how much additional propellant must be initially loaded per module for mission completion. The factors shown times the required propellant per module from figure 8-4 represent this initial propellant loading. Conversely, not loading additional propellant results in premature mission termination in the event of a subsystem failure. The factors for configuration 1 are identical to those for configuration 4. A new configuration 5 was also evaluated. Configuration 5 is defined as having six thrusters per module, forming two independent clusters of three engines each. Each engine cluster is connected to its own tankage and pressurant systems, thereby providing complete system redundancy in each module. If manifolded together, complete redundancy is retained in configuration 5 by providing two independent manifold systems. This differs from the technique of configurations 2 and 3 where all modules share a common manifold.

An engine-on failure appears to create the most adverse system impact. This type of failure, if not corrected, can cause premature mission termination. Methods of correcting this type of failure are discussed in section 3.5. More detailed failure analyses, using component criticality, would be required to determine the listed failure mode probabilities.

Based on these considerations of overall system operation, capability, and availability, bipropellant evaluation henceforth concentrated on two principal systems. These are four S-IVB/IB modules manifolded, and two S-IVB/V modules with redundant thrusters and tankage. The installation of these systems is discussed in the following sections. The symbols used in the schematic diagrams of the systems are shown in figure 8-6.

8.1.4.1 Manifolded S-IVB/IB Modules

Four S-IVB/IB modules, manifolded together, provide sufficient total impulse for mission accomplishment. Changes to the propulsion system are relatively minor, being essentially an exchange of low-minimum-impulse-bit engines for the existing ones. However, thermal-control power requirements are greater for this system than for two S-IVB/V modules, primarily due to inherent design characteristics and surface-to-volume ratios.

	VENT
	PRESSURE RELIEF VALVE
	3-WAY 2-POSITION SOLENOID VALVE
	PRESSURE REGULATOR
	CHECK VALVE
	FLANGE
	2-WAY SOLENOID VALVE
	SELF SEALING DISCONNECT
	FILTER
	TEMPERATURE TRANSDUCER
	PRESSURE TRANSDUCER
	CAPPED LINE
	CARTRIDGE VALVE
	BURST DIAPHRAGM
	LIQUID LEVEL SENSOR

Figure 8-6 Schematic Symbols

The structural modifications involved in providing new supports for the two additional modules are extensive as discussed in section 9. Flight control considerations make 90-degree spacings desirable about the aft skirt circumference. This implies placing the added APS modules on Position Planes II and IV, which interferes with an ullage rocket located on Plane IV. Relocation of this APS module can be tolerated if angular offset is not excessive. Retro-fit of this system would be difficult.

A four-module system also involves changes in control system logic. Differential signal mixing is required in all axes, requiring additional switching amplifiers and associated hardware.

For the above reasons, use of four S-IVB/IB modules is not recommended for the WACS.

8.1.4.2 S-IVB/V APS Modules - Bladder Tankage

Another APS configuration, considered for use on the OWS, is the basic Saturn V APS system, figure 8-7, which consists of bladder tankage, a cylindrical nitrogen storage tank, regulator, propellant-control module and pressurization system manifolding. A component change to the pressurization system is the addition of a cartridge valve between the check valve and high-pressure storage tank. The valve is activated at liftoff, and reduces leakage after launch. This change will increase the reliability of the nitrogen gas system. The nitrogen low-pressure module removal is recommended because of high leakage rates and failure encountered during testing. This module is replaced by the relief module which contains a burst disc, relief valve, and checkout pressure port. APS modifications required for the OWS are removal of existing 150-pound thrust and Gemini engines, and replacement with six model R-1E engines, plus designing new engine-mounting adaptors. The propellant system is manifolded 180 degrees around the stage to provide redundancy in the two-module concept. The propellant system in each module supplies propellant to an engine cluster in both modules by means of this manifolding. Recirculation provisions are incorporated into each module, which allows removal of gas from stage manifolds by cross-flowing propellant during loading, and bleeding in the opposite APS module.

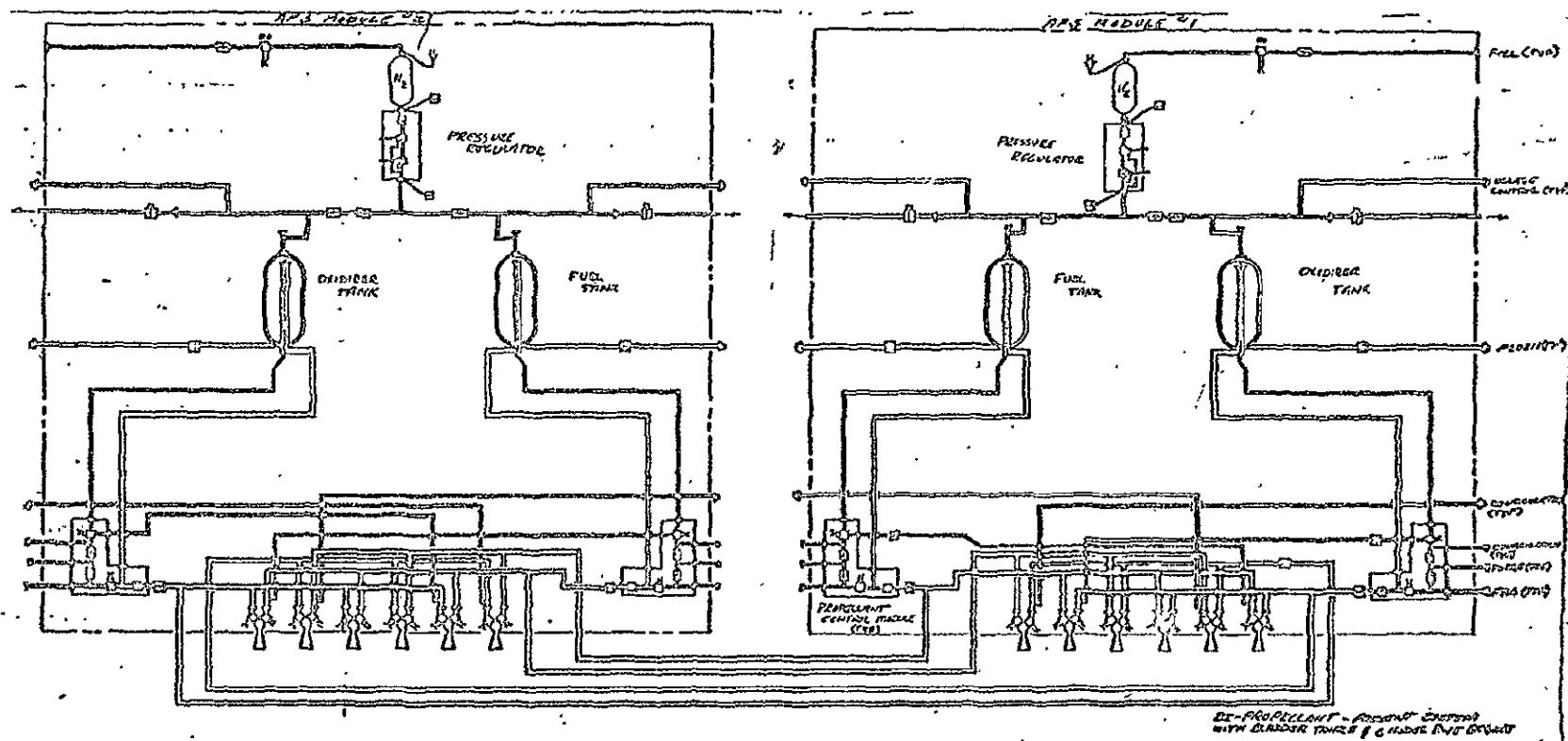


Figure B-7 Bi-Propellant : Present System With
Bladder Tanks & 6 Model R-1E Engines

The lack of confidence in the bladder tank for a 23 month mission, without extensive qualification testing, is the principal objection to this system. Testing could very likely impact the launch data; and for this reason this system is not recommended.

8.1.4.3 S-IVB/V¹ APS Modules - S-IVB/IB Tanks

The APS configuration selected for installation on the orbital workshop, figure 8-8, is composed of two independent propulsion systems in each module. Each system consists of S-IVB/IB propellant tankage, which includes a high-pressure nitrogen storage sphere, nitrogen pressure regulator, propellant-control modules, check valves, relief module, cartridge valves and oxidizer-fuel nitrogen manifolding.

The positive-expulsion propellant tankage utilizes a welded stainless-steel bellows for containing the propellants. The propellant-control modules are used to control propellant filling, recirculation, and unloading of propellant from the APS. These modules are qualified for both S-IVB/IB and S-IVB/V stage usage. The APS filling with propellant (both oxidizer and fuel) uses a common module and manifolding to fill both tanks. The manifolding contains two cartridge valves which are activated after recirculation and loading of propellants. The purpose of these valves is to isolate each system in a module; thus, each tank supplies only three engines.

The dual-regulator pressurization system is isolated after loading by an explosive cartridge valve located between the regulators. The nitrogen gas regulator consists of two regulators in series for regulating nitrogen gas to the propellant tank ullage volumes. The pressurized ullage of the propellant tankage provides the driving force for expelling propellant into the engine valve manifolding.

In order to achieve high reliability in the dual-system module application, it appears that adequate redundancy can be provided by manifolding each module 180 degrees apart on the aft skirt. The manifolding includes cartridge valves which are normally closed to separate the modules, and are opened either when propellant usage depletes either module to

ORBITAL WORK SHOP APS SYSTEM

4056

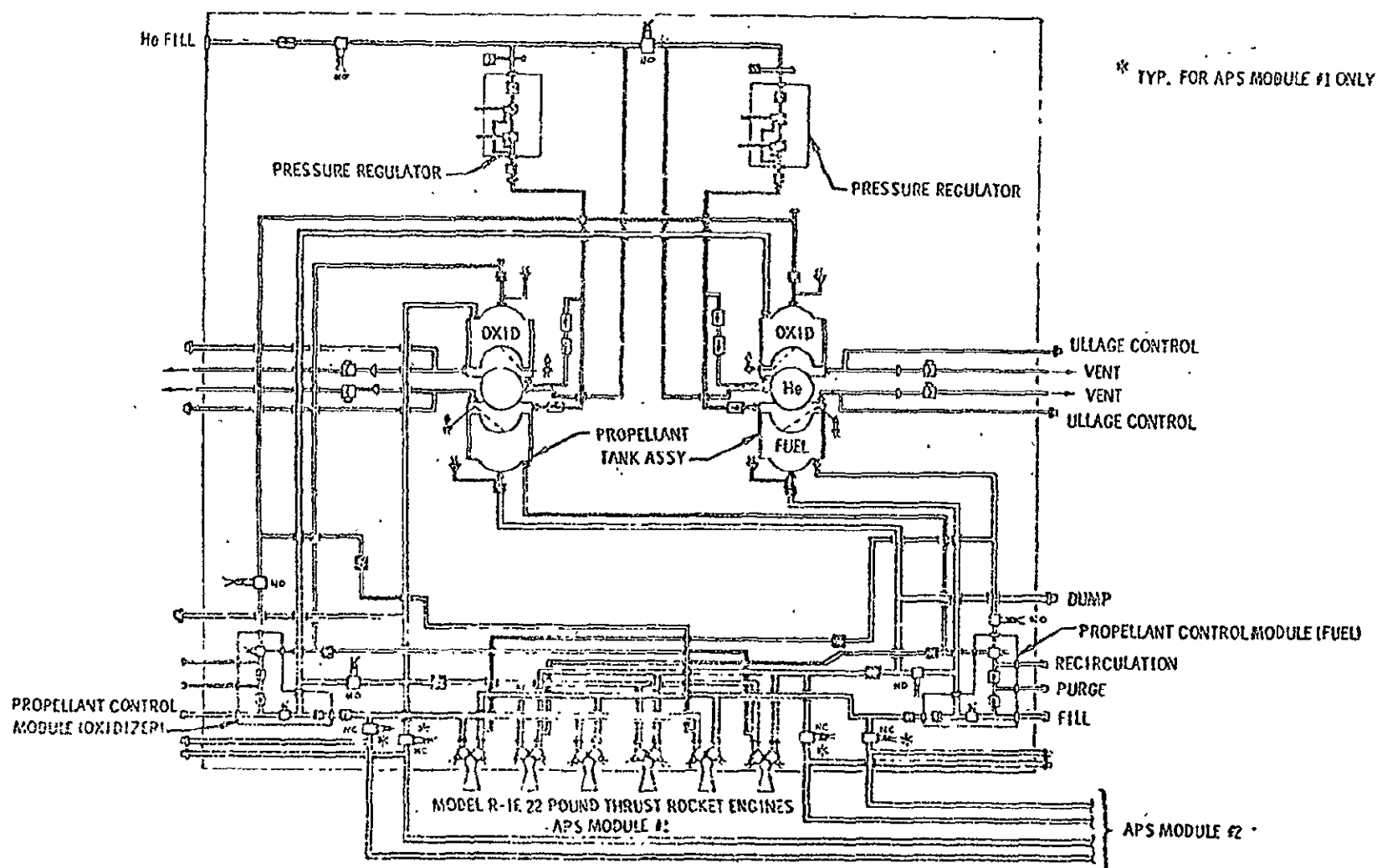


Figure 8-8 Orbital Workshop APS System

Hydrazine attitude-control systems have been successfully used on the Mariner program, and are presently being developed for the Titan Transtage. System simplicity, reliability, and relatively low development and production costs are its chief advantages over bipropellant systems. Major disadvantages for OWS application are low specific impulse, increased thermal control requirements, new technology to the Saturn program, and new ground support equipment at the Sacramento Test Center and KSC. The first two disadvantages affect overall system weight and become increasingly significant as total-impulse requirements increase. The other disadvantages may affect the ability to meet early launch dates. For these reasons, this system was rejected.

The following sections present the results of evaluation of monopropellant components and systems.

8.2.1 Monopropellant Thrustors

Thruster selection was based on the same requirements and design criteria used for bipropellant engines. Thrust levels of 17 pounds or greater, and minimum impulse bits of 0.6 pound-second or less, were the principal criteria.

Monopropellant thruster specific impulse is a strong function of duty cycle. Extended duration between pulses permits catalytic bed temperatures to drop. A significant amount of energy from a subsequent pulse is therefore expended in raising the catalyst temperature, and propellant decomposition may be incomplete. Both conditions degrade performance. The theoretical specific impulse of hydrazine is less than for the bipropellant (NTO/MMH) reactions. It is to be expected, therefore, that performance will be less than with bipropellant engines.

Table 8-3 presents data from a survey of monopropellant thrustors with the required operational characteristics. Rocket Research has a firm contract to develop the backup hydrazine thruster for the Titan Transtage. The thruster utilizes Shell 405 catalyst, and has the capability of blow-down operation between 300 and 150 pounds per square inch. These pressures correspond to thrusts of 26.7 and 16 pounds, respectively.

TABLE 8-3

 APS-35
 7-20-67

MONOPROPELLANT THRUSTOR SURVEY

MANUFACTURER	ENGINE	THRUST (LBS)	MIB (LB-SEC)	OPERATING LIFE (SEC)	CURRENT STATUS	DELIVERY TIME FROM ATP (MONTHS)
ROCKET RESEARCH	TRAN-STAGE	26/16	.2	1,000*	CONTRACT TRANSTAGE	12 BY DEC 67 24 BY APRIL 68
TMC	25 LB	25		AS REQUIRED	IN-HOUSE	3
KIDDE	TRAN-STAGE	26/16	.2	1,000	DEV. FOR TRANSTAGE	3-4
TRW	TRAN-STAGE	26/16	.2	1,000	DEV. FOR TRANSTAGE	--
HAMILTON STANDARD	R-14C	26/16	.2	1,000	DEV. FOR TRANSTAGE	4

*TRANSTAGE EMPLOYS A BLOWDOWN SYSTEM TESTED TO 2500 SECONDS, 5000 TO 7000 SEC. MINIMUM RELIABILITY EXPECTATION FOR TRANSTAGE.

At the S-IVB APS operating pressure of 196 pounds per square inch, 17.3 pounds of thrust is available. A desirable feature of this engine is the use of a series-redundant propellant valve.

The thruster lifetime specification is 1000 seconds, with 2500 seconds accumulated in testing to date. The minimum catalyst bed life is expected to be from 5000 to 7000 seconds. Qualified thruster delivery is possible by December 1967 with complete shipment by April 1968.

Additional efforts required for application of the thruster to the OWS are the addition of valve heaters to prevent freezing in space, and lifetime qualification testing.

The Marquardt thruster is neither developed nor designed; however, Marquardt has sufficient confidence attained from in-house activity to offer a fixed-price quotation and a 3-month delivery, utilizing their 46-watt solenoid valve. TRW, Kidde, and Hamilton Standard engines were developed under contract for the transtage. A full qualification program would be required for any of these thrusters, which could consume more time than is available for the OWS. The expense of a full qualification program does not appear to be warranted.

8.2.2 Monopropellant System Installation

Monopropellant-system specific impulse is generally below that of a bipropellant system. Therefore, those propellant-capacity-increase methods which proved inadequate for bipropellant systems were not reconsidered for the monopropellant system. The schemes utilizing four S-IVB/IB APS modules were eliminated for the same reasons as noted in the bipropellant case.

The baseline-monopropellant-system configuration considered is two S-IVB/V APS modules manifolded together and located in the present S-IVB/IB APS position. System variations primarily involve changes within the modules, such as different tankage or pressurization techniques.

Figure 8-9 presents three monopropellant configurations with propellant capacities and gross module weights.

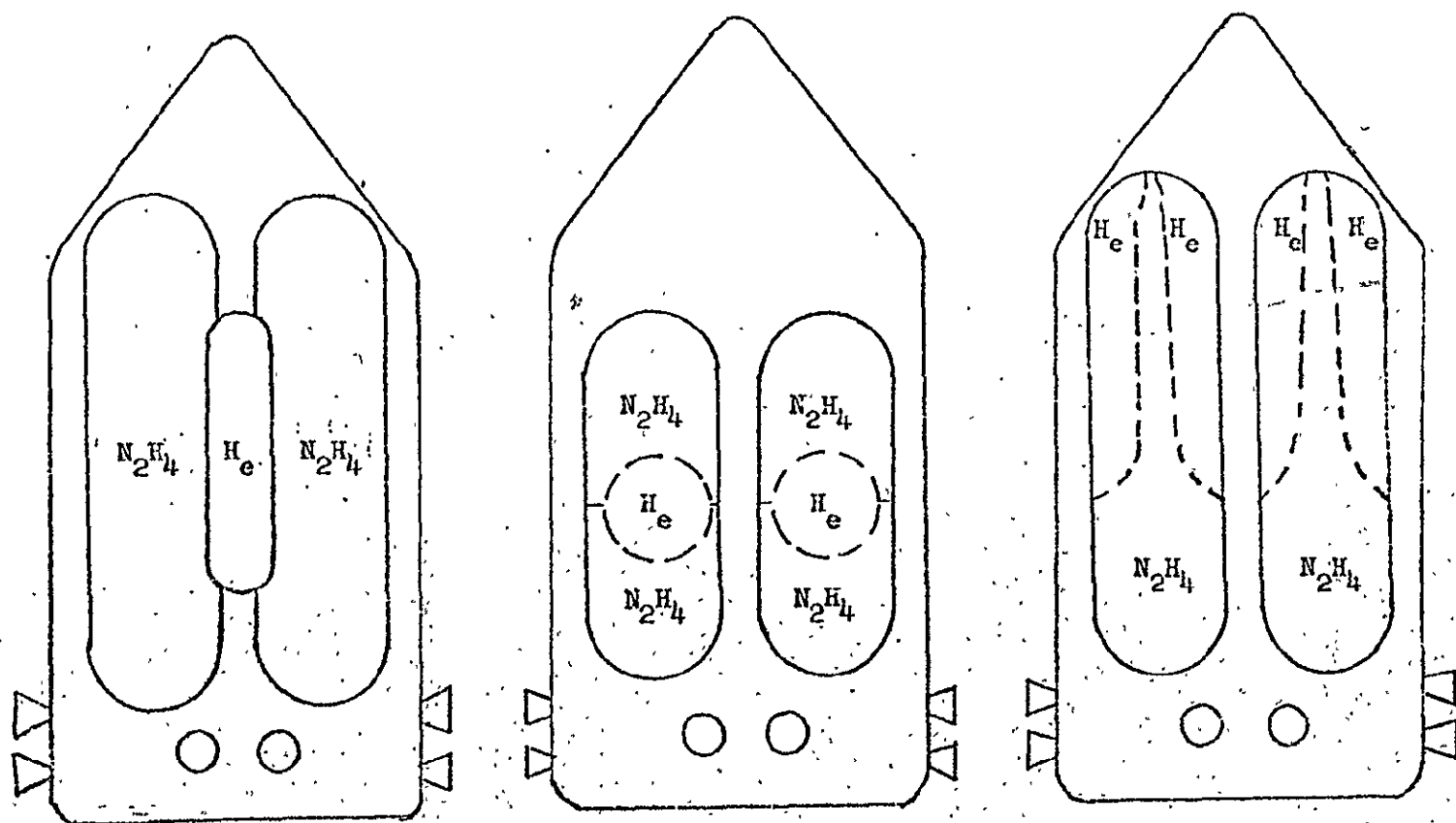
Configuration 1 retains the existing S-IVB/V tankage and pressure-regulated system. Several components can be deleted along with some piping and manifolding. Module dry weight has decreased from that of the S-IVB/V bipropellant APS module.

Configuration 2 uses two S-IVB/IB APS tanks installed similar to the configuration considered for the bipropellant system. The tank metal bellows are compatible with hydrazine. Six thrusters are used for system redundancy and S-IVB powered flight requirements.

Configuration 3 utilizes the S-IVB/V APS tankage with a blowdown pressurization system. Each tank is loaded half full, and pressurized to approximately 300 pounds per square inch. System pressure at propellant depletion is approximately 150 pounds per square inch. Thrustor operation is possible throughout the blowdown range. The modules are manifolded together, and there is complete redundancy of systems.

Figure 8-10 shows the total impulse and equivalent propellant loading per module to complete the mission. In computing the propellant loads per module, the effects of duty cycle were taken into consideration. During boost and passivation, the pulse width was kept at 0.065 seconds and a specific impulse of 175 seconds assumed. A steady-state specific impulse of 220 seconds was used. During minimum-impulse-bit operation, a cold bed specific impulse of 100 seconds was assumed. This reduced the minimum impulse bit from 0.6 to approximately 0.3 pound second. As a result, the pulsing impulse actually required during coast is reduced to 2000 pound-seconds. All configurations have sufficient impulse capability for the mission.

In the following sections, these systems are described in detail.



Configuration #1

Configuration #2

Configuration #3

Module

Saturn V/S-IVB

Saturn V/S-IVB

Saturn V/S-IVB

Tanks

Saturn V/S-IVB

Saturn V/S-IVB

Saturn V/S-IVB

Propellant

Capacity (Lbs.)

300

113

150

Dry Wt. (Lb.)

345

414

306

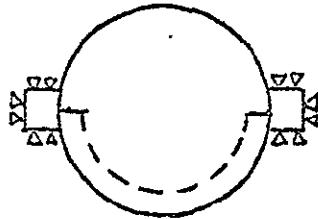
Total Wt. (Lb.)

645

527

456

Figure 8-9 MONOPROPELLANT CONFIGURATIONS



	Impulse (lb-sec)	Propellant (lb)
• Boost & Passivation		
Steady State	1,800	8.2 (Isp = 220)
Pulsing	5,000	28.6 (Isp = 175)
Subtotal	6,800	36.8
• Docking & Coast		
Steady State	6,850	31.2 (Isp = 220)
Pulsing	2,000	20.0 (Isp = 100)
Subtotal	11,250	75.2
TOTAL	15,650	92.0 (Isp = 155)
• Propellants Available		
W/S-IB Tanks (2/module)		113 lb
W/SV Tanks (Full)		300 lb
W/SV Tanks (1/2 full-blowdown)		150 lb

Figure 8-10. Monopropellant Impulse and Propellant Requirements (Per Module)

8.2.2.1 Regulated Monopropellant System Using Bladder Tanks

Figure 8-11 is a schematic of a monopropellant system which utilizes a regulated pressurization system and bladder tanks similar to those in the S-IVB/V APS. This system is basically the S-IVB/V APS with modifications.

The high-pressure system is identical to the S-IVB/V APS system, with the addition of a normally-open cartridge valve in the fill line. This valve is closed at liftoff, thereby reducing system leakage to allow mission completion.

The regulated-pressure system differs from the S-IVB/V APS configuration in that no check valves are required and the tanks are pressurized from a common line. Since the tanks are manifolded, only one relief module is required.

The propellant supply system is essentially one half of the standard S-IVB/V system. The tanks are manifolded together, permitting the use of one propellant control module and retaining present fill and recirculation procedures. The engine manifolds are simpler, requiring only one line per engine. However, to obtain system redundancy and allow equal usage (modular) of propellants, a manifold is utilized from each set of engines in one module to a similar set in the other module.

The tanks utilized are the present S-IVB/V APS tanks with new bladders of Butyl rubber or ethylene propylene rubber (EPR). The new bladder material offers better cycling, permeation characteristics, and better compatibility than the present Teflon bladder. The time required for the change of bladders is well within the scope of the program (3 months delivery).

8.2.2.2 Regulated Monopropellant System Using S-IVB/IB Bellows Tanks

A schematic of this system is shown in Figure 8-12.

This system is identical to the system described in section 8.2.2.1, except that S-IVB/IB bellows tanks are used in place of the bladder tanks. The

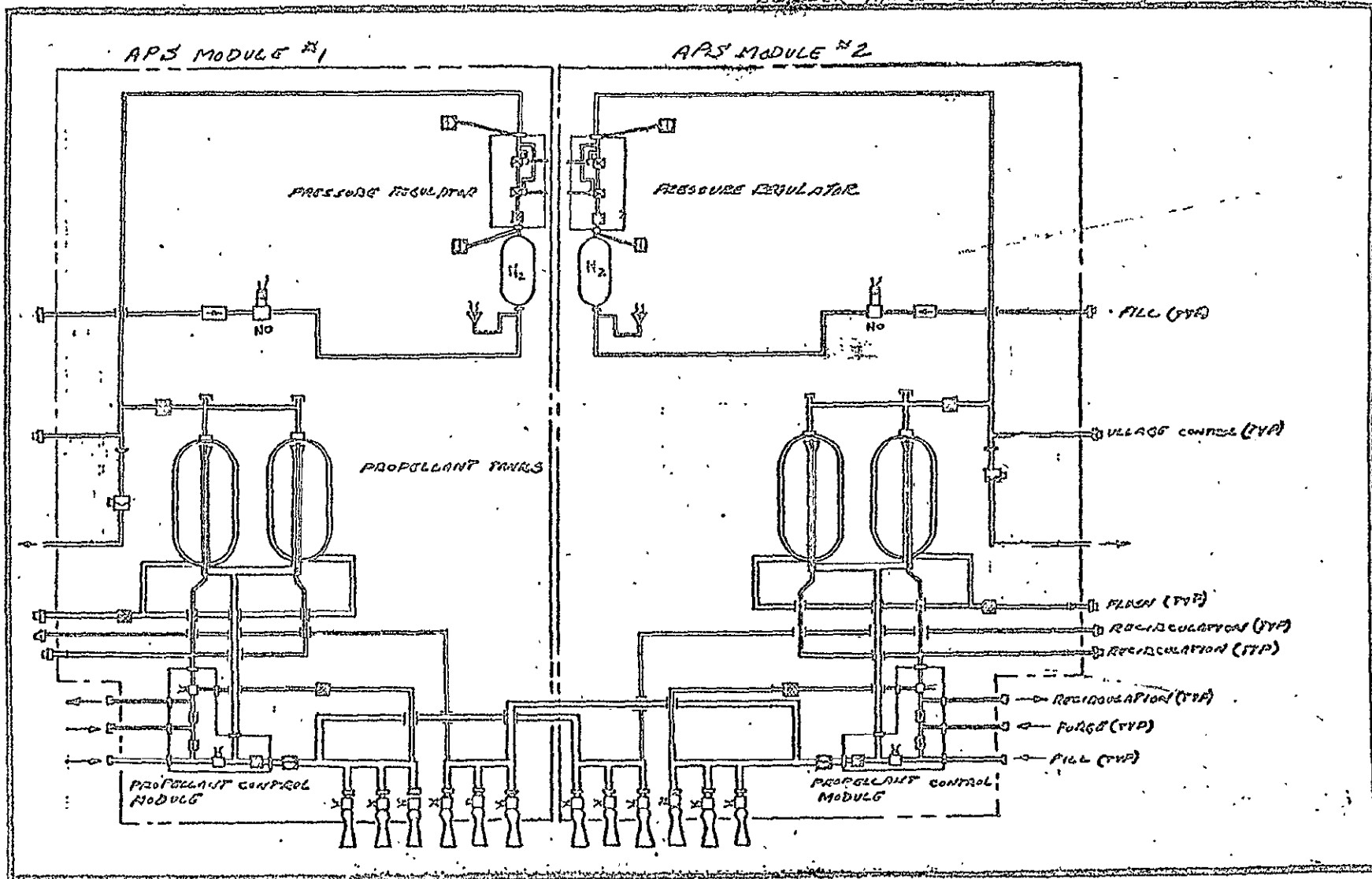


Figure 8-11 REGULATED MONOPROPELLANT SYSTEM BLADDER TANKS

PREPARED BY: L. Rejo DOUGLAS AIRCRAFT COMPANY, INC.
 CHECKED BY: _____ DIVISION _____
 DATE: 15-11-65 TITLE: S-V APS WITH S-1B BELLOWS TANK & 6 HYDRAZINE ENGINES
 REPORT NO: _____

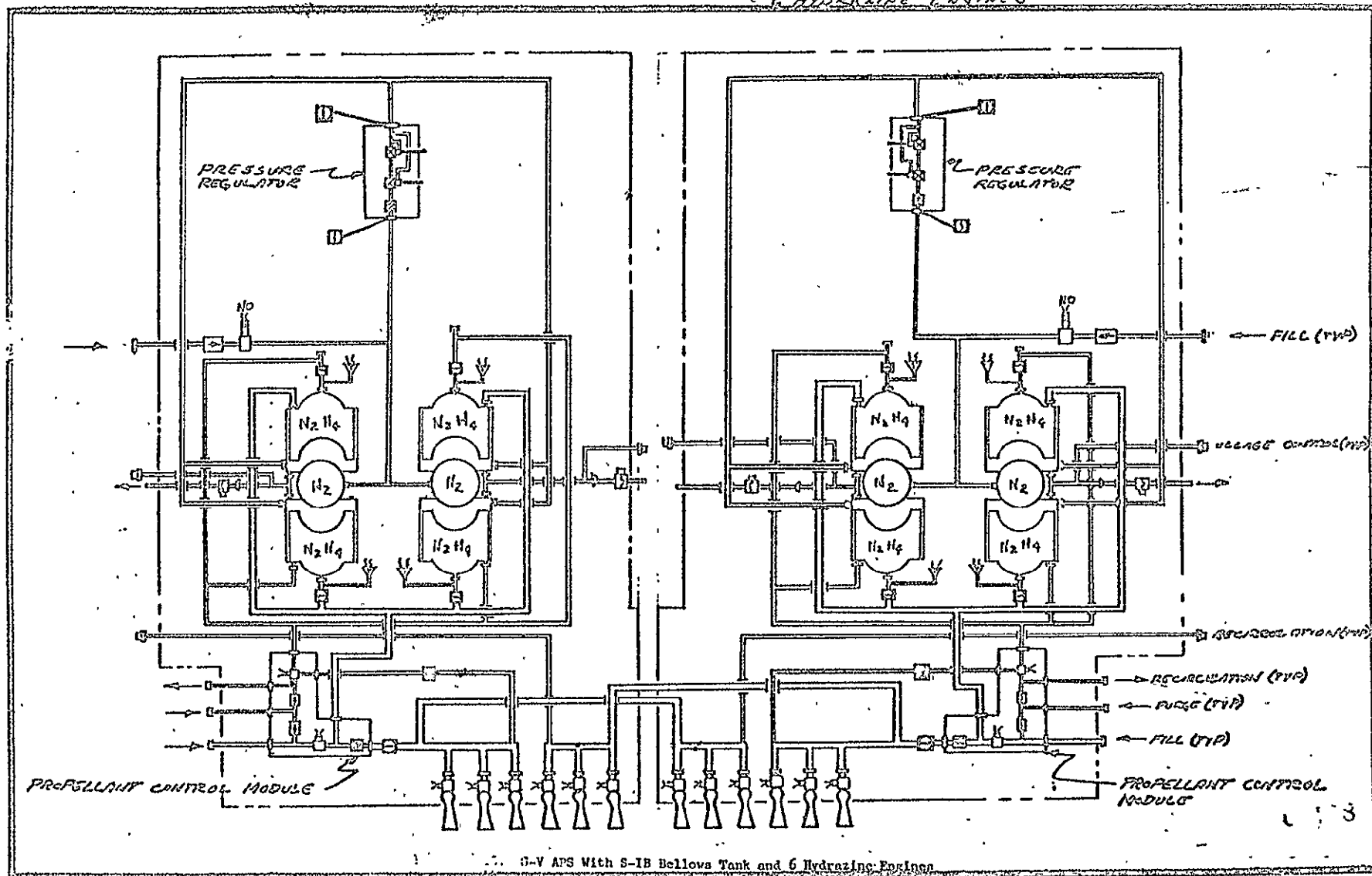


Figure 8-12 S-V APS WITH S-1B BELLOWS TANK AND 6 HYDRAZINE ENGINES

system still offers much simplicity, and the disadvantages associated with the development of bladder tanks have been eliminated.

8.2.2.3 Blowdown Monopropellant System Using Bladder Tanks

Figure 8-13 shows a schematic of a monopropellant system using bladder tanks and a blowdown-type pressurization system.

The pressurization of this system is accomplished by off-loading the bladder tanks to 50 percent capacity, and pressurizing the ullage space to 300 pounds per square inch absolute. Upon expelling propellant, this pressure will decay to 150 pounds per square inch absolute. The thrusters used are capable of operation throughout this range of supply pressures. As with the regulated systems, the pressurant fill line is sealed at liftoff by use of a normally-open cartridge valve.

The propellant supply, and engine systems are identical to those described in section 8.2.2.1. The tankage is also the same as described in section 8.2.2.1, and the associated disadvantages still apply.

The removal of the regulator from the system results in an even simpler system, and eliminates any problems associated with the use of a regulator. Although a reduction of engine thrust occurs when the tank pressure decreases, this occurs during the latter portions of the mission when it is acceptable and perhaps desirable.

8.3 Phase I Compressed-Gas System

Figure 8-14 indicates that a compressed-gas system is impractical for providing the total mission impulse requirement. System weight is approximately double that of an equivalent bipropellant or monopropellant system.

One advantage of using compressed gas in conjunction with a liquid propellant system lies in reducing thermal-control-power requirements during long-term coast. In such an application, liquid-propellant-system usage might be discontinued at the end of the manned visit. During the 6 to 12-month storage period, no thermal control would be used, and restabilization for a revisit might be accomplished by the gas system.

PREPARED BY: LR DOUGLAS AIRCRAFT COMPANY, INC.

CHECKED BY: _____ DIVISION _____

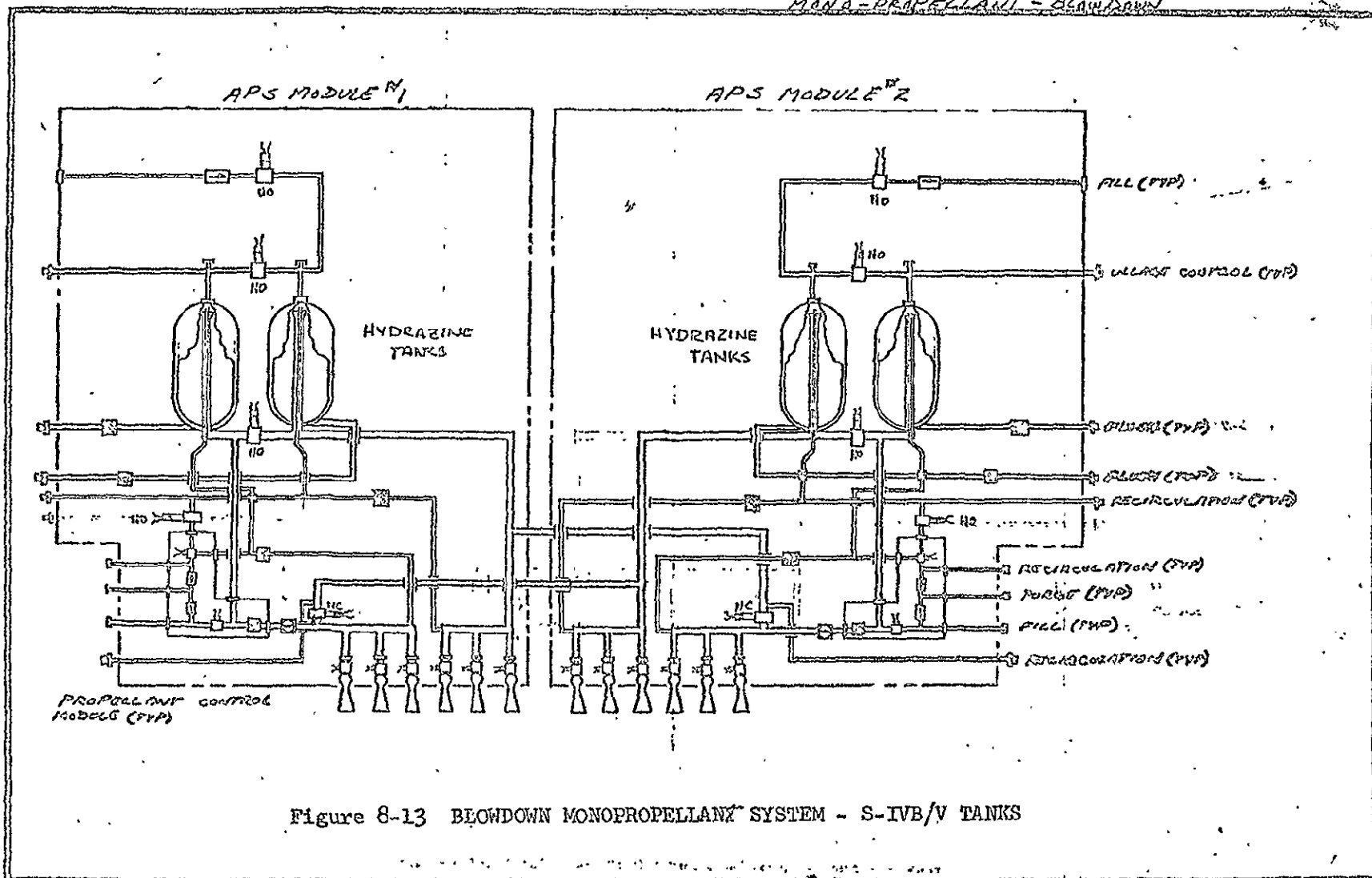
DATE: 3-26-7

TITLE: SATURN V APS MODULES - MANIFESTED
MONA - PROPELLANT - BLOWDOWN

PAGE _____

MODEL: _____

REPORT NO. _____



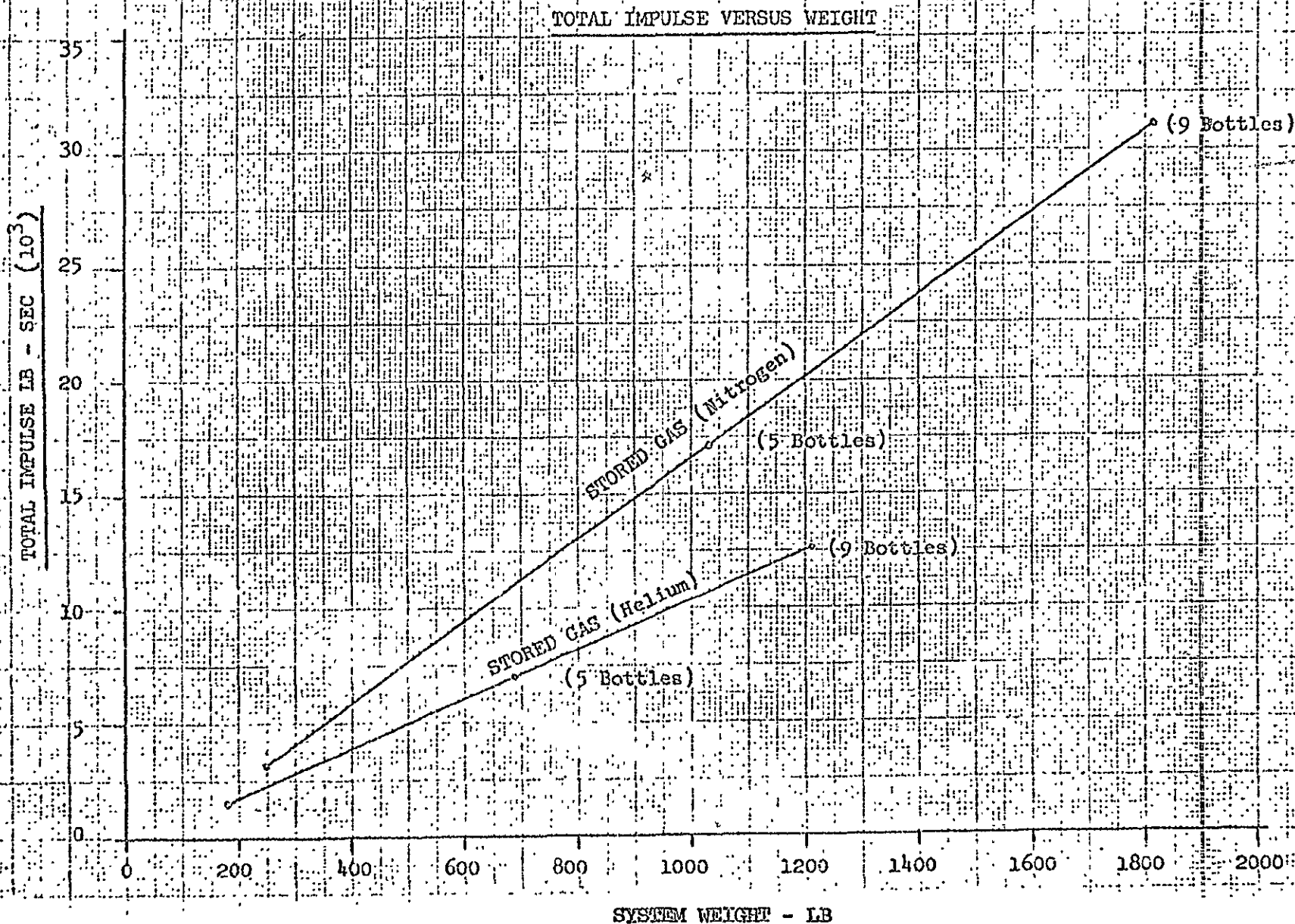


Figure 8-14 COMPRESSED GAS SYSTEM IMPULSE VS. WEIGHT

DOWGLAS

There are several disadvantages to this concept. Approximately 8,300 pound-seconds of impulse would be required to stabilize and permit docking to the OWS after both storage periods. This corresponds to a gas-system weight of about 550 pounds. There is some risk in permitting liquid propellant to freeze within confined systems. As the propellants freeze, their contraction may permit liquid to flow past check valves. Upon thawing the increasing propellant volume may deform valve seats, yield soft materials, or possibly rupture components. Unless refueling is used, liquid propellants will have to be aboard throughout the storage periods for use during revisits. With thermal protection required, even when not active, there is no advantage in introducing a compressed-gas system.

A second use considered for compressed gas was in providing the 0.6 pound-second minimum-impulse-bit capability. If that part of the mission having this requirement were done with the gas system, there would be no reason for changing engines. The existing ablative engines have sufficient lifetime, and quad propellant valve redundancy would probably eliminate the need for thruster redundancy, and automatically operated cluster shutoff valves. However, with the availability of a radiation-cooled thruster and its superior lifetime for system growth, development of the gas system does not appear to be merited. For these reasons, a compressed-gas system was rejected for use on the OWS.

8.4 Phase I System Selection

Of the bipropellant systems, the system in which most confidence is placed for meeting the early launch date is the S-IVB/V module with S-IVB/IB tanks. While not the system with highest propellant mass fraction, lightest weight, or greatest growth potential, it is the system which is considered as requiring a minimum of development. S-IVB/IB APS manufacturing and testing experience are directly applicable to this design. Three launches have proven the flight capability of many components proposed for this system. Potential bladder permeation problems are eliminated by using S-IVB/IB metal bellows tanks. While a Saturn IB flight capability is certainly not a conclusive test of a 23-month mission, it does represent a significant hardware maturity over new or modified designs not yet in existence.

To take advantage of monopropellant operation, a bladder-tank configuration is recommended for its lighter weight. The consequences of possible gas permeation, associated with bipropellant bladder tanks, are of little concern for monopropellants. Bubble entrainment within the propellant stream does not affect hardware integrity as it might in a mixture-ratio-sensitive system. Hydrazine space storage for months in rubber bladders has been demonstrated by Mariner IV.

With the use of S-IVB/V rubber-bladder tanks, sufficient tank volume is available to permit mission completion with an integral blowdown system. This is the recommended monopropellant system. Simplicity is obvious. No moving pressure-regulator parts, burst discs replacing heavier relief valves, and complete system redundancy within each module yield a light weight, reliable system. Thrustor blowdown capability is part of the transtage RCS engine specification. In making the final selection between the recommended monopropellant and bipropellant systems, a broad assessment of overall OWS program implications was made. Figure 8-15 presents monopropellant and bipropellant system weights as a function of available impulse.

The first point on the bipropellant curve represents two S-IVB/IB APS modules. By placing two S-IVB/IB APS tank assemblies in an S-IVB/V module, the impulse capability is doubled as shown by point 2. A further increase in available impulse is realized by utilizing the S-IVB/V modules with their present tankage shown by point 3. To obtain impulse in the 250,000 to 300,000 pound-second range, the use of four S-IVB/V APS modules with S-IVB/V APS modules with S-IVB/V tankage is required. This is shown by point 4.

On the monopropellant curve, point 5 represents the blowdown system utilizing two S-IVB/V APS modules with S-IVB/V tanks half full. To increase the system capability, the tanks may be completely filled and an external pressure source used; this situation is shown by point 6. The maximum impulse is obtained by using four S-IVB/V APS modules. This configuration is represented by point 7 on the monopropellant curve.

ALABAMA
LA 1782

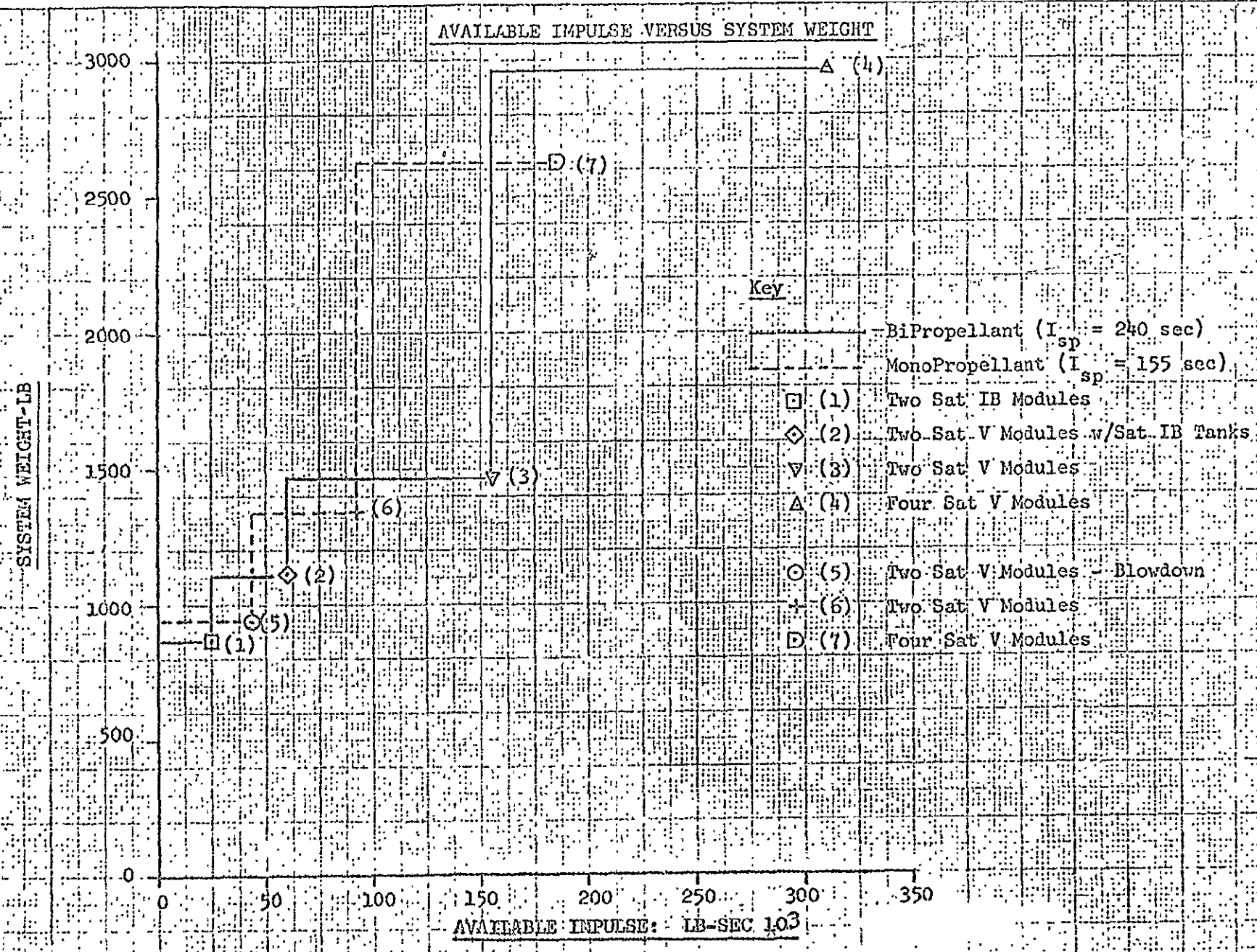


Figure 8-15 System Weight Versus Impulse for Various Systems

DOUGLAS

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It can be seen from the figure that the monopropellant systems are competitive with the bipropellant systems in the 30,000 to 100,000 pound-second impulse range. However, as the impulse requirements increase, the bipropellant system shows an increasing weight saving. This is due, primarily, to superior specific impulse.

Table 8-4 presents nine significant attitude-control-system considerations.

High on the list are component availability and testing required for qualification, both of which have direct bearing on the flight date. New thrusters are the pacing items in each case, with a bladder development required for monopropellant storage in the S-IVB/V tanks. The monopropellant system testing is expected to be more extensive than bipropellant development, because of the new tank qualification. Also, facilities and GSE changes created by the hydrazine system will require their own test and checkout. On the other hand, considerable related test experience is available for bipropellant testing, and the learning curve not as steep.

Reliability is affected by critical components in each system. The regulator and bladder are unique to the bipropellant and monopropellant systems, respectively. While previous APS experience is available to improve bipropellant system reliability, the inherent simplicity of the monopropellant system is expected to compensate.

Power availability is critical, and the difference between maintaining a system temperature of 20°F or 40°F is almost a doubling in electrical power for the latter. With freezing points of 34°F for hydrazine, 11°F for NTO, and -65°F for MMH, the bipropellant system will require less thermal-control power.

Modifications to GSE are undesirable, especially with tight development schedules. Existing KSC facilities are adequate for the bipropellant system, whereas at least new cleaning, and possibly some hardware modifications, are needed for hydrazine handling.

Table 8-4 PROPELLANT SYSTEM COMPARISON

	SELECTED BIOPROPELLANT SYSTEM	SELECTED MONOPROPELLANT SYSTEM
<u>AVAILABILITY</u>	- ALL SYSTEM COMPONENTS AVAILABLE - ENGINES PACING ITEM	- S-V TANK BLADDER CHANGE - ENGINES PACING ITEM
<u>RELIABILITY</u>	- REGULATOR RELIABILITY CRITICAL - PREVIOUS EXPERIENCE	- BLADDER RELIABILITY CRITICAL - SIMPLICITY
<u>TEST</u>	PREVIOUS TEST EXPERIENCE	- TANK QUALIFICATION TEST REQUIRED - FACILITIES AND GSE TESTING
<u>GROWTH</u>	- SUPERIOR ISP - LOWER TOTAL SYSTEM WEIGHT ABOVE 100,000 LB-SEC	GROWTH TO 250,000^{LB} LB-SEC NOT PRACTICAL WITHIN MODULES
<u>POWER</u>	MINIMUM PROPELLANT TEMP. ALLOWABLE 20°F	MINIMUM PROPELLANT TEMP. ALLOWABLE 40°F
<u>GSE</u>	EXISTING FACILITIES	REQUIRES GSE MODIFICATION FOR HYDRAZINE AT SACTO. AND KSC
<u>WEIGHT</u>		200# LIGHTER
<u>DESIGN</u>	SIMILARITY TO EXISTING DESIGN	SIMPLER SYSTEM
<u>MANUFACTURING</u>	IB TANK INSTALLATION REQUIRED	

For the 36,000 pound-second mission impulse requirement of Phase I, the monopropellant system is approximately 200 pounds lighter than the bipropellant system, due primarily to a lower dry weight.

Design or manufacturing differences are not considered to have extensive impact on the system selection. Similarity to existing design benefits the bipropellants, whereas, simplicity assists monopropellant design. Manufacturing will require changes in techniques where the S-IVB/IB tanks are placed in S-IVB/V modules. However, this is not expected to be any more severe than engine changes or high Performance Insulation (HPI) additions.

As a result of these considerations, the APS recommended for the WACS for Phase I is contained in two S-IVB/V modules, and manifolded together. Each module is composed of an S-IVB/V APS module outer shell (structure) modified for installation of two S-IVB/IB APS bellows tank assemblies, a dual nitrogen-pressurization subsystem, HPI, heater blankets, and a protective shroud. Each of the two engine clusters in each module include three Marquardt R-IE bipropellant thrusters. (See figure 8-16 and 8-17) A detailed description of this system is given in section 8.1.4.3.

Table 8-5 presents the pressurization system leakage of the selected system. The values listed under system leakage apply to one subsystem (two subsystems per module). Due to the reduction in leak rates realized, nitrogen was selected for pressurization. The initial pressure of 2800 pounds per square inch absolute was established to be compatible with present KSC redlines. The maximum allowable leak rate was determined to be 0.017 standard cubic inches per minute, assuming a temperature decay of 40°F. All values shown in the specification leakage column are leakage to the environment per component. The leakage of the relief module denoted by the double asterisk (**) is the leakage rate after the burst diaphragm has ruptured. With the cartridge valves closed and the burst diaphragm intact, the total system leakage is 0.0057 standard cubic inches per minute. The resulting system lifetime is 6 years. If a burst diaphragm were to rupture at liftoff, the system lifetime would be reduced to 2.6 years, which is still adequate.

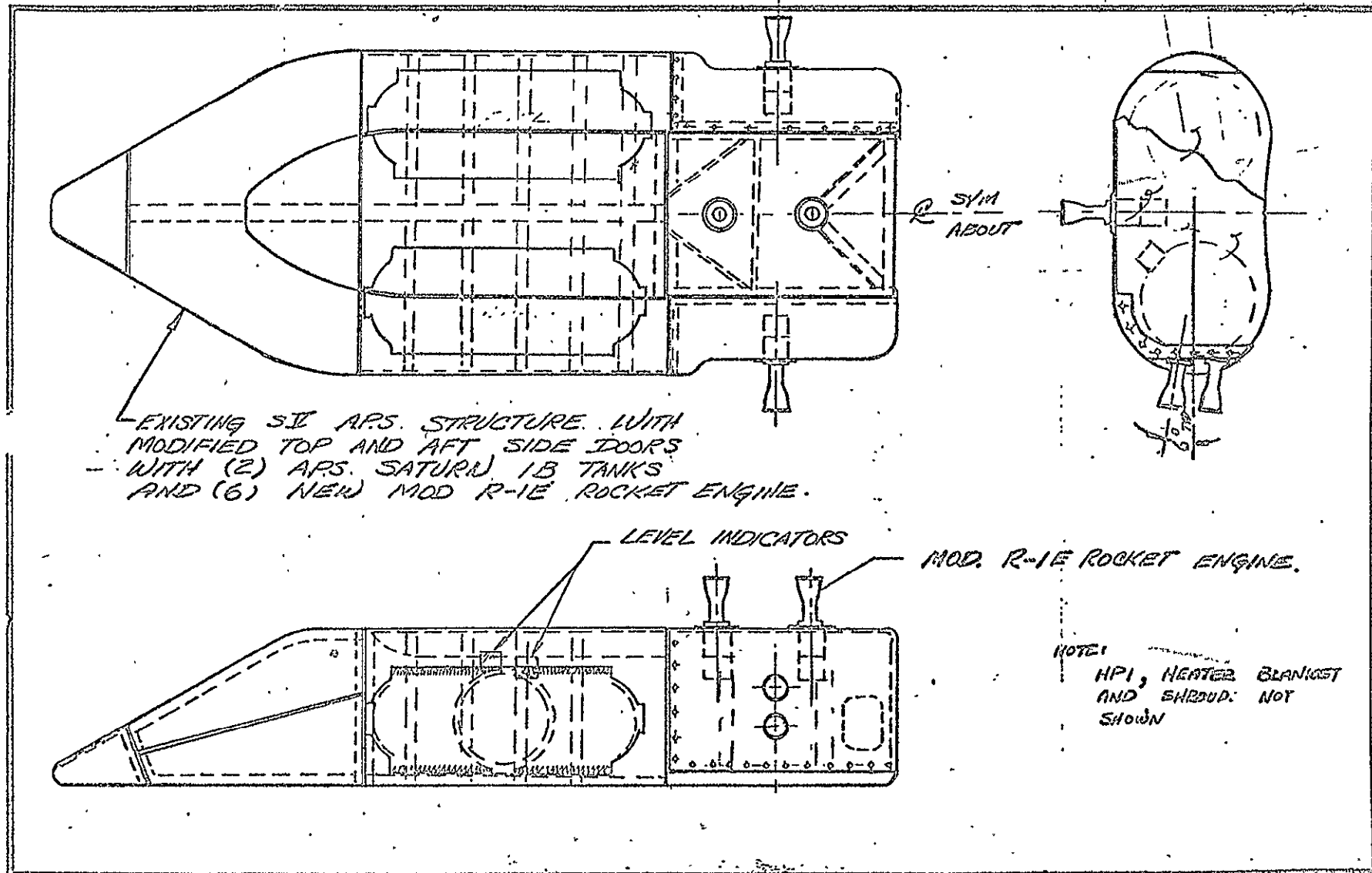


Figure 8-16 PROPOSED APS FOR ORBITAL WORKSHOP PHASE I (THREE-VIEW)

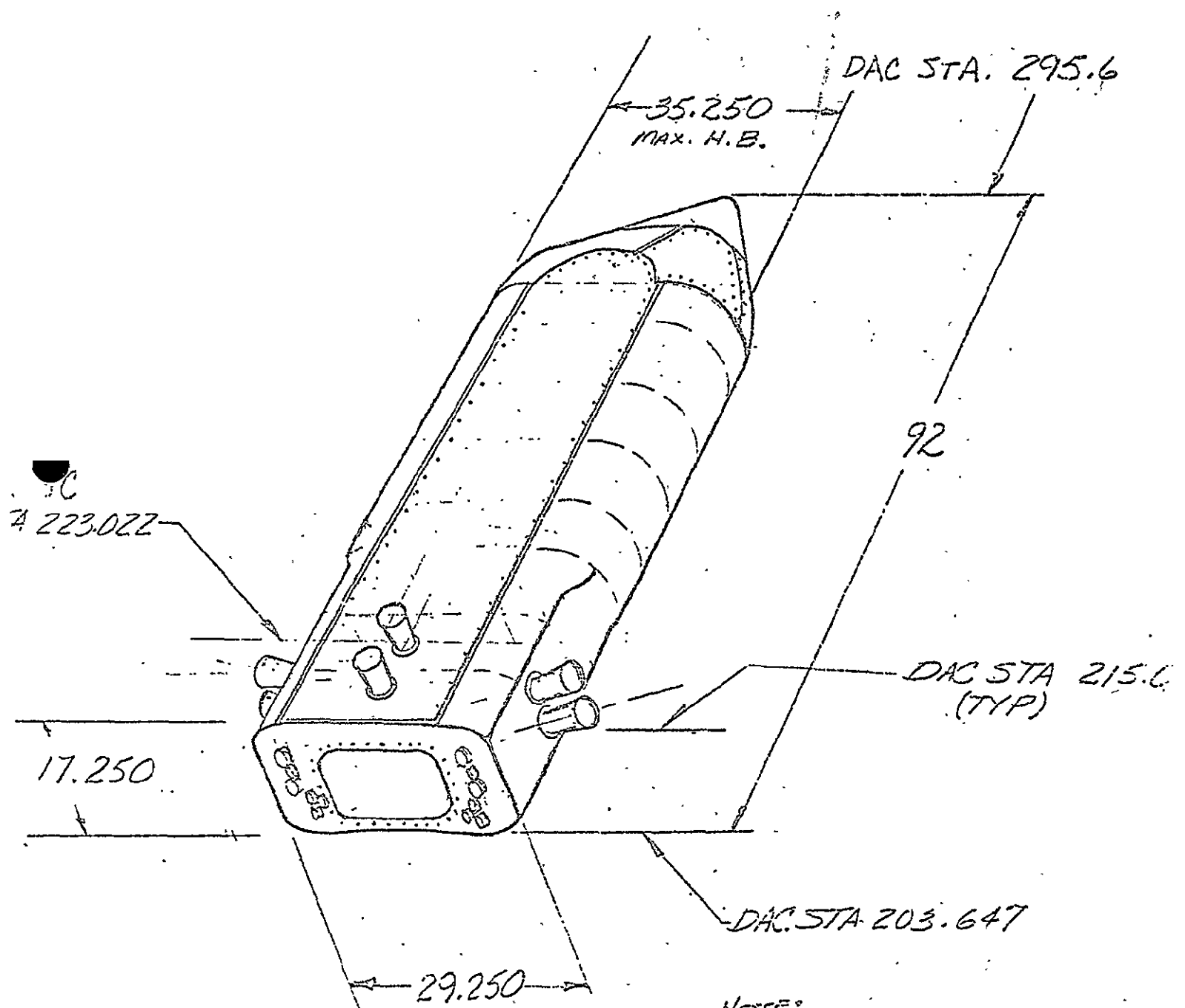


Figure 8-17 Proposed APS for Orbital Workshop - Phase I.
(Schematic)

The estimated dry weight of the total system is 1013 pounds, which is 438 pounds heavier than the present S-IVB/IB APS. The propellant capacity of 248 pounds is twice that of the S-IVB/IB APS. A detailed weight breakdown is given in table 8-7.

8.5 Phase II Auxiliary Propulsion System Studies

The Phase II studies evaluated auxiliary propulsion systems to meet the requirements for the vehicle solar orientations which are compatible with the environmental control and solar panel system requirements. In section 4 of this report several vehicle orientation schemes were analyzed and the total impulse required to accomplish each mode is summarized in section 4.2.4. These total impulse requirements range between 88,000 and 294,000 pound-seconds. This impulse range and the system design conclusions reached during the Phase I study, which were based on an allowable development cycle of 18 months, formed the guide lines for the Phase II study. A summary of the conclusions from the Phase I study is presented below as the ground rules for the Phase II study.

PHASE II STUDY GROUND RULES

1. Maximum utilization of existing APS hardware
2. Bipropellant System (N_2O_4/MMH)
3. Module Concept
4. Bellows type propellant tanks
5. Propellant tank manifolding
6. N_2 pressurant
7. Marquardt R-IE engines

The following sections present the results of the investigation made of the available bellows type propellant tanks, the Marquardt R-IE thruster, and the GN_2 pressurization system for their application into the selected system for the Phase II mission.

Pressurant - Nitrogen Initial Pressure = 2800 psia
 Maximum Allowable Leakage for 2 year Mission = 0.017 scim

Item	Manufacturer	Part Number	Specification Leakage	System Leakage (GN ₂)*
Regulator	Fairchild Stratos	65-168	1 scch of Helium	0.0004 scim
Check Valves (3)	Marotta	801602	1 scch of Helium	0.0012 scim
Tank Assembly Ph I	Douglas Ph I	1A93231	1.5×10^{-4} sccs of Helium	0.0002 scim
Cartridge Valve Ph II	Ball Ph II Rocketdyne	NA5-28069T4#1	10^{-6} sccs of Helium	Negligible
Relief Module	Calmec	805	10^{-5} sccs of Helium **(20 scch of Helium)	Negligible **(.0076 scim)
Conoseals (12)	Marman	---	1×10^{-5} sccs Helium	0.0002 scim
"B" Nuts (15)	---	---	.01 sccm Helium	0.0034 scim
Tubes	---	---	---	0.0003 scim
TOTAL				0.0057 scim

Based on the above leakages with a temperature degradation from 80°F to 40°F, the system lifetime is 6 years.**

* Leakage shown for one system of one module.

** With the burst diaphragm ruptured at liftoff, system lifetime is reduced to 2.6 years.

scch - standard cubic centimeters per hour

sccs - " " " " second

sccm - " " " " minute

scim - " " inches " minute

Table 8-5 Bipropellant Pressurization System Leakage

8.5.1 Tankage Applicable to Phase II

Considering the increased total impulse requirements of Phase II, the bellows type tanks evaluated during the Phase I study (see table 8-1) were reviewed for their application to the Phase II mission. In addition, a preliminary investigation was made into the development of two new tanks. The first new tank is essentially an extended Saturn IB bellows tank. The second new tank, referred to hereafter as the Saturn V APS bellows tank, would provide the optimum propellant bellows tank for use with the Saturn V APS module structure.

Table 8-6 lists each tank reviewed and provides the total impulse capability for a tank pair (oxidizer/fuel). The total impulse range has been computed for a pulsing specific impulse of 215 seconds; and a steady-state specific impulse of 280 seconds.

It should be noted that to attain the maximum mission total impulse of 294,000 pounds-seconds, each propellant tank pair (oxidizer/fuel) must be capable of supplying approximately 50,000 pound-seconds, assuming the use of six modules.

The S-IVB/IB APS tank, manufactured by Douglas, was listed for information only. Its propellant capacity is too limited to be given consideration for the Phase II mission.

The Agena target vehicle tank is described by its manufacturer, Bell Aerosystems, as being fully qualified and man rated. The tank has completed its developmental and qualification testing and has been successfully flight tested.

The overall tank dimensions are 45.1 inches long and 10.1 inches in diameter. This envelope will easily fit within the modified S-IVB/V APS module structure. A summary of the pertinent tank details is presented in table 8-6. Modifications are required to make provisions for propellant tank recirculation and a propellant measuring device. Bell has stated that they have developed a magnetic-microswitch propellant level sensor system with one percent accuracy, which could be easily installed on this tank. If necessary,

BELLOWS TANK SUMMARY

TANK	USABLE VOLUME (IN ³)	ASSEMBLY WEIGHT (LBS)	MAXIMUM OPERATING PRESSURE (PSIG)	TOTAL IMPULSE PER TANK PAIR (LB-SEC)		REMARKS
				I _{SP} = 215 SEC (PULSING)	I _{SP} = 280 SEC STEADY STATE	
SATURN S-IB APS BELLOWS (DOUGLAS)	775	63	275	13,350	17,350	FULLY QUALIFIED FOR SATURN IB
POST BOOST (BELL)	3,248	47.5	225	56,300	73,400	COMPLETE QUAL PROGRAM REQUIRES 12 MONTHS
EXTENDED S-IVB/IB BELLOWS (DOUGLAS - NEW)	2,930	38.0	275	50,700	66,100	FULL QUAL. PROGRAM REQUIRED 9 MONTHS
SATURN V APS BELLOWS (DOUGLAS - NEW)	4,301	58	275	74,531	47,065	FULL QUAL. PROGRAM REQUIRED 12 MONTHS
GEMINI TARGET VEHICLE (BELL)	2,840	32.0	225	49,500	64,400	FULLY QUALIFIED (MAN RATED)

Table 8-6

this tank could be modified to incorporate the proven S-IVB/IB APS potentiometer-type propellant level sensor. With the minor modification required to accommodate tank recirculation, the tank design will be compatible with the existing S-IVB APS loading techniques.

The tank modifications discussed above will require a limited qualification test program, but this requirement should not result in any program delays. The manufacturer has indicated that initial tank deliveries could begin 10 months after ATP.

This tank can satisfy all the requirements for the Phase II mission.

The Post-Boost tank shown in figure 8-18, has been designed by Bell Aerosystems, and is presently undergoing developmental and prequalification testing. The preliminary review of this tank concluded that it could satisfy the capacity and envelope criteria established for the OWS mission. However, after a detailed review, it was concluded that the crush-formed type bellows tank was not acceptable.

The following problem areas were uncovered during the detailed review. The tank design was based on a vacuum type loading system. This loading approach would require an extensive modification to the existing APS loading technique and hardware. The operating pressure to expel propellants is significantly higher for crush-formed bellows than for the welded design. This tank will require a differential pressure of 50 to 75 pounds per square inch across the bellows to expel the final 15 percent of tank capacity, compared to 10 pounds per square inch for the S-IVB/IB welded bellows. Expulsion cycle life of the crush-formed bellows is another limitation. According to the tank specification, the tank is limited to a maximum of six cycles. Based on Douglas experience during the S-IVB/IB program, the cycle requirements for checkout, propellant loading, etc., should be in excess of 75 cycles.

Experience at Douglas with 8-to-10-inch-diameter crush-formed bellows on propellant ducts has indicated that there is a critical allowable clearance between the housing and bellows. If sufficient clearance does not exist, a hammering effect results during transverse dynamic testing and cracking

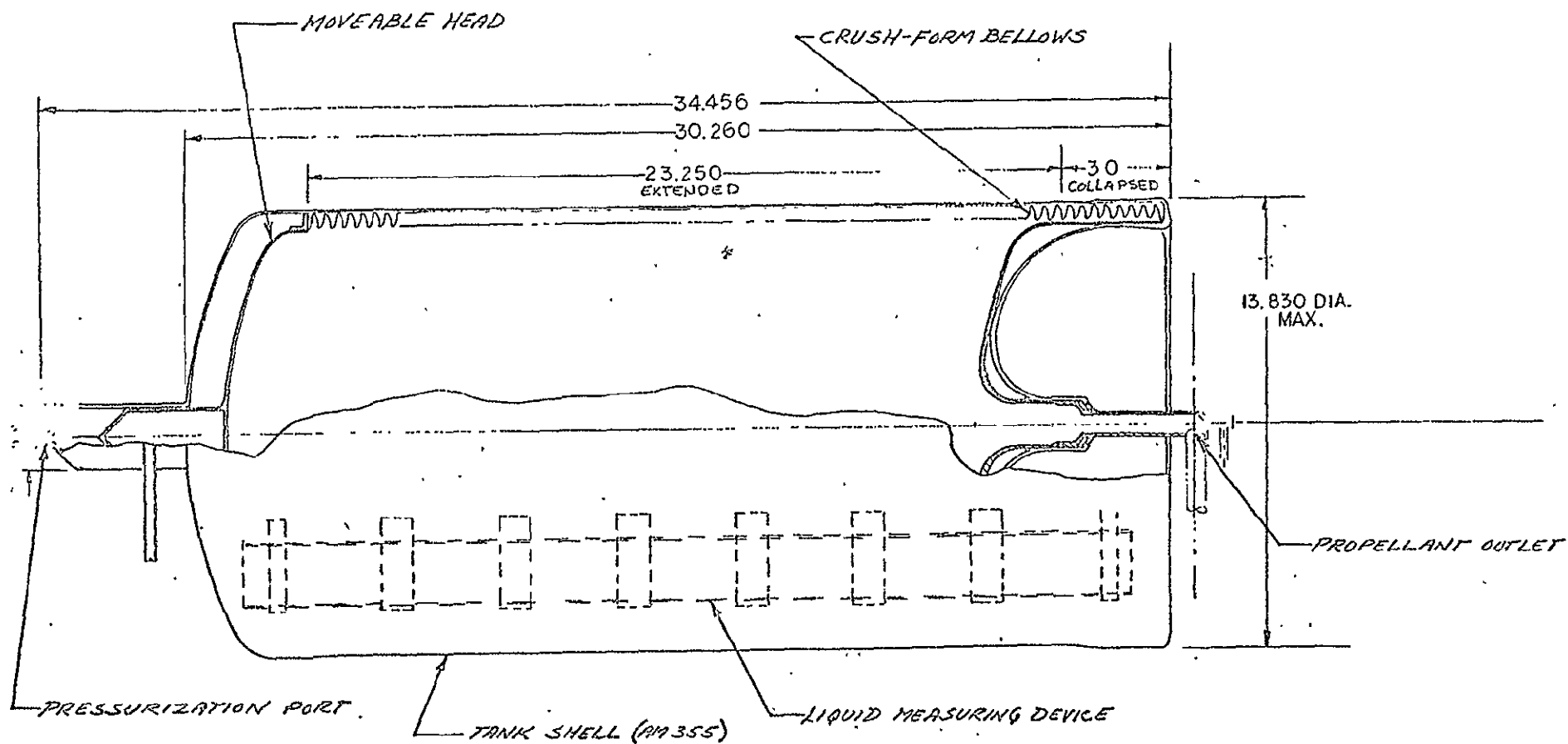


Figure 8-18 Bell Post Boost Tank.

results at the convolution inside diameter.

Considering the problems discussed above, the Bell Post-Boost tank is not considered acceptable.

The extended S-IVB/IB bellows tank is essentially four S-IVB/IB APS propellant tanks joined together as shown in figure 8-19. The philosophy behind this tank design is that it uses the same basic design concepts, i.e., materials and technology, of the existing S-IVB/IB APS bellows tank. Furthermore, the new tank would use existing bellows fixed head, bellows vibration dampener, bellows, bellows moveable head, tank ring and dome, and liquid measuring device. Although every effort was made to minimize the development cycle of this tank, it was concluded that, at best, the tank would not be available for approximately 9 months after ATP.

From table 8-6, it can be seen that the tank would supply the required total impulse of approximately 50,000 pound-seconds; however, the advantages of this new tank do not warrant a complete development program.

The Saturn V APS bellows tank is a new development. Since system weight is a prime consideration, it would be most advantageous to install the largest capacity tank possible within the S-IVB/V APS module. At present no tank with this capability exists. Therefore, it appears reasonable that if a sufficient time period would be available prior to the projected flight date, the development of an optimum volume tank should be undertaken. A bellows-type propellant tank has been investigated as a replacement for the S-IVB/V APS bladder tank. At present, preliminary lay-out drawings of this tank exist. The philosophy behind this tank is the same as that noted in the discussion of the tank using extended S-IVB/IB bellows. The tank incorporates the basic design concepts, materials and technology of the existing S-IVB/IB APS bellows tank. The expected development cycle for this new tank is approximately 12 months. From table 8-6, it can be seen that approximately a 40 percent total impulse increase could be obtained if this tank were developed. The growth potential afforded by this tank in both propellant capacity and lifetime would be applicable to other NASA programs.

Pressures and operating characteristics are compatible with the S-IVB, CSM and Lunar Orbiter.

Based on the discussions above, the only tank that satisfies the total impulse requirements, and provides the best possibility of meeting the flight date, is the Agena Target Vehicle Tank. This tank has been selected for the Phase II APS. If the launch date were to be delayed for 6 months or longer, serious considerations should be given to developing the proposed new S-IVB/V bellows tank for the OWS.

8.5.2 Thrustor Capability for Phase II

The thrustor requirements for the Phase II system are identical to those determined during Phase I and listed in section 8.1.2 except for an increase in burn life and a greater number of ignitions.

These requirements are:

1. Burn life - 200 minutes
2. Number of ignitions - 100,000

Marquardt has indicated that both of these increased requirements are within the capability of the R-IE Thrustor. These conclusions are based on the results of an R-IE life test program already completed.

During this engine life test program, thirteen engines were tested and a total burn time of 2101 minutes was accumulated. One engine accumulated a burn time of 219 minutes. No assessable failures, burnouts, or engine degradation was experienced during this test program. Practically speaking, the R-IE rocket engine has an unlimited life. The burn-life-limiting part of the engine is the molybdenum disilicide coating on the thrust chamber. The life capability of the coating has been analyzed and its life is a function of chamber wall temperature. The estimated life of the thrustor chamber based on torch tests and analytical predictions is over 50 hours. Although the demonstrated life of 219 minutes is only slightly greater than that of the mission requirement of 200 minutes, the predicted life is 15 times the required life.

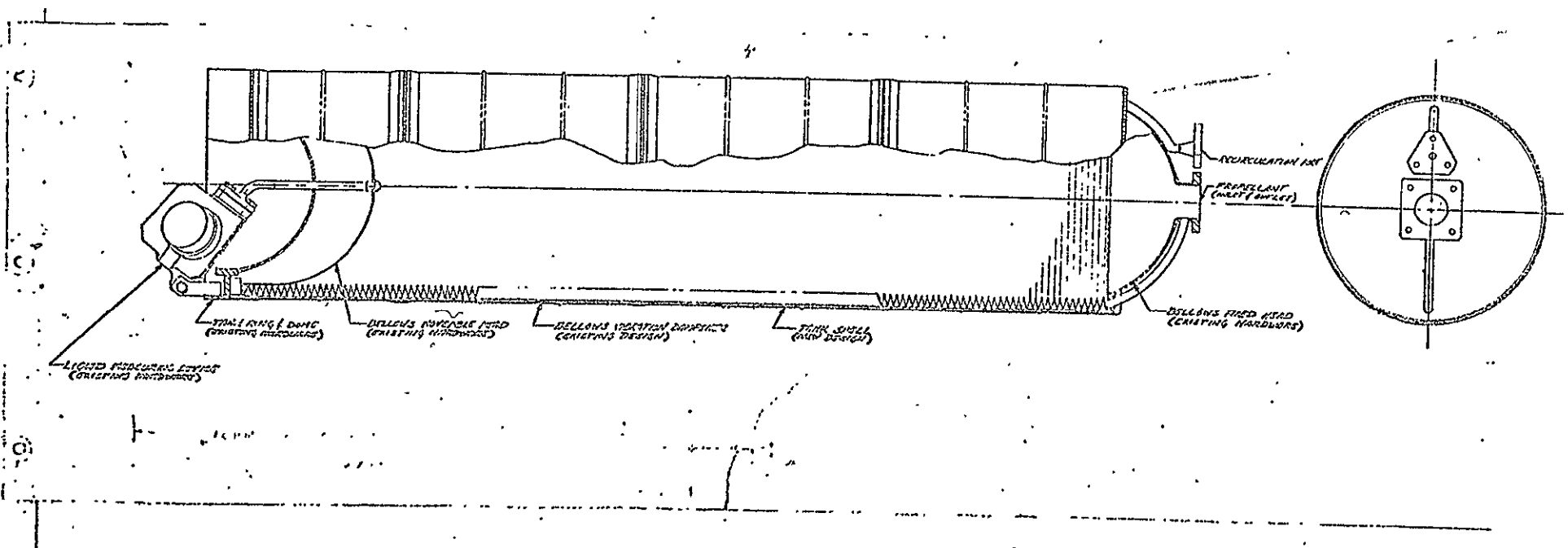


Figure 8-19 Extended S-IVB/IB Bellows Tank

During the same engine-life test program, the thirteen engines tested accumulated a total of 149,810 high altitude ignitions and 2,774 sea level ignitions. The maximum number of ignitions on one engine was 67,873. Although these cycles represent only two-thirds of the mission requirement of 100,000 cycles, other engines using valves of the same basic design, with the same materials, and using the same technology, have accumulated over 127,000 cycles.

The model R-IE engine is particularly well-suited for missions with durations measured in years. Deep-space storage capability was one of the important design criteria for the engine. Materials and clamping loads in the hardware have all been selected to provide no degradation from stress or galvanic corrosion over long periods of storage with or without engine firings. Space storage capability on the order of a year has been demonstrated on the model R-4D, 100-pound-thrust engine, on the Lunar Orbiter flights. During this program, the engine demonstrated zero valve leakage in extended space exposure as well as no ill effects due to the complete space environment (hard vacuum, micrometeorites, or solar radiation).

The R-IE engine is expected to perform equally well on long duration missions because the same technologies and design were used in its design as on the R-4D.

The engine reliability depends on the number of required engine ignitions, providing that burn time and structural requirements are much less than the engine capabilities. Statistical methods were used to predict minimum engine reliability from accumulated engine firing and valve test data. The engine reliability based on the orbital work shop mission requirements is .997.

A supplemental qualification program is recommended to qualify the engine for manned flight and extended mission requirements. The purpose of the program would be formal demonstration that the model R-IE would meet the extended mission duration requirements which were not demonstrated during the MOL Qualification Program. The program would consist of firing tests

which would accumulate 200,000 ignitions and over 400 minutes of burn time on each engine.

8.5.3 Pressurization Subsystem

It was concluded in the Phase I study, that the pressure regulated type pressurization system would best satisfy the requirements of the Phase I mission with minimum size and weight, and would provide even greater weight savings for larger systems. Based on these conclusions, reached in the Phase I study, the regulated type pressurization system was chosen for the Phase II APS.

8.5.4 Phase II System Description

Based on the conclusions reached during the Phase I study, the detachable modular concept was selected for the attitude control system. Because of the wide total impulse range for which the system had to be designed, a basic module and tanker module concept was adopted.

The basic module is composed of a Saturn V APS module structure modified for installation of two Agena target vehicle bellows-type tank assemblies, a nitrogen pressurization system and an engine package. The engine package is composed of six Marquardt R-1E engines and the engine support assembly as shown in figure 8-20.

Each basic module is composed of three major subsystems: gaseous nitrogen pressurization, positive expulsion propellant feed, and engine subsystem as shown in figure 8-21. The tanker module is identical to the basic module except that the propellant control modules and the engine package have been deleted. If the basic modules are used in a system without tanker modules, the two propellant shutoff valves in each module can be deleted.

From figure 8-21, the deletion of the propellant control modules permits single point propellant loading, thereby simplifying the loading of total system propellants. The module thermal conditioning system consists of a heater blanket, HPI and a propellant shroud as shown in figure 8-22.

WORKSHOP ATTITUDE CONTROL SYSTEM PHASE II APS CONFIGURATION

WACS-64

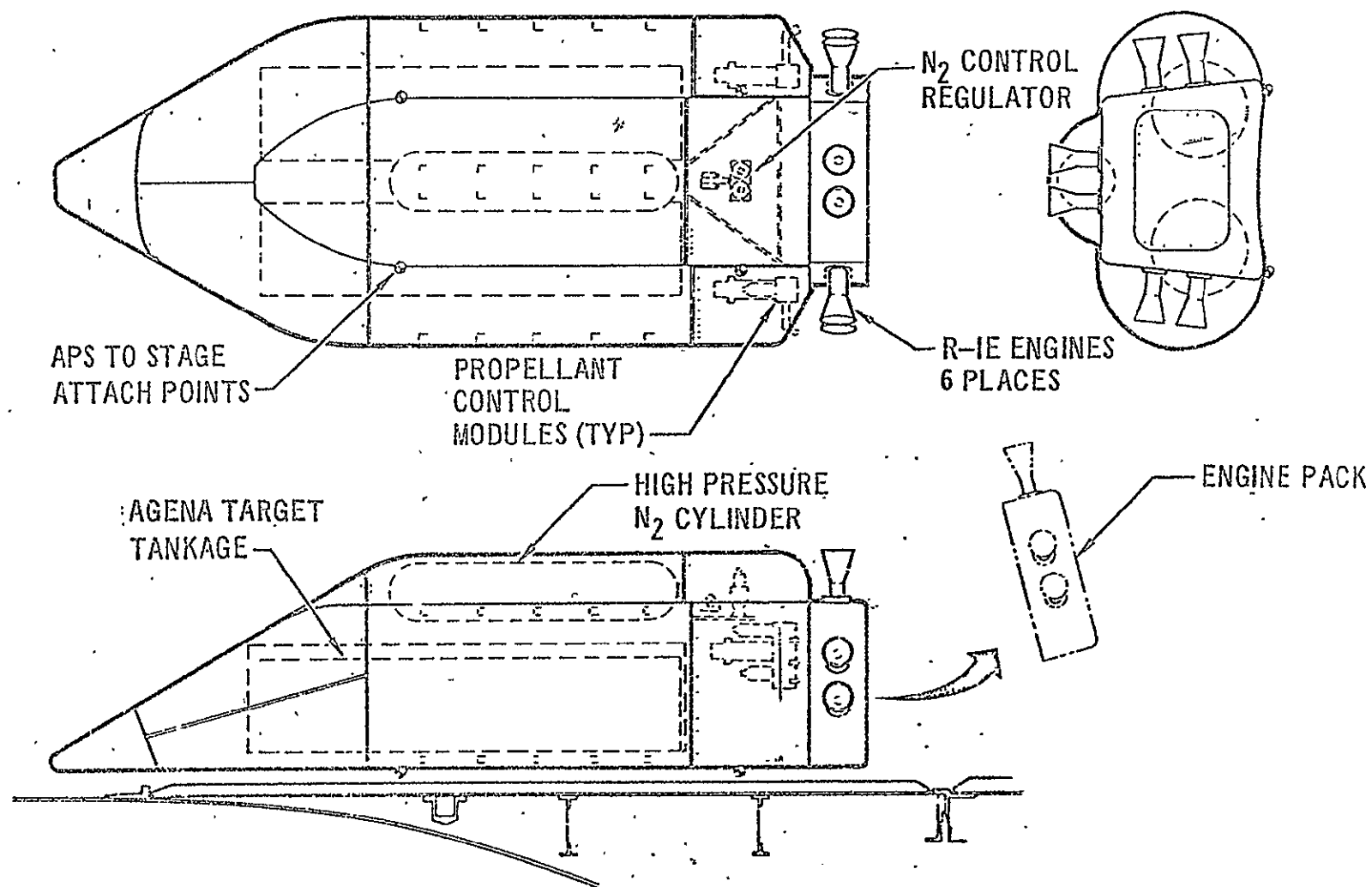


Figure 8-20 Phase II APS Configuration

8.5.4.1 Nitrogen Pressurization Subsystem

The GN_2 pressurization subsystem is composed of a S-IVB/V pressurant tank regulator, check valves, low pressure relief module and a cartridge isolation valve (see figure 8-21). The pressurant tank stores nitrogen at approximately 3000 pounds per square inch gauge in a volume of 925 cubic inches. This nitrogen is fed to a regulator assembly. The gas entering the assembly first passes through a filter and then through two regulators in a series, both of which sense downstream pressure. During the normal operation the first, or primary, regulator maintains a downstream pressure of 196 ± 3 pounds per square inch gauge. Should the primary regulator fail open, the secondary regulator begins operating and maintains downstream pressure at 200 ± 3 pounds per square inch gauge. Ambient pressure sensing ports, provided on both regulators, furnish the necessary ambient pressure references and provide a checkout capability for enabling the regulated outlet pressure settings to be checked. Regulator performance is evaluated by pressure transducers installed immediately before and after the regulators. Regulated nitrogen is fed through one check valve to the fuel tank ullage and two check valves to the oxidizer tank ullage. These check valves are a safety feature used to prevent contact of fuel and oxidizer vapors in the pressurization subsystem should both positive expulsion bellows develop a leak. Two valves are used in the oxidizer line because of the lower valve reliability in this environment.

The lower pressure relief module consists of a pressure control port, a relief valve, and a burst disc. Two relief modules are employed in the pressurization subsystem, one connected to each propellant tank ullage volume. Should the burst fail due to overpressurization from pressure surges or regulator leakage, the relief valve will prevent total depletion of the pressurant gas. During ground operation, a GSE solenoid dump valve connected to the pressure control port is opened to pressurize the tank ullage volume in order to collapse or extend the bellows during checkout and servicing.

The cartridge valve at the inlet of the pressurant tank assembly isolates the pneumatic system and provides a positive means of reducing pressurant leakage. This cartridge valve will be fired closed prior to vehicle liftoff.

The leakage rates for the Phase II pressurization system are identical to those for the Phase I system given in table 8-3. However, because of the larger pressurization system volume, a much greater lifetime results. With the burst disc intact, the lifetime is 24 years. If the burst disc were to rupture at liftoff, the system lifetime would be 10 years.

8.5.4.2 Positive-Expulsion Propellant-Feed Subsystem

The positive expulsion propellant feed subsystem provides for hypergolic propellant transfer to the engines under zero and random gravity conditions. Each subsystem consists of two Agena target vehicle propellant tank assemblies containing metal bellows for propellant expulsion, propellant control modules with propellant filters and auxiliary ports for servicing operations (purging and flushing), and propellant manifolds for distribution of propellants to the engines. A more detailed description of these components is presented in the following paragraphs.

The bellows is installed in a relaxed position and can be moved to either of its extreme positions (collapsed or extended) through the application of a small differential pressure across the bellows movable end. Pressurized nitrogen, when applied to the bellows movable end, compresses the bellows and expels propellant to the engines on demand. When the bellows is completely collapsed, the movable hemispherical end comes to rest against the mating tank bottom. This close stacking of the mating parts permits an expulsion efficiency of 98 percent or greater.

The propellant quantity measuring system for the Agena target vehicle tank was discussed in section 8.5. This will be used during propellant loading and as a source of information for flight monitoring and malfunction detection.

The propellant-control modules filter propellants and provide access into the propellant subsystems for servicing. The module contains a propellant-transfer valve, a recirculation valve, facility-lines-purge check-valves, and a system filter. The propellant-transfer valve is a direct-operating, normally-closed solenoid valve. This valve is employed to fill and drain the APS module. The propellant recirculation valve is a direct acting,

normally closed solenoid valve with two independent poppets and seats. This valve is used during propellant recirculation only. All propellant flowing to the engines and tanks passes through filters. Each filter has a 10-micron nominal and 25-micron absolute rating.

The propellant tank manifold system serves two important functions. First, it compensates for unequal module propellant consumption, and second, it provides for engine cluster redundancy (see engine subsystem).

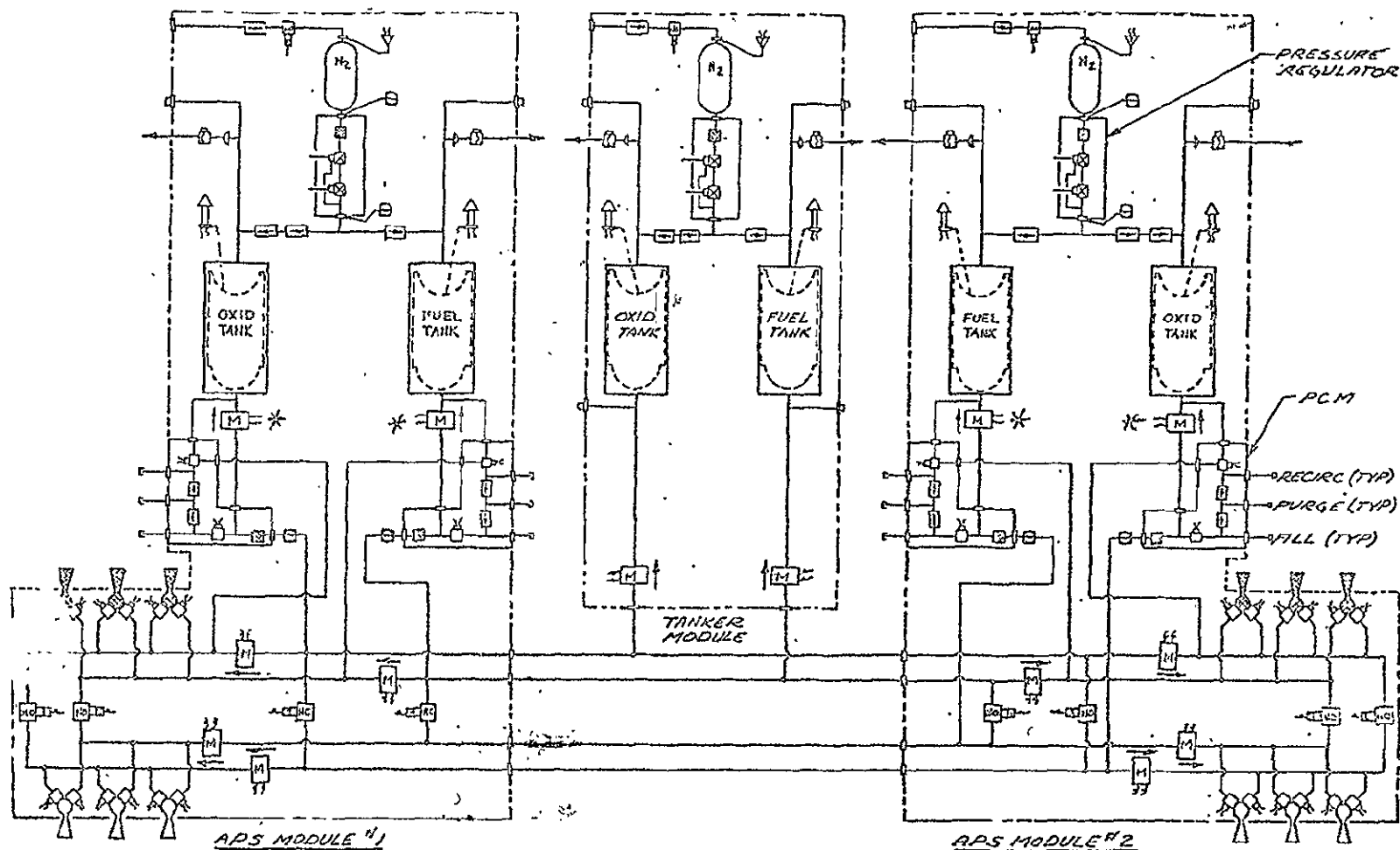
The manifold system is composed of two 360 degree manifolds. One manifold is used for the oxidizer system and the other for the fuel system. The six normally open cartridge valves shown in figure 8-21 permit the recirculation in the manifolds and engines during propellant loading. To separate the engine groups into two independent systems, these cartridge valves will be closed at liftoff.

8.5.4.3 Engine Subsystem

Six Marquardt R-1E engines will be installed in an engine package and attached to the aft end of the basic module (see figure 8-20). The engines in the package will be arranged into two groups of pitch and yaw-roll engines. One group of three engines will be connected to the propellant tanks of the module to which the engine package is attached, the other group of three engines will be connected to the propellant tanks of the other module. This engine group arrangement in conjunction with propellant tank manifolding forms two independent clusters of pitch, and yaw-roll engines in each module.

Motor operated shutoff valves will be installed in the fuel and oxidizer manifolds of each engine group. (See figure 8-21) These shutoff valves will be used to isolate the engine group should a malfunction in that group be detected. This valve was used on the Gemini spacecraft and is fully qualified and man rated.

The group shutoff valve philosophy was required to provide a system backup capability for the single engine propellant shutoff valves. The alternatives to the shutoff valve approach would be series-redundant or quad-redundant valves



* Motor valves indicated (*) can be deleted if a tanker module is not used.

FIGURE 8-21 Phase II Propulsion System Schematic

WSP WORKSHOP ATTITUDE CONTROL SYSTEM PROPOSED THERMAL CONTROL INSTALLATION

WACS-58
8-22-67

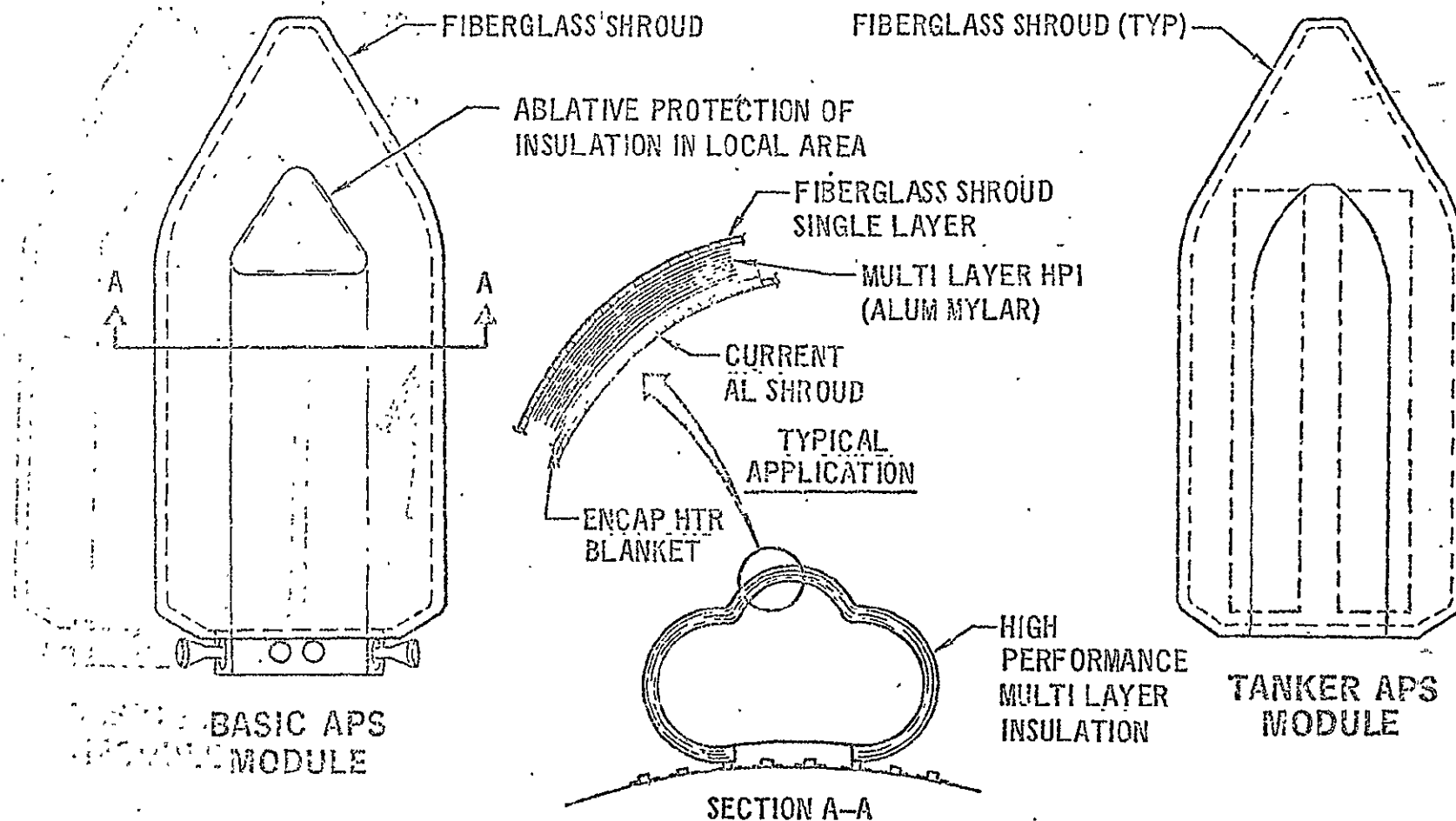


FIGURE 8-2 Figure 8-22 Proposed Thermal Control Installation

on each engine. However, the development and qualification of quad-redundant valves could not be completed within the required schedule.

The two normally closed cartridge valves between the two fuel and two oxidizer manifolds are used to interconnect the two independent propellant manifolds. Should an engine malfunction be detected and the respective cluster shutoff valve is closed, firing these cartridge valves interconnects the propellant manifolds and permits maximum utilization of propellants.

8.5.4.4 Auxiliary Propulsion System Installation

Figure 8-23 shows the approximate location of the basic and tanker modules on the S-IVB aft skirt. It should be noted, that the basic modules have been located on fin planes I and III which are approximately 90° from the solar array location (II-IV). This location for the basic modules will minimize the APS engine plume impingement upon the arrays.

8.6 Auxiliary Propulsion System Weight Summary

Table 8-7 presents the propulsion system weight breakdown and impulse capability of the Saturn IB, Phase I and Phase II auxiliary propulsion systems. The weights noted in the table are based on S-IVB/IB and S-IVB/V weight data. The purpose of this table is to illustrate the weight increase over the S-IVB/IB system.

As an example, for the two module configuration of Phase II, the weight increase over the S-IVB/IB system of two modules is 555 pounds, of which 336 pounds are propellant. If a tanker module was required, the total weight increase over the S-IVB/IB would be 555 pounds plus 694 pounds or 1199 pounds, of which 566 lbs. are propellant.

8.7 Testing

Three types of testing are required to qualify the APS for manned flight. A summary of each type, i.e., Vibration, Environmental and preflight rating (PFRT), is presented below.

WORKSHOP ATTITUDE CONTROL SYSTEM APS MODULE INSTALLATION

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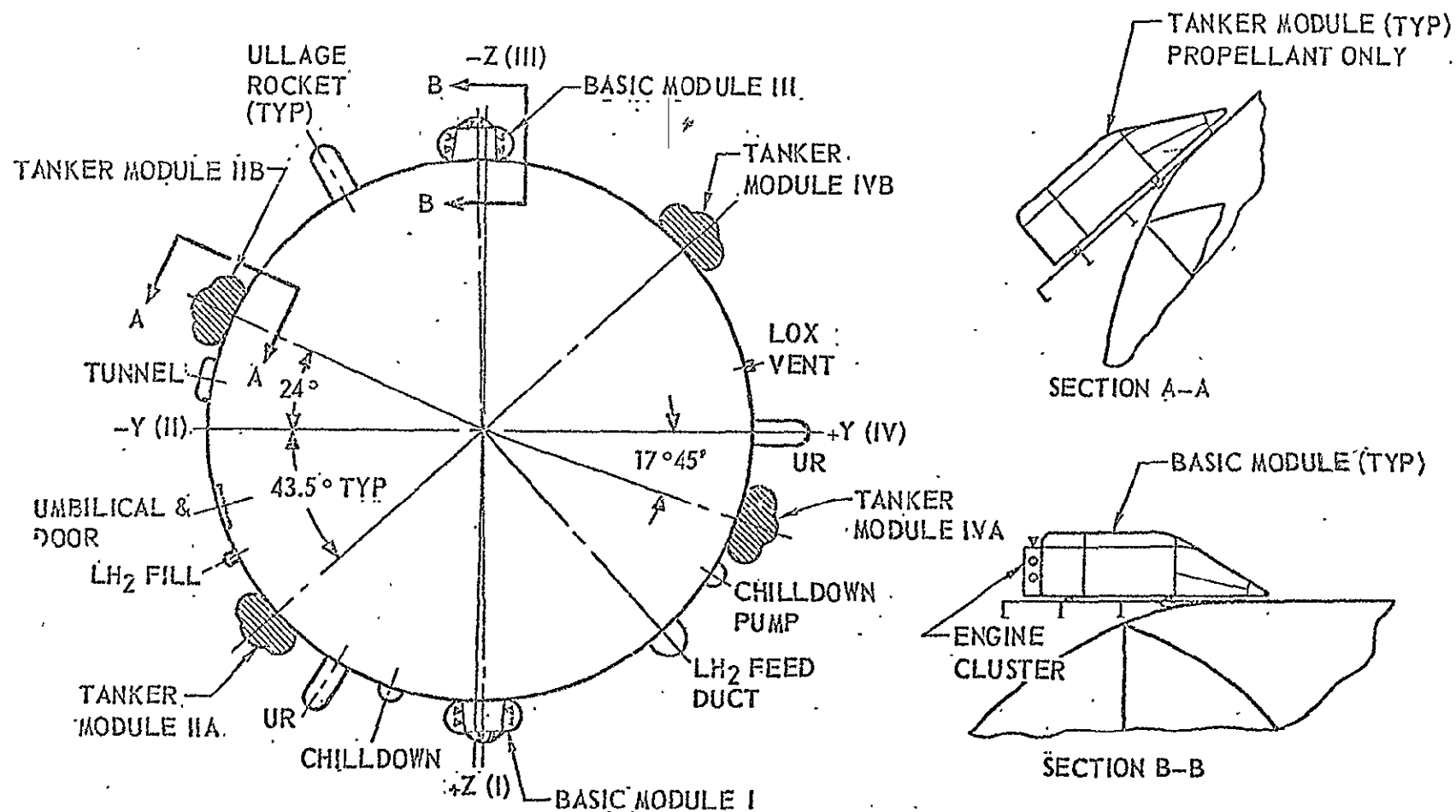


Figure 8-23 APS Module Installation

Table 8-7

WORKSHOP ATTITUDE CONTROL SYSTEM

APS WEIGHT AND IMPULSE SUMMARY

WACS-54
8-22-67

	SATURN IB APS (2 MODULES) (LBS)	PHASE I SELECTION (2 MODULES) (LBS)	PHASE II SELECTION	
			BASIC (2 MODULES) (LBS)	TANKER (2 MODULES) (LBS)
FAIRING		305	251	251
PROPELLANT TANKS		252	128	128
PRESSURANT TANK			39	39
ENGINES		38	38	
ENGINE SUPPORT ASSY			19	
ENGINE MANIFOLD AND BRACKETS			9	
MISC HARDWARE		249	249	249
SHROUD, HTI, HEATER BLANKET		110	90	30
MODULE DRY WEIGHT	699	954	823	757
PROPELLANT WEIGHT	124	248	460	460
MODULE LOADED WEIGHT	823	1202	1283	1217
MODULE MOUNT STRUCTURE BEEF-UP WEIGHT	0	14	50	66
MANIFOLD INCLUDING THERMAL CONTROL	0	45	45	2
TOTAL APS WEIGHT LOADED & INSTALLED	823	1261	1378	1285
WEIGHT INCREASE OVER S-IB MODULES (2)		438	555	
TOTAL IMPULSE ASSUMING:				
Isp = 215 SEC (PULSING)	26,700	53,400	99,000	99,000
Isp = 280 SEC (STEADY STATE)	34,700	69,400	129,000	129,000

Vibration testing of the APS module can be conducted at Douglas facilities in Santa Monica, California. This testing will provide assurance that the modifications made to the S-IVB/V module for OWS Application do not cause any dynamic problems. Since the APS structure has not been altered significantly from the S-IVB/V, no trouble is anticipated.

The testing will consist of three phases; sinusoidal, random, and shock. All testing will be performed in accordance with design requirements in each of the three primary axes: longitudinal, radial (to vehicle centerline), and 2nd transverse at 90 degrees to the radial axis. The shock test phase will consist of three shocks per axis at the specified peak acceleration level.

The environmental tests can be conducted in a space simulator at the Douglas Space Systems Center in Huntington Beach, California. This testing will result in thermodynamic qualification, and provide an evaluation of the APS module operational and heat transfer characteristics. In addition, system leakage, compared with design specifications, will be one of the primary test objectives.

Testing will be conducted in accordance with known conditions, where available, and otherwise per theoretical calculations. The vacuum environment will be at 10^{-5} to 10^{-9} Torr. Heat flux will be measured at the APS surfaces, and temperature simulation will range to deep space levels. Inert fluids will be used to simulate propellants, and propellant utilization will be programmed.

Preflight rating tests can be performed at the Douglas "Gamma Site" facilities in Sacramento, California. These tests will give pre-service evaluation of system performance, checkout and test procedures, loading conditions, malfunction effects, and possible problems associated with disassembly inspection, and cleaning. Although no problems are expected, these tests will be conducted to establish system design confidence.

8.8 Effect on GSE and Facilities

Based on an evaluation of the selected system requirements, it was concluded that no significant changes will be required to ground support equipment or

facilities except as noted below.

The APS propellant loading requirements will be increased, but assuming the worst case of two basic modules and four tanker modules, the existing S-IVB/IB APS services can easily satisfy the total propellant requirements. If a tanker module is necessary, minor modifications to the propellant recirculation panels at KSC will be required to permit recirculation of the tanker module. The change from a helium pressurant to a nitrogen pressurant will require the addition of a gaseous nitrogen umbilical at KSC, but since GN_2 is available in the pneumatic consoles for console valve operation no significant problem is anticipated.

If a tanker module is required to satisfy the requirements for total mission impulse, modifications to accommodate the tanker module during the APS module PFRT testing at the Gamma site at SACTO will be required.

STRUCTURAL REVISIONS

The study of revisions to the Attitude-Control System installation on the Orbital Workshop Vehicle included three approaches. These are:

- a. Replacement of two existing Saturn IB APS modules with two Saturn V APS modules: at their present location.
 1. With all modifications performed on the aft skirt assembly.
 2. With all modifications performed on the APS modules.
 3. With modifications performed on both assemblies.
- b. Installation of two additional Saturn IB APS modules, located approximately 90 degrees from the existing APS modules.
- c. Installation of up to four additional Saturn V APS modules on the aft skirt.

Modifications of the above listed approaches were also considered.

9.1 Replace Existing IB APS Modules with Saturn V Modules

9.1.1 All Modifications Made to Aft Skirt Assembly

To provide the exchange of APS modules on the Saturn IB aft skirt requires revision to the support structure, seal support structure, and electrical and air-conditioning provisions. The Saturn IB skirt used for comparison is Drawing No. 1B29825 (per vehicle 211); support structure installation is Drawing No. 1A97480.

Changes required at the forward end are:

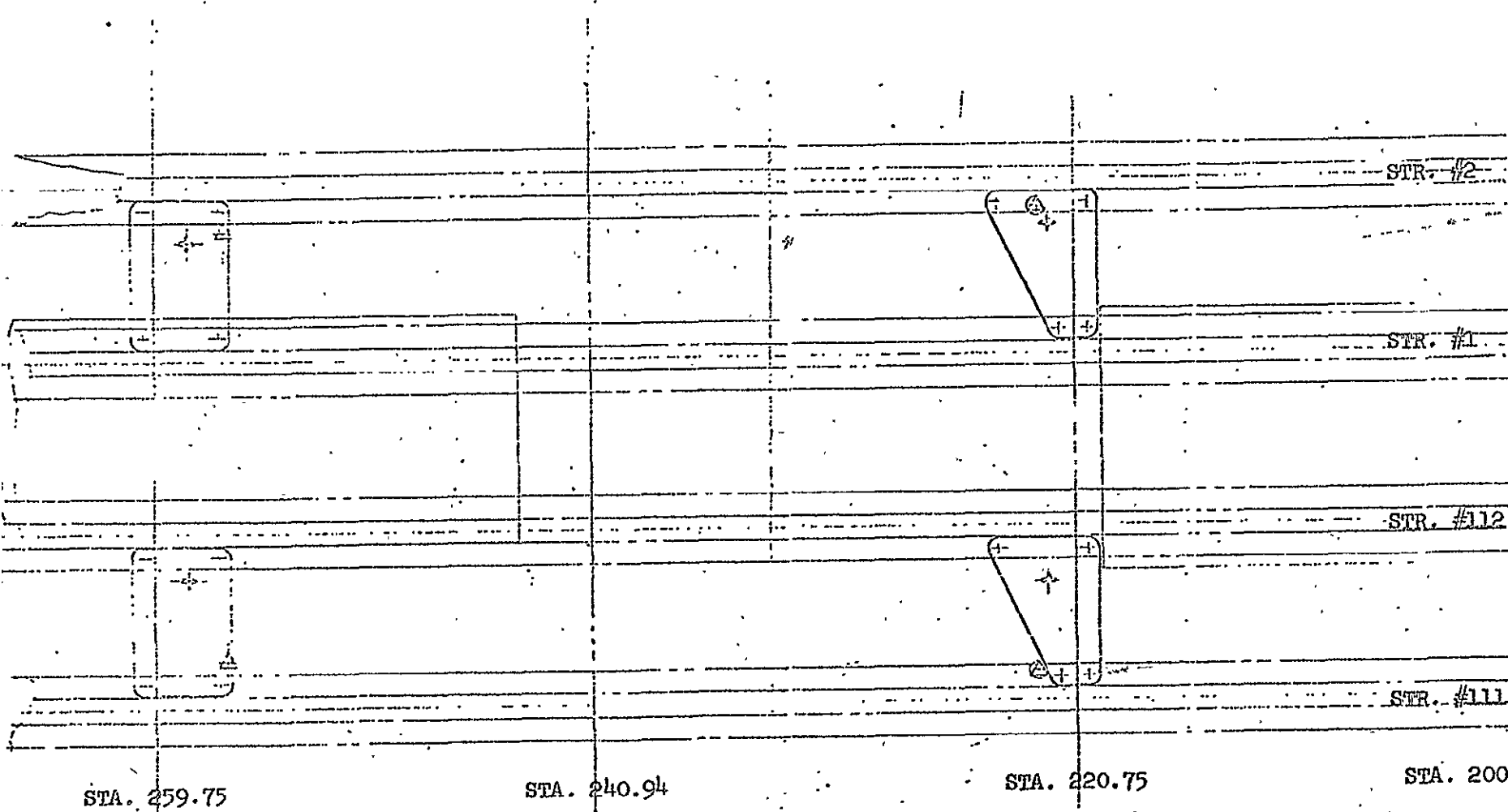
- a. Cut a hole in the skin forward of Station 256 large enough to provide access for tools required to remove the bolt and nut holding the fittings. Remove the 1A84790 fittings.

Note: Access to the area bounded by the aft dome, aft skirt, Station 240 and Station 286 can only be attained by cutting a hole in the skin.
- b. Remove the 1A97480-13 plate and its seal.

- c. Locate the centerline of the Saturn V APS 1.437 inches toward Stringer 112 (or 56) from the existing centerline of Saturn IB APS.
 - d. Machine new fittings to provide Saturn V APS support points on Saturn IB skirt structure interface. See figure 9-1.
 - e. Install new fittings.
 - f. Install new seal assembly similar to 1B39246-23 except:
 - 1. Part to be approximately 14 inches square and located aft of fittings.
 - 2. Flanges should be removed to accommodate Saturn IB stringers.
 - g. Replace the skin area cut out for access with a patch and doubler combination.
- Note: Fittings and seal assembly would be installed with "blind" fasteners.

Changes required at the aft end of the modules are:

- a. Remove the 1A84792 and 1A84793 mounts. This area is accessible from inside the skirt.
- b. Remove the 1A97480-3 plate and its seal.
- c. Locate the centerline of the Saturn V APS 1.437 inches towards Stringer 112 (or 56) from existing centerline of Saturn IB APS.
- d. Machine new fittings to provide Saturn V APS support points, and maintain existing attach location in skirt.
- e. Install new fittings and insulators.
- f. Install new seal assembly similar to 1B39246-19, except omit flanges which interfere with Saturn IB stringers.
- g. Add extra cap material to the inboard cap of Station 220 frame to prevent local crippling. Investigate need for added frame web stiffeners at centerline of APS area.



Points Indicated Thus are Existing S-IB APS.
 Points Indicated Thus are New S-V APS.

FIGURE 9-1 APS MOUNTING POINTS

9.1.2 All Modifications Made to APS Modules

This method of modification would require that permanently attached adapter fittings be provided on the APS structure. This can be done physically, but there are several problems which immediately become evident.

- a. The entire APS module would be moved outboard approximately 2.5 inches.
- b. The added distance outboard would increase the loads on the skirt frames.
- c. The air-conditioning seals would interfere with the adapters.
- d. A fairing would be required between the new APS position and the skirt. The Structures redesign for the aft skirt effort would be:
 1. Remove the 1A97480-3 and -13 plates and seal assemblies.
 2. Install new seal assemblies similar to 1B39246-19 and -23 plates with items listed below.
 3. Add new sheet metal frames to raise seals up to the APS skin (approximately 2.5 inches high). Provide cutout and seal around supports at Stringers 111 and 55 at Station 220.
 4. Add inboard cap beef-up locally on frame at Station 220. The 2.5-inch gap between existing skirt structure and APS skin would be covered by the HPI installation and shroud described in section 8.5.4.4. All work noted above could be accomplished from outside the vehicle.

9.1.3 Modifications Made to APS Modules and Aft Skirt

This method for providing support for Saturn V APS modules on Saturn IB aft skirt encompassed portions of each of the previous modifications. An adapter would be provided for the forward attach areas on the APS Module. New fittings would be provided on the skirt in the aft mounting area. Specifically, a minimum-height plate at the "blade fitting" station of the APS skin will be added to the APS. Onto this plate will be attached the Saturn IB "blade fittings" in the Saturn IB location.

The skirt structure will be modified as follows to accept the modified Saturn V APS module:

a. Forward End

1. Remove the 1A97480-13 plate and its seal assembly.
2. Install a new seal plate assembly similar to 1B39246-23 except:
 - a) Plate to be approximately 14 inches square and located aft of the fittings.
 - b) Seals to be spaced off the plate equal to the thickness of adapter on APS.
 - c) Plate to be flanged to accommodate the Saturn IB stringers.

b. Aft End

1. Remove the 1A84792 and 1A84793 mounts. This hardware is accessible from outside.
2. Remove the 1A97480-3 plate and its seal.
3. Locate the centerline of the Saturn V APS 1.437 inches toward Stringer 112 (or 56) from the existing centerline of the Saturn IB APS.
4. Machine new fittings to provide Saturn V APS support points on Saturn IB skirt structure interface. Provide new insulator at Stringers 111 and 55. See figure 8-1.
5. Install new fittings and insulators.
6. Install new seal assembly similar to 1B39246-19, except omit flanges which interfere with Saturn IB stringers.
7. Add extra cap material to the inboard cap of the closing frame (Station 220.750) to prevent local crippling. Investigate need for added frame web stiffeners at centerline of APS area.

It is felt that the third method (joint APS-aft skirt modification) is the more desirable. There is a minimum of rework required at the forward attach location, thereby minimizing the potential damage to the purge plenum and dome. The APS module is "raised" outboard by a minimum amount which is compatible with the addition of HPI.

9.2 Installation of Additional APS Modules

As an alternative to replacing Saturn IB APS modules with Saturn V APS modules, a study was made of adding two additional Saturn IB APS modules to the aft skirt. The one major criterion was that the modules should be 90 degrees apart. This would imply placing the added APS modules on Position Planes II and IV. Position Plane IV is the present location for an ullage rocket. The module can be installed approximately 10 degrees from Position Plane IV toward Position Plane I, and clear the ullage rocket which is jettisoned soon after S-IVB ignition.

The addition of the module would require backup structure at the "hat frame" Station 256 location. This is a restricted access area for rework, when the skirt is attached to the fuel tank. At the aft end, additional intercostal structure must be provided. Again, this area is highly inaccessible with the equipment panels installed and the wire harnesses connected.

An alternate plan would locate the two additional APS modules midway between Position Planes III and IV. This would eliminate interference with the ullage rocket. The difficulty of installation of this modification dictates that structural considerations indicate preference for replacing the present IB APS Modules with Saturn V APS Modules if possible.

9.3 Installation of Additional Saturn V APS Modules

Additional Saturn V APS Modules can be installed on the aft skirt as shown in figure 8-21. Structural modifications similar to those described in the previous sections would be required. The most recent location for the solar panels is at the locations shown for Modules IIB and IVA in figure 8-21, thus, two additional tanker modules can be added at the locations identified as IIA and IVB.

9.4 Meteoroid Shield

Modifications will be required to either, or both, the nose fairing and meteoroid shield. The nose fairing of Saturn V APS extends beyond the aft end of meteoroid shield.

CONCLUSIONS AND RECOMMENDATIONS

10.1 Phase I Studies

Alternate attitude control system configurations and components, to meet Phase I requirements, have been evaluated in the areas of:

- a. Propulsion Systems
- b. Thermal and Environmental Control Systems
- c. Control Logic Electronics
- d. Control Sensors (i.e., rate gyros, horizon sensors, etc.)

These systems and components were evaluated for relative reliability, weight, schedule compatibility, cost, power requirements, and vehicle installation. Recommendations of subsystem configuration and component selection have been made in each of the study areas.

- a. The recommended propulsion subsystem, which makes maximum use of existing Saturn S-IVB Auxiliary Propulsion System (APS) components, consists of two S-IVB/V APS modules with two S-IVB/IB APS (bellows) tanks in place of the Saturn V tanks in each module, and six Marquardt R-IE 22-pound-thrust engines in place of the present engines. These S-IVB/V APS modules will replace the existing S-IVB/IB modules on the S-IVB aft skirt, and be employed during boost and for the workshop missions.
- b. The recommended thermal control for the APS modules and propellant manifolds is a combination active-passive system. It consists of high-performance insulation (HPI) on the exterior of the APS modules, which is covered with a shroud for protection during boost. In addition, electrical heater blankets, installed on the inside of the modules, are needed to keep propellants and other components in acceptable temperature ranges. Heater strips and insulation are also used on the propellant manifolds.

The recommended thermal control for the electronics subsystem consists of a passive control system to maintain equipment below its high-temperature limit, and an active control system (heater elements) to add the energy required to maintain the equipment above its low-temperature limit.

- c. The recommended control-logic electronics utilizes S-IVB type (IU) spatial amplifiers to furnish commands to the thrusters, based on attitude errors and attitude rates derived from the sensors. The recommended electronics assembly requires a new package, and includes relays for switching of sensors, gains, and commands for the various mode of operation. This electronics assembly is similar in mechanization to the Saturn Flight Control Computer, but it is expected to be smaller and lighter. A power-distribution assembly includes relays for controlling power distribution to the sensors, thermal-conditioning heaters, and electronics assembly. All units can be installed in the S-IVB aft skirt.
- d. The recommended control sensors consist of slightly modified versions of the Barnes Lunar Planetary Horizon Sensor for sensing the local vertical, and the Honeywell Apollo Block II Rate Gyro Package. An Agena-type inverter, made by Gulton Industries, is recommended to supply power to the gyros.

0.2 Phase II Studies

Several orientations, which provide greater impingement of solar energy on the solar panels and OWS, have been evaluated, using the same guidelines as for Phase I to determine:

- a. Control impulse requirements.
- b. Changes or additions to mechanization in the areas of
 1. Propulsion Systems
 2. Thermal and Environmental Control System
 3. Control Logic Electronics
 4. Control Sensors

The following conclusions and recommendations are based on control system and solar power considerations evaluated during the Phase II study.

- a. The optimum attitude orientation for solar orientation is one which orients the vehicle to minimize the effects of gravity-gradient torques. The two best orientations in order of increasing impulse usage are;

1. Case 5; the x-axis is perpendicular to the orbit plane with OWS roll control and solar panel articulation about the pitch axis for maximum solar power. The end-docked CSM is optimum and small deadbands in pitch and yaw minimize impulse usage. This case requires a total mission impulse of 91,000 pound-seconds plus an appropriate margin of safety.
 2. Case 3; the z-axis is sun-oriented with a constant yaw rate of 1.0 degree per second. The CSM side-docked is optimum, and larger deadbands decrease impulse usage. This case requires a total mission impulse of 118,000 pound-seconds plus an appropriate margin of safety.
- b. The recommended propulsion system, which provides 150,000 pound-seconds of impulse, makes maximum use of Saturn S-IVB Stage APS components, consists of two S-IVB/V APS modules with two Bell Aerospace Company Agena-Target-Vehicle bellows tanks in each module in place of the present tanks, and six Marquardt R-TE 22-pound thrust engines in place of the present engines. These modules will replace the present APS modules in the S-IVB aft skirt. In addition, one tanker module which is identical to those described except that the engines will be omitted, will be installed on the S-IVB aft skirt. The propulsive and tanker modules will be connected by propellant manifold lines so that each cluster of three engines in each module will be supplied from tanks in separate modules. The tanks in the tanker module will be connected to only one propellant supply system. The APS modules will be used during boost, passivation and the Orbital Workshop Mission A and B.
- c. The recommended thermal control system is identical to that for Phase I. However, propellant will be used from the tanker module first so that its supply will be depleted early in the mission and thermal control of this module discontinued.
- d. The recommended control logic is identical to that for Phase I except that provisions are required for accepting signals from the additional sensors in the solar orientation. This requires the addition of timers from obtaining sequencing signals and electronic integrators for obtaining attitude errors from rate gyro output signals.

- e. The control sensors recommended for Phase I are also applicable to the Phase II system. In addition, LM/ATM rate gyros are an alternative to the Apollo Block II Rate Gyro Package.
- f. Development of the recommended system is feasible within the mission schedule constraints as noted in figure 10.1.

10.3 Areas for Further Study

The studies to date have concentrated on preliminary system requirements and definition of system mechanization. Several areas in which further study is required became evident during the study. A listing of these areas follows.

a. Operations:

1. The operational maneuvering of the vehicles, particularly during rendezvous and docking, should be more completely defined since these operations have an effect on horizon sensor location and mode-of-operation switching.
2. Malfunction correction requirements and the extent to which the Astronaut and ground-based Flight Controller can participate in this function should be clearly defined.

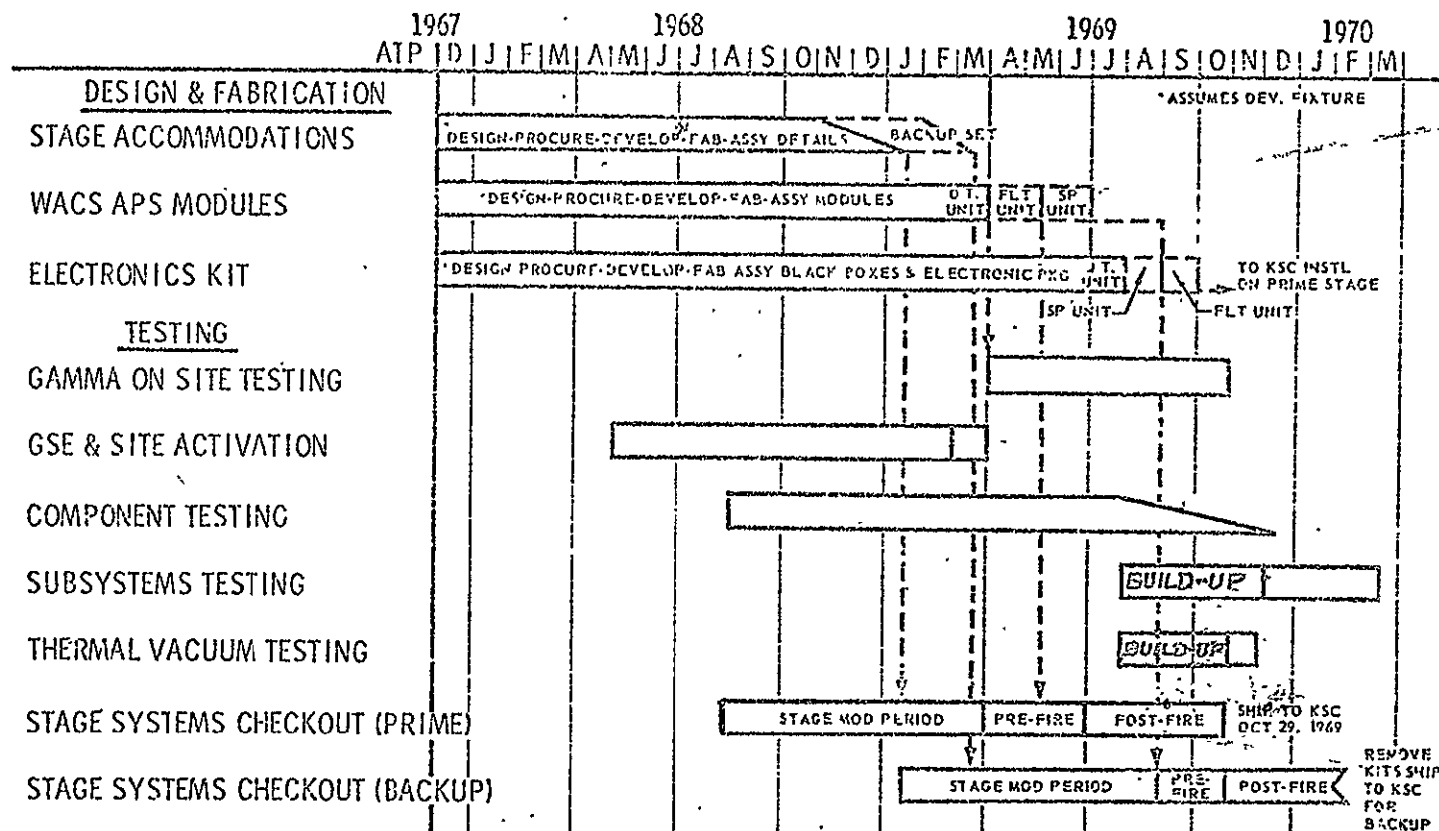
b. Design:

1. Investigate malfunction-detection and correction systems and procedures, both automatic and manual.
2. Investigate periodic earth pointing capability for experimental purposes.
3. Investigate onboard beta angle calculation and derivation of attitude commands.
4. Investigate alternate methods of updating yaw-attitude reference for Case 5 orientation.
5. Determine optimum values of rate and attitude deadbands and other control logic parameters for optimum performance and minimum sensitivity to tolerances and failures.

6. Evaluate effect of variations in mass properties on system operation including impulse usage.
7. Evaluate S-IVB/Airlock interface including command decoding and logic methods.
8. Evaluate gyro spin-up and warm-up time in relation to malfunction correction capability.
9. Investigate methods of installation and alignment of sensors on the vehicle.
10. Investigate installation of electronic equipment in the vehicle in more detail.
11. Investigate integration of vehicle attitude control and solar panel control systems.
12. Evaluate the reliability of various control system schemes in the areas of sensor and signal shaping redundancy and malfunction correction schemes.
13. Continue the market survey for wide-angle sun sensors.
14. Investigate optimized APS tankage.
15. Investigate internal installation of tanker module.
16. Evaluate effect of disturbances on attitude control system operation.
17. Analyze thermal characteristics of solar arrays to predict solar cell voltage and current instantaneously throughout entire orbits.
18. Investigate the effects of docking structure and solar array dynamics, including nonlinearities, on operation of WACS.
19. Because of the change in launch schedule, which allows more development time, the desirability of utilizing a more optimum propulsion system should be evaluated.
20. The best system configuration for use on AAP-4 based on cost and payload considerations should be determined.

WORKSHOP ATTITUDE CONTROL SYSTEM SCHEDULE

WACS-72
REV. TO
ACS-72



NOTES: THE ELECTRONICS KIT OF THE WACS WILL NOT BE AVAILABLE IN TIME FOR INSTALLATION PRIOR TO THE PRIME STAGE STATIC FIRING. THEREFORE THE BACKUP STAGE WILL BE MODIFIED IN TIME TO ACCEPT THE WACS KIT. THE SPARE ELECTRONICS KIT AND APS WILL BE INSTALLED ON THE BACKUP STAGE TO UNDERGO STATIC FIRING ENVIRONMENT APPROXIMATELY FOUR (4) MONTHS SUBSEQUENT TO THE PRIME STAGE STATIC FIRING AND APPROXIMATELY FOUR (4) MONTHS PRIOR TO THE PRIME STAGE LAUNCH. THE FLIGHT WACS APS MODULES CAN BE FIT CHECKED & INSTALLED ON THE PRIME STAGE FOR STATIC FIRING ENVIRONMENT ONLY.

Figure 10-1 Workshop Attitude Control System Schedule

11. REFERENCES

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CASE FILE DOCUMENT WITHDRAWAL

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