NAS CR- 111792

FINAL REPORT

for

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

VOLUME II - SYSTEM AND SUBSYSTEM TECHNICAL RELEASES

4 SEPTEMBER 1970

AVSD-0555-70-RR

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

for

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

NAS CR-111792

FINAL REPORT

for

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

VOLUME II - SYSTEM AND SUBSYSTEM TECHNICAL RELEASES

4 SEPTEMBER 1970

AVSD-0555-70-RR

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

for

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

PREFACE

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

The report is presented in two volumes:

Volume I - Summary

Volume II - System and Subsystem Technical Releases

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

Prepared by:

Stanley J. Brzeski
Study Project Manager

Approved by:

Ronald J. Massa
Director, Space Programs

CONTENTS

SYMBOLS AND UNITS

SYSTEM AND SUBSYSTEM TECHNICAL RELEASES

Number	Subject
L910-FWG-70-5	Baseline System Definition
F210-70-CKW-107	Trajectory Associated Characteristics for the 4-Stage Scout Launch Vehicle
F220-JEM-70-42	Definition of Requirements for Attitude Control Study
F220-JEM-70-47	Inertial Constraints on Gravity Gradient Stabilized Vehicles
F220-JEM-70-55	Vehicle Disturbance Torques
F220-JEM-70-58	Active Control of a Gravity Gradient Configuration
F220-EL-70-61	Stability of Gravitational Systems
F220-JEM-70-60	Sizing of Nutation Damper and Yo-Yo Despin Device
F220-RL-70-115	Attitude Determination Subsystem Analyses
KA-70-20	Data Handling Options
KA-70-22	Cost Considerations of Data Handling Options
KA-70-23	Data Handling - Scope of Study, Accomplishments, Summary of Results and Key Parametric Data
KA-70-24	STADAN Compatibility - Data Handling
KA-70-25	Recommended Design Approach, Data Handling and Control
ESDM-F440-4086	Solar Power Availability
ESDT/R-F440-4097	Communications
ESDT/R-F440-4002	STADAN Compatibility: Present and Predicted by Mid 1974

SYMBOLS AND UNITS

The following lists those symbols and units used throughout Volume I and Volume II of this study report.

a	Nutation damper tube radius.
a_{ij}	The element of the matrix $[A]$ in the ith row and the jth column.
A	1. The transverse tip displacement of a boom in the "2" axis direction of the satellite reference frame (F220-EL-70-61).
	2. Aerodynamic profile area of the satellite (F220-JEM-70-55).
$[A]$	Transformation matrix to transform vector components measured in a satellite fixed reference frame into vector components measured in the local vertical reference frame.
b	1. Nutation damper toroid radius (F220-JEM-70-60).
	2. A coefficient in the yaw-roll characteristic equation (F220-JEM-70-47).
B	The transverse tip displacement of a boom in the negative "1" axis direction of the satellite reference frame.
$\vec{B}$	The Earth's magnetic field intensity.
c	A coefficient in the yaw-roll characteristic equation.
C	A dimensionless constant, $C = \frac{I}{mR^2} + 1$.
C_D	Aerodynamic drag coefficient.
$d\vec{F}$	Differential force vector element.
dm	Differential mass element.
$d\vec{M}$	Differential torque vector element.
D	Maximum diameter of nutation damper.
e	Orbital eccentricity.
E	1. Young's modulus for the boom material (F220-EL-70-61).
	2. Pitch angular error resulting from orbital eccentricity (F220-JEM-70-55).

F	Tensile force in Yo-Yo cord.
F_c	Linear force analogy to $\tilde{\tau}_c$.
H_R	The relative Hamiltonian.
H_p	Angular momentum of a solar panel.
$\vec{i}$	A unit vector element of the body fixed cartesian triad $(\vec{i}; \vec{j}; \vec{k})$.
I	Polar moment of inertia of a satellite.
I_A	The area moment of inertia of the boom cross-section.
I_B	Polar moment of inertia of the large body in a two-body analysis.
I_i	The ith component of the principal mass moments of inertia.
$\bar{I}_i$	The ith component of the normalizes principal mass moments of inertia.
I_p	Moment of inertia of the solar panel.
I_R	Rotor moment of inertia.
I_s	Moment of inertia about the spin axis.
I_T	Moment of inertia about a transverse axis (normal to the spin axis).
I_x	Roll moment of inertia.
I_y	Pitch moment of inertia.
I_z	Yaw moment of inertia.
$\vec{j}$	A unit vector element of the body fixed cartesian triad $(\vec{i}; \vec{j}; \vec{k})$.
$\vec{k}$	A unit vector element of the body fixed cartesian triad $(\vec{i}; \vec{j}; \vec{k})$.
K_B	The linear spring constant analogy to K_B.
$\widehat{K}_B$	Spring constant of relative angular motion of the tip mass caused by boom bending.
K_G	The linear spring constant analogy to the gravity gradient spring constant.
ℓ	The length of the boom (F220-EL-70-61).
ℓ_F	A moment arm.
L	1. A Liapunov function (F220-EL-70-61).

2. Yo-Yo cord length (F220-JEM-70-60).

L_B The length of the boom (F220-JEM-70-55).

L_T The part of the Liapunov function due to relative kinetic energy.

L_V The part of the Liapunov function due to relative potential energy.

m Total Yo-Yo mass.

$\dot{m}$ Mass flow rate.

m_T Tip mass (F220-JEM-70-58).

M Mean anomoly of an eccentric orbit.

$\vec{M}$ 1. Generated magnetic moment (4.4.1.1).

2. The body moment vector (M_x; M_y; M_z).

M_B The body mass analogy to I_B.

M_F Mass of the nutation damping fluid.

M_M Mass of metal in the nutation damper.

M_T Tip mass (F220-JEM-70-55).

M_x Roll component of $\vec{M}$.

M_{xG} Roll axis gravity gradient moment.

M_y Pitch component of $\vec{M}$.

M_{yG} Pitch axis gravity gradient moment.

M_z Yaw component of $\vec{M}$.

M_{zG} Yaw axis gravity gradient moment.

q 1. Any generalized coordinate (F220-EL-70-61).

2. Orbital dynamic pressure (F220-JEM-70-55).

r The radius of a boom cross-section.

$\vec{r}$ Radius vector from the center of the Earth to a point on the satellite.

R Radius of Yo-Yo cord winding.

$\vec{R}$ The position vector of the satellite center of mass measured from the center of the Earth.

R_x Normalized roll inertia $R_x = I_x/I_y$.

R_z Normalized yaw inertia $R_z = I_z/I_y$.

s Laplace operator.

t 1. Wall thickness of the hollow tubular boom (F220-EL-70-61).

2. Wall thickness of the nutation damper tube (F220-JEM-70-60).

3. Time (F220-JEM-70-60).

T_R The relative kinetic energy.

v Kinematic viscosity of damper fluid.

$\vec{V}$ The velocity vector of the satellite.

V_E Mass exit velocity of nozzle.

V_R The relative potential energy.

V_s The strain energy stored in boom bending.

W Total nutation damper weight.

x A body fixed axis coordinate.

x_B Linear analogy to θ_B.

x_I An inertially fixed coordinate.

x_o A coordinate of the orbital reference frame.

x_T Linear analogy to θ_T.

y A body fixed axis coordinate.

y_I An inertially fixed coordinate.

y_o A coordinate of the orbital reference frame.

$\{y_{iL}\}$	The position of a point measured in a local vertical reference frame.
$\{y_{is}\}$	The position of a point measured in a satellite fixed reference frame.
z	A body fixed axis coordinate.
z_I	An inertially fixed coordinate.
z_o	A coordinate of the orbital reference frame.
α	1. Linear coefficient of thermal expansion for the boom material (F220-EL-70-61).
	2. Roll angle in a pitch, roll, yaw set of rotations (F220-EL-70-61).
	3. Pitch angle with respect to an inertial reference (F220-JEM-70-47).
β	2γ (F220-JEM-70-47).
γ	1. $-\theta$ = pitch angle measured from the local vertical (F220-JEM-70-47).
	2. The true anomoly in an orbit (F220-JEM-70-55).
Γ'	A nutation damper effectiveness factor.
δ	Time length of a torque pulse.
δ_{cp}	Center of pressure uncertainty.
ΔI	An inertia difference.
ΔT_o	Temperature difference across a boom.
Z	The shape function for axial motion of a point on the boom due to transverse motion under the constraint of constant boom length.
η	Absolute magnitude of the maximum relative angular deflection of the boom tip due to bending.
θ	The pitch Euler angle relative to the orbital frame.
θ_B	Body angular position.

θ_s Relative angular position of the tip mass caused by bending.

θ_T Angular position of tip mass.

$\dot{\theta}_c$ Angular coasting rate during limit cycling.

λ $(I_s/I_T) - 1$.

$\bar{\lambda}$ A frequency normalized with respect to orbital frequency.

μ Gravitational constant for Earth.

$\vec{\rho}$ Position vector of a point in the satellite with respect to the center of mass.

ρ_F Density of nutation damper fluid.

ρ_M Density of nutation damper metal.

τ Nutation damper time constant.

τ_A Aerodynamic torque.

τ_C Control torque.

τ_D Disturbance torque.

τ_{GG} Gravity gradient torque.

τ_I Torque impulse = $\int |\tau| \, dt$.

ϕ Roll Euler angle.

χ First mode shape function for boom bending.

ψ Yaw Euler angle.

ω Angular rate.

ω_f Final angular rate.

ω_n The fundamental frequency of the boom.

ω_o 1. Orbital rate (F220-JEM-70-55) and (F220-JEM-70-47).

2. Initial angular rate (F220-JEM-70-60).

ω_R	Rotor angular rate.
ω_s	Spin rate.
ω_x	Roll rate.
ω_y	Pitch rate,
ω_z	Yaw rate.
Ω_o	Orbital rate.
Ω_{iR}	The component of angular velocity of the satellite relative to the local vertical reference frame.

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X]

L910-FWG-70-5

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	F. W. Griebel	L910	8-21-70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-L91-0022		

SUBJECT

Baseline System Definition

DISTRIBUTION

List "B"

SIGNED F W Griebel

APPROVED S J Brzeski

INFORMATION REQUESTED / RELEASED

1.0 INTRODUCTION

As a result of the engineering studies and analyses conducted under Contract No. NAS 1-10014 for NASA Langley Research Center (LRC), a small satellite system concept has been established. This concept will provide a quick response capability for flight test and evaluation of single, or at most dual, earth pointing experiments dedicated to applications, science, and technology disciplines.

This TR is a baseline system definition document which contains 1) an identification and description of the mission elements, element interfaces, mission profile, mission sequence, and mission constraints, and 2) an identification and functional description of the small satellite system and its constituent equipments, subsystems, and assemblies including preliminary design data sheets for selected assemblies, subassemblies, and components.

2.0 MISSION OBJECTIVES

Increasing interest in the exploitation of demonstrated space expertise and technology for the direct benefit of mankind in such areas as earth resources, communications, navigation, national security, science and technology, and international participation has resulted in the initiation of investigation and development of a plethora of sensors and experiments for detection and measurement of various physical phenomena. This activity has created a potential requirement for small quick reaction satellite which will accept an available sensor and/or experiment without major modifications, enable the sensor and/or experiment to be flown in its required orbit, and, thereby, permit an evaluation of the sensor and/or experiment prior to its intended usage in one of the planned high cost, slow reaction, multi-purpose satellite programs, such as Nimbus, ERTS, ATS, etc. Therefore, the primary objective of the Small Satellite System defined herein will be to provide a versatile, low cost, quick reaction test bed for evaluation of sensors and experiments. The secondary or detailed mission objectives will vary with each flight and will be a function of the particular sensor and/or experiment carried by the satellite.

In order to achieve the primary objective, the Small Satellite must 1) be stable in an earth pointing attitude, 2) have two and/or three axis attitude control, 3) have high accuracy attitude determination capability, 4) have flexible data handling and control capability, 5) have a standardized experiment satellite interface, 6) have provisions for simple, easy, and rapid experiment integration and checkout, and 7) be capable of functioning at a range of orbits for periods of time sufficient to provide an adequate evaluation of the particular sensor and/or experiment.

3.0 MISSION REQUIREMENTS

3.1 MISSION ELEMENTS

The successful accomplishment of a Small Satellite System mission requires the integration and operation of the five (5) systems identified in Figure 3-1. Each of these systems is defined below and will consist of the hardware, software, facilities, and personnel required for mission implementation.

3.1.1 Primary Payload System

The Small Satellite System (described herein in detail will be the primary payload system consisting of the flight spacecraft with its experiment, as development and flight hardware, and its associated ground support equipment (GSE); design and control documentation and other required software, all facilities and personnel to design, develop, fabricate, assemble, inspect, and test the flight spacecraft and GSE as well as to support pre-launch, launch and flight operations.

3.1.2 Launch Vehicle System

The launch vehicle system will be comprised of the four stage Scout launch vehicle with its guidance subsystems, spacecraft adapter, and ascent heat shield plus the supporting GSE, software, and associated manpower as described in the Scout Users Manual, Volumes 1 through 5, dated 1 April 1969.

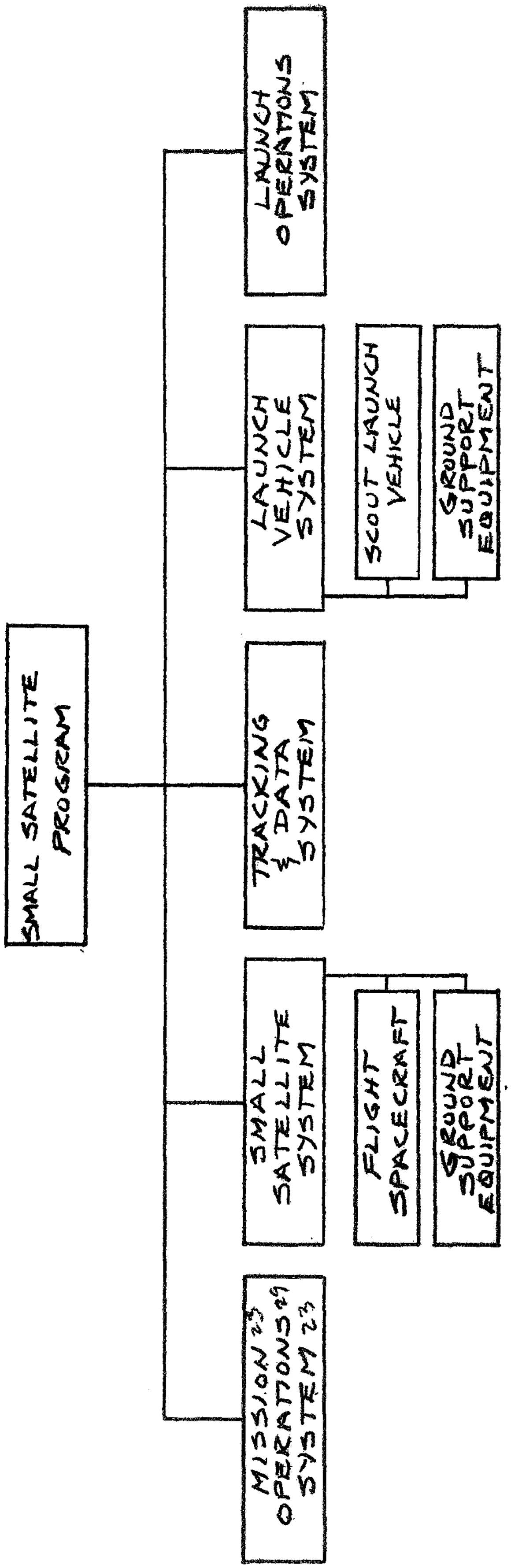


FIGURE 3-1
MISSION ELEMENT IDENTIFICATION

3.1.3 Mission Operations System (MOS)

The mission operations system will be located at NASA Langley Research Center and will consist of 1) NASA management and operation teams that plan, direct, control, and execute flight operations after injection of primary payload spacecraft in to orbit, 2) appropriate procedures and documentation, and 3) communications and data reduction facilities.

3.1.4 Tracking and Data System (TDS)

The tracking and data system will be the NASA Space Tracking and Data Acquisition Network (STADAN); the appropriate NASA Communications (NASCOM) and other circuits assigned to handle mission data and commands; and other NASA facilities, equipment, and personnel assigned to handle mission data, to command, and to support the mission operations.

3.1.5 Launch Operations System (LOS)

Depending on the particular sensor and/or experiment requirements, the launch operations system will be Wallops Island, San Marco Island, or the USAF Western Test Range including its launch complex, missile and spacecraft assembly facilities, and tracking equipment with the operating and support personnel responsible for planning and executing the pre-launch and launch-to-injection phases of the mission

3.1.6 Interfaces

The types of interfaces between the mission elements are identified in Figure 3-2. Each of these interfaces will be defined in detail during the Small Satellite System design, development, and operational cycle (Phases C/D) and will be documented in Joint Operating Plans (JOP's), Program Requirements Documents (PRD's), etc. prior to final acceptance of the Small Satellite System at the Launch Operations System (LOS).

3.2 MISSION PROFILE

The Small Satellite System mission will consist of the following phases:

a) Pre-launch Phase
b) Launch Phase
c) Injection Phase
d) Re-orientation Phase
e) Scientific Mission Phase
f) Mission Termination Phase

A preliminary sequence of events for the above mission phases has been formulated and is shown in Table 3-I.

FIGURE 3-2 MISSION ELEMENT INTERFACES

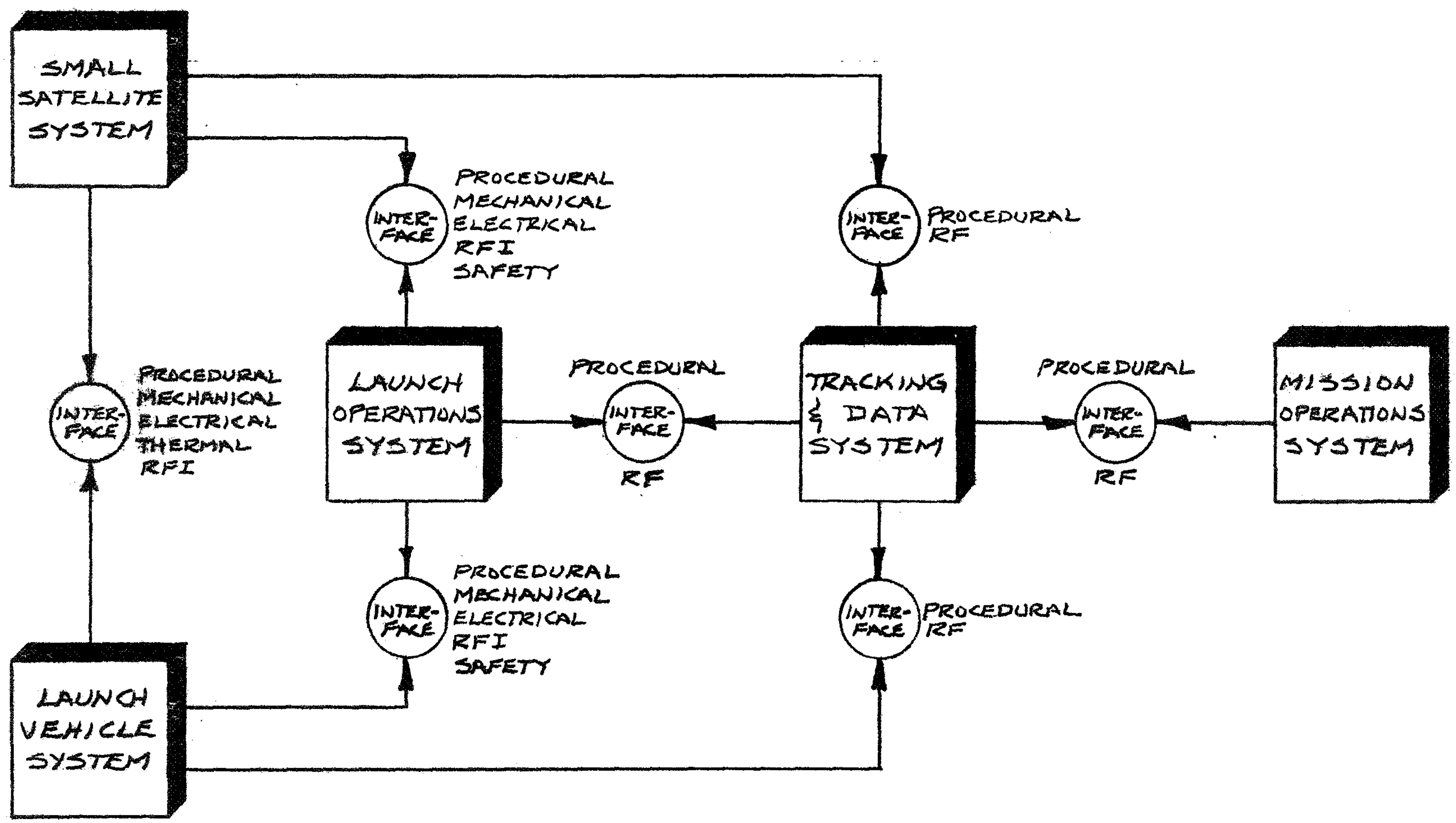


TABLE 3-I SEQUENCE OF OPERATIONS

MISSION PHASE	EVENT NO.	EVENT	TIME	REMARKS
PRELAUNCH	1	MOTOR RECEIVING AND INSPECTION	R-33 DAYS	AT MISSILE ASSY BLDG
	2	VEHICLE UNCRATING & INSPECTION	R-28	AT MISSILE ASSY BLDG
	3	SPACECRAFT UNLOADING & INSPECTION	R-28	AT S/C CHECKOUT FACILITY
	4	S/C CHECKOUT	R-26	DETAILS TO BE DETERMINED
	5	LV BUILD-UP (STGS 1, 2, & 3)	R-26	AT MISSILE ASSY BLDG
	6	LV SYSTEMS CHECKS	R-15	" " " "
	7	S/C RFI CHECK	R-10	" " " "
	8	PREP. FOR SPIN CHECK	R-10	SPIN TEST FACILITY
	9	REC. 4TH STG MOTOR	R-10	" " "
	10	INSTALL MOTOR & SPIN ADAPTER	R-10	" " "
	11	PRELIM. BALANCE	R-10	" " "
	12	INSTALL S/C ON 4TH STG.	R-9	" " "
	13	SPIN BALANCE	R-9	" " "
	14	TRANSPORT VEHICLE TO PAD (STGS. 1, 2, & 3)	R-8	
	15	MATE VEHICLE TO LAUNCHER (STAGES 1, 2, & 3)	R-8	ON-PAD OPERATION
	16	LOS ELECTRONIC FUNCT. TEST	R-7	" " "
	17	TRANSPORT 4TH STG PLUS S/C TO LAUNCH PAD	R-6	" " "
	18	INSTALL 4TH STG & S/C ON LV	R-6	INCLUDES CONN. OF HARDLINE UMBILICAL & BATT. ISOL. PLUG
	19	INSTALL S/C ORDNANCE W/SHORTING PLUGS	R-5	ON-PAD OPERATION
	20	INSTALL ASCENT H/S	R-5	" " "
	21	INSTALL LV ORDNANCE	R-5	" " "
	22	FINAL INSPECTION	R-3	" " "
	23	DRY RUN INCL. UMBR. RETRACT.	R-2	" " "
	24	PREP. FOR LAUNCH	R-1	" " "
	25	COUNTDOWN INITIATION	T-405 MINS.	
	26	LV ELECTRONICS CHECKOUT	T-375	
	27	S/C ELECTRONICS CHECKOUT	T-375	SEQ. & TIME TO BE DETERMINED
	28	LV RCS FUELING	T-225	
	29	LV/LAUNCHER SECURING & ERECTION	T-150	
	30	LV IGNITION & DESTRUCT SYSTEM CHECKOUT	T-60	
	31	COUNTDOWN EVALUATION	T-45	BUILT-IN HOLD
LAUNCH	32	INITIATE TERMINAL COUNTDOWN	T-30 MINS.	
	33	SEPARATE S/C UMBILICAL	T-3	
	34	VERIFY INTERNAL S/C POWER ON	T-3	
	35	1ST STAGE IGNITION	T-0 SECS.	
	36	START TIMER	T+0.03	
	37	1ST STAGE BURNOUT	T+76	
	38	2ND STAGE IGNITION	T+86	
	39	2ND STAGE BURNOUT	T+126	
LAUNCH (CONTINUED)	40	REMOVE ASCENT H/S	T+130	
	41	ACTIVATE "C" BURN CONTROLS	T+131	
	42	3RD STG SQUIB IGNITION	T+134	
	43	3RD STG IGNITION	T+134	SEPARATE 2ND STAGE
	44	3RD STG BURNOUT	T+171	
	45	ACTIVATE "C" COAST CONTROLS	T+175	
	46	SPIN MOTOR IGNITION	T+464	
	47	3RD STAGE SEPARATION	T+485	
	48	RETRO FORCE COMMAND	T+486	
INJECTION	49	4TH STG IGNITION	I=0 SECS.	T+491
	50	4TH STG BURNOUT		
	51	4TH STG SEPARATION		
	52	ARM S/C DESPIN ASSY		
	53	INITIAL DETERMINATION OF ORBITAL ELEMENTS		RANGE TRACKING NETWORK
	54	VERIFY SEPARATION		S/C SEPARATION SWITCH
RE-ORIENTATION	55	TURN ON S/C XMITTERS	R=0 MINS.	I+
	56	VERIFY S/C STATUS	—	
	57a	ACTUATE DESPIN ASSY	—	SEQUENCE OF EVENTS AND TIMES TO BE DETERMINED
	57b	DEPLOY SOLAR PADDLES	—	
	57c	EXTEND BOOM TO POS. #1	—	
	57d	EXTEND BOOM TO POS. #2	—	
	58	TURN ON ATTITUDE DETERMINATION SUBSYST.	—	
	59	VERIFY SPIN RATE IS 0 & S/C STABILIZED IN EARTH POINTING ATTITUDE WITHIN 5°	—	
	60	ENABLE RCU	—	
	61	TRIM & CORRECT S/C ATTITUDE TO < 1°	—	
	62	VERIFY ATTITUDE & STABILITY	—	
	63	TURN OFF RCU	—	
SCIENTIFIC	64	TURN ON EXPERIMENT.	M=0	SEQUENCE OF EVENTS AND TIMES TO BE DETERMINED
	—			
	—			
	—			
	—			
	—			
TERMINATION	—	SHUT DOWN S/C		SEQUENCE OF EVENTS AND TIMES TO BE DETERMINED

3.2.1 Pre-Launch Phase

The pre-launch phase begins with the receipt of the Scout launch vehicle rocket motors at the LOS, encompasses the Launch Vehicle and Small Satellite System flight spacecraft assembly, integration, balance, and functional checkout activities, and terminates at the initiation of the final operational countdown. The duration of this phase will be approximately 33 days. In addition, other activities such as launch control checkout, range tracking verification, etc. leading to commitment to launch, will be completed during this time period.

3.2.2 Launch Phase

The launch phase commences with initiation of final countdown and terminates with third stage separation. Generally, final countdown will be followed by first stage ignition, launch vehicle lift-off, and first stage burnout; second stage ignition, separation of first stage, burnout of second stage and ascent heat shield jettison; third stage ignition, second stage separation, third stage burnout, spin-up of the third stage vehicle to approximately 180 rpm and third stage separation. The time required to accomplish these events will be approximately 38 minutes.

3.2.3 Injection Phase

The injection phase starts with ignition of the fourth stage vehicle and terminates with separation of the fourth stage from the Small Satellite System spacecraft. The fourth stage burn will place the spacecraft into the proper orbit. The time for accomplishment of this phase is a function of the selected orbit and launch site and will be determined from the requirements of the particular sensor and/or experiment in the spacecraft.

3.2.4 Re-orientation Phase

The re-orientation phase begins with fourth stage separation, includes initiation of the despin mechanism to reduce the spacecraft spin rate and deployment of the gravity gradient boom and the solar paddles, and terminates when the satellite attitude has been stabilized with the spacecraft yaw axis pointed toward earth. The time for accomplishment of this phase is dependent on the particular sensor and/or experiment requirements and may range from three-quarters of an orbit to several orbits.

3.2.5 Scientific Mission Phase

The scientific mission phase will commence with the completion of the satellite reorientation and stabilization and will terminate when the specific mission has been completed. During this phase, the sensor and/or experiment will be operated, satellite attitude will be maintained, and data collection, processing, and transmission, as well as command reception will be accomplished. The duration of this phase will be from 3 weeks to 3 months depending on the particular sensor and/or experiment being evaluated. The exact sequence of events and their times will be determined prior to each Small Satellite System flight.

3.2.6 Mission Termination Phase

Mission termination will occur 1) upon a failure of one or more critical components of the flight spacecraft or 2) by command of competent authority. Specific shutdown procedures will be developed during subsequent program phases.

3.3 MISSION CONSTRAINTS

The following items are the major constraints placed on the Small Satellite System by the mission and will be considered fixed.

3.3.1 Mission Schedule

For the Small Satellite System, the mission schedule will be flexible and will be limited only by 1) the time necessary to design, develop, qualify, and build a suitable flight spacecraft, 2) the availability of candidate sensors and/or experiments, and 3) the launch date and time determined by mission analysis to be the optimum for satisfaction of the detailed Small Satellite System mission objectives. At this time, there is no established schedule for a Small Satellite System program.

3.3.2 Commitment to Launch

The Small Satellite System will utilize instrumentation and checkout procedures necessary to detect malfunctions and non-standard system operation. In the event of non-standard performance, the Small Satellite System contractor will be capable of evaluating its effect on mission performance and will affirm commitment to launch or request hold for maintenance as prescribed by the mission launch and hold criteria in effect at the time of launch.

3.3.3 Mission Life

The mission life of the Small Satellite System spacecraft flights will normally be from 3 weeks to 3 months in orbit. The spacecraft, however, will be designed to have a minimum orbital lifetime of one (1) year except that the storage capacity of expendables, such as the gas for the Attitude Control Subsystem, will be sized for three (3) months of operation.

3.3.4 Orbit

The orbits for all Small Satellite System missions will be dependent on the requirements of the particular sensor and/or experiment in the spacecraft and will range from 200 to 1000 nautical miles with inclinations of 3° to 99°.

3.3.5 Environment

All Small Satellite System equipment will be required to operate and/or survive, as appropriate, in the natural or induced environments shown in Table 3-II.

3.3.6 Launch Vehicle Constraints

The Small Satellite System flight spacecraft must be designed to be compatible with the launch vehicle constraints specified in Section 4, Volume III, Scout Users Manual dated 1 April 1969. The spacecraft design factors affected by these constraints will generally consist of the following:

a) Weight, geometry, and center of gravity.
b) Mechanical attachment to and separation from the "E" Section spacecraft adapter.
c) Location, mounting, and selection of the spacecraft checkout umbilical connector.
d) Electromagnetic interference.
e) Static and dynamic mass unbalance of the spacecraft when attached to the fourth stage motors.
f) Operational sequences and procedures.
g) Ability to withstand flight environments.

3.3.7 Launch Operations Constraints

The Small Satellite System design will be compatible with the launch operations constraints specified in the Wallops Station Handbook and the Pacific Missile Range Manual, Range User's Handbook, Volume II.

3.3.8 Tracking & Data System Constraints

Only those ground stations of the NASA STADAN tied by microwave or land line to the Mission Operations System will be used to transmit commands to the spacecraft and to receive data from the Small Satellite System spacecraft.

Tracking of the spacecraft will be accomplished by the NASA Minitrack System (part of STADAN) and the RF output of the spacecraft transmitter beacon must be compatible with the applicable requirements of NASA GSFC Report X539-64-159, Satellite Tracking and Data Acquisition Network Facilities Report (STADAN).

TABLE 3-II ENVIRONMENTAL CONDITIONS-TYPICAL SATELLITE MISSION

MISSION DESCRIPTION			ENVIRONMENTAL CONDITIONS										ELECTROMAGNETIC INTERFERENCE								
PHASE	FUNCTION	DURATION	TEMPERATURE (°F)	PRESSURE (mm Hg)	RELATIVE HUMIDITY	SAND & DUST	PRECIPITATION	CORROSIVE ATMOSPHERE	FUNGUS	ATMOS. WIND	SOLAR WIND	METEROIDS & DUST	GENERATED	SOLAR	PARTICLE	MAGNETIC FIELDS	SPIN	VIBRATION	SHOCK	ACCELERATION	ACOUSTIC NOISE
MANUFACTURING AND ASSEMBLY	MANUFACTURING, ASSEMBLY, AND TEST OF SUBSYSTEMS INTO FLIGHT SPACECRAFT FOR ACCEPTANCE TEST; INTEGRATE EXPERIMENTS; CHECKOUT AND ACCEPTANCE TEST OF SPACECRAFT	APPROX. 6 MONTHS	60 TO 80	760	40-50%	NEGLIGIBLE	NOT APPLICABLE	NONE	NON-FUNGUS NUTRIENTS TO BE USED FOR ALL EQUIPMENTS	NOT APPLICABLE	NOT APPLICABLE	NOT APPLICABLE	VARIOUS FORMS OF ELECTROMAGNETIC FIELDS MAY BE GENERATED BY THE SPACECRAFT, GSE, AND PROJECT FACILITY EQUIPMENT REFER TO MIL-I-26600	NEGLIGIBLE	NEGLIGIBLE	EARTH FIELD 0.4 GAUSS	SPIN BALANCE AT 250 RPM	NEGLIGIBLE	NEGLIGIBLE	EARTH GRAVITY $G_E = 32.17\ FT/SEC^2$	NEGLIGIBLE
STORAGE AND TRANSPORTATION	GSE PROVIDED TO PROTECT SPACECRAFT AND TO MAINTAIN CLEAN ROOM CONDITIONS DURING ANY OPERATIONS NOT CONDUCTED IN CLEAN ROOM	APPROX. 2 WEEKS	-35 TO +150 INCLUDES 25° SOLAR RADIATION RISE	760 TO 87 INCLUDES AIR TRANSPORT	0 TO 100%	10^{-4} TO 3 mm DIA DUST BLOWN BY 60 MPH WIND	RAIN:- MAX. RATE OF 10 cm/hr BLOWN BY 60 MPH WIND ICE:- UP TO 1 cm FROZEN ON EXPOSED SURFACES SNOW:- 1 TO 3 mm DIA. CRYSTALS AT MAX. RATE OF 7.5 cm/hr BLOWN BY 60 MPH WIND HAIL:- 0.5 TO 5.0 mm DIA. FROZEN WATER PELLETS AT MAX. RATE OF 10 cm/hr BLOWN BY 60 MPH WIND FOR 15 MIN. ABOVE 32°F SLEET:- 0.75 TO 1.5 mm DIA. ICE PELLETS AT MAX. RATE OF 10 cm/hr BLOWN BY 60 MPH WIND FOR 30 MIN. BELOW 32°F	SALT FOG OF 5% SALT, 95% WATER AT SPEC. GRAVITY 1.023 TO 1.037 AND PH OF 6.5 TO 7.2 AT 95°F, 85% R.H.		0 TO 60 KNOTS			VARIOUS FORMS OF ELECTROMAGNETIC FIELDS MAY BE GENERATED BY TRANSPORTATION EQUIPMENT	INTENSITY VARIES SINUSOIDALLY FROM 0 TO 0.140 WATTS/cm^2 IN 24 HOUR PERIOD SPECTRUM PERCENTAGE 8% - ULTRAVIOLET 41% - VISIBLE 51% - INFRARED	NEGLIGIBLE	EARTH FIELD 0.4 GAUSS	NOT APPLICABLE	$\pm 1.3\ G_E$ @ 2-26 Hz $\pm 5.0\ G_E$ @ 52-300 Hz	RELATED TO TRANSPORTATION AND HANDLING	EARTH GRAVITY $G_E = 32.17\ FT/SEC^2$	RELATED TO TRANSPORTATION EQUIPMENT
PRELAUNCH	SPACECRAFT CHECKOUT IN HANGAR SPACECRAFT INTEGRATION, TEST, AND CHECKOUT WITH SCOUT LAUNCH VEHICLE	APPROX. 33 DAYS	60 TO 80 (INDOORS) 20 TO 150 INCLUDES 25° SOLAR RADIATION RISE (LAUNCH PAD)	760	40-50% (INDOORS) 10-100% (LAUNCH PAD)	NEGLIGIBLE (INDOORS) 10^{-4} TO 3 mm DIA. DUST BLOWN BY 60 MPH WIND (LAUNCH PAD)		NONE (INDOORS) SALT FOG OF 5% SALT, 95% WATER (LAUNCH PAD)		NONE (INDOORS) 0 TO 60 KNOTS (LAUNCH PAD)			VARIOUS FORMS OF ELECTROMAGNETIC FIELDS MAY BE GENERATED BY THE SPACECRAFT, GSE, LAUNCH VEHICLE, AND LAUNCH FACILITY EQUIPMENT		NEGLIGIBLE	EARTH FIELD 0.4 GAUSS	SPIN BALANCE WITH 4TH STAGE BOOSTER - 250 RPM	NEGLIGIBLE	NEGLIGIBLE	EARTH GRAVITY $G_E = 32.17\ FT/SEC^2$	NEGLIGIBLE
LAUNCH	SPACECRAFT BOOST BY FIRST THREE STAGES OF SCOUT LAUNCH VEHICLE. ASCENT HEATSHIELD REMOVED AFTER SECOND STAGE BURNOUT SPIN UP OF FOURTH STAGE AND SPACECRAFT BY THIRD STAGE AFTER MAIN ENGINE BURNOUT	APPROX. 38 MINUTES	20 TO 150 INCLUDES 25° SOLAR RADIATION RISE (LAUNCH PAD) FLIGHT TEMP. TO BE DETERMINED FROM ATMOSPHERIC ASCENT HEATING, SOLAR RADIATION, INTERNAL EQUIP. OPERATION, SPACE SHADOW EFFECTS MATERIALS ABSORPTIVITY EMMISIVITY CHARACTERISTICS AND RELATED CONFIGURATION	760 TO 5×10^{-8}	10-100% (LAUNCH PAD) THROUGH NONE IN FLIGHT	10^{-4} TO 3 mm DIA. DUST BLOWN BY 60 MPH WIND TO NONE		SALT FOG OF 5% SALT, 95% WATER TO NONE		VELOCITY VS ALTITUDE PER ARDC HANDBOOK	AT 1 AU (DYNES/cm^2) 10^{-7} QUIET SUN 10^{-5} AVERAGE 10^{-3} MAJOR ACTIVITY			MAXIMUM 0.140 WATTS/cm^2 AT 1 AU ALBEDO FACTOR, EARTH = 0.40 LIGHT PRESSURE 2×10^{-7} LB/FT^2 AT 1 AU PARTICLE WAVELENGTH FRACTIONS OF TOTAL ENERGY VARY THROUGHOUT ENTIRE SPECTRUM	NEGLIGIBLE TO 10^6 ELECTRONS/cm^2/SEC WITH E > 0.5 Mev	VARYING FROM 0.2 GAUSS AT THE EQUATOR TO 0.36 GAUSS AT THE POLES	SPIN UP BY 3RD STAGE TO APPROX. 180 RPM	REFER TO SCOUT USER'S MANUAL	REFER TO SCOUT USER'S MANUAL	REFER TO SCOUT USER'S MANUAL	REFER TO SCOUT USER'S MANUAL
INJECTION	SPACECRAFT INJECTION INTO ORBIT AND SEPARATION OF FOURTH STAGE	TBD		5×10^{-8} TO 10^{-10}	NONE	NONE	NONE	NONE	NOT APPLICABLE	NOT APPLICABLE		NEAR EARTH MODEL $N = 10^{-12} m^{-3/4}$ WHERE N = IMPACTS/m^2/SEC. OF PARTICLES OF MASS, m (grams) OR GREATER AT AN AVERAGE VELOCITY OF 30 km/sec AND A DENSITY OF — 8 gm/cm^3 - 2% 3.5 gm/cm^3 - 10% 0.44 gm/cm^3 - 88%	VARIOUS FORMS OF ELECTROMAGNETIC FIELDS MAY BE GENERATED BY THE SPACECRAFT AND THE 4TH STAGE OF THE LAUNCH VEHICLE REFER TO MIL-I-26600				SPIN RATE - APPROX. 180 RPM				NOT APPLICABLE
SPACECRAFT IN ORBIT	STABILIZATION OF SPACECRAFT. REORIENTATION OF SPACECRAFT TO EARTH POINTING ATTITUDE. SCIENCE DATA ACQUISITION. MISSION TERMINATION	APPROX. 3 WEEKS TO 3 MONTHS			NONE	NONE	NONE	NONE	NOT APPLICABLE	NOT APPLICABLE			VARIOUS FORMS OF ELECTROMAGNETIC FIELDS MAY BE GENERATED BY THE SPACECRAFT REFER TO MIL-I-26600				STABILIZE SPACECRAFT YAW AXIS CONING FROM MAX. HALF ANGLE OF 8° TO 0.1° DESPIN SPACECRAFT TO 0 RPM	NEGLIGIBLE	NONE	NONE	NOT APPLICABLE

Preliminary orbital analysis of the Small Satellite System spacecraft has indicated that the average view time per orbit from any ground station is small. Therefore, it is desirable that the spacecraft by equipped with data storage and dump capability rather than to rely on real time data only. This capability will enhance the ability to satisfy the detailed mission objectives of the Small Satellite System.

3.3.9 Mission Operations System Constraints

The Small Satellite System flight spacecraft will contain the necessary equipment to enable the Mission Operations System to perform the following functions:

a) Determine spacecraft separation, location time, and attitude as a function of actual orbit parameters.

b) Determine spacecraft time sequence of events to provide initiation commands for those events which will require updating as the mission progresses.

4.0 SYSTEM DESCRIPTION

It has been determined that the Small Satellite System identified and shown in Figure 4-1 will 1) satisfy the mission objectives stated in Section 2.0 above and 2) be compatible with the mission constraints discussed in paragraph 3.3. The equipments comprising this selected baseline system are described below.

4.1 FLIGHT SPACECRAFT

The flight spacecraft will be a small, three axis stabilized, earth pointing, orbiting satellite vehicle consisting of the following subsystems: -

a) Structure
b) Power
c) Command and Telemetry
d) Data Handling and Control
e) Attitude Determination
f) Attitude Control
g) Experiments

The basic spacecraft will be a right circular cylinder approximately 30 inches diameter by 36 inches long and will weigh between 150 and 300 pounds depending on the particular sensor and/or experiment payload. The selected spacecraft configuration is shown in Figure 4-2 and features 1) a fixed location for the spacecraft support subsystems, 2) a structural base suitable for a large body mounted solar array with provision for attachment of deployable solar paddles, 3) approximately 7 cubic feet of unobstructed volume for "universal" mounting and integration of up to 100 lbs. of sensors and/or experiments within the

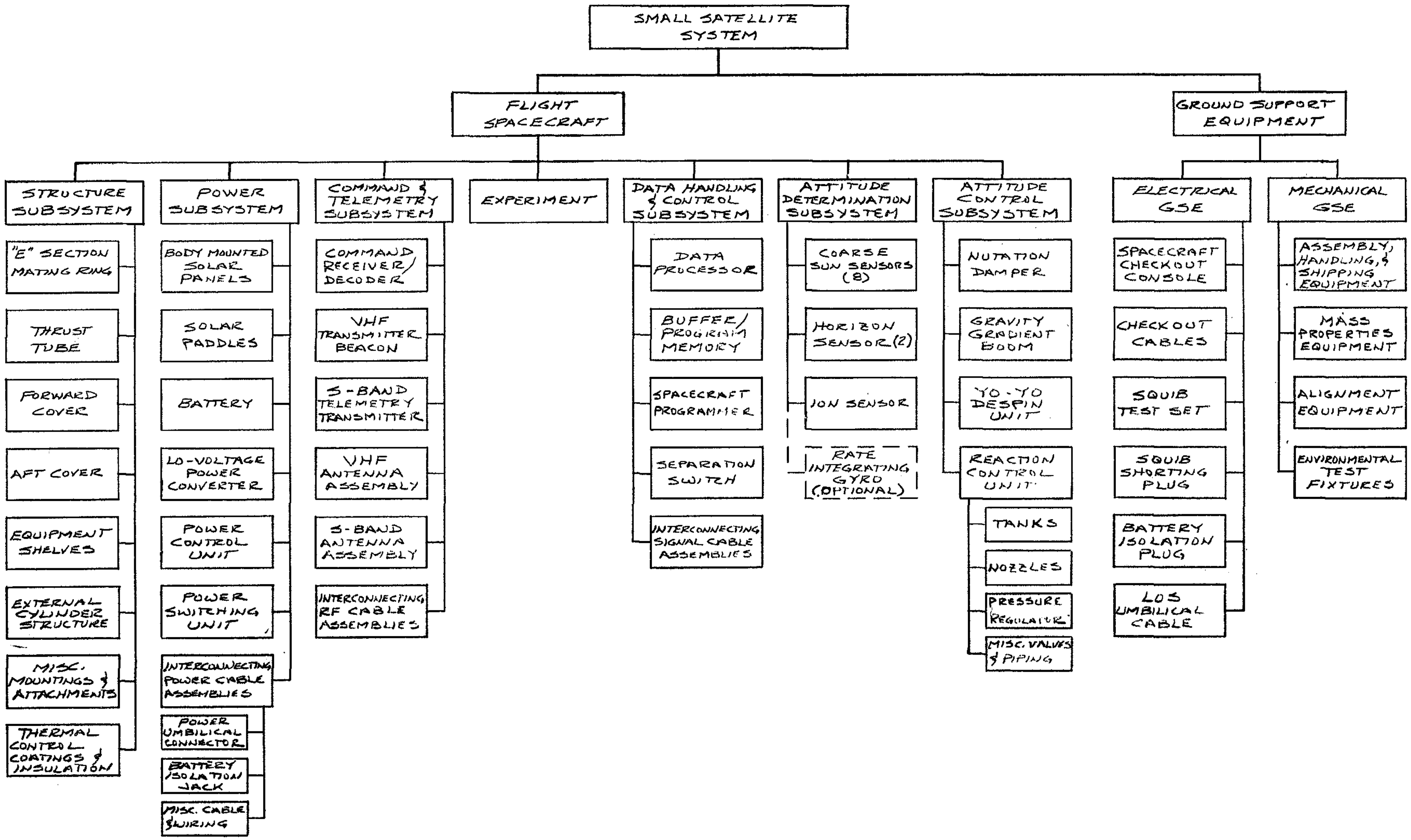


Figure 4-1 SMALL SATELLITE SYSTEM HARDWARE TREE

LEGEND

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

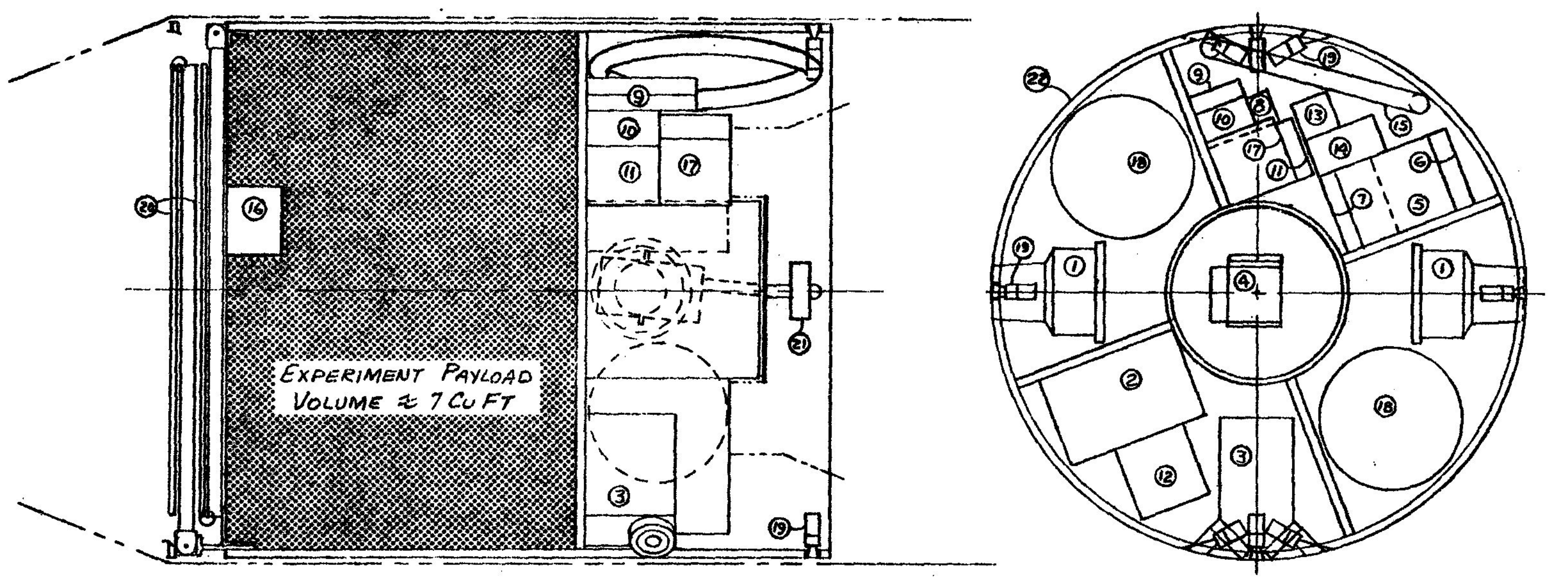


FIGURE 4-2 BASIC SATELLITE WITH FORWARD SOLAR PADDLES

structure, and 4) provision for attachment of booms and/or boom mounted sensors and/or experiments, as required.

4.1.1 Structure Subsystem

The primary function of the structure subsystem will be to 1) provide mechanical rigidity for the satellite, 2) support and hold the satellite subsystems and experiments, 3) mechanically interface with the launch vehicle adapter, and 4) provide passive thermal control.

The structure subsystem will weigh approximately 25 lbs. and will consist of, but not be limited to, the following: -

a) "E" Section Mating Ring
b) Thrust Tube
c) Forward Cover
d) Aft Cover
e) Equipment Shelves
f) External Cylinder Structure
g) Miscellaneous Mountings & Attachments
h) Thermal Control Coatings & Insulation

The external shape of the structure will be that of a right circular cylinder with provision for easy access to and removal of the experiments and other subsystem components. Maximum flexibility in the mounting and attachment of each equipment will be provided to assure attainment of the required direct and solid angles of view without obstruction from the basic structure or its appendages. This flexibility will be achieved by fabricating the external cylinder structure and the equipment shelves of aluminum honeycomb. All structural loads will be transmitted to the centrally located aluminum thrust tube and the "E" Section mating ring which attaches directly to the top flange of the Scout Launch Vehicle "E" Section payload adapter. The payload separation mechanism is contained in and furnished as part of the "E" Section.

The spacecraft thermal control will be passive. Special coatings and insulations will be used to maintain the temperature of all equipments within their acceptable operating limits under orbital conditions.

4.1.2 Power Subsystem

The primary function of the power subsystem will be to provide sufficient electrical power for continuous operation of the spacecraft under orbital conditions of spin, cell failure, occultation, and variation of solar aspect angle.

The selected power subsystem will be sized to provide 20 watts minimum to 100 watts maximum and will consist of, but not be limited to, the following:

a) Body Mounted Solar Panels
b) Deployable Solar Paddles
c) Power Control Unit (PCU)
d) Power Switching Unit (PSU)
e) Low Voltage Power Converter
f) Interconnecting Power Cable Assemblies

Preliminary descriptions and performance characteristics of the constituent assemblies of the power subsystem are presented in Preliminary Design Data Sheets, Tables 4-I through 4-VI. The preliminary interface design requirements for the total power subsystem will be as follows: -

a) Inputs

1. +28 VDC through an umbilical power connector for ground operation of the satellite.

2. A GSE battery isolation jack to disconnect the battery during ground checkout operations.

3. GSE battery charging and monitoring capability through the umbilical power connector.

4. Solar radiation energy into the solar panels during flight to generate prime power.

5. Control signals from the spacecraft programmer to effect the required power switching operations

b) Outputs

1. +28 ± 4 VDC main bus power, switched and continuous

2. Low voltage power of +6, ± 10, and +20 volts DC regulated to 1%, continuous

3. Temperature, voltage, and current "housekeeping" monitor data.

Interconnecting power cable assemblies will be provided to 1) interconnect the elements of the power subsystem, and 2) interconnect the power subsystem to all other spacecraft subsystems. In addition, a spacecraft power umbilical connector will be provided to 1) disconnect the battery and solar panels from

TABLE 4-I

DESIGN DATA SHEET

Item Nomenclature: -

Body Mounted Solar Panel

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ________

Procurement: -

☐ Make
☒ Buy   Suggested Mfgr **EOS**   Part No. ________

Status: - ☐ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ____ Levels

☒ Other **Solar Cells are qualified but not in the required configuration**

Functional Description: - **The solar array will consist of solar cells mounted on the cylinder honeycomb structure (external cylinder structure of the structure subsystem).**

The array of cells will provide the prime source of spacecraft power during sunlit operation as well as providing the power required to recharge the battery.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ____ Hz
Weight **10 lbs.**	Average Power ____ Watts at ____ Volts
Volume ________	Peak Power ____ Watts at ____ Volts
Dimensions ________	Duty Cycle ________

Temperature Requirements	Heat Dissipation
Operating Range **-105°C to +105°C**	Average **Orbit Dependent**
Non-operating Range ________	Peak **500 watts based on 10% cell efficiency**

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The solar cell array will consist of approximately 60 strings of 2 x 2 cm solar cells. Each string will have a maximum of 88 cells to provide a nominal bus voltage of +28 volts D. C. The number of cells per string is dependent upon the surface area which must be made available for instrument view ports. The parallel connection of strings will be connected to the Power Control Unit, each string isolated from all others by diodes. Temperature monitors will be attached at selected points of the solar array and their outputs fed to the Data Processor for formatting and ultimate transmission to the ground station.

TABLE 4-II

DESIGN DATA SHEET

Item Nomenclature: -

Deployable Solar Paddles

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ____

Procurement: -

☐ Make
☒ Buy Suggested Mfgr EOS Part No. ____

Status: - ☐ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ____ Levels

☒ Other Solar cells are qualified but not in the required configuration

Functional Description: - The deployable solar paddles will be used to provide additional spacecraft power when 1) orbits and attitudes present unfavorable solar aspect angles to the body mounted array and 2) the body mounted array will not generate sufficient power to satisfy specific experiment requirements. The quantity and location of the solar paddles will vary with the mission objectives and orbital parameters. The characteristics specified below are typical for a single paddle.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ____ Hz
Weight 6 lbs.	Average Power ____ Watts at ____ Volts
Volume 270 cu. in. (including substrate)	Peak Power ____ Watts at ____ Volts
Dimensions 29.4 in. diam. x 0.25 in. thick	Duty Cycle ____

Temperature Requirements	Heat Dissipation
Operating Range -105°c to +45°c	Average Orbit Dependent
Non-operating Range ____	Peak 340 watts based on 10% cell efficiency

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The solar paddle array will consist of 12 strings of 2 x 2 cm solar cells mounted on one side of a flat, circular honeycomb substrate. Each string will contain 64 cells. The parallel connection of strings will be connected to the body mounted array through isolation diodes. The output of the paddle array will be 38 watts at normal sun incidence. Temperature monitors will be attached at selected points of the paddle array and their outputs will be fed to the Data Processor for formatting and ultimate transmission to the ground station.

Paddle deployment and positioning will be accomplished with an electric motor drive.

TABLE 4-III

DESIGN DATA SHEET

Item Nomenclature: -

Battery (Ni-Cd)

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ________

Procurement: -

☐ Make
☒ Buy Suggested Mfgr General Electric Part No. ________

Status: - ☐ New Design ☒ Existing Design ☒ Flight Proven/Qualified to ________ Levels

☐ Other ________

Functional Description: - The battery will provide the prime source of spacecraft power during 1) launch and orbital insertion, 2) orbital sun occultation, and 3) orbital operations when peak power requirements exceed the solar array capacity.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ____ Hz
Weight 2.3 lbs.	Average Power ____ Watts at ____ Volts
Volume 34 cu. in.	Peak Power ____ Watts at ____ Volts
Dimensions 4 in. x 5 in. x 1.7 in.	Duty Cycle ____

Temperature Requirements	Heat Dissipation
Operating Range -20°C to +40°C	Average ____
Non-operating Range -60°C to +60°C	Peak ____

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The battery characteristics are based on providing 20 watts of continuous power at +24 volts DC during a 35 minute sun occultation period with a 50% discharge depth.

Battery inputs will consist of 1) external battery charge during ground checkout and 2) internal battery charge from the Power Control Unit during flight.

Battery outputs will be 1) battery power fed to the Power Control Unit and 2) battery temperature also fed to the Power Control Unit and Data Processor.

TABLE 4-IV

DESIGN DATA SHEET

Item Nomenclature: -

Power Control Unit (PCU)

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy Suggested Mfgr EOS Part No. ______

Status: - ☐ New Design ☒ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☐ Other ______

Functional Description: - The PCU will: 1) switch battery power to main bus whenever the solar array cannot maintain over 24 volts, 2) provide for battery charging when excess solar power is available, 3) provide solar array voltage limiting, and 4) provide battery undervoltage sensing and an appropriate output warning signal.

Physical Characteristics: -

Weight 2.2 lbs.
Volume 40 cu. in.
Dimensions 4 in. x 5 in. x 2 in.

Power Requirements: - ☒ DC ☐ AC ______ Hz

Average Power 1 Watts at 28 ± 4 Volts
Peak Power 1 Watts at 28 ± 4 Volts
Duty Cycle Selectable

Temperature Requirements

Operating Range ______
Non-operating Range ______

Heat Dissipation

Average ______
Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

PCU inputs are:
1) External GSE power during ground checkout
2) Solar array power during flight
3) Battery power during flight

PCU outputs are:
1) Main bus power of +28 ± 4 volts D.C.
2) Battery recharge power
3) Voltage & current "housekeeping" monitors to Data Processor for formatting and subsequent transmission to the ground
4) Undervoltage signal monitor to the Data Processor

TABLE 4-V

DESIGN DATA SHEET

Item Nomenclature: -

Power Switching Unit (PSU)

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☒ Make
☐ Buy Suggested Mfgr ______ Part No. ______

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☐ Other ______

Functional Description: - The PSU will provide all power ON/OFF switching functions for the spacecraft subsystems and the experiment payload. The switching functions will be initiated by control signals from the spacecraft programmer or automatically by the Power Control Unit (undervoltage or overcurrent).

Physical Characteristics: -

Weight 0.35 lbs.
Volume 7.1 cu. in.
Dimensions 4.75 in. x 1.5 in. x 1.0 in.

Power Requirements: - ☒ DC ☐ AC ______ Hz

Average Power 0.04 Watts at 28 ± 4Volts
Peak Power 0.06 Watts at 28 ± 4Volts
Duty Cycle Selectable

Temperature Requirements

Operating Range ______
Non-operating Range ______

Heat Dissipation

Average ______
Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The physical characteristics and power requirements specified above are based on approximately 15 switching functions. Detailed design will be required to establish the final requirements.

PSU Inputs

1) Main bus power from the PCU
2) Safed main bus power from the separation switch in the Data Handling & Control Subsystem
3) Control signals from the Spacecraft Programmer

PSU Outputs

1) Switched main bus power
2) Switched safed main bus power

TABLE 4-VI

DESIGN DATA SHEET

Item Nomenclature: -

Low Voltage Converter

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Spacetac Part No. ______

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☒ Other Constructed of flight proven components

Functional Description: - The converter will supply regulated low voltage power to the Data Processor, Buffer/Program Memory, and Spacecraft Programmer which require voltages other than the nominal 28 volt D.C. main bus.

Physical Characteristics: -

Weight 0.5 lbs.
Volume 4.5 cu. in.
Dimensions 3 in. x 1.5 in. x 1 in.

Power Requirements: - ☒ DC ☐ AC ______ Hz

Average Power 2.5 Watts at 28 ± 4 Volts
Peak Power 2.5 Watts at 28 ± 4 Volts
Duty Cycle Continuous

Temperature Requirements

Operating Range ______
Non-operating Range ______

Heat Dissipation

Average ______
Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The converter input will be 28 ± 4 volts D.C. main bus power. The output will consist of 1% regulated voltages required by specific users. The voltages will include +6, ±10, and +20 volts D.C. A converter temperature "housekeeping" monitor will be provided to the Data Processor. The converter will contain over-current protection to minimize damage to the using equipment.

the spacecraft circuits and connect a ground auxiliary power supply to the spacecraft for checkout, and 2) connect a conditioner and monitor to the spacecraft battery for ground changing. Since certain spacecraft functions continuously require power, a battery isolation jack will be provided to disconnect the spacecraft battery from the spacecraft circuits until final spacecraft checkout on the Scout Launch Vehicle. The plug in the jack will be removed during terminal countdown (when all pyrotechnics are armed) thereby assuring that the spacecraft battery is fully charged at lift-off and will be able to satisfactorily operate the spacecraft functions until the solar panels can begin to generate sufficient power for these functions.

4.1.3 Command and Telemetry Subsystem

The primary function of the command and telemetry subsystem will be to 1) receive and decode commands from the ground network, 2) transfer the decoded command signals to the spacecraft data handling and control subsystem, 3) receive and transmit engineering, scientific and spacecraft attitude data from the spacecraft data handling and control subsystem to the ground receiving network, and 4) provide a beacon signal of sufficient magnitude to enable ground station tracking of the spacecraft.

This subsystem will consist of, but not be limited to, the following: -

a) Command Receiver/Decoder
b) VHF Transmitter Beacon
c) S-Band Telemetry Transmitter
d) VHF Antenna Assembly
e) S-Band Antenna Assembly
f) Interconnecting RF Cable Assemblies

The preliminary design requirements for these assemblies are presented in Design Data Sheets, Tables 4-VII through 4-XI.

Interconnecting RF cable assemblies are also required and will be provided to 1) interconnect the assemblies of the command and telemetry subsystem and 2) to interconnect the command and telemetry subsystem to the data handling and control subsystem.

4.1.4 Data Handling and Control Subsystem

The primary function of the data handling and control subsystem will be to 1) receive command signal from the command decoder and provide appropriate control signals to the experiment(s) and other functional spacecraft subsystems, 2) provide safing and arming of the spacecraft pyrotechnics, and 3) acquire, handle, format, store, and program engineering, scientific, and attitude data and send these data to the S-Band telemetry transmitter.

This subsystem will consist of, but not be limited to, the following: -

TABLE 4-VII

DESIGN DATA SHEET

Item Nomenclature: -

Command Receiver/Decoder

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Avco AED Part No. RCVR: AED 301A DCDR: AED 403A

Status: - ☐ New Design ☐ Existing Design ☒ Flight Proven/Qualified to OSO, ISIS Levels
☐ Other

Functional Description: - The command receiver/decoder will receive and decode ground commands for the operation of specific support equipment(s) on board the spacecraft.

Physical Characteristics: -

Weight RCVR: 1.25 lbs DCDR: 3.3 lbs
Volume RCVR: 24.2cu. in. DCDR: 77.8cu. in.
Dimensions RCVR 3.64in. x5.64in. x1.18in
DCDR 5.0in. x3.625in. x4.29in

Power Requirements: - ☒ DC ☐ AC ____ Hz

Average Power 0.40 Watts at 28 ± 4 Volts
Peak Power 6.8 Watts at 28 ± 4 Volts
Duty Cycle Ave. Power: continuous
Peak: Selectable

Temperature Requirements

Operating Range -35°C to +80°c
Non-operating Range

Heat Dissipation

Average
Peak

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

	Inputs	Outputs	Operating Parameters
Receiver	VHF Antenna Assembly via diplexer Bus voltage 28±4 VDC	Single cable to decoder input	Standby power of 0.40 watts always ON. Power increase to 0.60 watts during command reception.
Decoder	Receiver Output Bus Voltage 28±4 VDC	All decoder outputs terminate in the Data Processor and S/C Programmer	Standby power of .005 watts always ON. Increase to 6.3 watts during Interrogation lasting ≈35 millisecs

Decoder includes protection from reception of spurious commands. Maximum of 70 simple real time commands can be realized.

TABLE 4-VIII

DESIGN DATA SHEET

Item Nomenclature: -

VHF Transmitter Beacon

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ________

Procurement: -

☐ Make

☒ Buy Suggested Mfgr Conic Part No. CTB-2000 Series

Status: - ☐ New Design ☒ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☐ Other ________

Functional Description: - The VHF transmitter beacon will provide the tracking signal for the STADAN Minitrack Interferometer.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ______ Hz
Weight 0.4 lbs.	Average Power 0.5 Watts at 28 ± 4 Volts
Volume 8.4 cu. in.	Peak Power 0.5 Watts at 28 ± 4 Volts
Dimensions 3.6 in. x 1.8 in. x 1.3 in.	Duty Cycle Continuous

Temperature Requirements	Heat Dissipation
Operating Range -30°C to +80°C	Average ________
Non-operating Range ________	Peak ________

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Inputs

1) Switched main bus power of 28 ± 4 volts D.C.

Outputs

1) RF signal fed to the diplexer of the VHF antenna assembly for radiation via the antenna to the ground tracking station.
2) Temperature "housekeeping" monitor signal fed to the Data Processor.

TABLE 4-IX

DESIGN DATA SHEET

Item Nomenclature: -

S-Band Telemetry Transmitter

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Cubic Corp. Part No. T-150 Series

Status: - ☐ New Design ☒ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☐ Other ______

Functional Description: - The S-band transmitter will provide the RF link between spacecraft and ground station for transmission in the 2200-2300 MHz frequency band of all engineering, scientific, and attitude data.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ____ Hz
Weight 1.9 lbs.	Average Power 0.66 Watts at 28 ± 4 Volts
Volume 18 cu. in.	Peak Power 0.66 Watts at 28 ± 4 Volts
Dimensions 6 in. x 3 in. x 1 in.	Duty Cycle Selectable

Temperature Requirements	Heat Dissipation
Operating Range -25°C to +85°C	Average 0.56 watts
Non-operating Range ______	Peak 0.56 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Transmitter inputs: 1) Switched main bus power of 28 ± 4 volts D.C.
2) Modulation signal from the Data Processor

Transmitter Outputs: 1) RF telemetry signal to the S-band antenna
2) Temperature "housekeeping" monitor signal fed to the Data Processor

TABLE 4-X

DESIGN DATA SHEET

Item Nomenclature: -

VHF Antenna Assembly

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other

Procurement: -

☒ Make (Avco)

☒ Buy Suggested Mfgr Diplexer - TRF Part No.

Status: - ☐ New Design ☐ Existing Design ☐ Flight Proven/Qualified to Levels

☒ Other Antenna assy not existing and/or qualified. Design is proven

Functional Description: - The antenna assembly consisting of 4 quarter wavelength whips in the 148-150 MHz frequency band in a turnstile configuration and diplexer will provide the antenna radiation coverage required for communication between spacecraft and ground station. The diplexer permits using the same antenna configuration for command reception and beacon signal transmission while providing the isolation necessary for simultaneous occurrence of the events.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ____ Hz
Weight See below	Average Power ____ Watts at ____ Volts
Volume See below	Peak Power ____ Watts at ____ Volts
Dimensions See below	Duty Cycle

Temperature Requirements	Heat Dissipation
Operating Range	Average
Non-operating Range	Peak

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Antenna assembly inputs:
1) RF output of beacon transmitter.
2) RF Command signals generated by ground station transmitter.

outputs:
1) Radiated tracking signal over the radiation sphere provided by the turnstile antenna.
2) Command signals to the command receiver/decoder

Maximum gain achieved along the yaw axis of the right circular cylinder

Physical Characteristics

Whip Antenna (1) Weight = 0.3 lbs.
Dimensions = 20 in. x 0.005 in. x 0.5 in.

Diplexer Weight = 1.0 lbs.
Volume = 9 cu. in.
Dimensions = 6 in. x 2 in. x 0.75 in.

TABLE 4-XI

DESIGN DATA SHEET

Item Nomenclature: -

S-Band Telemetry Antenna

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ________

Procurement: -

☒ Make
☐ Buy Suggested Mfgr ________ Part No. ________

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☐ Other ________

Functional Description: - The S-Band Antenna Assembly will be a circular, open-ended wave guide, flush mounted antenna and will provide the required down link coverage necessary to radiate the spacecraft data to the ground station receiving antennas.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ______ Hz
Weight <1.0 lbs.	Average Power ______ Watts at ______ Volts
Volume 10.75 cu. in.	Peak Power ______ Watts at ______ Volts
Dimensions 3.7 in. diam. x 3.0 in. long	Duty Cycle ________

Temperature Requirements	Heat Dissipation
Operating Range ________	Average ________
Non-operating Range ________	Peak ________

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The antenna assembly input is the modulated RF telemetry signal from the S-band telemetry transmitter. The output is the radiated RF telemetry signal over the radiation sphere provided by the flush mount design.

a) Data Processor
b) Buffer/Program Memory
c) Spacecraft Programmer
d) Separation Switch
e) Interconnecting Signal Cable Assemblies

The preliminary design requirements for these assemblies are presented in Design Data Sheets, Tables 4-XII to XV.

Interconnecting signal cable assemblies will be required to 1) interconnect the data processor, buffer/program memory, and spacecraft programmer and 2) interconnect all functional subsystems; i.e., power, command and telemetry, attitude determination, attitude control, and the experiment(s) to the data handling and control subsystem.

4.1.5 Attitude Determination Subsystem

The primary function of the attitude determination subsystem will be to accurately determine the spin rate, orientation, and location of the spacecraft.

The baseline subsystem will consist of, but not be limited to, the following: -

a) Coarse Sun Sensors (8)
b) Horizon Sensors (2)
c) Ion Sensor

For those sensors and/or experiments requiring pitch, roll, and yaw rate control and information, a Rate Integrating Gyro Assembly will be supplied in place of the Ion Sensor. The preliminary design requirements for the above assemblies, including the optional Rate Integrating Gyro are presented in Design Data Sheets, Tables 4-XVI through 4-XIX.

4.1.6 Attitude Control Subsystem

The primary function of the attitude control subsystem will be to 1) provide damping of the dynamic disturbances and perturbations caused by separation of the satellite from the launch vehicle, 2) despin the spacecraft after separation from the launch vehicle, 3) re-orientate and stabilize the spacecraft in an earth pointing attitude; i.e., yaw axis coincident with the local vertical, and 4) control and maintain the spacecraft in an accurate, non-spinning, 3-axis stabilized, earth pointing attitude.

To achieve the above functions, the baseline subsystem will consist of, but not be limited to, the following: -

a) Nutation Damper
b) Gravity Gradient Boom
c) Yo-Yo Despin Unit
d) Reaction Control Unit

TABLE 4-XII

DESIGN DATA SHEET

Item Nomenclature: -

Data Processor

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ____

Procurement: -

☐ Make
☒ Buy   Suggested Mfgr Spacetac, Inc.   Part No. ____

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ____ Levels

☐ Other Constructed of flight proven components

Functional Description: - This assembly will sample, format and prepare for transmission, data from the experiment, attitude determination data, engineering and diagnostic data. In addition, the unit provides control signals and timing signals for other assemblies and subsystems and operates in conjunction with the memory, programmer and command decoder.

Physical Characteristics: -

Weight 4.5 lbs.
Volume ~110 cu. in.
Dimensions 5.5 in. x 4.5 in. x 4.5 in.

Power Requirements: - ☒ DC ☐ AC ____ Hz

Average Power 1.0 Watts at ±10&+6 Volts
Peak Power 1.0 Watts at ±10&+6 Volts
Duty Cycle Continuous

Temperature Requirements

Operating Range -20°C to +50°C
Non-operating Range -40°C to +85°C

Heat Dissipation

Average 1.0 watts
Peak 1.0 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Analog inputs: 0-5 VDC
Digital inputs (parallel): Binary "0" = 0 ± 0.5 V; "1" = +5 ±2V
Digital inputs (serial): Same as above
Pulse inputs, positive, width 0.5 μs ± 0.2 μs; T_r = 0.1 μs; t_f = 0.5 μs
PCM outputs: compatible with Stadan and transmitter requirements
Binary or discrete outputs: "0" = 0 ± 0.5 volts; "1" = +5 ± 1 volt
Bit rate: selectable to 200K BPS
Modulation: split phase
Power input: regulated low voltage power at ± 10 and +6 volts D.C.

TABLE 4-XIII

DESIGN DATA SHEET

Item Nomenclature: -

Buffer/Program Memory

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Honeywell or Spacetac Part No. ______

Status: - ☐ New Design ☐ Existing Design ☒ Flight Proven/Qualified to Scout Levels

☒ Other Modification of existing design

Functional Description: - This assembly will be a 1024 x 16 magnetic core memory and will be used to store program instructions, commands, and data.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ______ Hz
Weight 3.0 lbs.	Average Power 0.5 Watts at +20&+6 Volts
Volume ~75 cu. in.	Peak Power 3.0 Watts at +20 Volts
Dimensions 5.5 in. x 4.5 in. x 3 in.	Duty Cycle 5%

Temperature Requirements	Heat Dissipation
Operating Range -20°C to +50°C	Average 0.5 watts
Non-operating Range -55°C to +85°C	Peak 3.0 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Address: 10 bits parallel; random
Data: 16 bits parallel; IN/OUT
Cycle Time: 3.0 μs max
Access Time: 1.2 μs max
Control inputs: such pulse and level commands required to operate the memory
Power Input: regulated low voltage power at +20 and +6 volts D.C.

TABLE 4-XIV

DESIGN DATA SHEET

Item Nomenclature: -

Spacecraft Programmer

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy   Suggested Mfgr Spacetac, Inc.   Part No. ______

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels
☒ Other Constructed of flight proven components

Functional Description: - The programmer will sequence and control all spacecraft functions, such as, turning equipments ON and/or OFF, correcting spacecraft attitude, extending booms, deploying solar paddles, etc. The programmer will function in any or all of the following modes:

1) Automatic sequencing to a pre-set program
2) Automatic sequencing as modified by command in flight
3) Command sequencing only

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ______ Hz
Weight 1.6 lbs.	Average Power 0.5 Watts at +6 Volts
Volume 4.5 cu. in.	Peak Power ______ Watts at ______ Volts
Dimensions 5.5 in. x 4.5 in. x 1.75 in.	Duty Cycle Continuous

Temperature Requirements	Heat Dissipation
Operating Range -20°C to +50°C	Average 0.5 watts
Non-operating Range -55°C to +85°C	Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The programmer operates in conjunction with the command decoder and data processor.

Inputs

1) Real time commands from command decoder
2) Delayed commands from the memory via the data processor
3) Timing signals from the data processor
4) Regulated low voltage power of +6 volts D.C.
5) Attitude data signal from the attitude determination subsystem

Outputs

1) Control signals to the PSU to switch power and turn equipments ON and OFF, fire pyrotechnics, deploy booms and solar paddles
2) Control signals to RCU electronic trigger circuits for attitude control
3) Control/calibration signals to operate the experiment(s)
4) Control signals to the data processor for data handling, storage, and playout.

TABLE 4-XV

DESIGN DATA SHEET

Item Nomenclature: -

Separation Switch

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Texas Instruments Part No. AT4-3

Status: - ☐ New Design ☐ Existing Design ☒ Flight Proven/Qualified to Titan Levels
☐ Other

Functional Description: - The switch will indicate separation of the spacecraft from the launch vehicle and will arm the pyrotechnic circuits of the Attitude Control Subsystem.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ____ Hz
Weight 0.03 lbs.	Average Power ____ Watts at ____ Volts
Volume	Peak Power ____ Watts at ____ Volts
Dimensions 0.375 in. dia. x 1.13 in. long	Duty Cycle

Temperature Requirements	Heat Dissipation
Operating Range -65°F to +275°F	Average 0.1 watts
Non-operating Range	Peak 0.1 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The switch is normally open while satellite and launch vehicle are mated.

The current capacity of the switch will be 5 amperes resistive at 28 volts D.C. continuous.

Contact resistance will be 0.05 ohms at 2 amperes

TABLE 4-XVI

DESIGN DATA SHEET

Item Nomenclature: -

Coarse Sun Sensor

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Ball Bros. Part No. ______

Status: - ☐ New Design ☒ Existing Design ☐ Flight Proven/Qualified to Tiros Levels

☐ Other ______

Functional Description: - Each of the eight (8) sensors will be a silicon photovoltaic cell. The sensors will be used in pairs electrically connected in parallel opposition to produce a current output that will be approximately a sinusoidal function of the sun line angle. Thus, providing a measure of the spacecraft spin rate and the position of the yaw axis.

Physical Characteristics: -

Weight 0.02 lbs.
Volume 0.[illegible]2 cu. in.
Dimensions 0.87 in. dia x 0.53 in. long

Power Requirements: - ☐ DC ☐ AC ______ Hz

Average Power ______ Watts at ______ Volts
Peak Power ______ Watts at ______ Volts
Duty Cycle ______

Temperature Requirements

Operating Range -40°F to +185°F
Non-operating Range -40°F to +200°F

Heat Dissipation

Average ______
Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Each two (2) elements will be mechanically mounted 90° apart and 45° each from the vehicle axis, and electrically connected in parallel opposition.

The output of each pair of sensors will be fed into a signal amplifier. The amplifier input impedance will be in the order of a few hundred ohms and will be uniquely adjusted for optimum temperature compensation.

The scale factor will be 0.1 volts per degree. Amplifier power consumption will be 0.050 watts.

The amplifier output signals will be fed to 1) the Data Processor for transmission to the ground as data, and 2) the spacecraft programmer for use as attitude control signals.

TABLE 4-XVII

DESIGN DATA SHEET

Item Nomenclature: -

Horizon Sensor

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other

Procurement: -

☐ Make
☒ Buy Suggested Mfgr Barnes Engineering Part No. 13-151

Status: - ☐ New Design ☒ Existing Design ☒ Flight Proven/Qualified to Gemini Levels

☐ Other

Functional Description: - The assembly will consist of two (2) IR sensors utilizing conical scans. The sensor assembly will be used to determine spacecraft pitch and roll. The sensor will operate in the 15-16 micron spectral range, thus essentially eliminating the effects of sun light and clouds.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ____ Hz
Weight 22 lbs.	Average Power 22 Watts at +28 Volts
Volume 222 cu. in.	Peak Power 22 Watts at +28 Volts
Dimensions Head 4.8in. dia. x4.82 in. long Electronics 3 in. x 8in. x 9 in.	Duty Cycle

Temperature Requirements	Heat Dissipation
Operating Range 0°F to +140°F	Average 22 watts
Non-operating Range -40°F to +180°F	Peak 22 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

The sensors will operate over an altitude range of 80-6000 nautical miles and will be installed to scan about the spacecraft roll and pitch axes. The sensor optical axes will be tilted with respect to the above spacecraft axes by an angle (not critical) that is a function of altitude.

The sensors will accomplish 12.5 scans per second, thereby providing a solution every 80 milliseconds. Filtering can be used to achieve greater accuracy.

The pitch and roll output signals will be fed to 1) the Data Processor for transmission to the ground as data and 2) the Spacecraft Program for use as attitude control signals. The scale factor will be 1 volt per degree.

TABLE 4-XVIII

DESIGN DATA SHEET

Item Nomenclature: -

Ion Sensor

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy   Suggested Mfgr Avco Tulsa   Part No. ______

Status: - ☐ New Design ☒ Existing Design ☒ Flight Proven/Qualified to Gemini Levels
☐ Other ______

Functional Description: - The ion attitude sensor produces an output voltage indicating the angle between the spacecraft yaw axis and the actual direction of flight. The IAS measures this angle by detecting ion in the space through which the vehicle is traveling. Comparison of the number of ions entering two apertures are used in computing this angle.

Physical Characteristics: -

Weight 5 lbs.
Volume 2000 cu. in.
Dimensions 5 in. x 5 in. x 8 in.

Power Requirements: - ☒ DC ☐ AC ______ Hz

Average Power 3 Watts at +28 Volts
Peak Power 3 Watts at +28 Volts
Duty Cycle ______

Temperature Requirements

Operating Range -20°F to +160°F
Non-operating Range -60°F to +180°F

Heat Dissipation

Average 3 watts
Peak 3 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Input axis of unit is along roll axis of spacecraft and must have unobstructed view forward.

Output is a linear function between ± 25 degrees. Scale factor is 0.25 volts per degree.

TABLE 4-XIX

DESIGN DATA SHEET

Item Nomenclature: -

Rate Integrating Gyro (Optional)

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ____

Procurement: -

☐ Make
☒ Buy   Suggested Mfgr U. S. Time   Part No. 16140

Status: - ☐ New Design ☒ Existing Design ☒ Flight Proven/Qualified to Minuteman Levels
☐ Other ____

Functional Description: - The assembly will consist of a triad of rate integrating gyros mounted in a common housing with mutually orthogonal axes, the gyros will be fundamentally positioned units fitted with torque motors permitting rates to be commanded. Auxiliary electronics will permit operating the gyros in a mode that permits very accurate rates to be measured.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ____ Hz
Weight 4.5 lbs.	Average Power 15 Watts at +28 Volts
Volume 70 cu. in.	Peak Power 20 Watts at +28 Volts
Dimensions 4.9 in. x 4.1 in. x 3.5 in.	Duty Cycle Variable

Temperature Requirements	Heat Dissipation
Operating Range -20°F to +140°F	Average 15 watts
Non-operating Range -40°F to +180°F	Peak 20 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

As position sensors range will be ± 3 degrees with a resolution 10^{-3} degrees
Time constant 10^{-3} seconds
As rate sensors threshold will be 10^{-7} rad sec^{-1}
As rate sensors, null uncertainty = 10^{-5} rad sec^{-1}
Without calibration 5×10^{-6} rad sec^{-1}

Input axes of gyro are coincident with spacecraft roll, pitch and yaw axes.

The preliminary design characteristics for these assemblies are presented in Design Data Sheets, Tables 4-XX through 4-XXIII.

4.1.7 Experiments

All experiments to be evaluated on a Small Satellite System flight spacecraft will be furnished by the Government. Each spacecraft will be capable of carrying one or more candidate sensors and/or experiments within the volume and weight limitations specified herein above. It is anticipated that the candidate sensors and/or experiments to be evaluated will examine the physical phenomena associated with the following applications areas.

a) Agriculture/Forestry
b) Geology/Hydrology
c) Oceanography/Marine Technology
d) Geography
e) Atmospheric Science and Technology

To accommodate such a wide variety of experiments and yet provide a low cost flight spacecraft, it will be necessary to provide a "standardized" experiment/spacecraft interface. For the spacecraft design described above, it is not possible, at this time, to define a detailed experiment/spacecraft interface, however, this interface can be characterized as follows: -

a) Mechanical

By providing a large, unobstructed volume surrounded by honeycomb structure, a vast spectrum of experiment shapes, sizes, and mounting arrangements can be accommodated. Inserts can be imbedded in the honeycomb structure, as required, additional equipment shelves can be added for experiments comprised of several "black boxes", and view ports can be cut into the external cylinder structure. Thus, the mechanical interface is very flexible.

b) Thermal

Since the baseline spacecraft design will utilize passive thermal control, it will be relatively easy, after analyzing the experiment thermal requirements, to specify and provide the necessary conductive paths, thermal coatings, and insulations necessary to maintain the required thermal environment for the particular experiment.

c) Electrical

As currently configured, the baseline spacecraft will provide the following electrical connections to the experiment interface: -

TABLE 4-XX

DESIGN DATA SHEET

Item Nomenclature: -

Nutation Damper

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☒ Make
☐ Buy Suggested Mfgr ______ Part No. ______

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☐ Other ______

Functional Description: -

Ring filled with viscous fluid to damp out satellite coning motion.

Physical Characteristics: -	Power Requirements: - ☐ DC ☐ AC ______ Hz
Weight 1.9 lb.	Average Power ______ Watts at ______ Volts
Volume 32 cu. in.	Peak Power ______ Watts at ______ Volts
Dimension 13 in. dia. x 1.0 in. dia.	Duty Cycle ______

Temperature Requirements	Heat Dissipation
Operating Range -20°C to +85°C	Average ______
Non-operating Range -40°C to +100°C	Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Damping time constant: 350 sec at 18 rad/sec.

TABLE 4-XXI

DESIGN DATA SHEET

Item Nomenclature: -

Gravity Gradient Boom

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☐ Make
☒ Buy   Suggested Mfgr Fairchild Hiller   Part No. ______

Status: - ☐ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels
☒ Other Scaling of Existing Design

Functional Description: - This assembly will consist of a motor driven furlable one piece metal boom with a tip weight. The assembly will be mounted on the end of the spacecraft yaw axis. When fully extended, this boom will provide a moment of inertia change such that the spacecraft will be stable in an earth pointing attitude within $\approx 5°$ of the local vertical.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ______ Hz
Weight 7.1 lbs.	Average Power ______ Watts at ______ Volts
Volume 84 cu. in.	Peak Power $\approx$24 Watts at +28 Volts
Dimensions 4 in. x 3 in. x 7 in.	Duty Cycle Two periods of 60 seconds each

Temperature Requirements	Heat Dissipation
Operating Range ______	Average ______
Non-operating Range -40° to +100°C	Peak 24 watts

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Extended length = 60 ft.
Boom diameter = 0.5 in. min. to 1.0 in. max.
Rigidity (EI) = 2000 lb-in^2 to 16000 lb-in^2
Tip weight mass = 5 lbs.

TABLE 4-XXII

DESIGN DATA SHEET

Item Nomenclature: -

Yo-Yo Despin Unit

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ____

Procurement: -

☒ Make
☐ Buy Suggested Mfgr ____ Part No. ____

Status: - ☐ New Design ☒ Existing Design ☐ Flight Proven/Qualified to ____ Levels

☐ Other ____

Functional Description: - This assembly will consist of two (2) small tip weights and two tethers attached to opposite sides of the spacecraft. The tethers will be wound around the exterior of the spacecraft and each tip weight will be restrained by a squib actuated quick release mechanism. When the squibs are fired, the centrifugal force of the spinning spacecraft will start unwinding the tethers. This action will slow down the rotation of the spacecraft. When the spacecraft rpm reaches zero, the tethers and tip weights will be fully unwound, extended, and completely detached from the spacecraft.

Physical Characteristics: -	Power Requirements: - ☒ DC ☐ AC ____ Hz
Weight 1 to 2 lbs.	Average Power ____ Watts at ____ Volts
Volume TBD	Peak Power TBD Watts at +28 Volts
Dimensions TBD	Duty Cycle ____

Temperature Requirements	Heat Dissipation
Operating Range TBD	Average ____
Non-operating Range TBD	Peak ____

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

To be determined.

TABLE 4-XXIII

DESIGN DATA SHEET

Item Nomenclature: -

Reaction Control Unit

Indenture Level: -

☐ Subsystem ☒ Assembly ☐ Subassembly ☐ Component ☐ Other ______

Procurement: -

☒ Make
☐ Buy   Suggested Mfgr ______   Part No. ______

Status: - ☒ New Design ☐ Existing Design ☐ Flight Proven/Qualified to ______ Levels

☒ Other Constructed of flight proven components

Functional Description: - The assembly will be a standard design cold gas (nitrogen or Freon) reaction control unit consisting of two (2) gas storage tanks, 12 gas nozzles, and the necessary regulators, valves, and piping. The nozzles will be mounted on the spacecraft in such a manner as to provide pitch, roll, and yaw control, as required. This assembly will be used for fine attitude adjustment of the spacecraft and will provide earth pointing to $<1°$ of the local vertical and stabilization of all 3 axes of 0.01°/sec.

Physical Characteristics: -

Weight 19.4 lbs.
Volume Tanks 720 cu. in.
Dimensions Tanks 8 in. dia.

Power Requirements: - ☒ DC ☐ AC ______ Hz

Average Power ______ Watts at ______ Volts
Peak Power 1.5 Watts at +28 Volts
Duty Cycle four 10 second intervals/hour

Temperature Requirements

Operating Range TBD
Non-operating Range TBD

Heat Dissipation

Average ______
Peak ______

Performance Characteristics: - (Specify Inputs, Outputs, Operating Parameters, etc.)

Total impulse = 320 lb-sec.
Thrust level = 0.001 lbs. per nozzle.

1. Switched main bus power at +28 volts D. C. (Power regulation and conversion will be supplied by the experimenter)

2. Experiment control/calibration signals at +6 volts D. C. (The quantity, sequencing, and timing to be determined)

3. Diagnostic data circuits (quantity and type-analog and/or digital to be determined)

4. Science data circuits (quantity and type-analog and/or digital to be determined)

All data processing, formatting, and handling will be done by the spacecraft Data Handling and Control Subsystem. Sufficient capacity has been provided to handle the data requirements for the experiments known to be under development and/or consideration at this time. In the event, additional capacity is required, it can be added as additional modules within the currently planned spacecraft assemblies.

4.1.8 Functional Description

The functional interrelationships of the candidate flight spacecraft subsystems are shown in Figure 4-3. The functional command list and the exact functional sequences have not yet been developed, but the flight spacecraft will generally function as described below.

As a prime payload, the flight spacecraft will be launched in the "power on" condition. Therefore, the command assembly, low voltage power converter, data processor, buffer/program memory, and spacecraft programmer will be energized by the spacecraft battery from lift-off until the solar panels begin to generate sufficient power. At this time, the PCU will maintain the control and balance of power between the solar panels and the battery.

After orbital injection has been achieved, the spacecraft will be separated from the Scout fourth stage booster by the "E" Section Adapter separation mechanism. As soon as separation has been achieved, the separation switch in the data handling and control subsystem will close and arm the yo-yo despin unit and the squib valves of the attitude control subsystem. At this point in the flight, the spacecraft will be spinning at approximately 180 rpm, and the yaw axis will be tangential to the orbital path.

Immediately after spacecraft separation, the VHF transmitter beacon and the S-Band telemetry transmitter will be turned on to permit the STADAN ground stations to track the spacecraft and to enable spacecraft "housekeeping" monitor data to be transmitted to the ground for evaluation of spacecraft status.

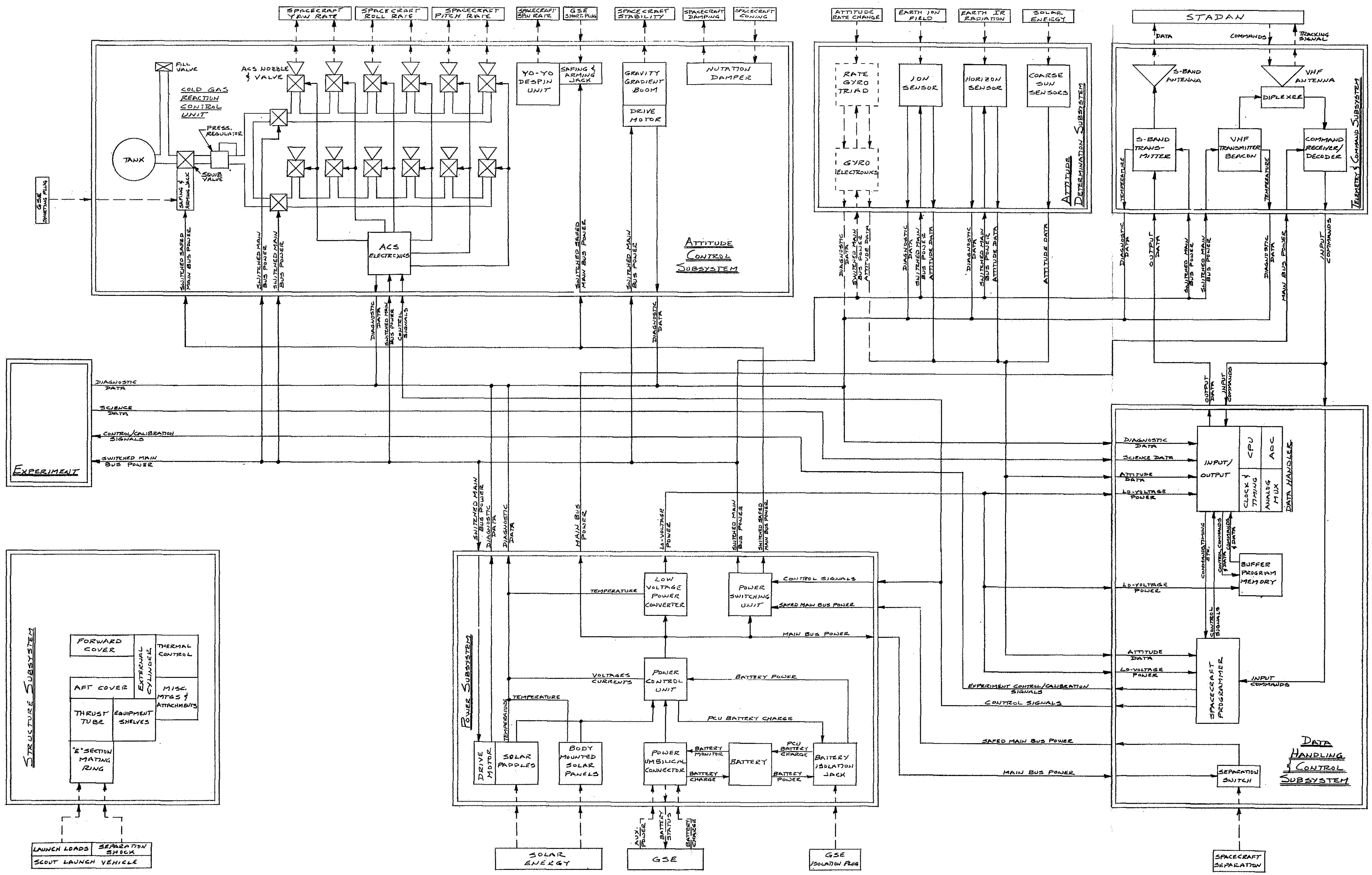


Figure 4-3 FUNCTIONAL BLOCK DIAGRAM FOR A TYPICAL FLIGHT SPACECRAFT

Approximately three quarters of an orbit after spacecraft separation, the yo-yo despin unit will be actuated, the solar paddles will be deployed, and the gravity gradient boom will be extended to the 50% position. One quarter of an orbit later, the gravity gradient boom will be fully extended. These functional events will decrease the spacecraft spin rate to zero rpm and will stabilize the spacecraft in an earth pointing attitude with the yaw axis within 5° of the local vertical. It must be noted that 1) the above reorientation and stabilization maneuver can be delayed for several orbits if required for a particular mission and 2) the required sequence of events can be initiated automatically by the spacecraft programmer, "manually" by ground command, and/or any combination thereof.

After the spacecraft has been stabilized in the earth pointing mode, the sensors of the attitude determination subsystem will be turned on, and the spacecraft attitude and/or attitude rates will be evaluated. The next event will be to enable the reaction control unit (RCU) and by energizing the appropriate thrusters to trim and correct the spacecraft attitude to less than 1° of the local vertical and to stabilize the spacecraft pitch, roll, and yaw rate to 0.01° per second, if required.

Once the required spacecraft attitude has been attained, the RCU will be turned off, and the experiment will be turned on and scientific data will be collected and transmitted to the STADAN ground stations. From this point, the spacecraft operation will be dependent on evaluation of scientific, attitude and housekeeping data on the ground. Various data processing modes will be utilized depending on the scientific data requirements and attitude control will be executed as necessary.

The flight spacecraft will continue to be operated in this manner until a major failure occurs or it is decided that sufficient scientific data has been obtained. At this time, the spacecraft will be turned off. It should be noted that the command assembly, data processor, buffer/program memory, and spacecraft programmer will still be energized, and the gravity gradient boom will maintain the spacecraft attitude to within 5° of the local vertical. Thus, it will be possible to reactuate the spacecraft at a later date to obtain additional data, if required.

4.2 GROUND SUPPORT EQUIPMENT

The ground support equipment (GSE) for the Small Satellite System will consist of the electrical and mechanical equipment required for assembly, checkout, handling, test, and transportation of the flight spacecraft during factory operations through the prelaunch mission phase. Preliminary identification of these equipments is presented below.

4.2.1 Electrical Ground Support Equipment

To be truly effective, a spacecraft functional checkout technique should 1) be capable of being used during the in-plant testing program as well as during the launch site operations, and 2) closely simulate orbital operation. During in-plant assembly of the spacecraft and prior to installation of the solar panels, checkout of the electrical assemblies and/or subsystems can be accomplished using standard laboratory test equipment; e.g., oscilloscopes, voltmeters, etc.; connected directly into test points on the electrical boxes. However, once the assembly of the spacecraft has been completed, access to these test points becomes very difficult, if not impossible. Further, this type of electrical check-out does not simulate orbital operation. Therefore, the use of an RF link method for checkout of the spacecraft has been selected.

As shown in Figure 4-1, the electrical GSE for the independently launched satellite concept will consist of, but not be limited to, the following:

a) Spacecraft Checkout Console
b) Checkout Cables
c) Battery Isolation Plug
d) Squib Shorting Plug
e) Squib Test Set
f) LOS Umbilical Cable

The above equipments will be used to calibrate, exercise, and operationally checkout the flight spacecraft during in-plant test operations as well as during pre-launch operations at the launch site. It has been assumed that the Government will supply any specialized equipment needed to exercise or test the experiments other than through the spacecraft command and telemetry. Provisions for integrating such experiment GSE will be included in the space-craft checkout console design.

The satellite checkout console will provide the following functions:

a) It will provide a source of spacecraft primary power for exercising the spacecraft functions and will charge and monitor the spacecraft battery. This capability is needed 1) to prevent the spacecraft battery from being discharged by repeated spacecraft checkouts, and 2) because the solar panels cannot generate power without a solar simulator. This is not to say that spacecraft tests using battery power and/or solar panel power cannot or will not be conducted. The ground power will be provided by hard line into an umbilical connector mounted permanently to the spacecraft. This umbilical plug on the spacecraft will be used both during in-plant system tests and during system checkouts at the launch site.

b) It will provide for complete exercise of the spacecraft through the command and telemetry subsystem.

The recommended console will have a 1) command section consisting of a command test set, RF generator, power amplifier, and transmitting antenna which will permit exercising the spacecraft functions under command control; 2) receiver section consisting of a receiving antenna, telemetry receiver, PCM decommutator and readout devices, such as strip chart recorder or tape recorders; 3) auxiliary power supply of sufficient capacity to power the spacecraft; 4) experiment GSE, if any, and 5) battery conditioner and monitor to charge and monitor the spacecraft battery. The above equipments will be rack-mounted in a manner compatible with blockhouse facilities, and will be totally self contained.

The checkout cables listed above will be used to interconnect the satellite checkout console to the spacecraft. As currently envisioned, these cables should handle power only.

A Battery Isolation Plug will be to shut off the battery power to the command assembly, low voltage power converter, data processor, buffer/program memory, and spacecraft programmer during in-plant operations as well as launch site operations, thereby assuring that the spacecraft battery is fully charged at lift-off. The plug will be a non-conducting unit which when inserted into the spacecraft jack (Reference Figure 4-3) will open the battery output circuit. This GSE plug will be equipped with a large flag and will be in place until after spacecraft checkout on the launch pad (approximately 5 hours prior to launch).

Since the baseline spacecraft concept has a squib actuated despin assembly and squib actuated valves in the RCU, it will be necessary to provide a squib shorting plug to prevent inadvertent detonation of the squib by short circuit, RF or other means. This plug is removed approximately 5 hours prior to launch. In addition, a separate portable squib test set will be necessary. The squib test set will basically be used to determine the continuity and status of the pyrotechnic devices during various phases of system test and up to 5 hours prior to launch.

As a primary payload, the baseline spacecraft concept can be "live" at launch. Therefore, a Launch Operations System (LOS) Umbilical Cable will be required. This cable will be used to interconnect the Spacecraft Checkout Console (located in the Launch Control Center) to the spacecraft umbilical connector to keep the spacecraft battery charged and monitor its status until approximately 3 minutes prior to launch at which point all umbilicals will be disconnected.

4.2.2 Mechanical Ground Support Equipment

As shown in Figure 4-1, the mechanical GSE for the independently launched satellite concept will consist of, but not be limited to, the following:

a) Assembly, Handling, and Shipping Equipment

1) Structure Build-up Fixture
2) Mobile Assembly Fixture
3) Satellite Handling Fixture
4) Satellite Hoist Sling
5) Spacecraft Shipping Container
6) Checkout Console Shipping Container

b) Mass Properties Equipment

1) Moment of Inertia Yoke
2) Spin Balance Adapter

c) Alignment Equipment

1) Precision Index Head
2) Optical Instruments

d) Environmental Test Fixtures

1) Vibration and Shock Adapter
2) Accelerator Adapter
3) Adapter and Support Stand

The Mobile Assembly Fixture will serve as the mainstay for supporting the spacecraft during assembly and functional test operations. This fixture will hold the structure at the separation plane and will have the capability of multi-position rotation for ease of assembly of the spacecraft subsystems into the structure. The Mobile Assembly Fixture will have hard rubber dolly wheels with locking capability, thus permitting the spacecraft to be transported around the plant area, as necessary.

The Satellite Handling Fixture together with the Satellite Hoist Sling will be utilized to hold the spacecraft during 1) removal from the Mobile Assembly Fixture and 2) installation and removal of the spacecraft in the various test equipment and facilities, and 3) installation and removal of the spacecraft in the shipping container. The Satellite Handling Fixture will also serve as a bumper ring for the satellite while in the shipping container.

The Spacecraft Shipping Container will be a simple rectangular box provided with suitable devices to protect the spacecraft during transportation. The container will be configured for lifting by crane or forklift and transport by ground or air. It will also contain a sealed barrier to provide protection of the spacecraft from humidity, rain, salt spray, sand, dust, insects, and fungus during transportation.

The Checkout Console Shipping Container will be used to transport the Satellite Checkout Console and Checkout Cables. It will be configured and designed in a similar manner to the Spacecraft Shipping Container and will have the same capabilities and protective features.

The Moment of Inertia (MOI) Yoke will be an L-shaped device which will attach to the spacecraft at its separation plane and will enable the spacecraft to be supported in a vertical or horizontal position. The fixture will provide precise adjustable attachments on two orthogonal surfaces to provide the spacecraft support while attached to the torsional pendulum moment of inertia measurement system. The MOI Yoke will also be used to support the spacecraft during determination of weight and location of the longitudinal center of gravity.

The Spin Balance Adapter will be a simple fixture which will clamp the orbiting satellite at its separation plane and enable the satellite to be mounted on the spin balance machine.

To achieve the mission objectives, strict alignment procedures will be required during spacecraft assembly in order to accurately determine 1) the alignment of the experiments with respect to the sensors of the Attitude Determination Subsystem and 2) the alignment of the experiment sensors and attitude determination sensors with respect to the X, Y, and Z axes of the spacecraft. A high precision index head and precise optical alignment instruments will be used for this program.

Installation of the spacecraft into the vibration, shock, acceleration, and thermal vacuum facilities does not present any serious handling problems. Standard fixtures for mounting equipment in these facilities are currently available. Therefore, the vibration and shock adapter, accelerator adapter, and adapter and support stand will be simple mechanical structural equipments designed to mate the spacecraft to these standard test fixtures.

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X]

F210-70-CKW-107

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	1910	C. K. Wilkinson	F210	13 July 1970

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control Study for Small Satellites and Related Subsystems	K159-F210-020 K159-F210-030		

SUBJECT

"Trajectory Associated Characteristics for the 4-Stage Scout Launch Vehicle

DISTRIBUTION	
List B, Files (5 copies to S. J. Brzeski)	SIGNED C K Wilkinson C. K. Wilkinson
	APPROVED Donald P. Fields D. P. Fields

INFORMATION ~~REQUESTED~~ / RELEASED

This T/R summarizes the payload capabilities, sequence of events, and injection accuracies of the Scout Launch Vehicle for launches from the Western Test Range (WTR or VAFB), Wallops Island, and San Marco. Also included are orbital lifetime characteristics. The information presented is taken from and intended to briefly summarize the following References:

1. Scout Planning Guide, LTV, October 1968
2. Scout User's Manual, LTV, 1 April 1969

1. Payload Capabilities

The Scout payload capability for circular orbits is presented in Figure 1 for due east (maximum payload) launches from San Marco and Wallops Island and for a polar orbit launched from VAFB. The variation in payload is caused by the earth's rotation and is maximized for due east launches at the equator. For a 200 lb. payload, circular orbit altitudes of 450, 600 and 650 N.M. are attainable with launches from VAFB, Wallops Island, and San Marco respectively. These increase to 800, 950 and 1000 N.M. respectively, if the payload is decreased to 100 lbs. The azimuth-inclination relationship with associated range safety constraints is given in Figure 2 for the three Launch facilities. The effects of inclination on payload for selected circular orbit altitudes are given in Figure 3, 4, and 5 for launches from VAFB, Wallops Island, and San Marco respectively.

Payload capabilities for elliptic orbits are not presented here but are available in References 1 and 2.

2. Sequence of Events

A sequence of events for a typical orbital mission is given in Table 1. The associated loads and trajectory parameters during the boost phase are given in Figure 6.

TABLE 1. TYPICAL SEQUENCE OF EVENTS - ORBITAL

PAYLOAD = 286 POUNDS
LAUNCH ELEVATION ANGLE = 85.4°
LAUNCH AZIMUTH = 90°
LAUNCH SITE = WALLOPS ISLAND

TIME SECONDS	EVENT
0.0	STAGE 1 IGNITION
0.03	START TIMER
76.1	STAGE 1 BURNOUT
80.5	STAGE 2 IGNITION, ACTIVATE "B" CONTROLS, SEPARATE FIRST STAGE, REMOVE FIRST STAGE CONTROLS
119.5	STAGE 2 BURNOUT
122.8	SEPARATE PAYLOAD HEAT SHIELD, ACTIVATE "C" BURN CONTROLS, THIRD STAGE SQUIB IGNITION
124.5	STAGE 3 IGNITION SEPARATE SECOND STAGE, REMOVE SECOND STAGE CONTROLS
159.4	STAGE 3 BURNOUT
164.4	ACTIVATE "C" COAST CONTROLS
607.9	SPIN MOTOR IGNITION, FOURTH STAGE SQUIB IGNITION
609.4	EXPLOSIVE BOLT IGNITION, SEPARATE THIRD STAGE
610.4	RETRO FORCE COMMAND
613.9	STAGE FOUR IGNITION
645.4	STAGE FOUR BURNOUT

3. Orbit Accuracy

The accuracy information presented here is based on an accuracy analysis published by LTV in 1963. It agrees well with errors attained on actual flights since 1963. Figure 7 presents the one-sigma injection errors in velocity, altitude and flight path angle for circular orbits. The injection altitude deviation increases from approximately 10 to 20 NM as the injection altitude increases from 100 to 700 N.M. The effect of injecting with velocity slightly in excess of circular velocity on apogee and perigee deviation is shown in Figure 8 for 100, 300, 500 and 700 N.M. injection altitudes. These one-sigma apogee and perigee deviations can combine to give a deviation of .01 in orbital eccentricity.

3. Orbit Accuracy (Cont)

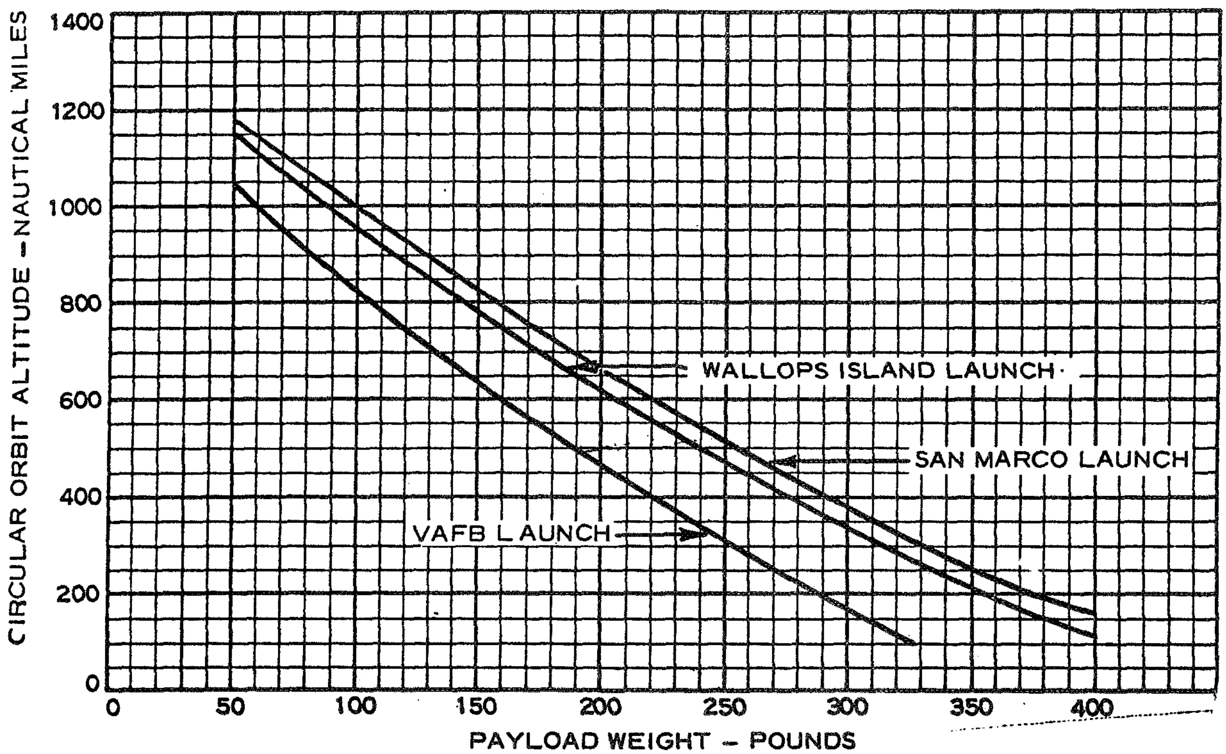
Figures 9 and 10 show the inclination accuracy for circular orbit altitudes of 300, 500 and 700 N.M. for Launches from Wallops and WTR respectively. The one-sigma deviation increases with inclination but is always less than one-third of a degree. Since the flight path angle deviation (Figure 7) is also about one-third of a degree and since the fourth stage accounts for a little more than one-third of the total velocity increment, the attitude of the vehicle during the fourth stage boost must average to at most a one degree deviation one-sigma from its nominal attitude which is approximately along the local horizontal in the trajectory plane.

4. Probable Orbit Lifetime.

The probable orbit lifetime curves presented here are based on a theory developed by King-Hele and are applicable to orbits with eccentricities of 0.2 or less. The assumptions are made that the earth and its atmosphere are spherically symmetric and rotate at the same rate; that the atmospheric density, vehicle mass, and drag coefficient do not vary with time; and finally, that the density is satisfactorily simulated in the neighborhood of perigee by the well known isothermal approximation. These are the usual assumptions made in the development of an analytical formula for lifetime, and are reasonably accurate over all but the final decay phase.

Approximate lifetimes of earth satellite orbits are shown in Figures 11 and 12 for inclinations of 45 and 90 degrees respectively, and a ballistic coefficient, (W/C_DA), of unity. The two figures differ only slightly in detail. As an example in the use of the curves, a circular orbit of 200 NM altitude would result in a lifetime of 70 days for a vehicle with a W/C_DA of 10 lbs/ft^2 (7 days for $W/C_DA = 1$).

SAN MARCO LAUNCH - DUE EAST
WALLOPS ISLAND LAUNCH - DUE EAST
VAFB LAUNCH - POLAR ORBIT

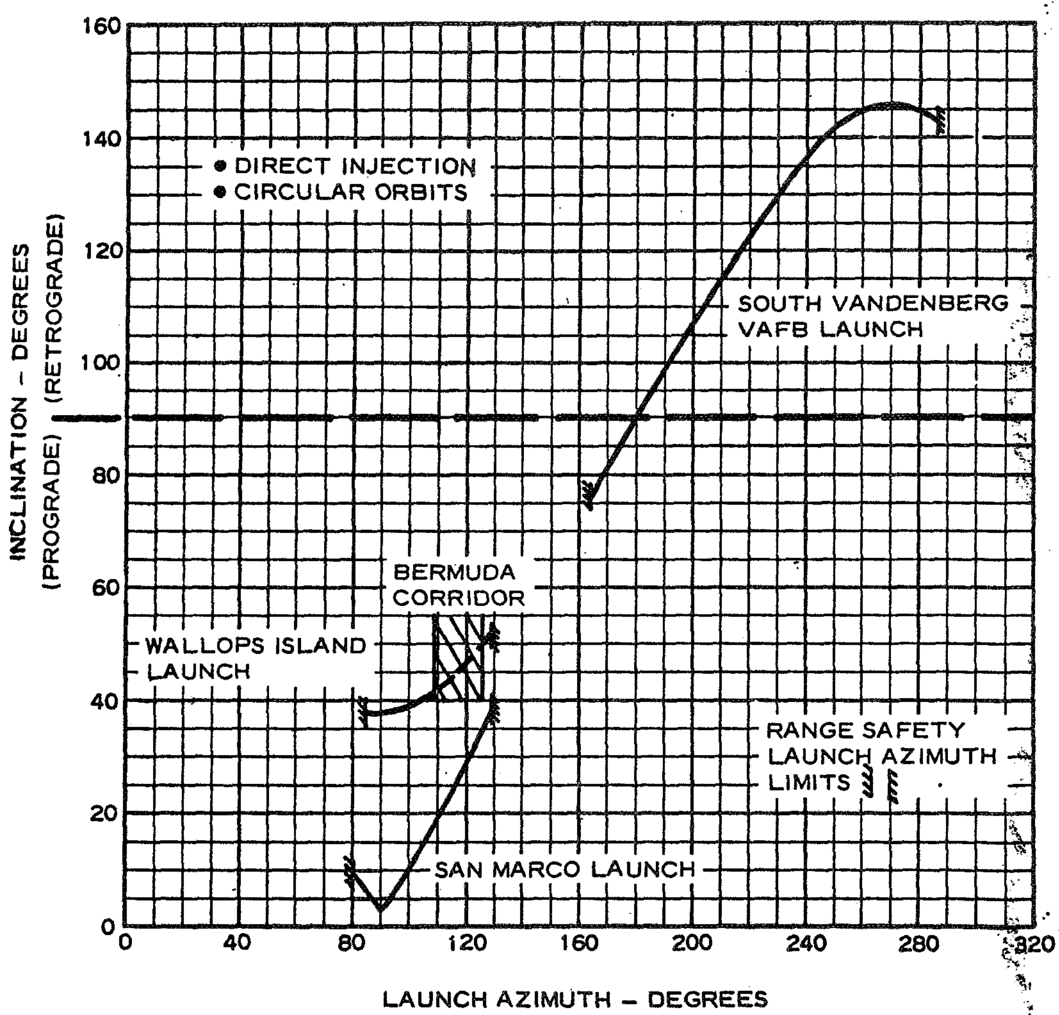


CIRCULAR ORBIT PERFORMANCE

FIGURE 1

ORBIT MISSIONS

AZIMUTH-INCLINATION RELATIONSHIP



LAUNCH AZIMUTH LIMITS

WALLOPS STATION = 85 DEGREES THROUGH 109 DEGREES;
126 THROUGH 129 DEGREES

WESTERN TEST RANGE = 164 DEGREES THROUGH 287 DEGREES

SAN MARCO = 80 DEGREES THROUGH 130 DEGREES

FIGURE 5-15. AZIMUTH-INCLINATION RELATIONSHIP (CIRCULAR ORBITS)

FIGURE 2

ORBIT MISSIONS

PAYLOAD - INCLINATION RELATIONSHIP

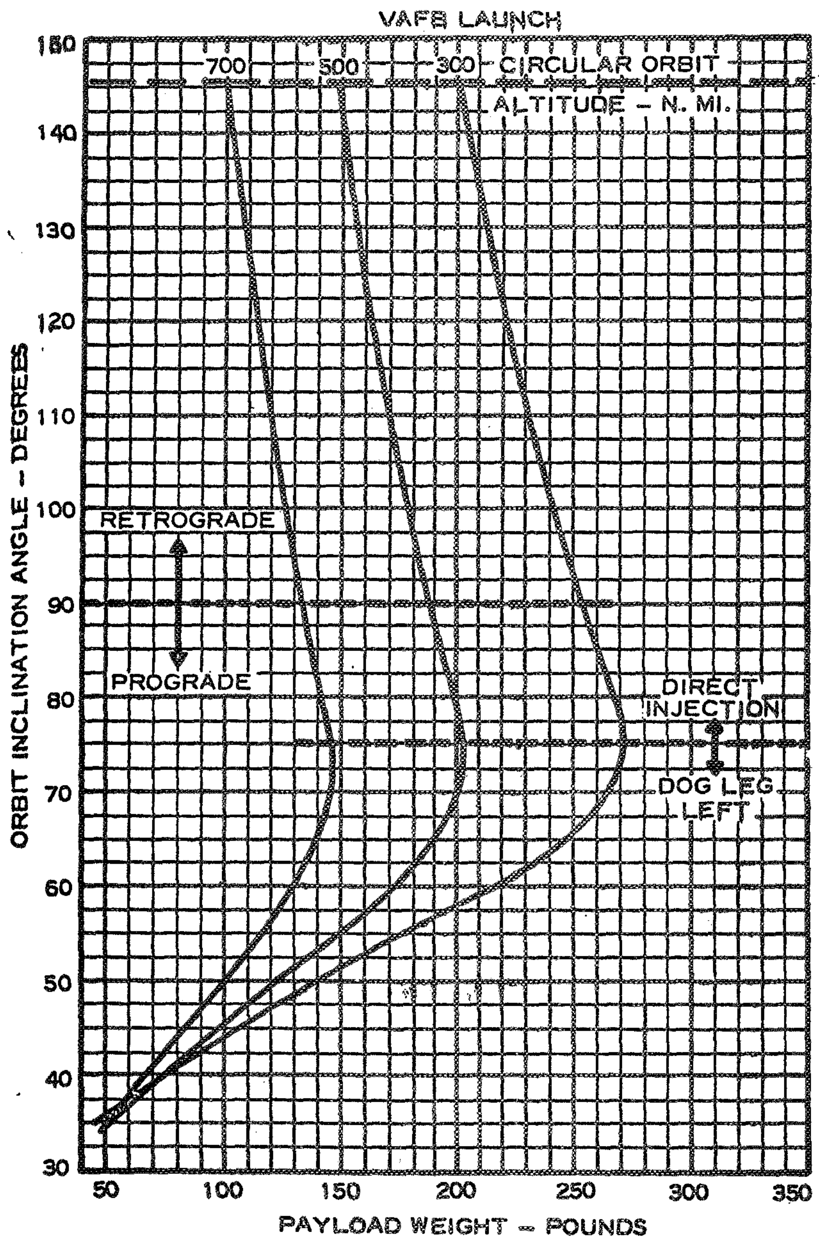
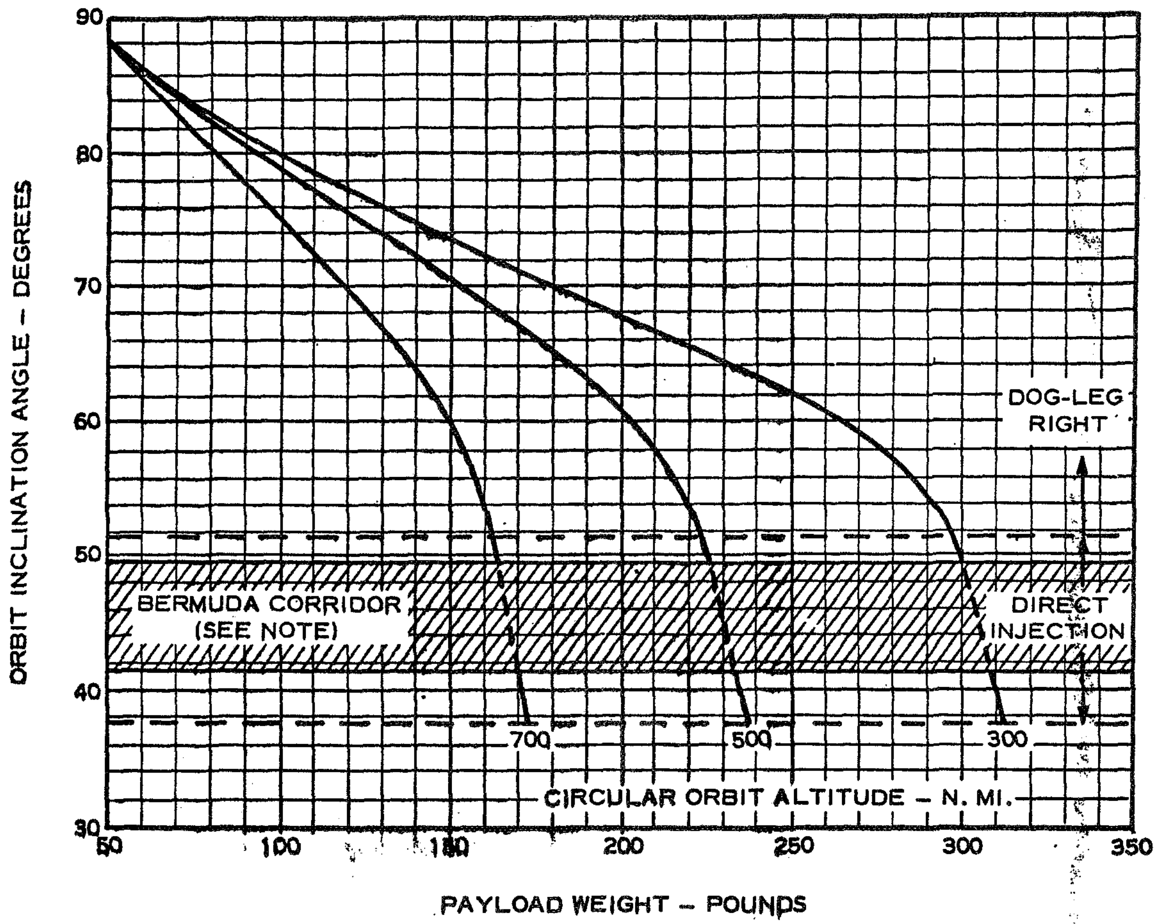


FIGURE 5-16. EFFECT OF ORBIT INCLINATION ON PAYLOAD CAPABILITY - VAFB

WALLOPS STATION LAUNCH



NOTE:
THESE INCLINATIONS NOT AVAILABLE
BY DIRECT INJECTION.

FIGURE 5-17. EFFECT OF ORBIT INCLINATION ON PAYLOAD CAPABILITY - WALLOPS STATION

ORBIT MISSIONS

PAYLOAD - INCLINATION RELATIONSHIP

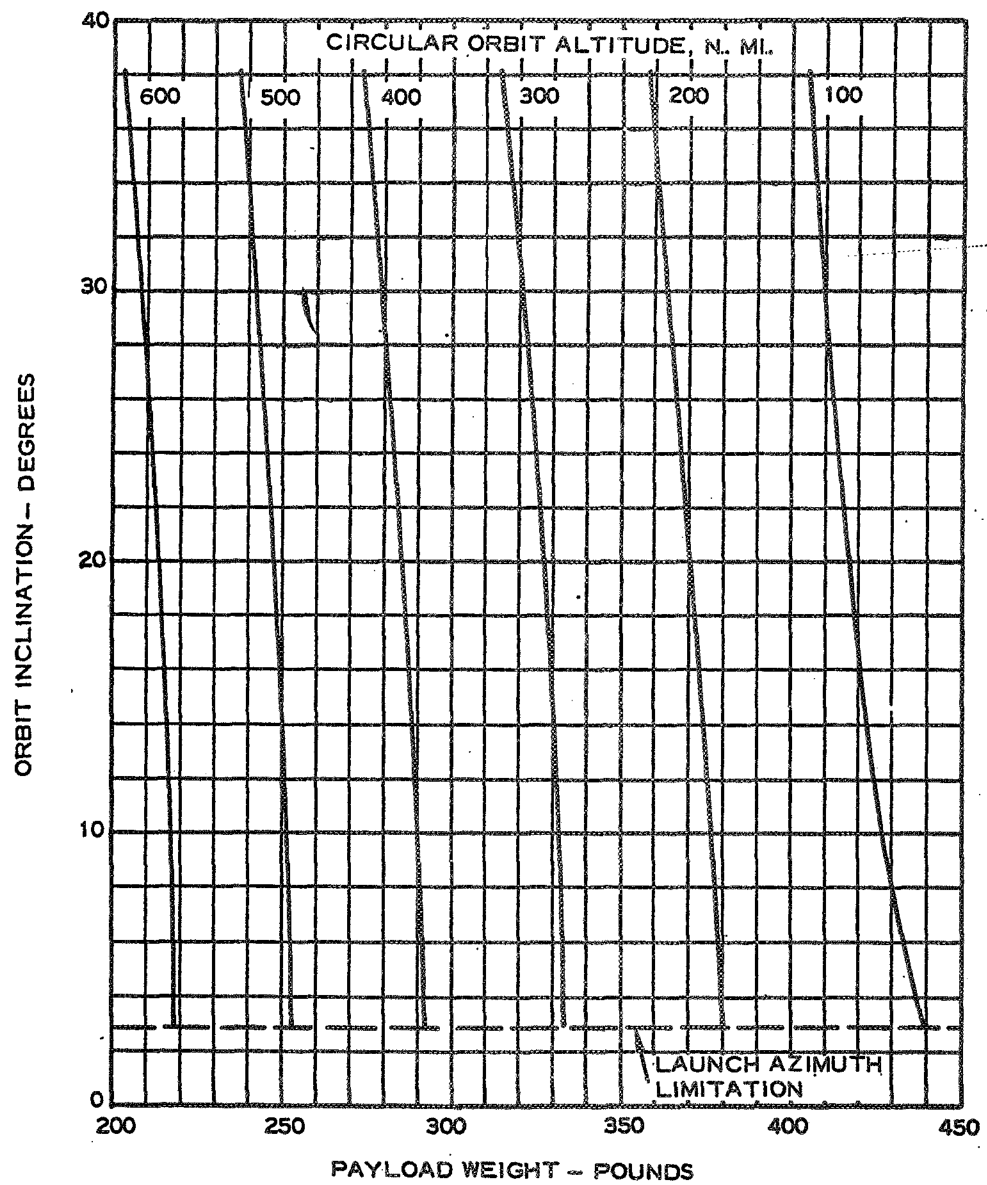
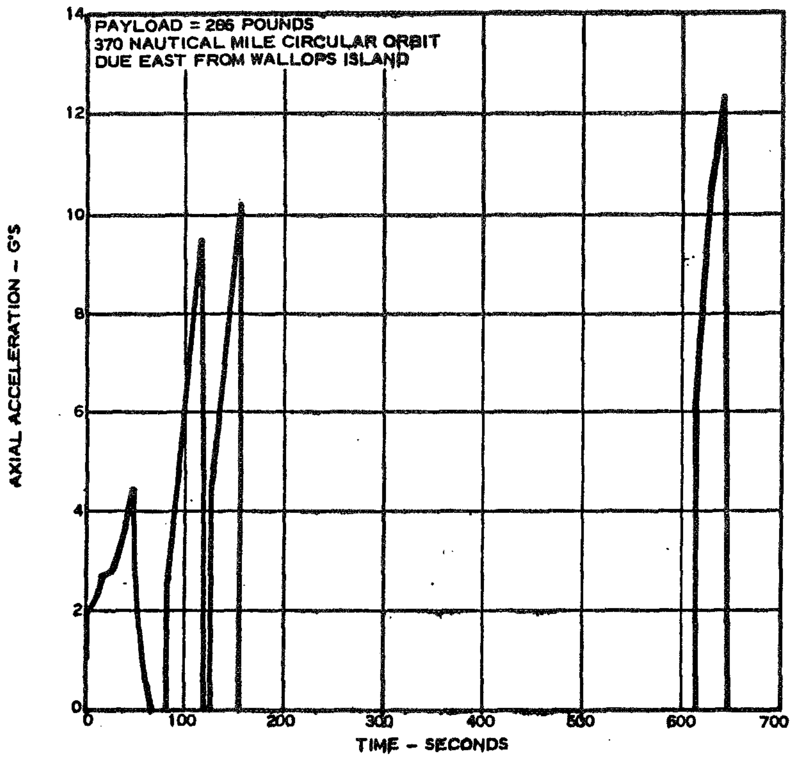
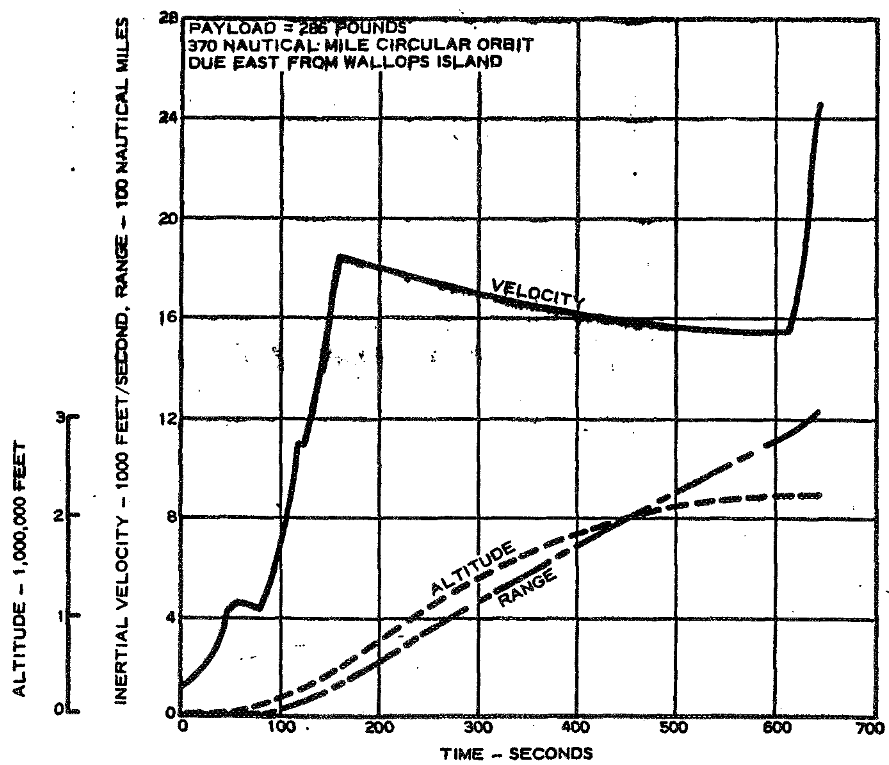


FIGURE 5-40 D. EFFECT OF ORBIT INCLINATION ON PAYLOAD CAPABILITY - SAN MARCO (SHEET 1)



TYPICAL FLIGHT LOADS



TYPICAL BOOST PERFORMANCE

TYPICAL BOOST PERFORMANCE DURING ORBITAL ASCENT

FIGURE 6

ORBIT MISSIONS

INJECTION ACCURACY

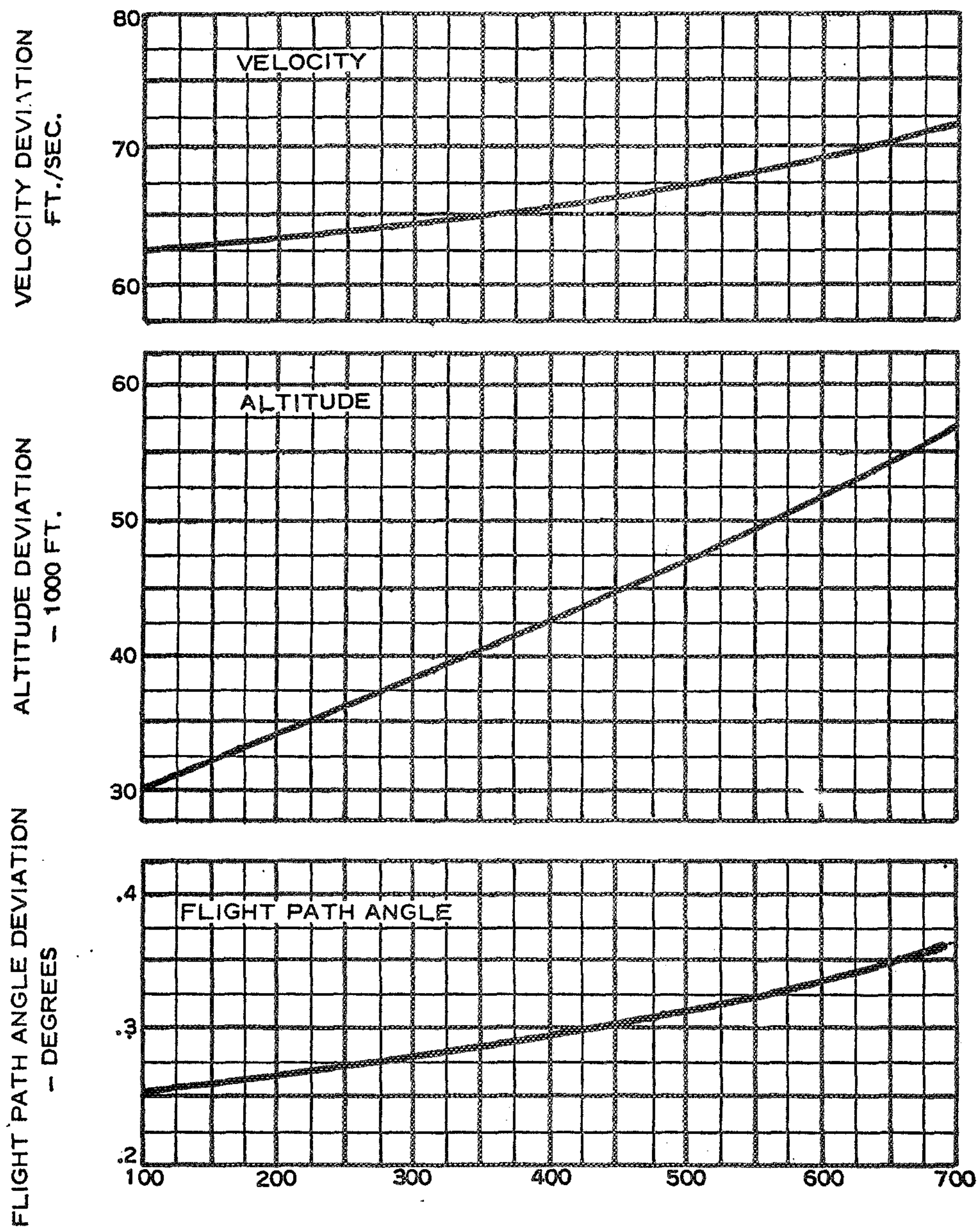


FIGURE 5-18. ONE-STANDARD DEVIATION IN INJECTION CONDITION

FIGURE 7.

ORBIT MISSIONS

ORBIT ACCURACY

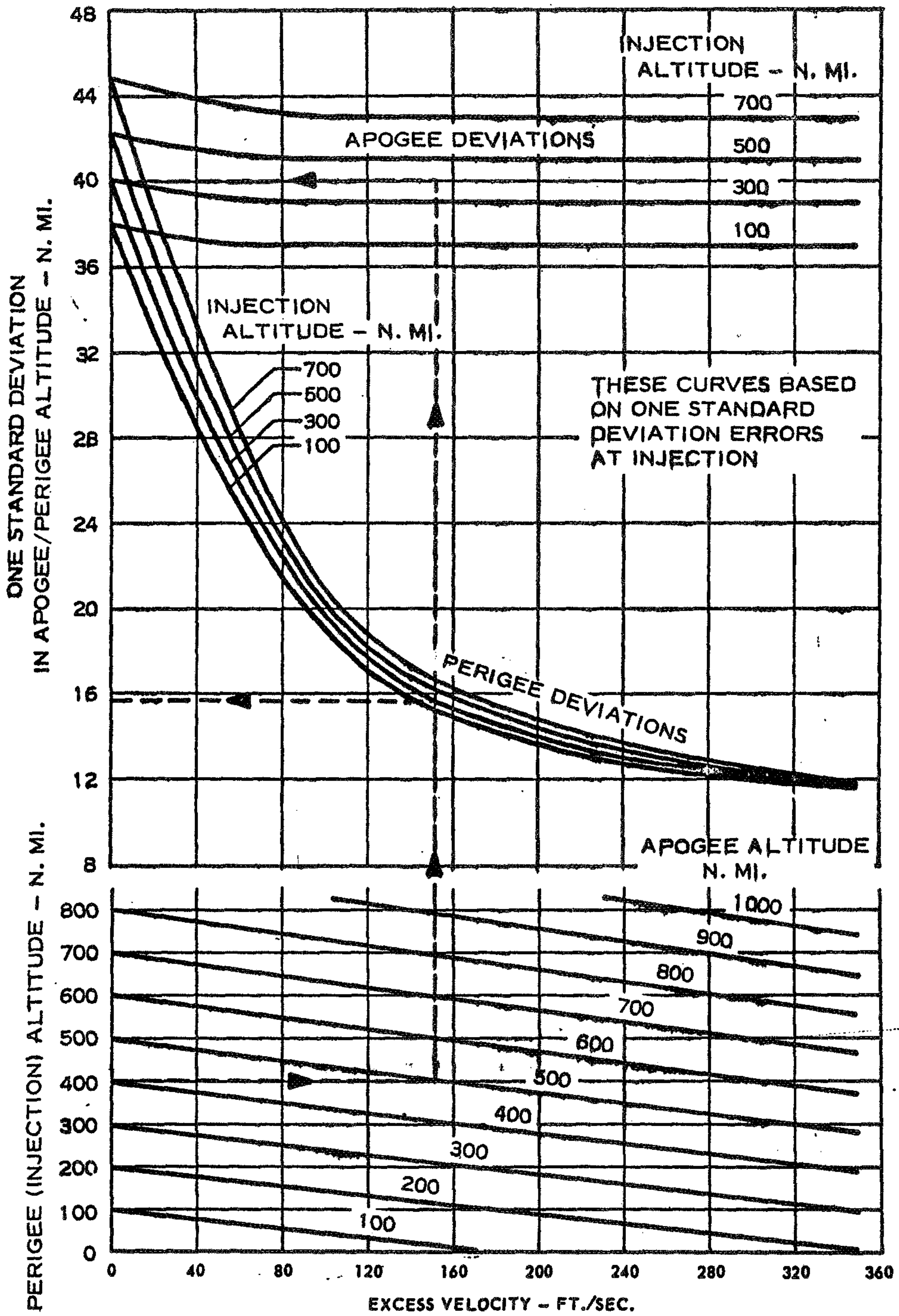


FIGURE 5-21. ORBIT ACCURACY (SHEET 1)

ORBIT MISSIONS

INCLINATION ACCURACY

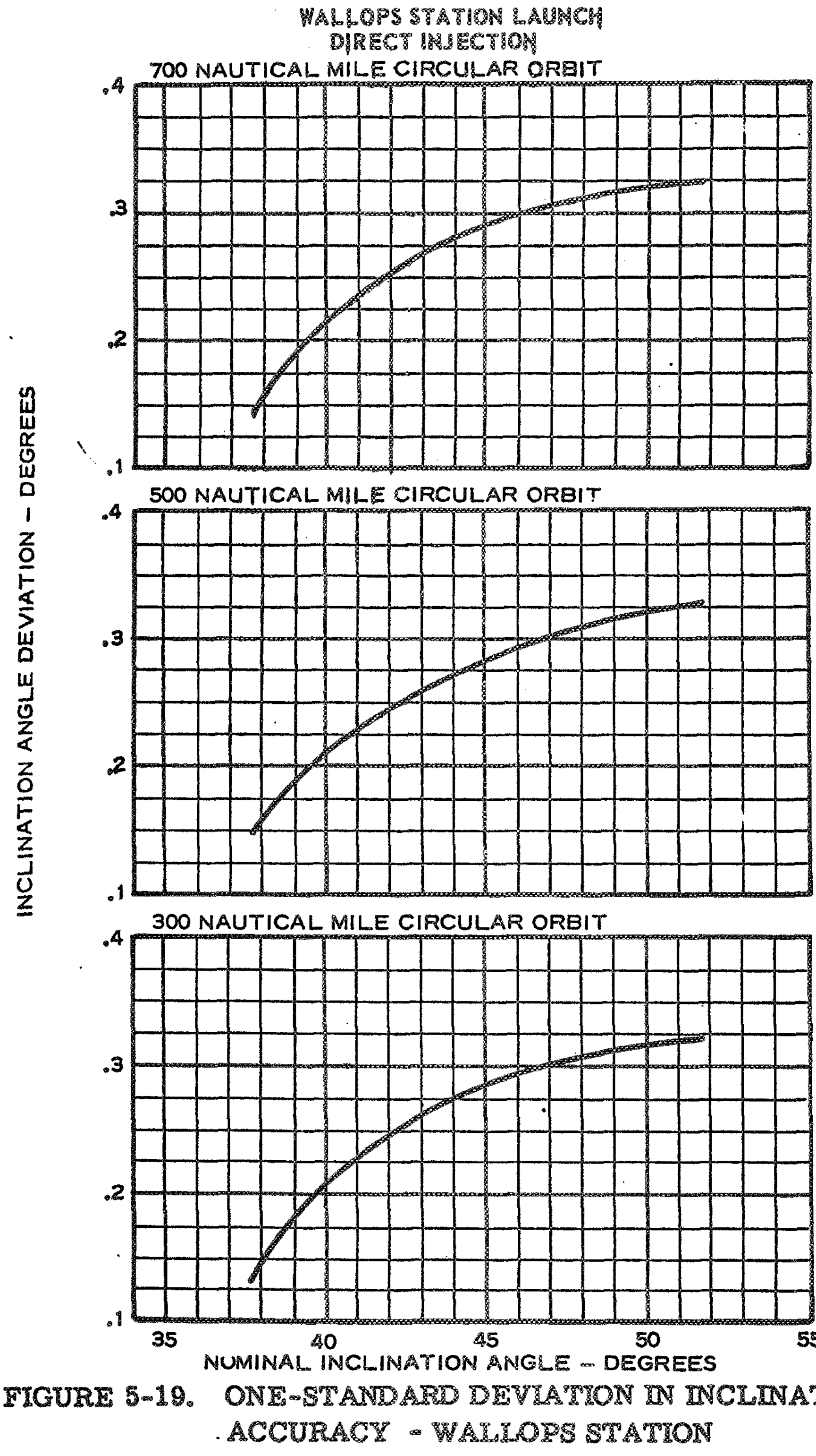


FIGURE 5-19. ONE-STANDARD DEVIATION IN INCLINATION ACCURACY - WALLOPS STATION

FIGURE 9

ORBIT MISSIONS

INCLINATION ACCURACY

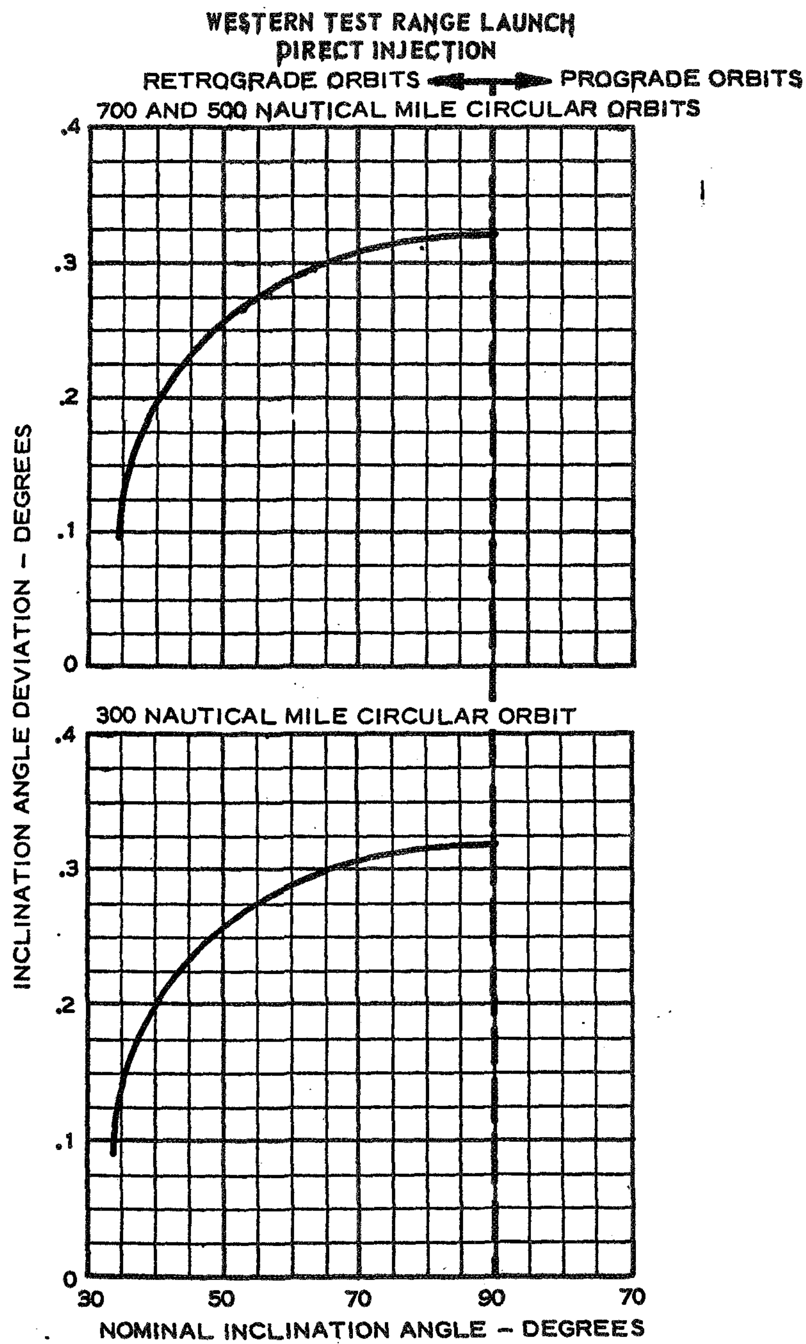


FIGURE 5-20. ONE-STANDARD DEVIATION IN INCLINATION ACCURACY - WTR

FIGURE 10

ORBIT MISSIONS

ORBIT LIFETIME

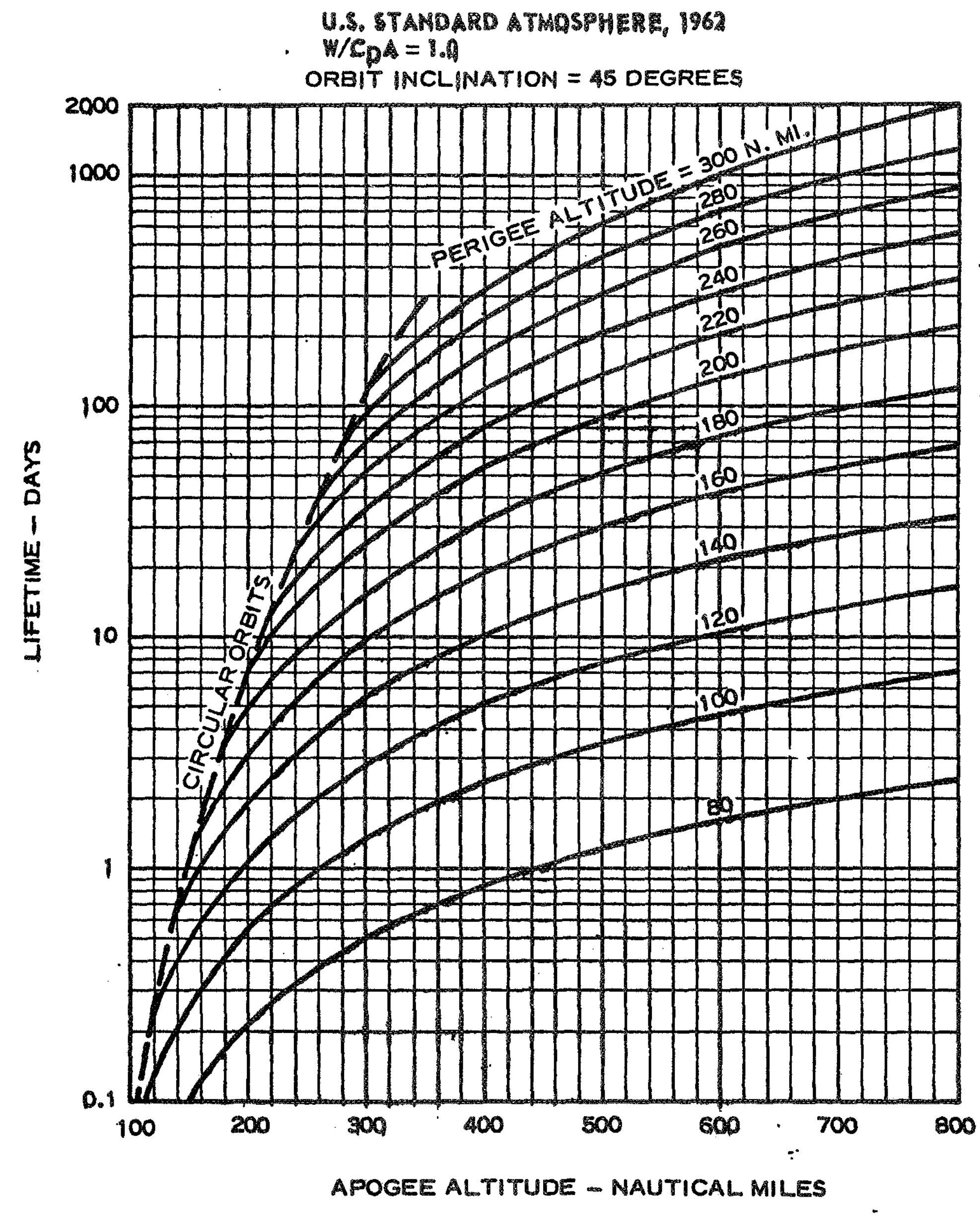


FIGURE 5-22. APPROXIMATE LIFETIME OF EARTH SATELLITE ORBITS (SHEET 1)

FIGURE 11

ORBIT MISSIONS

ORBIT LIFETIME

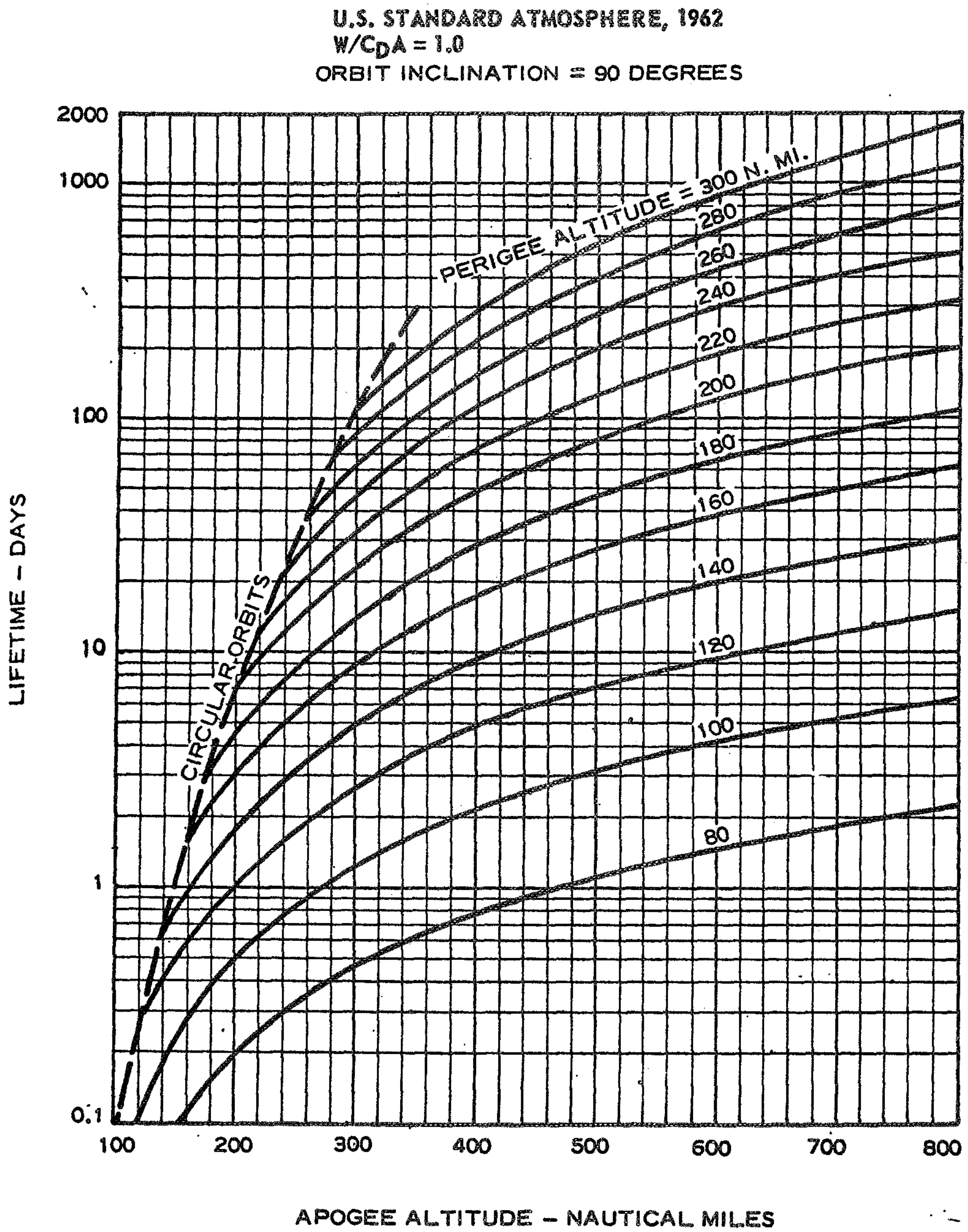


FIGURE 5-22. APPROXIMATE LIFETIME OF EARTH SATELLITE ORBITS (SHEET 2)

FIGURE 12

6-0445
C-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ]
RELEASE [X] F220-JEM-70-42

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	J. E. Mozzicato	F220	6/25/70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control Study for Small Satellites and Related Subsystems	W159-F220-030		

SUBJECT

Definition of Requirements for Attitude Control Study

DISTRIBUTION	SIGNED / APPROVED
K. Arnesen, T. N. Banks, D. P. Fields, W. C. Hailey, R. E. Herskind, W. J. Kubicki, E. J. Lawlor, R. Litte, Files	SIGNED: [illegible signature] APPROVED: F. H. Scammell

INFORMATION REQUESTED / RELEASED

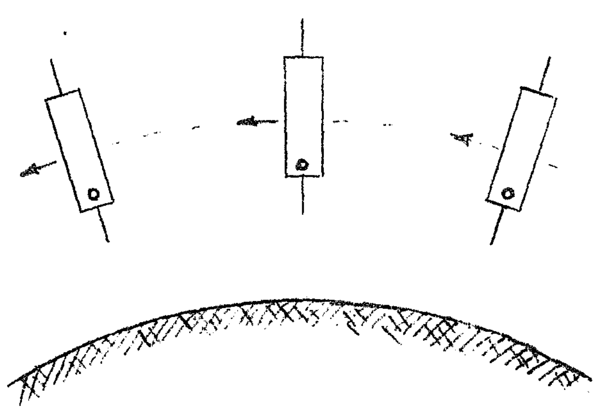
This T/R gives the current definition of the system requirements to be used for the attitude control study.

1. Coordinate Frame

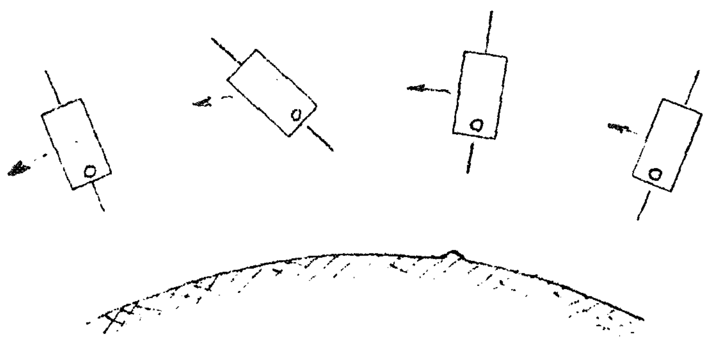
The reference coordinate frame will be a local orbital frame such that $\vec{Z}_o$ (yaw axis) is directed from the CG of the vehicle toward the center of the Earth, $\vec{x}_o$ (roll axis) is in the orbital plane and orthogonal to $\vec{Z}_o$ and is positive in the direction of orbital flight, and $\vec{y}_o$ (pitch axis) is orthogonal to both $\vec{x}_o$ and $\vec{Z}_o$ in a right handed frame such that $\vec{x}_o \times \vec{y}_o = Z_o$. (Note that this not an inertial frame).

2. Vehicle Attitude

The baseline ACS shall stabilize the satellite in the orbital frame as shown below.



Temporary deviations from this attitude may be required; for example it may be desirable to orient a vehicle axis at a fixed point on the Earth's surface in the orbital plane as shown below.



Other attitude modes such as inertial, spin, or solar stabilization are not considered to be attitude requirements for this study.

3. Attitude and Rate Accuracy

The ACS will maintain the desired attitude within $\pm$ .25° 3σ on each axis and will maintain the desired rate within $\pm$ .05°/sec 3σ on each axis.

4. Orbital Range

Altitude: 200 nm to 1,000 nm
Inclination: 0 to 50°
Eccentricity: (only unintentional eccentricities)
Lifetime: 3 weeks to 3 months

5. Satellite Weight and Size

Weight: 75 to 400 pounds
Shape: Cylindrical
Max. Diameter: 30"
Max. Length: 36"
Inertias: To be Determined

6. Initial (Acquisition) Maneuver Requirement

   a. Despin from up to 215 RPM
   b. Cancel separation rates
   c. Roll up to $\pm$ 90^{o}
   d. Pitch or yaw up to $\pm$ 90^{o}
   e. Stabilize within 7 to 30 days depending on experiment

7. Attitude Determination Accuracy

   To be established

8. Magnetic Field

   a. 10 gauss max. within vehicle when a coil is energized.
   b. 10 gamma max. at 2 meters from vehicle C. G. when coil is not energized.

9. Mass Expulsion

The expulsion of mass by the ACS may be limited by consideration of experiment cleanliness requirements.

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒ F220-JEM-70-47

TO	DEPT.	FROM	DEPT.	DATE
S. Brzeski	L910	J. E. Mozzicato	F220	7/2/70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-F220-032		

SUBJECT

Inertial Constraints on Gravity Gradient Stabilized Vehicles

DISTRIBUTION

K. Arnesen, D. Fields, W. C. Hailey, E. J. Lawlor, Jr., R. Litte, F. Scammell, G. Kunkel, G. Pfeiffer, M. Weinberger, Files

SIGNED J. E. Mozzicato

APPROVED F. H. Scammell

INFORMATION REQUESTED / RELEASED

This Technical Release presents a brief development of the expressions for Gravity Gradient Torque followed by an analysis which establishes the inertial constraints for stability of a gravity gradient stabilized vehicle and an analysis which establishes the inertial constraints for capture into the gravity gradient mode.

1. Gravity Gradient Torques

Consider a reference orbital coordinate frame at the center of gravity of an Earth satellite such that:

$\vec{x}_o$ is in the direction of the velocity vector for a circular orbit.

$\vec{y}_o$ is normal to the orbital plane and directed such that $(\vec{x}_o \times \vec{y}_o)$ is toward the center of the Earth.

$\vec{z}_o$ is directed toward the center of the Earth.

Let the body fixed axes be a right handed triad i, j, k obtained by the following sequence of Euler rotations:

(1) Yaw about the z axis by the angle ψ

(2) Pitch about the y axis by the angle θ

(3) Roll about the x axis by the angle ϕ

Figure I shows the Euler transformation.

If we represent the distance from the center of the Earth to the center of gravity of the satellite by R then the vector $\vec{R}$ from the center of the Earth to the satellite C. G. is given in body coordinates by:

$$\vec{R} = R(i \sin\theta - j\cos\theta\sin\phi - k\cos\theta\cos\phi)$$

Let the body coordinates of a mass element of the satellite (dm) be given by the vector $\vec{\rho}$ such that

$$\vec{\rho} = ix + yj + zk$$

Therefore, the vector $\vec{r}$ from the center of the Earth to the mass element (dm) is:

$$\vec{r} = \vec{R} + \vec{\rho}$$

and the gravity force on element (dm) is given by the force vector $d\vec{F}$ where

$$d\vec{F} = -\frac{\mu\, dm}{|r|^3}\vec{r}$$

μ is the Earth's gravitational constant.

Now the moment vector $d\vec{M}$ about the C.G. created by $d\vec{F}$ is given by:

$$d\vec{M} = \vec{\rho} \times d\vec{F}$$

or

$$d\vec{M} = -\frac{\mu\, dm}{|r|^3}\left(\vec{\rho} \times [\vec{R}+\vec{\rho}]\right)$$

But

$$\vec{\rho} \times [\vec{R}+\vec{\rho}] = \vec{\rho} \times \vec{R}$$

Therefore,

$$d\vec{M} = -\frac{\mu\, dm}{|r|^3}\left(\vec{\rho} \times \vec{R}\right)$$

The scalar quantity $\frac{1}{|r|^3}$ is obtained as follows:

$$|r|^2 = (\vec{R}+\vec{\rho})\cdot(\vec{R}+\vec{\rho})$$

$$|r|^2 = |R|^2 + |\rho|^2 + 2\vec{R}\cdot\vec{\rho}$$

$$|r|^2 = |R|^2\left(1 + \left(\frac{|\rho|}{|R|}\right)^2 + \frac{2\vec{R}\cdot\vec{\rho}}{|R|^2}\right)$$

The term $\left(\frac{|\rho|}{|R|}\right)^2$ is second order and may be neglected; so, we may write:

$$\frac{1}{|r|^3} = \frac{1}{|R|^3}\left[1 + \frac{2\vec{R}\cdot\vec{\rho}}{|R|^2}\right]^{-\frac{3}{2}}$$

Since $\frac{2\vec{R}\cdot\vec{\rho}}{|R|^2} << 1$ we may write:

$$\frac{1}{|r|^3} = \frac{1}{|R|^3}\left(1 - 3\frac{\vec{R}\cdot\vec{\rho}}{|R|^2}\right)$$

Therefore, the moment vector is

$$d\vec{M} = \frac{-\mu\, dm}{|R|^3}\left(1 - 3\frac{\vec{R}\cdot\vec{\rho}}{|R|^2}\right)(\vec{\rho}\times\vec{R})$$

or $$\vec{M} = -\frac{\mu}{|R|^3}\left\{\int(\vec{\rho}\times\vec{R})dm - \frac{3}{|R|^2}\int(\vec{R}\cdot\vec{\rho})(\vec{\rho}\times\vec{R})\,dm\right\}$$

The term $\int(\vec{\rho}\times\vec{R})dm$ consists of only terms which contain $\int\rho$ dm but by the definition of the center of gravity. This integral is identical to zero therefore:

$$\vec{M} = \frac{3\mu}{|R|^5}\int(\vec{R}\cdot\vec{\rho})(\vec{\rho}\times\vec{R})\,dm$$

Expanding in body coordinates yields:

$$M_x = \frac{3\mu}{|R|^3}\int(x\sin\theta - y\cos\theta\sin\phi - z\cos\theta\cos\phi)(-y\cos\theta\cos\phi + z\cos\theta\sin\phi)\,dm$$

$$M_y = \frac{3\mu}{|R|^3}\int(x\sin\theta - y\cos\theta\sin\phi - z\cos\theta\cos\phi)(z\sin\theta + x\cos\theta\cos\phi)\,dm$$

$$M_z = \frac{3\mu}{|R|^3}\int(x\sin\theta - y\cos\theta\sin\phi - z\cos\theta\cos\phi)(-x\cos\theta\sin\phi - y\sin\theta)\,dm$$

If we let the body axes be the principal axes of inertia, all terms which contain $\int$x y dm or $\int$x z dm or $\int$y z dm are defined to be equal to zero.

The body moments now become:

$$M_x = \frac{3\mu}{|R|^3}\int(y^2 - z^2)\cos^2\theta\cos\phi\sin\phi\,dm$$

$$M_y = \frac{3\mu}{|R|^3}\int (x^2 - z^2)\sin\theta\cos\theta\cos\phi\, dm$$

$$M_z = \frac{3\mu}{|R|^3}\int (y^2 - x^2)\sin\theta\cos\theta\sin\phi\, dm$$

But

$$\int (y^2 - z^2)\, dm = \int (y^2 + x^2)\, dm - \int (z^2 + x^2)\, dm = I_z - I_y$$

$$\int (x^2 - z^2)\, dm = \int (x^2 + y^2)\, dm - \int (z^2 + y^2)\, dm = I_z - I_x$$

$$\int (y^2 - x^2)\, dm = \int (y^2 + z^2)\, dm - \int (x^2 + z^2)\, dm = I_x - I_y$$

Let

$$\frac{\mu}{|R|^3} = \omega_o^2$$

Where ω_o = orbital rate (rad/sec)

Let $\cos\phi \sin\phi = \dfrac{\sin(2\phi)}{2}$

Let $\cos\theta \sin\theta = \dfrac{\sin(2\theta)}{2}$

The body moments now become

$$\boxed{\begin{aligned} M_x &= \frac{3}{2} W_o^2 (I_z - I_y)\sin(2\phi)\cos^2\theta \\ M_y &= \frac{3}{2} W_o^2 (I_z - I_x)\sin(2\theta)\cos\phi \\ M_z &= \frac{3}{2} W_o^2 (I_x - I_y)\sin(2\theta)\sin\phi \end{aligned}}$$

2. Gravity Gradient Stability

Consider a satellite which is nearly stabilized in the orbitalframe. That is, the Euler angles (ψ , θ, ϕ) are small. If we establish an inertial reference frame (x_I, y_I, z_I) such that

$$x_o = x_I \cos W_o t - z_I \sin W_o t$$

$$y_o = y_I$$

$$z_o = x_I \sin W_o t + z_I \cos W_o t$$

where W_o is the orbital rate, we may conduct the following linear stability analysis.

The body rates in terms of the orbital and Euler rates are

$$W_x = \dot{\phi} - \dot{\psi} \sin \theta + W_o \cos \theta \sin \psi$$

$$W_y = \dot{\theta} \cos \phi + \dot{\psi} \sin \phi \cos \theta + W_o (\sin \phi \sin \theta \sin \psi + \cos \phi \cos \psi)$$

$$W_z = - \dot{\theta} \sin \phi + \dot{\psi} \cos \phi \cos \theta + W_o (\cos \phi \sin \theta \sin \psi - \sin \phi \cos \psi)$$

Linearizing for small angles yields

$$W_x = \dot{\phi} + W_o \psi$$

$$W_y = \dot{\theta} + W_o$$

$$W_z = \dot{\psi} - W_o \phi$$

The gravity gradient torques are

$$M_{xG} = \frac{3}{2} W_o^2 (I_z - I_y) \sin(2 \phi) \cos^2 \theta$$

$$M_{yG} = \frac{3}{2} W_o^2 (I_z - I_x) \sin(2 \theta) \cos \phi$$

$$M_{zG} = \frac{3}{2} W_o^2 (I_x - I_y) \sin(2 \theta) \sin \phi$$

Linearizing for small angles yields

$$M_{xG} = 3\,W_o^2\,(I_z - I_y)\,\phi$$

$$M_{yG} = 3\,W_o^2\,(I_z - I_x)\,\theta$$

$$M_{zG} = 0$$

Eulers equations of motion are:

$$M_x = I_x\,\dot{W}_x + (I_z - I_y)\,W_y\,W_z$$

$$M_y = I_y\,\dot{W}_y + (I_x - I_z)\,W_x\,W_z$$

$$M_z = I_z\,\dot{W}_z + (I_y - I_x)\,W_x\,W_y$$

Substituting for $\vec{M}$ & $\vec{W}$ and linearizing yields:

$$3W_o^2\,(I_z - I_y)\phi = I_x\,\ddot{\phi} + I_x\,W_o\,\dot{\psi} + (I_z - I_y)(W_o\,\dot{\psi} - W_o^2\,\phi)$$

$$3W_o^2\,(I_z - I_x)\theta = I_y\,\ddot{\theta}$$

$$0 = I_z\,\ddot{\psi} - I_z\,W_o\,\dot{\phi} + (I_y - I_x)(W_o\,\dot{\phi} + W_o^2\,\psi)$$

Note that the pitch axis is independent of yaw and roll, but yaw and roll are coupled.

The stability constraint about the pitch axis is $\dfrac{I_x - I_z}{I_y} \geqslant 0$

The roll-yaw characteristic equation is

$$0 = S^4 + S^2 W_o^2\left(\frac{2I_xI_z - 3I_z^2 - I_xI_y + 2I_yI_z + I_y^2}{I_x\,I_z}\right) + 4W_o^4\left(\frac{(I_y - I_x)(I_y - I_z)}{I_x\,I_z}\right)$$

Let $R_x = \dfrac{I_x}{I_y}$ and $R_z = \dfrac{I_z}{I_y}$

Then the stability constraint about the pitch axis is $\left(R_x - R_z\right) \geq 0$ (1)

and the roll-yaw characteristic equation is

$$0 = S^4 + S^2 W_o^2\left(\frac{2R_xR_z - 3R_z^2 - R_x + 2R_z + 1}{R_xR_z}\right) + 4\,W_o^4\left(\frac{(1-R_x)(1-R_z)}{R_xR_z}\right)$$

This equation has the form

$$0 = S^4 + b\,S^2 + C$$

where

$$\frac{b}{W_o^2} = \frac{2R_xR_z - 3R_z^2 - R_x + 2R_z + 1}{R_xR_z}$$

and

$$\frac{C}{4W_o^4} = \frac{(1-R_x)(1-R_z)}{R_xR_z}$$

The stability constraints on this equation are:

$$b \geq 0 \qquad (2)$$

$$c \geq 0 \qquad (3)$$

$$\frac{b^2}{4} - C \geq 0 \qquad (4)$$

Since the body is spinning about the y axis, it can be shown that with internal energy dissipation the satellite will be stable only in $I_y \geq I_x$ and $I_y \geq I_z$. These two constraints can be specified as:

$$R_x \leq 1.0 \qquad (5)$$

and $R_z \leq 1.0$ (6)

For any rigid body the following inequalities hold true:

$$I_x + I_y \geq I_z$$

$$I_y + I_z \geq I_x$$

$$I_z + I_x \geq I_y$$

or

$$R_z - R_x \leq 1 \tag{7}$$

$$R_x - R_z \leq 1 \tag{8}$$

$$R_z + R_x \geq 1 \tag{9}$$

For system stability these nine constraints must be satisfied simultaneously. On the entire R_z, R_x plane (which covers all possible rigid bodies), the only stable area is that shown on Figure II.

3. Gravity Gradient Capture

Consider the planar problem of a satellite which is not near the gravity gradient stabilized pitch angle and not near the gravity gradient stabilized pitch rate. This analysis will establish the inertial constraints which assure that the satellite will be captured into the desired gravity gradient stabilized mode. Figure III defines the angles used in this analysis.

$$M_y = I_y \ddot{\alpha} \qquad \text{(equation of motion)}$$

$$M_y = 1.5\, W_o^2\, (I_z - I_x) \sin 2\theta \qquad \text{(gravity gradient)}$$

or $\quad M_y = 1.5\, W_o^2\, (I_x - I_z) \sin 2\gamma$

Therefore, $I_y \ddot{\alpha} = 1.5\, W_o^2\, (I_x - I_z) \sin 2\gamma$

$$W_o t = \alpha + \gamma$$

$$\alpha = W_o t - \gamma$$

$$\dot{\alpha} = W_o - \dot{\gamma}$$

$$\ddot{\alpha} = -\ddot{\gamma}$$

Therefore, $-I_y \ddot{\gamma} = 1.5 W_o^2 (I_x - I_z) \sin 2\gamma$

Let $\beta = 2\gamma$ Then $\ddot{\gamma} = .5\ddot{\beta}$

$$-\frac{I_y}{2}\ddot{\beta} = 1.5\, W_o^2 (I_x - I_z) \sin\beta$$

$$\ddot{\beta} = -3\, W_o^2 \left(\frac{I_x - I_z}{I_y}\right) \sin\beta$$

$$\ddot{\beta} = -3\, W_o^2 (R_x - R_z) \sin\beta$$

$$\ddot{\beta} = \frac{d\dot{\beta}}{dt} = \frac{d\dot{\beta}}{d\beta}\,\frac{d\beta}{dt} = \dot{\beta}\,\frac{d\dot{\beta}}{d\beta}$$

Therefore, $\dot{\beta}\, d\dot{\beta} = -3\, W_o^2 (R_x - R_z) \sin\beta \;\; d\beta$

or

$$\frac{(\dot{\beta})^2}{2} - \frac{(\dot{\beta}_o)^2}{2} = 3W_o^2 (R_x - R_z)(\cos\beta - \cos\beta_o)$$

$$(\dot{\beta})^2 = (\dot{\beta}_o)^2 + 6\, W_o^2 (R_x - R_z)(\cos\beta - \cos\beta_o)$$

or

$$4(\dot{\gamma})^2 = 4\,(\dot{\gamma}_o)^2 + 6\, W_o^2\, (R_x - R_z)(\cos 2\gamma - \cos 2\gamma_o)$$

We can establish a capture bound by the statement that whenever the satellite achieves its maximum deviation from the stable position (or when $\dot{\gamma} = 0$), it is required that γ be within $\pm 90^o$. We can solve for the capture limit at $\dot{\gamma} = 0$ & $\gamma = 90^o$ or $\cos(2\gamma) = -1$

$$0 = 4\,(\dot{\gamma}_o)^2 - 6\, W_o^2 (R_x - R_z)(\cos(2\gamma_o) + 1)$$

or $$\left(\frac{\dot{\gamma}_o}{W_o}\right)^2 = 1.5\,(R_x - R_z)(\cos(2\gamma_o) + 1)$$

Figure II shows that $(R_x - R_z)$ should be limited such that

$$0 \leq (R_x - R_z) \leq 1.0$$

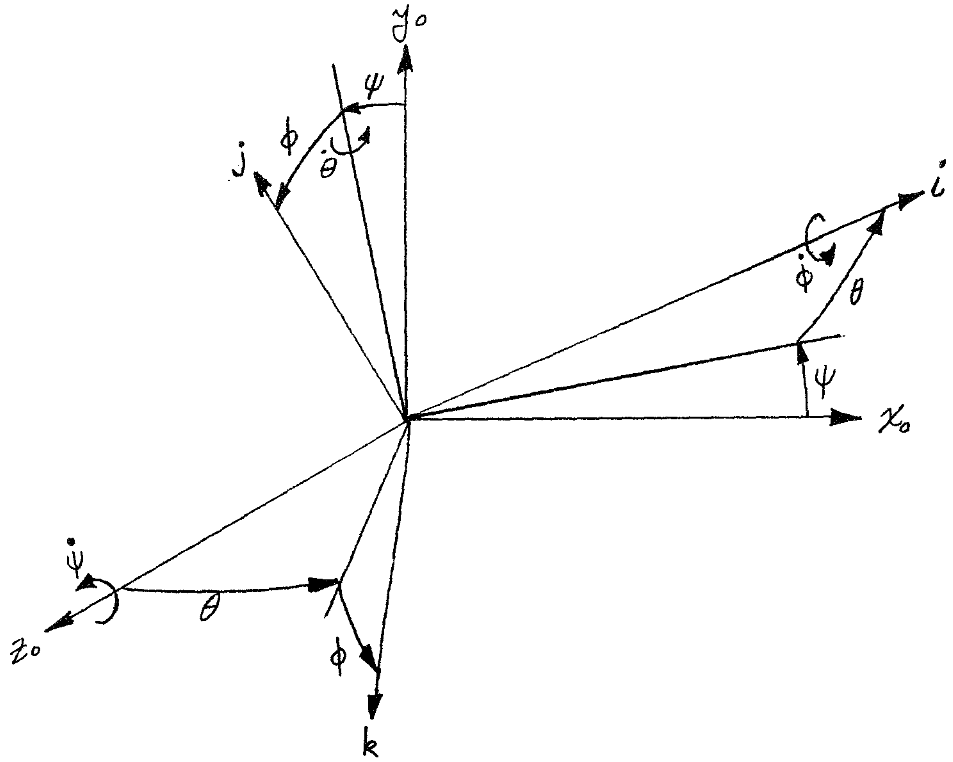
Figure IV is a plot of $\left(\frac{\dot{\gamma}_o}{W_o}\right)^2$ vs $(R_x - R_z)$ and $\pm \gamma_o$.

For a specific example let us assume that at t = 0 the satellite is in the proper attitude for gravity gradient stabilization ($\gamma_o = 0$) but that its inertial pitch rate is zero ($\dot{\alpha}_o = 0$) or $\dot{\gamma}_o = W_o$.

$$\left(\frac{\dot{\gamma}_o}{W_o}\right)^2 = 1.0$$

and from Figure IV $(R_x - R_z) = \frac{1}{3}$ minimum. Figure V shows the new stability zone with this capture constraint added.

1 - 1691



EULER TRANSFORMATION FROM THE ORBITAL COORDINATES $(\vec{x}_o ; \vec{y}_o ; \vec{z}_o)$ TO THE BODY COORDINATES $(i ; j ; k)$ VIA THE SEQUENCE YAW (ψ), PITCH (θ), AND ROLL (ϕ).

FIGURE I

1 - 1691

GRAVITY GRADIENT STABLE ZONE

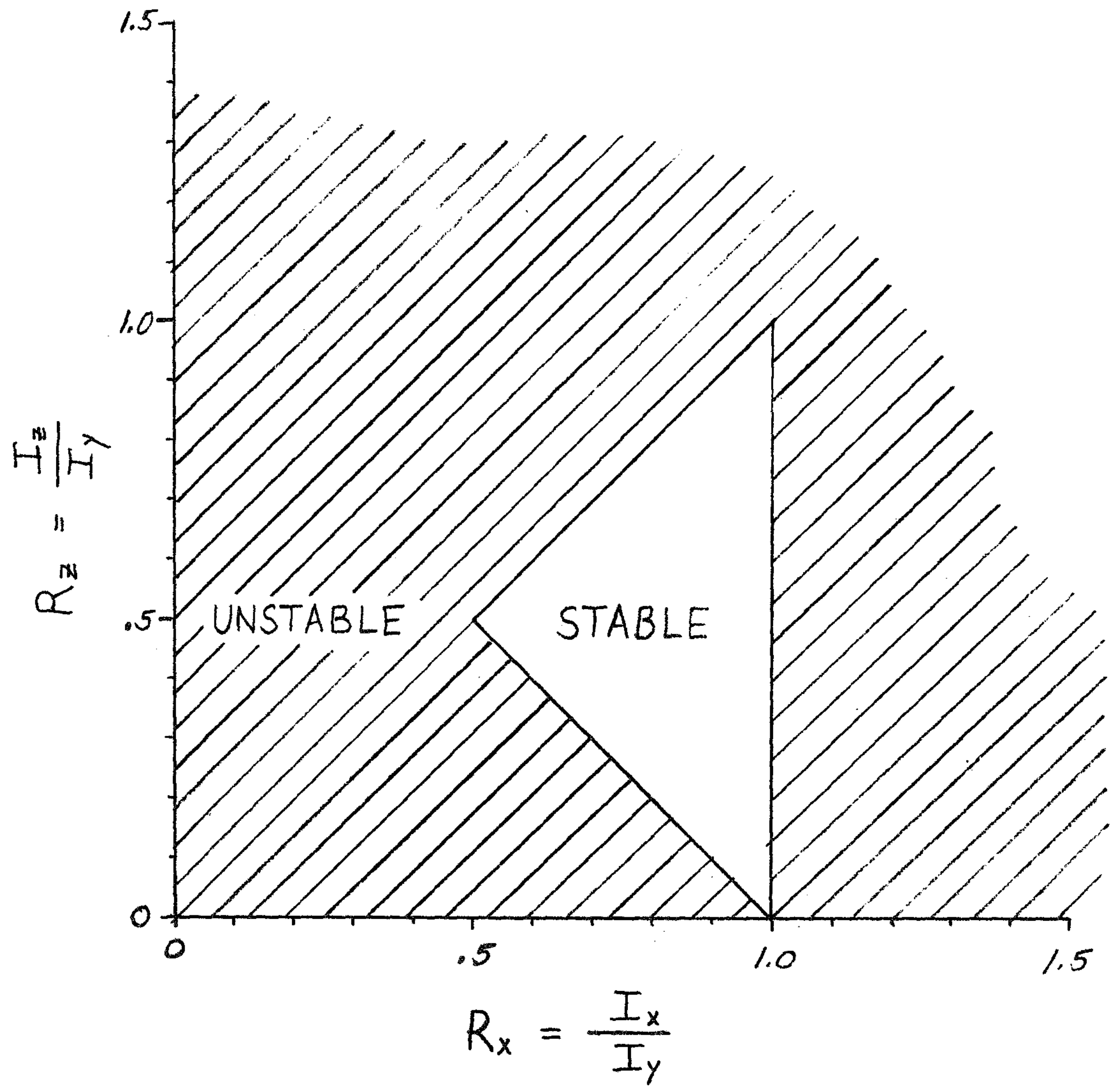
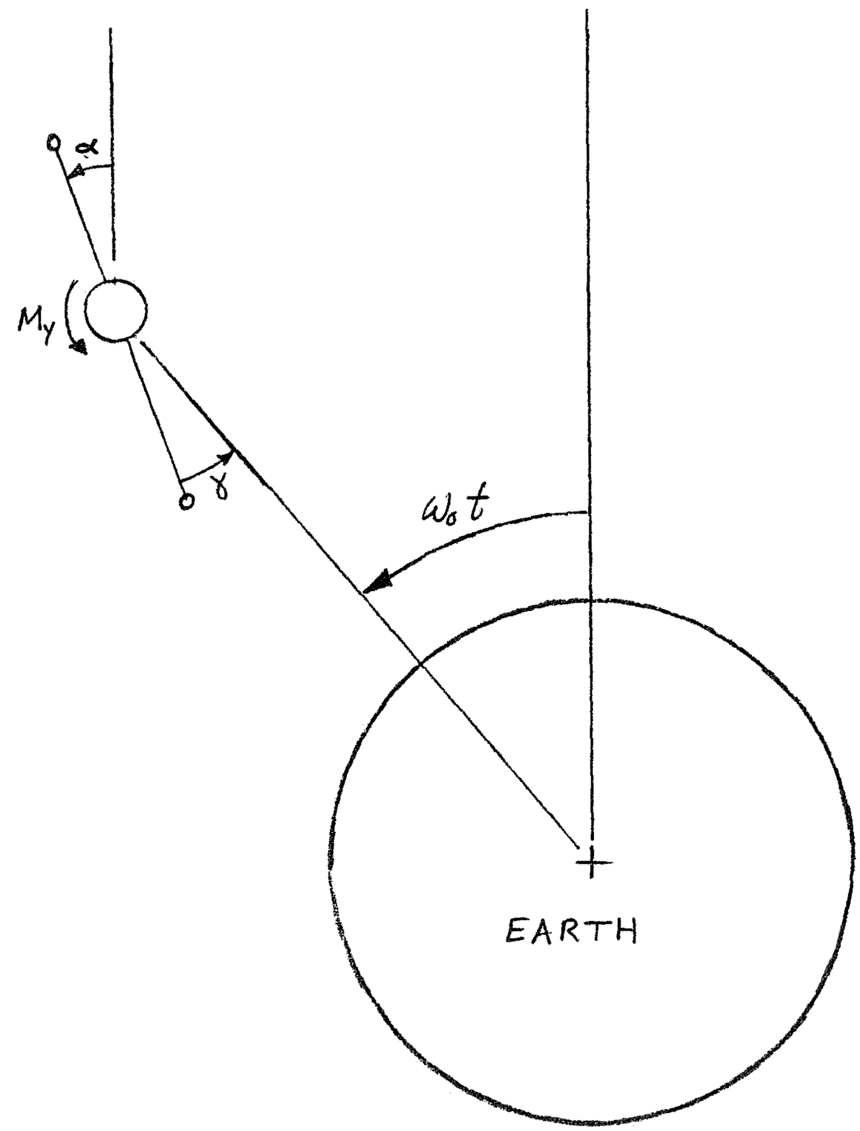


FIGURE II

ANGLE DEFINITIONS



NOTE: $\gamma = -\theta$ WHERE θ = PITCH EULER ANGLE REFERED TO ORBITAL FRAME

FIGURE III

LIMIT OF GRAVITY GRADIENT CAPTURE

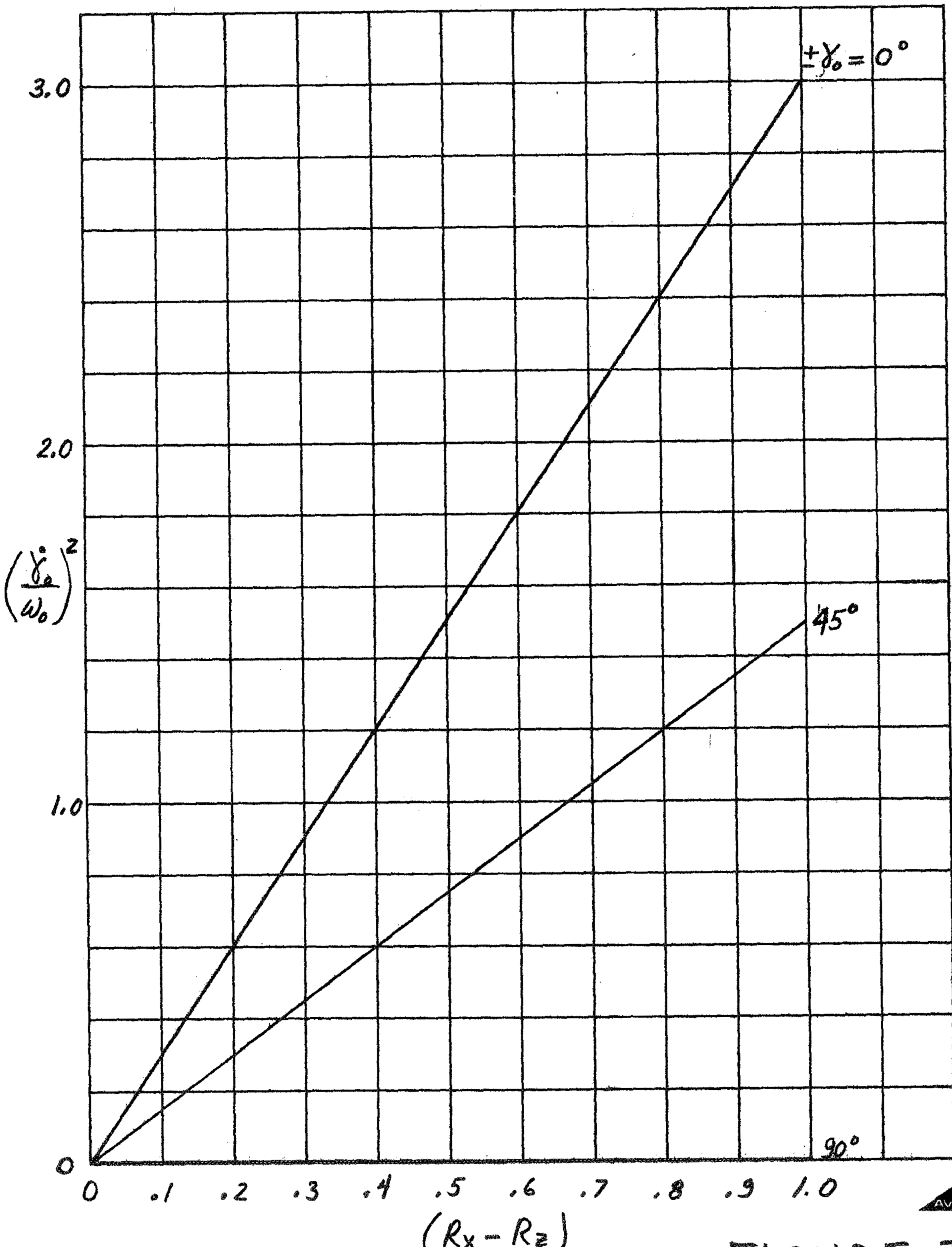


FIGURE IV

1 - 1691

GRAVITY GRADIENT STABLE AND CAPTURE ZONE

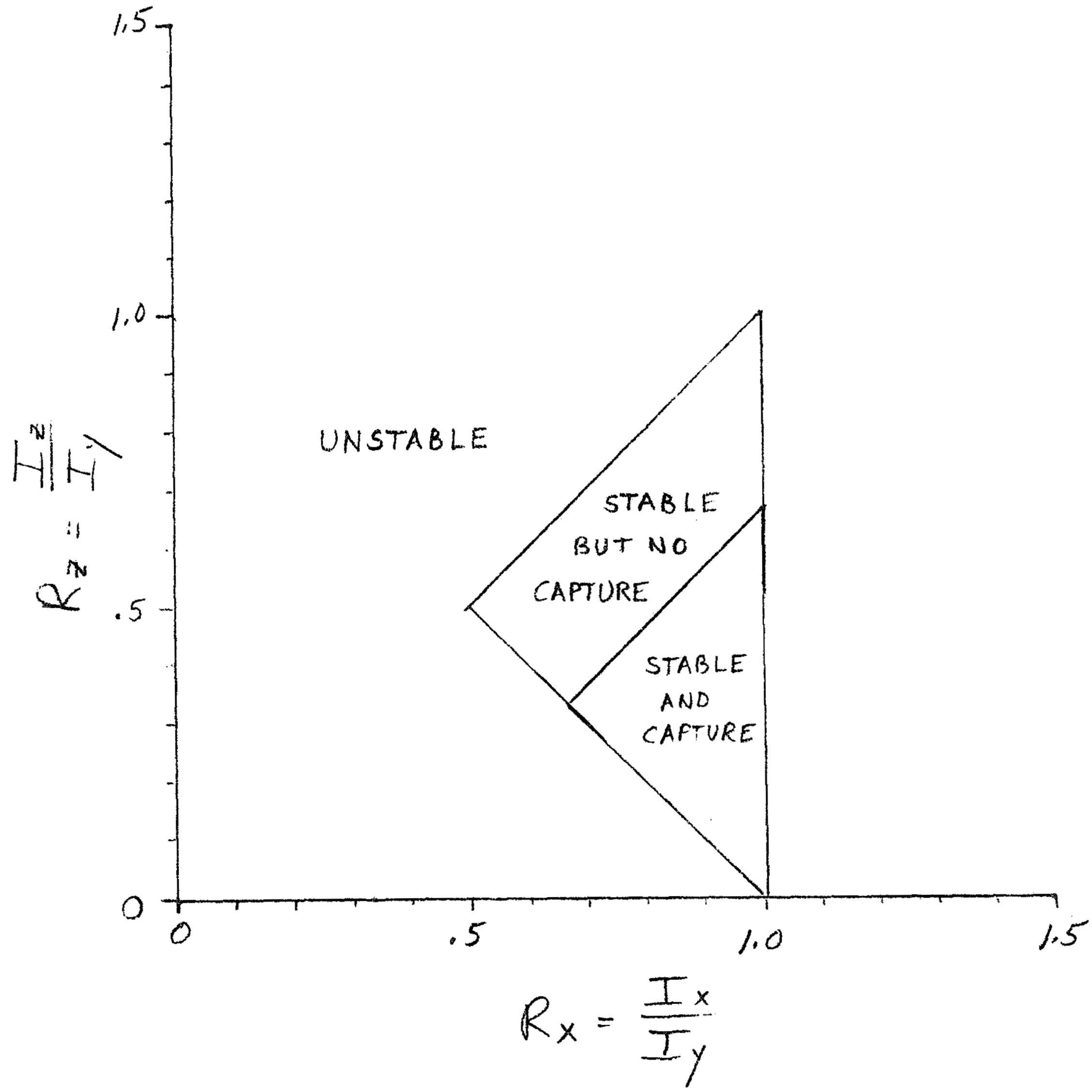


FIGURE V

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒ F220-JEM-70-55

TO	DEPT.	FROM	DEPT.	DATE
S. Brzeski	L910	J. E. Mozzicato	F220	7/29/70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control For Small Satellite and Related Subsystems	W159-F220-033		

SUBJECT

Vehicle Disturbance Torques

DISTRIBUTION

K. Arnesen, D. Fields, W. C. Hailey, E. Lawlor, R. Litte, F. Scammell, G. Kunkel, G. Pfeiffer, M. Weinberger, Files

SIGNED

J. E. M[illegible]

APPROVED

F. H. [illegible]

INFORMATION REQUESTED / RELEASED

This T/R presents an estimate of the disturbance torques which will be encountered by a typical (200#) vehicle.

1. Aerodynamic Torque

Two vehicle configurations will be considered in establishing the aerodynamic torque levels. These are (1) the balanced vehicle where theoretically the center of mass can be located on the center of pressure but torque is a result of uncertainty in the actual location of the center of pressure and (2) the unbalanced vehicle where the design provide a nominal displacement between the center of mass and the center of pressure.

To establish the vehicle area we shall consider the following external configuration.

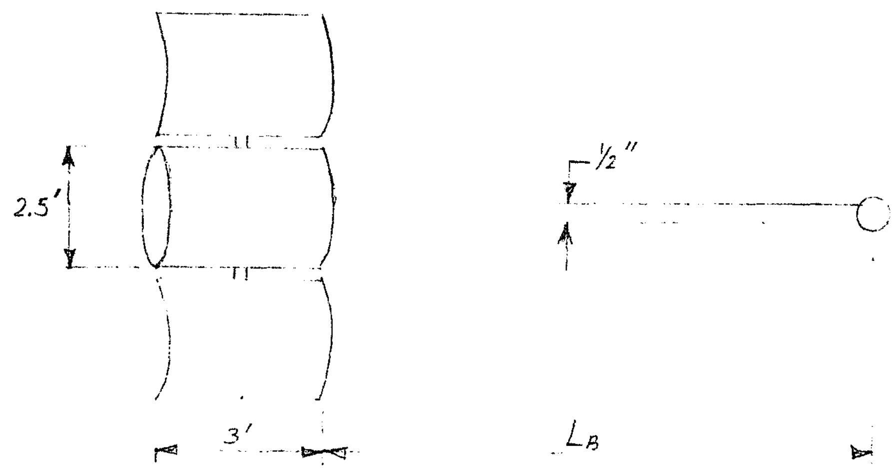


Figure I shows the total profile area of this configuration as a function of boom length. The figure shows that the boom length does not have a very strong effect on area and for the remainder of this exercise an area of 25 feet^2 will be assumed.

For the balanced vehicle we shall assume that the uncertainty of the C.P. location will be less than 5% of the length of the principle area source (.05 X 3 = .15 feet).

Therefore for the balanced vehicle the torque equation is:

$\tau_A = C_D \ A \ q \ \delta cp$

τ_A = Aerodynamic Torque (FT#)

C_D = Drag Coeff = 2.5

q = Orbital Dynamic Pressure (#/FT^2)

A = Profile Area (25 FT^2)

δcp = CP Uncertainty (.15 FT)

$\tau_A = 9.375 \ q$

Figure II shows τ_A vs orbital altitude for the balanced configuration. The q vs altitude profile used here is from figure 24 of NASA CR-831.

To estimate the maximum torque from the unbalanced vehicle we shall assume that the vehicle with panels deployed can be balanced to 5% of its 3 foot length. Therefore A δcp = 22.5 X .15 = 3.375 FT^3.

To this unbalance we will add the total unbalance of the boom.

(Total) A $\delta cp = (.04167 \ L_B) \left(\frac{L_B}{2} + 1.5 \right) + 3.375$

Figure II shows the torque on an unbalanced vehicle for various boom lengths.

An estimate of the required moment of inertia for the vehicle can be established by balancing the aerodynamic torque with a gravity gradient torque at some reasonable angle (Say 5°)

$$\tau_{GG} = 3 \ Wo^2 \ (\Delta I) \ \frac{5}{57.29}$$

τ_{GG} = Gravity Gradient Torque (FT#)

ω_0 = Orbital Rate (RAD/SEC)

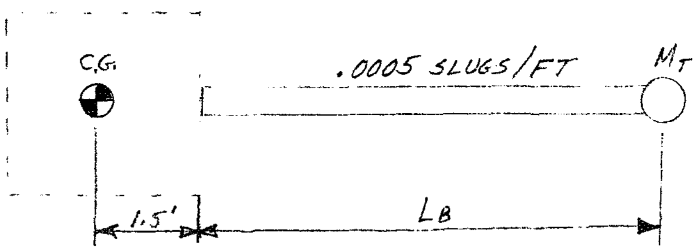
ΔI = Required moment of inertia (Slug FT^2)

ALT (nm)	$\frac{\Delta I}{\tau_{GG}} \left(\frac{SLUG\ FT^2}{FT\#}\right)$
200	2.9354×10^6
300	3.1841×10^6
400	3.4464×10^6
500	3.7225×10^6
600	4.01359×10^6
700	4.3190×10^6
800	4.6396×10^6
900	4.9756×10^6
1000	5.3275×10^6

Figure III is a plot of ΔI vs altitude.

An estimate of the configuration required to achieve the proper moment of inertia can be made as follows. Assume the vehicle itself is relatively symmetric and therefore its inertia does not enter into the ΔI calculation.

ΔI can be established from the following boom and tip mass configuration:



L_B = Boom Length (FT)
M = Tip Mass (Slugs)

$$\Delta I = \frac{.0005}{12} L_B^3 + .0005\ L_B \left(\frac{L_B}{2} + 1.5\right)^2 + M_T \left(L_B + 1.5\right)^2$$

Figure IV shows the resulting tip weight requirements vs. boom length for both the balanced and unbalanced configurations.

2. ORBIT ECCENTRICITY

For a vehicle in an eccentric orbit there is an error between the inertial angle, (M = Mean Anomaly) defined by integrating a constant rate which is equal to the mean orbital rate, and the true anomaly (γ= true anomaly) defined by the vehicle - earth line.

The true anomaly can be expanded into the following series:

$$\gamma = M + 2e \text{ SIN } M + \frac{5}{4} e^2 \text{ SIN} 2 M + . . .$$

For $e < .1$ we may say that:

$$\gamma = M + 2e \text{SIN } M$$

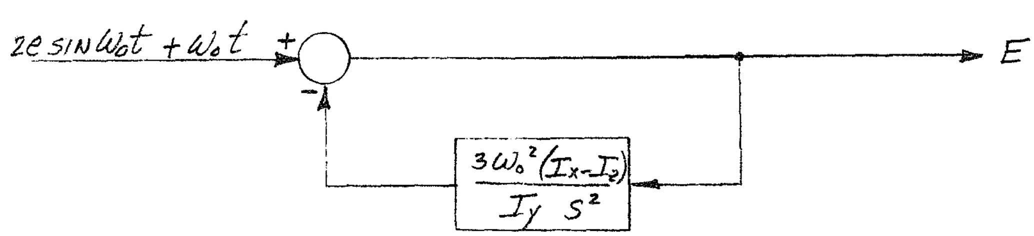
But $M = \omega_o t$

Where ω_o = mean orbital rate.

Therefore $\gamma = M + 2e \text{ SIN } \omega_o t$

Consider the following linear-planar analysis.

If we let the pitch error (E) be the error between vehicle position and local vertical we may construct the following diagram:



Considering only the sinusoidal input in the steady state, the pitch error will be given by:

$E \sin \omega_0 t$

Figure V shows E as a function of vehicle inertia and orbital eccentricity e. For slender vehicles

$$\frac{Ix}{Iy} - \frac{Iz}{Iy} \approx 1.0$$

Therefore the pitch error can now be expressed as

$e \sin \omega_0 t$ (Radians) or

$57.3\, e \sin \omega_0 t$ (Degrees)

3. OTHER DISTURBANCES

This paragraph will serve to show that all other disturbances are very small compared to the effects of aerodynamics and eccentricity.

a. Solar Pressure

The solar pressure for earth orbits is 9.65×10^{-8} #/FT^2 which is equivalent to the aerodynamic pressure encountered at 425 nm. and is two orders of magnitude lower than the aerodynamic pressure at 200 nm.

b. Magnetic Interaction

Based upon the approximations given in NASA SP-8018 for a 200# Class III non spinning vehicle, the magnetic torque should be less than 1.72×10^{-5} FT# for an equatorial orbit and 3.433×10^{-5} FT# for a polar orbit. This is about one decade lower than the aerodynamic torque at 200 nm.

c. Meteorite Impact

Based upon the data presented in NASA CR-831 it can be estimated that there is only a 2% chance of a meteorite impact causing an angular deviation of 1° (See Fig. VI).

d. Electromagnetic Emission

Electromagnetic forces which are of the order of 7.5×10^{-10} # /Watt and 1.6×10^{-10} # / BTU/HR are completely negligible.

e. Internal Mass Shifts

The only significant mass motion which must be considered is that of solar panels tracking the sun.

If we assume that total panel inertia is 10% of the satellite inertia with the boom retracted ($I_p \approx .5$ slug FT^2) and that it has an angular rate with respect to the body of the order of orbital rate (ω_o), the panel has an angular momentum relative to the vehicle of

$$H_p = .5\,\omega_o$$

If the panels were locked and all of the momentum was transferred to the vehicle having an inertia of 600 slug FT^2 then the vehicle angular rate would change by

$$\Delta W = \frac{.5\,\omega_o}{600}$$

If this induced rate causes the vehicle to oscillate at the pitch natural frequency

$$\omega_N = \omega_o \sqrt{3} \sqrt{\frac{I_x}{I_y} - \frac{I_z}{I_y}} \approx \sqrt{3}\,\omega_o$$

then the resulting rate ($\dot{\theta}$) will be

$$\dot{\theta} = \frac{.5\,\omega_o}{600} \sin \sqrt{3}\,\omega_o t$$

$$\text{OR} \quad \theta = -\frac{.5}{600\sqrt{3}} \cos \sqrt{3}\,\omega_o t$$

$$\therefore \theta_{MAX} = \frac{(.5)(57.29)}{(600)\sqrt{3}} = .0275^\circ$$

This indicates that solar panels tracking the sun should not introduce significant angular deviations.

4. Conclusions

a. At altitudes less than 350 nm the vehicle must be aerodynamically balanced to limit aerodynamic torques

b. Orbital eccentricity is a significant disturbance at all altitudes.

c. An aerodynamically balanced vehicle with a 60 feet boom and a 5 pound tip mass represents a reasonable base line design.

d. Solar pressure and magnetic torque do not become significant disturbances until an altitude of 400 nm is reached.

e. Other disturbances can be neglected.

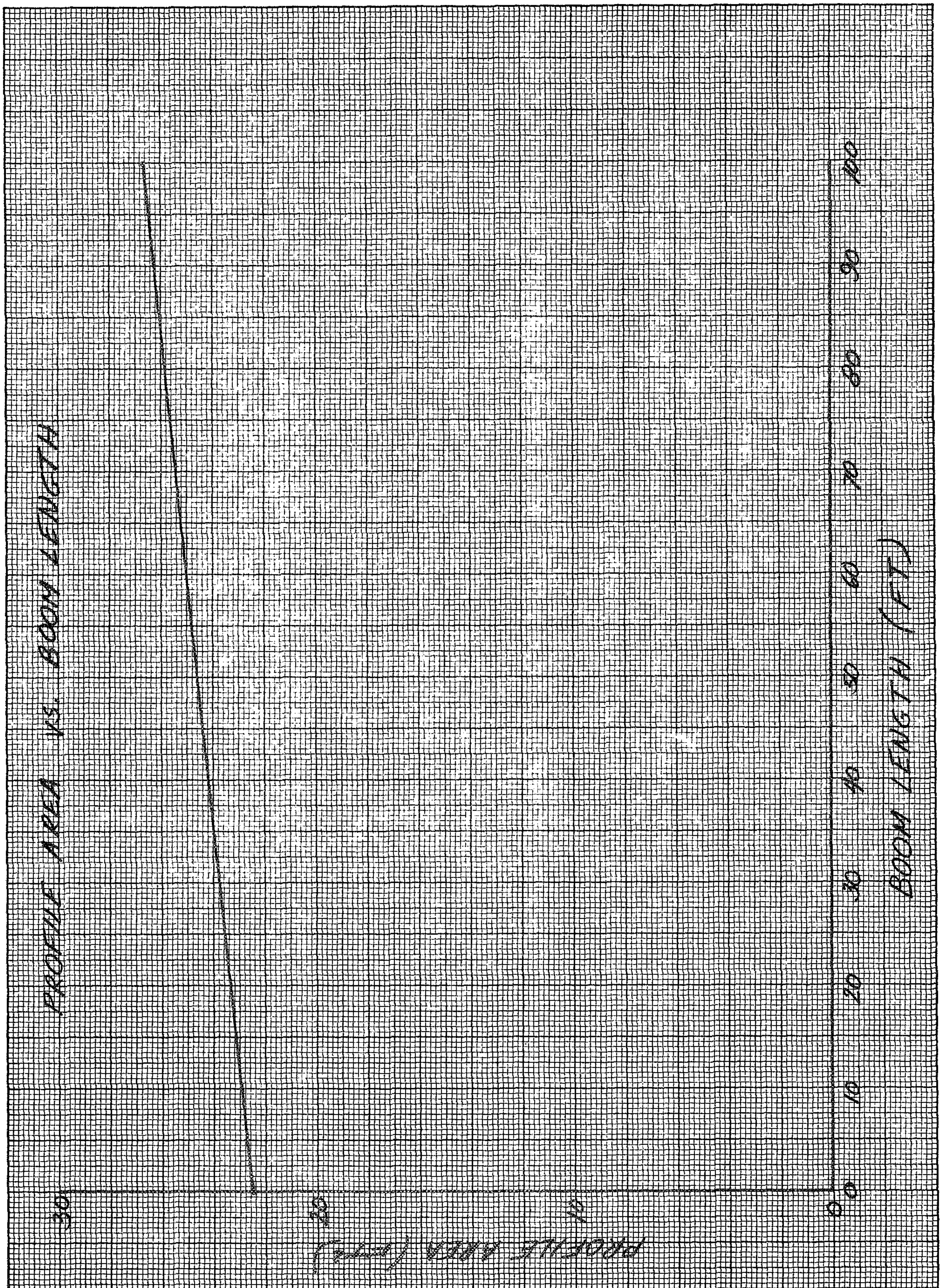

PROFILE AREA VS. BOOM LENGTH
BOOM LENGTH (FT)
0
10
20
30
40
50
60
70
80
90
100
PROFILE AREA
0
10
20
30

FIGURE I

AERODYNAMIC TORQUE vs. ALTITUDE

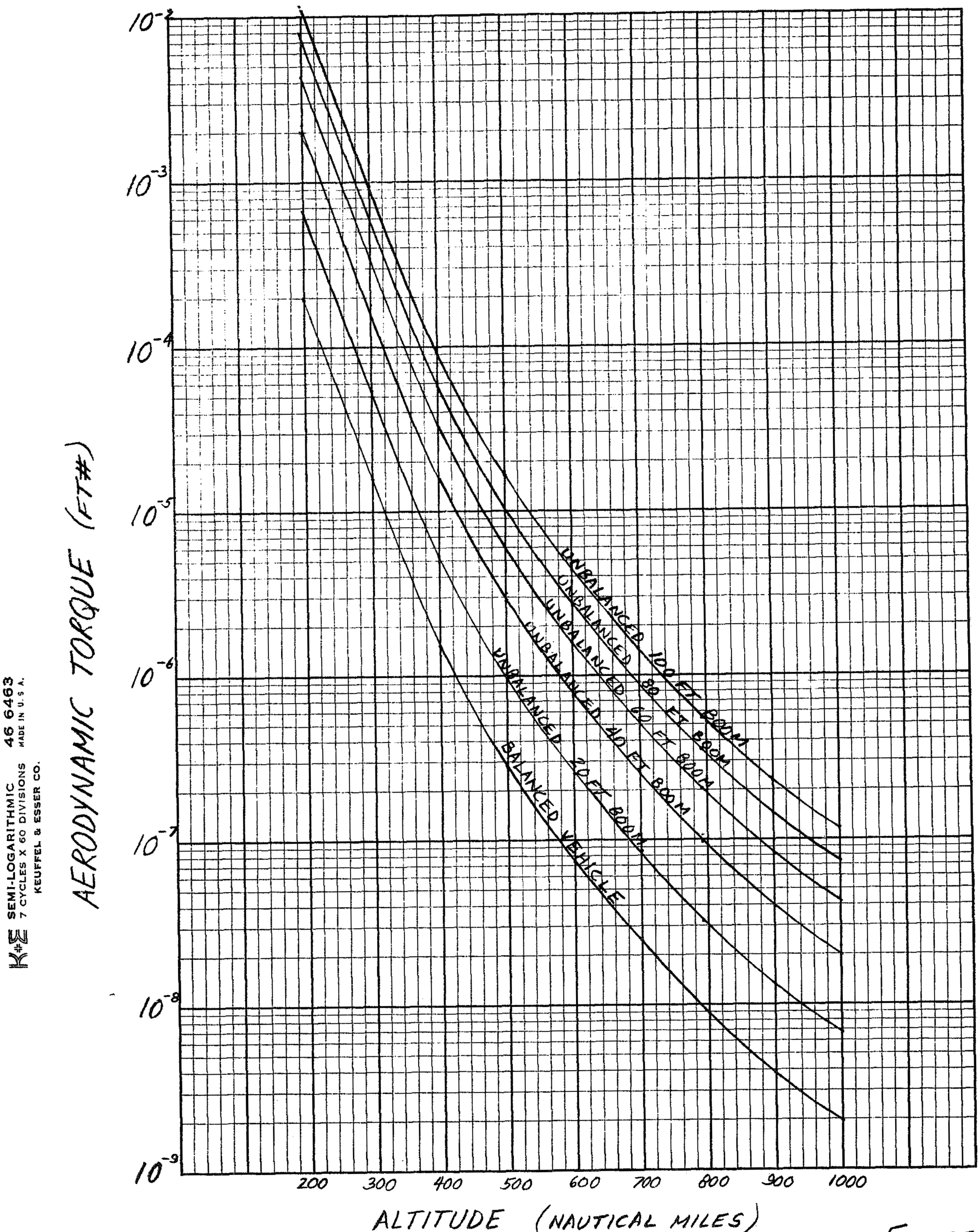


FIGURE II

INERTIA REQUIRED TO BALANCE AERO-TORQUE AT 5°

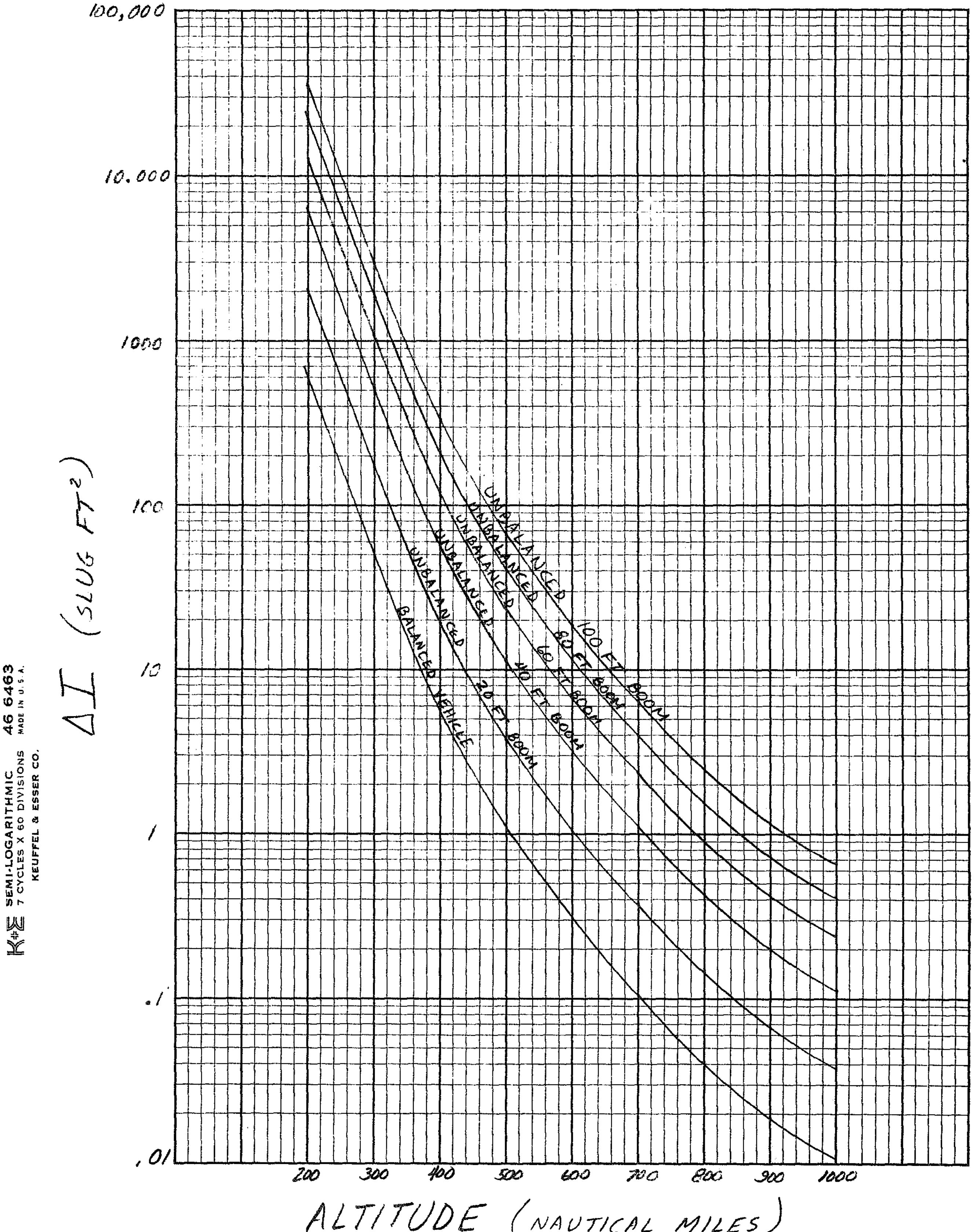
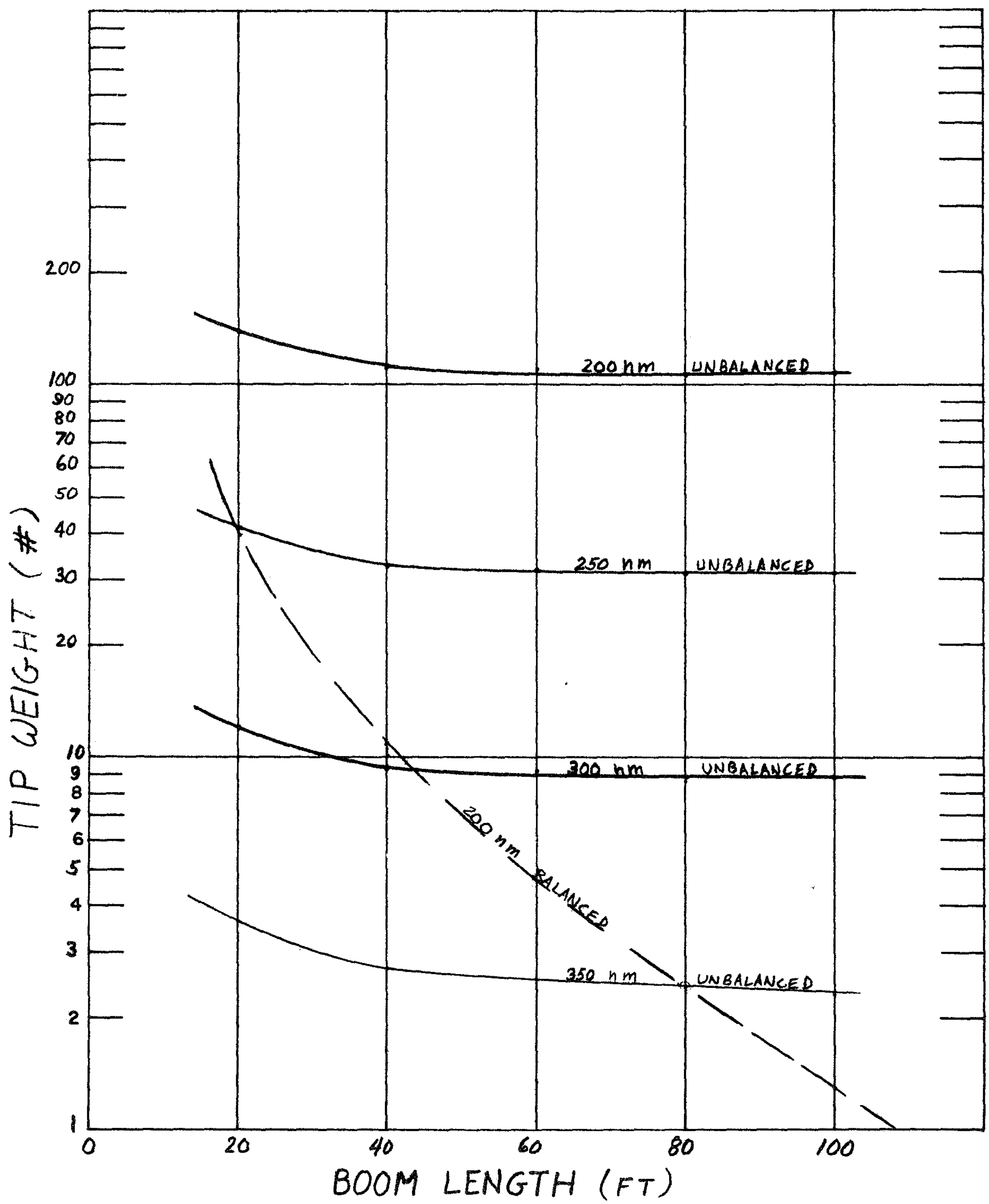


FIGURE III

TIP WEIGHT vs. BOOM LENGTH



AVCO

FIGURE IV

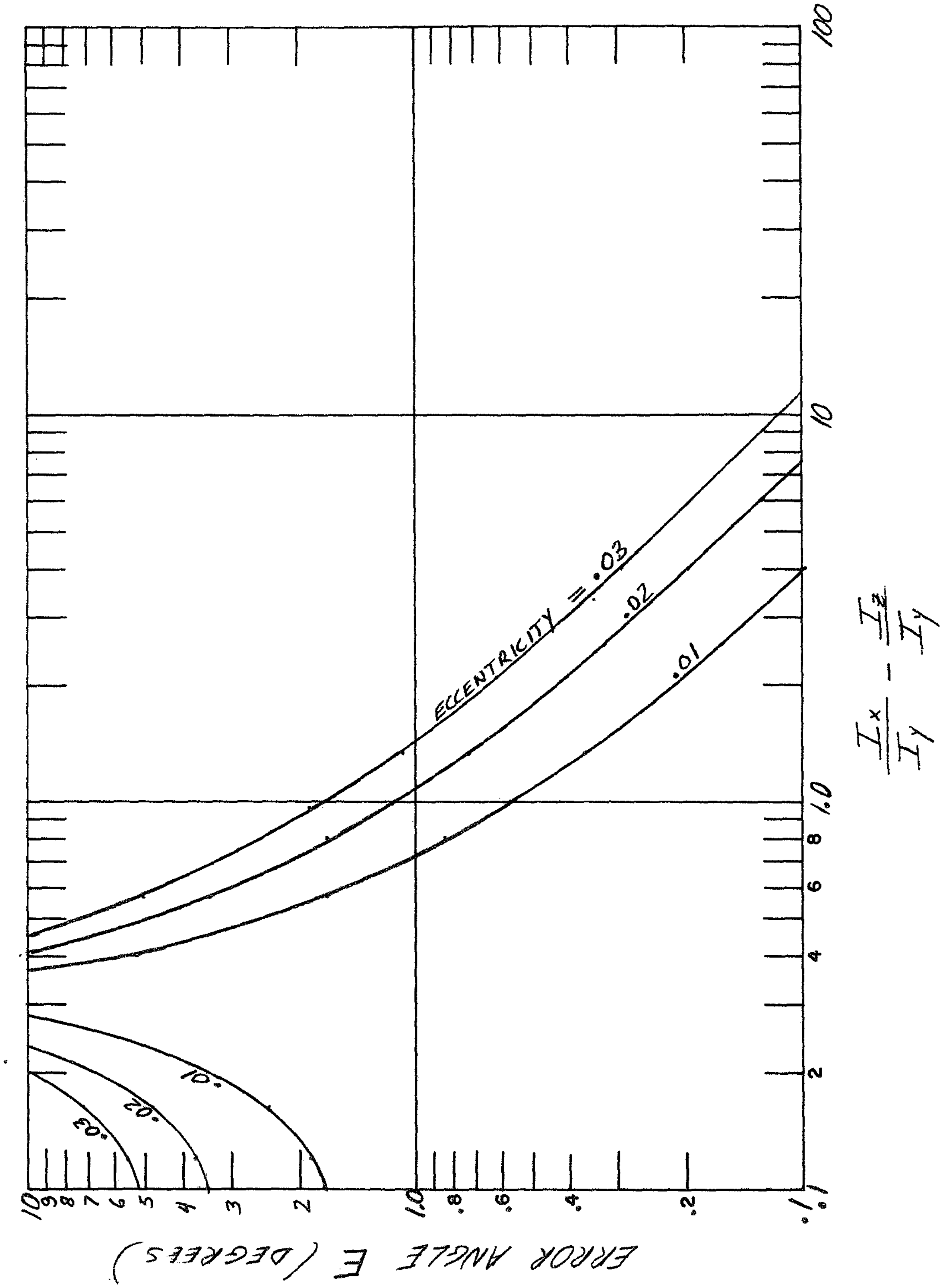

ECCENTRICITY = .03
.02
.01
.01
.02
.03
ERROR ANGLE E (DEGREES)
$\frac{I_x}{I_y} - \frac{I_z}{I_y}$

FIGURE V

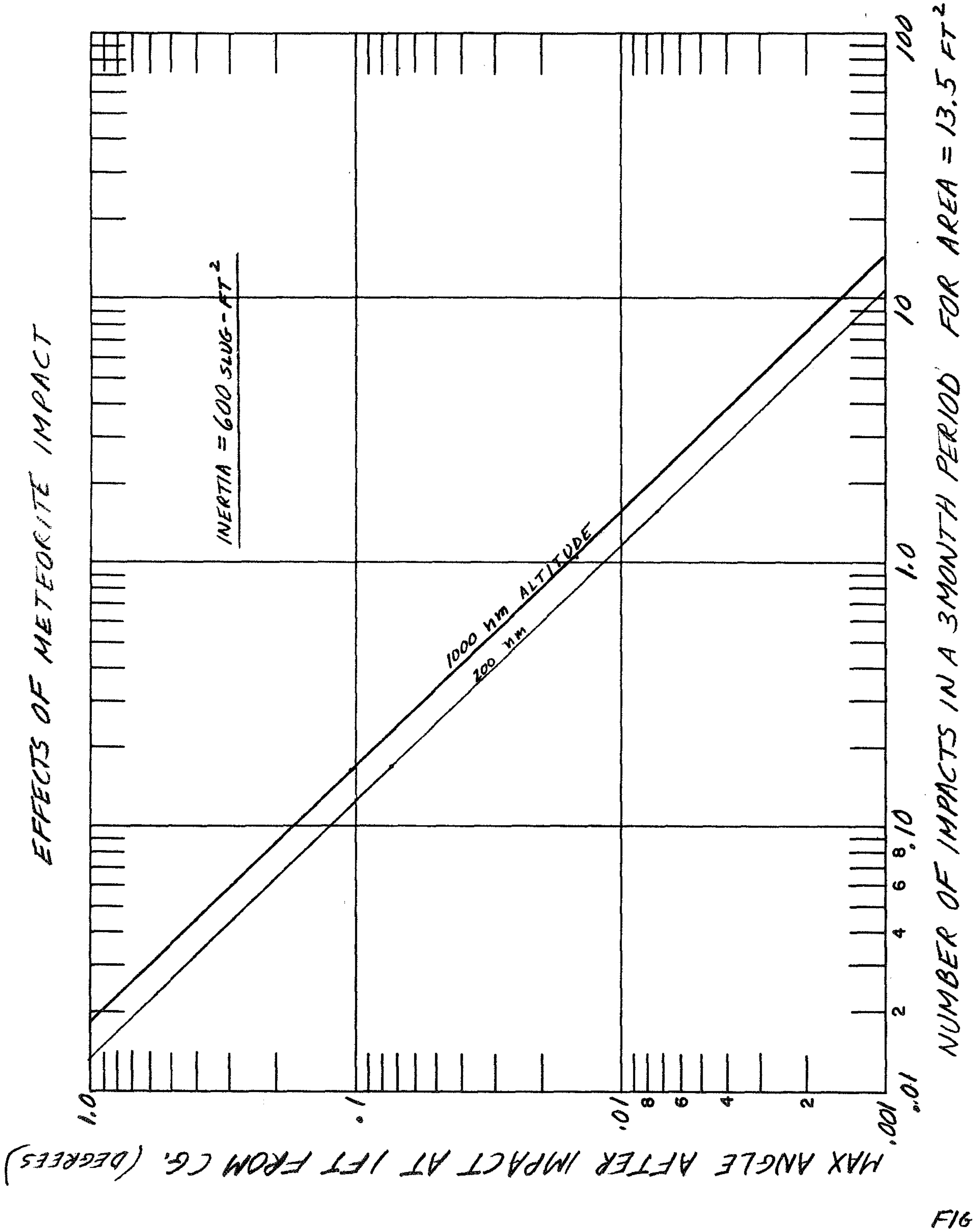

EFFECTS OF METEORITE IMPACT
INERTIA = 600 SLUG - FT²
1000 NM ALTITUDE
200 NM
NUMBER OF IMPACTS IN A 3MONTH PERIOD FOR AREA = 13.5 FT²
.01 2 4 6 8 .10 1.0 10 100
MAX ANGLE AFTER IMPACT AT 1FT FROM CG. (DEGREES)
1.0 .1 .01 8 6 4 2 .001

FIGURE VI

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X] F220-JEM-70-58

O	DEPT.	FROM	DEPT.	DATE
S. Brzeski	L910	J. E. Mozzicato	F220	8/12/70

ROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-F220-033		

JBJECT: Active Control of a Gravity Gradient Configuration

STRIBUTION	SIGNED / APPROVED
K. Arnesen, D. Fields, W.C. Hailey, E. Lawlor, R. Litte, F. Scammell, G. Kunkel, G. Pfeiffer, M. Weinberger, Files	SIGNED: J. E. Mozzicato; APPROVED: F. F. [illegible]

FORMATION REQUESTED / RELEASED

The following calculations establish the order of magnitude of the torque required and the total impulse required to provide active attitude control for a single boom gravity gradient configuration.

Conclusions

1. The use of a .5" diameter boom is marginal. A 1" diameter boom would be preferred.

2. Thrust levels should be of the order of .001 to .002 pounds.

3. A total impulse capability of 360 # sec is required for a 90 day mission.

Calculations

Several constraints must be observed when applying torque to the vehicle. The torque level must be sufficient to overcome the maximum disturbance expected (e.g., Aero torque = 2×10^{-4} ft #). The torque level must be low enough to avoid significant boom deflections. The limit cycle must be set to insure that the impulse expenditure is not excessive.

Consider a limit cycle in pitch which has a period which is 1/2 the orbital period (2745 sec) and which is maintaining the angle at $\pm 1^o$. If the limit cycle is mostly coast (constant angular rate) the average rate will be given by

$$\dot{\theta}_c = \frac{4 \times 1^o}{2745} \approx .001457 \text{ deg/sec}$$

During each torque application the rate changes by $2\,\dot{\theta}_c$ = .002914 deg/sec.

The torque impulse required to establish this rate change for a vehicle with an inertia of 588 slug ft^2 is given by

$$\tau_I = \frac{(588)(.002914)}{57.29} = .0299 \text{ ft \# sec}$$

Every cycle (2745 sec) this impulse is expended twice and therefore over a 90 day mission the total torque impulse for the pitch axis is

$$\tau_I = \frac{(2)(.0299)(90)(24)(3600)}{(2745)} = 169.4 \text{ ft \# sec}$$

For a moment arm of 1.5 ft the total pitch axis impulse is 112.9 # sec for 90 days. The roll axis will require a similar impulse expenditure and the yaw axis will require only about 17 # sec for a total impulse expenditure of 243 # sec. For low altitudes the pitch axis will be countering a secular aerodynamic torque rather than operating in a two-sided limit cycle. Therefore, the pitch impulse requirement will be as shown in Figure I. The total impulse requirement is

Pitch	230 # sec
Roll	113 # sec
Yaw	17 # sec
Total	360 # sec

PITCH IMPULSE REQUIREMENT

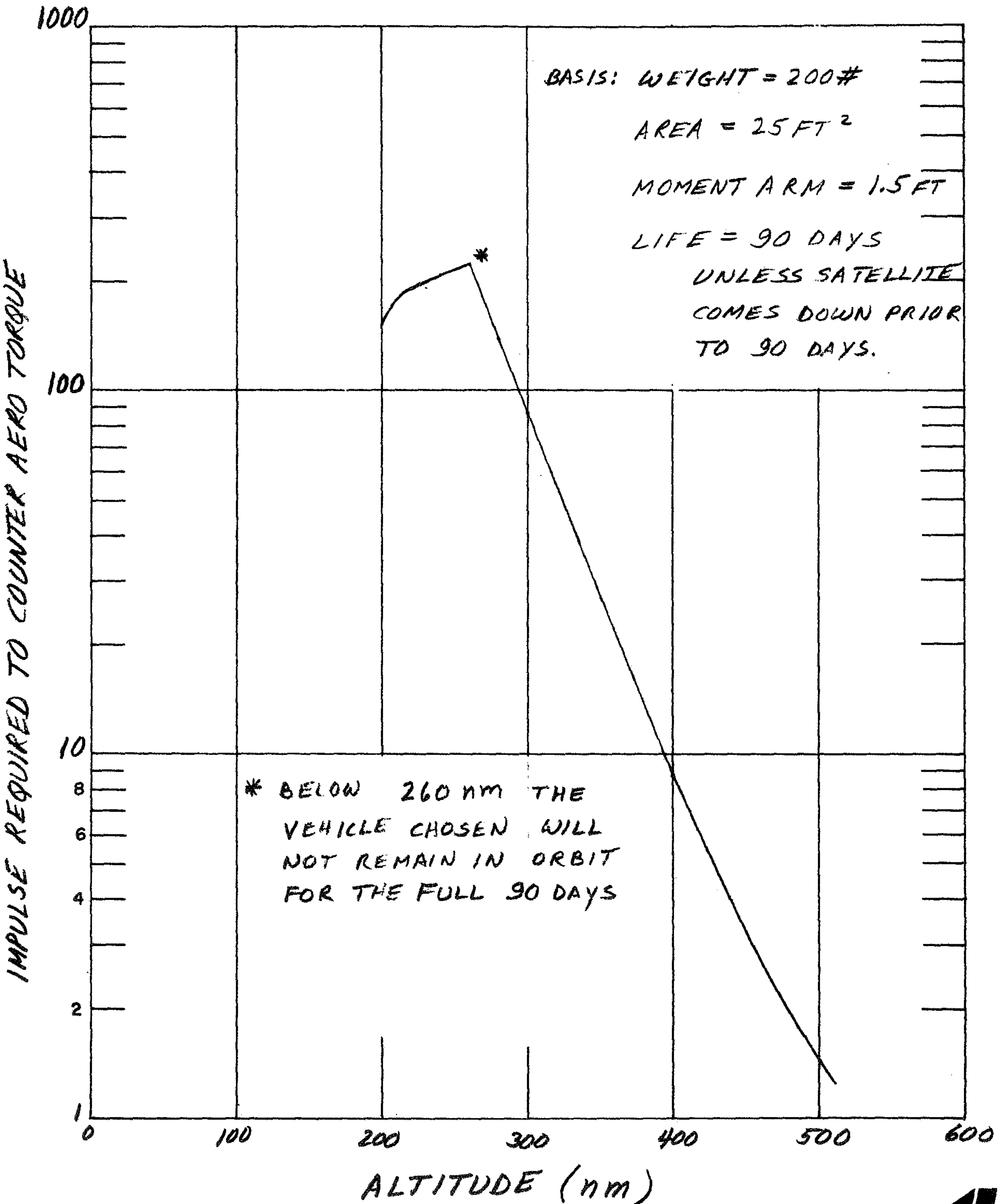
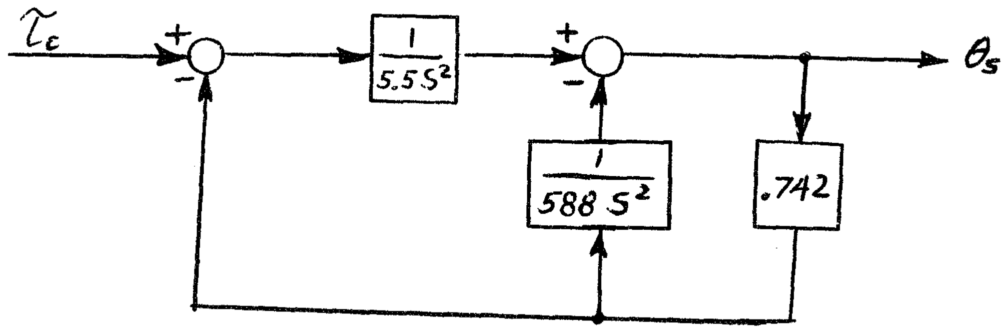


FIGURE I

The following planar linear analysis will be used to establish the maximum body acceleration which can be tolerated for a system with a boom. Figure II defines the basic parameters. Figure III is a spring mass analogy to Figure II and Figure IV is a block diagram of the vehicle. Since the gravity gradient natural frequency ($\sqrt{3}\,w_o$) is very low compared to the bending natural frequency the ($3\,w_o^2\,m_T\,L_B^2$) term may be neglected for this part of the analysis.

The block diagram now becomes:



where

θ_s = angular boom deflection (rad)

τ_c = control torque (ft #)

1 - 1691

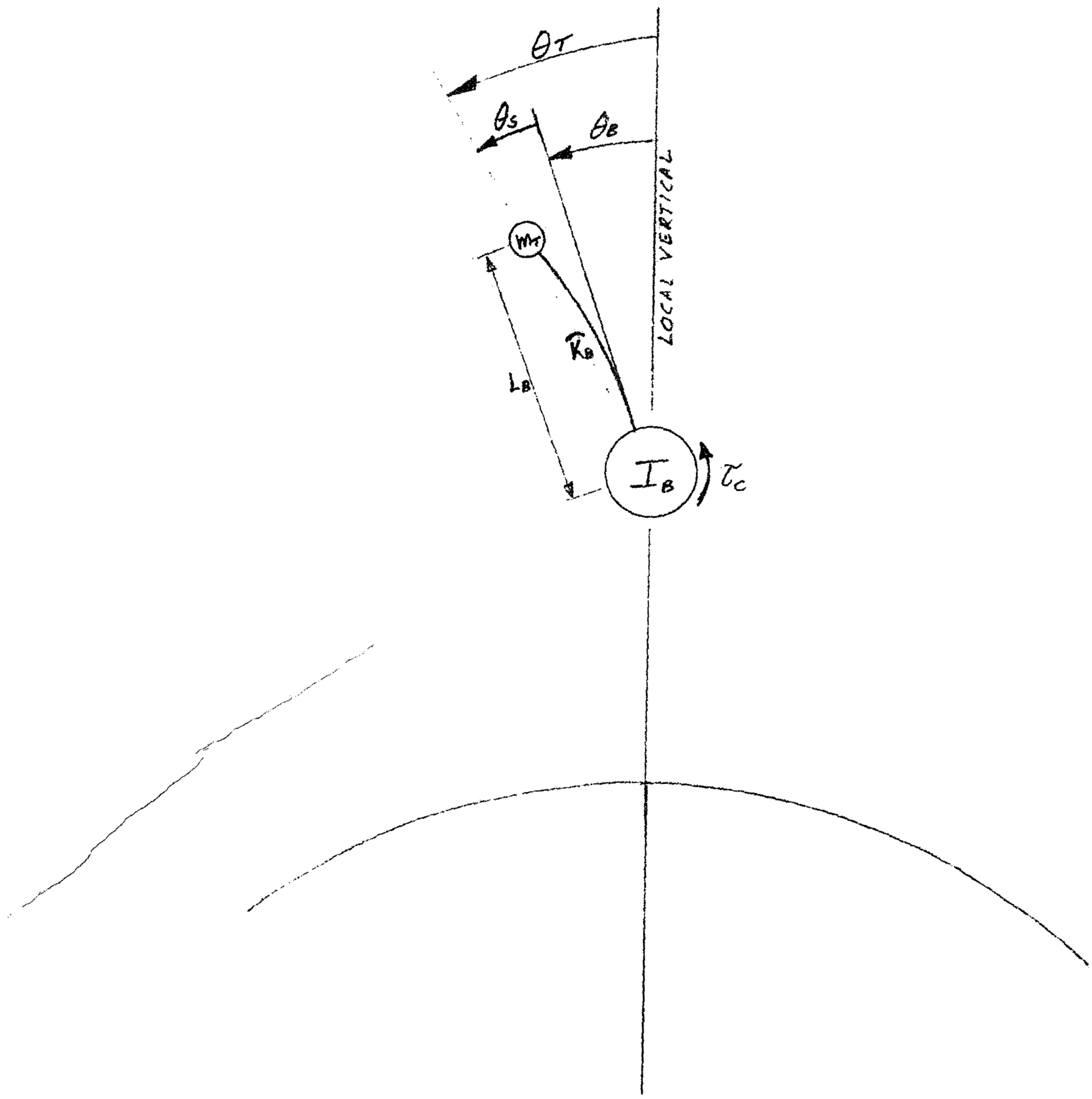
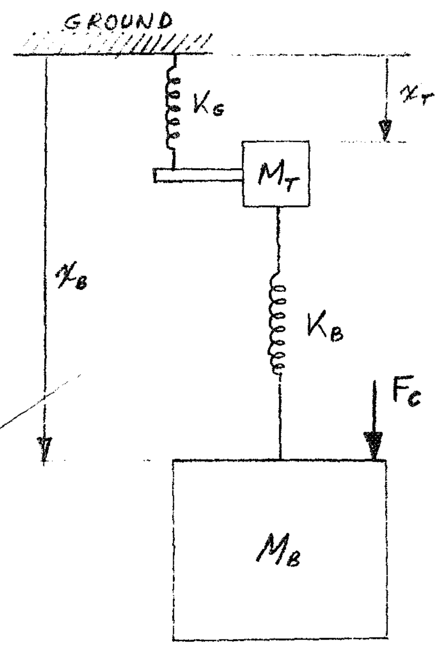


FIGURE II



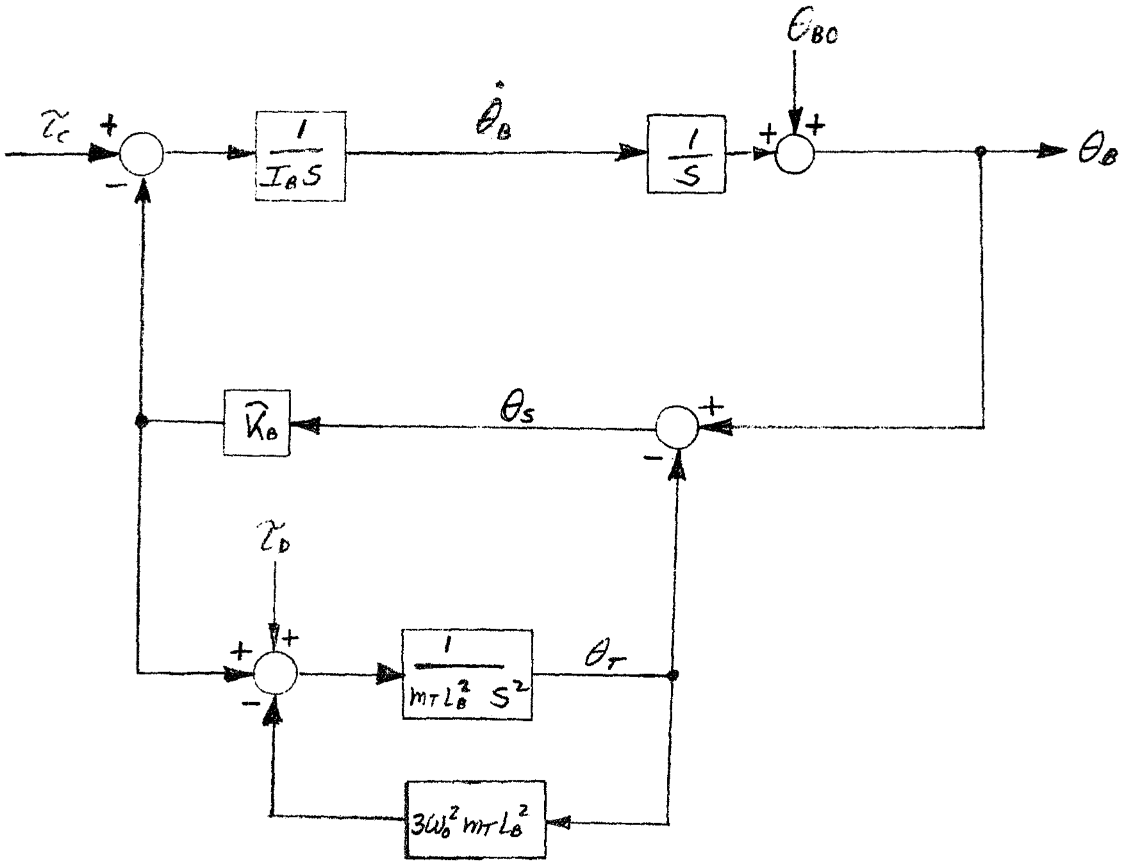
ANALOGY LIST

$M_B \longrightarrow I_B$	$K_B \longrightarrow R_B$
$M_T \longrightarrow m_T\, L_B^2$	$K_G \longrightarrow 3\omega_0^2\, m_T\, L_B^2$
$x_T \longrightarrow \theta_T$	GROUND $\longrightarrow$ LOCAL VERTICAL
$x_B \longrightarrow \theta_B$	$F_c \longrightarrow \tau_c$
$x_T - x_B \longrightarrow \theta_S$	

AVCO

FIGURE III

1 - 1691



LET:

$I_B = 5.5$ SLUG FT^2

$\hat{K}_B = .742$ FT#/RAD

$m_T L_B^2 = 588$ SLUG FT^2

$3\omega_0^2 m_T L_B^2 = .00229$ FT#/RAD

AVCO

FIGURE IV

$$\frac{\theta_s}{\tau_c} = \frac{.1818}{S^2 + (.370)^2}$$

Let $\tau_c = \frac{\tau}{S} (1 - e^{-\delta s})$

$$\theta_s = \frac{.1818\, \tau\, (1 - e^{-\delta s})}{S\, (S^2 + (.370)^2)}$$

$$\theta_s = \frac{.1818\, \tau}{(.370)^2} \left[1 - \cos .370t - 1 + \cos .370\, (t - \delta) \right]$$

$$\theta_s = \frac{.1818\, \tau}{(.370)^2} \left[\cos\, (.370 [t - \delta]) - \cos .370\, t \right]$$

$$\therefore \quad \theta_s = \frac{.1818}{(.370)^2} \left[b \sin\, (.370\, t + \psi) \right]$$

where $b = 2 \sin\left(\frac{.370\, \delta}{2}\right)$

$$\therefore \quad \theta_s = \pm\, 2.6559\, \tau \sin\, (.185\, \delta) \quad \text{(rad)}$$

But for the limit cycle chosen $\tau\delta = .0299$ ft # sec

$$\therefore \quad \theta_s = \pm \frac{.0794 \sin\, (.185\, \delta)}{\delta} \quad \text{(rad)}$$

$$\text{or} \quad \theta_s = \pm \frac{4.5494}{\delta} \sin\, (.185\, \delta) \quad \text{(degrees)}$$

$$\text{or} \quad \theta_{s_{max}} = \frac{4.5494}{\delta} = 152.15\, \tau \quad \text{(degrees)}$$

These calculations are based upon a .5 inch diameter, 60 ft long boom.

Figure V shows $\theta_{s_{max}}$ vs τ for .5 inch, 1. inch and 1.5 inch diameter booms.

For a maximum boom deflection of $.1^{o}$ and a thrust application radius of 1.5 ft., the thrust level vs boom diameter is shown on Figure VI.

The calculation indicates that for a 60 ft long, .5'' diameter boom with a 5 # tip mass, the allowable thrust level range would be .00013 to .00044 #. This indicates that the use of a .5 inch diameter boom is possible but marginal and that the use of a 1. inch diameter boom with thrust levels of .001 to .002 pounds would be preferable.

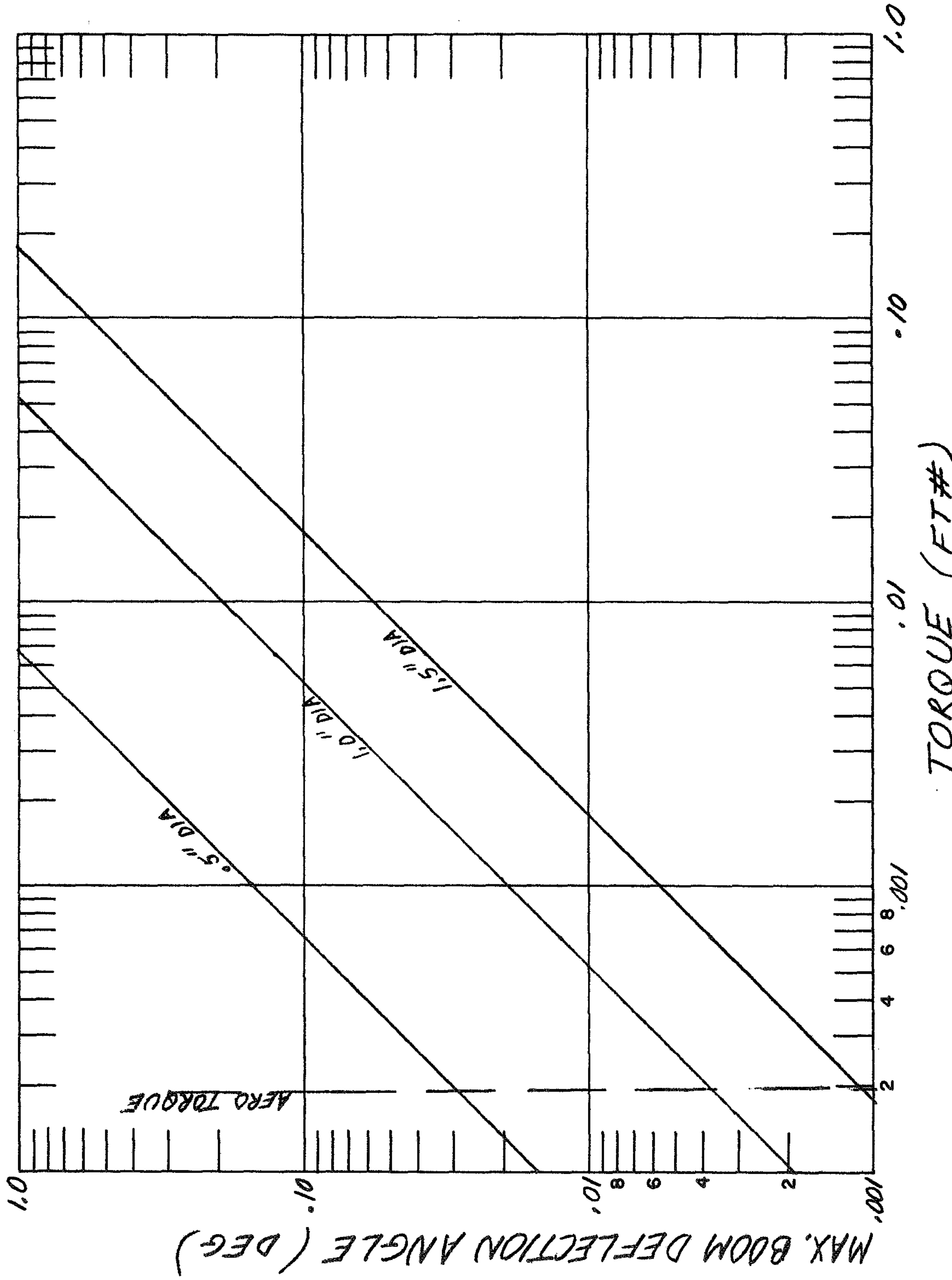

1.0
.10
.01
.001
8
6
4
2
TORQUE (FT#)
.10
.01
.001
MAX. BOOM DEFLECTION ANGLE (DEG)
1.5" DIA
1.0" DIA
.5" DIA
AERO TORQUE

FIGURE V

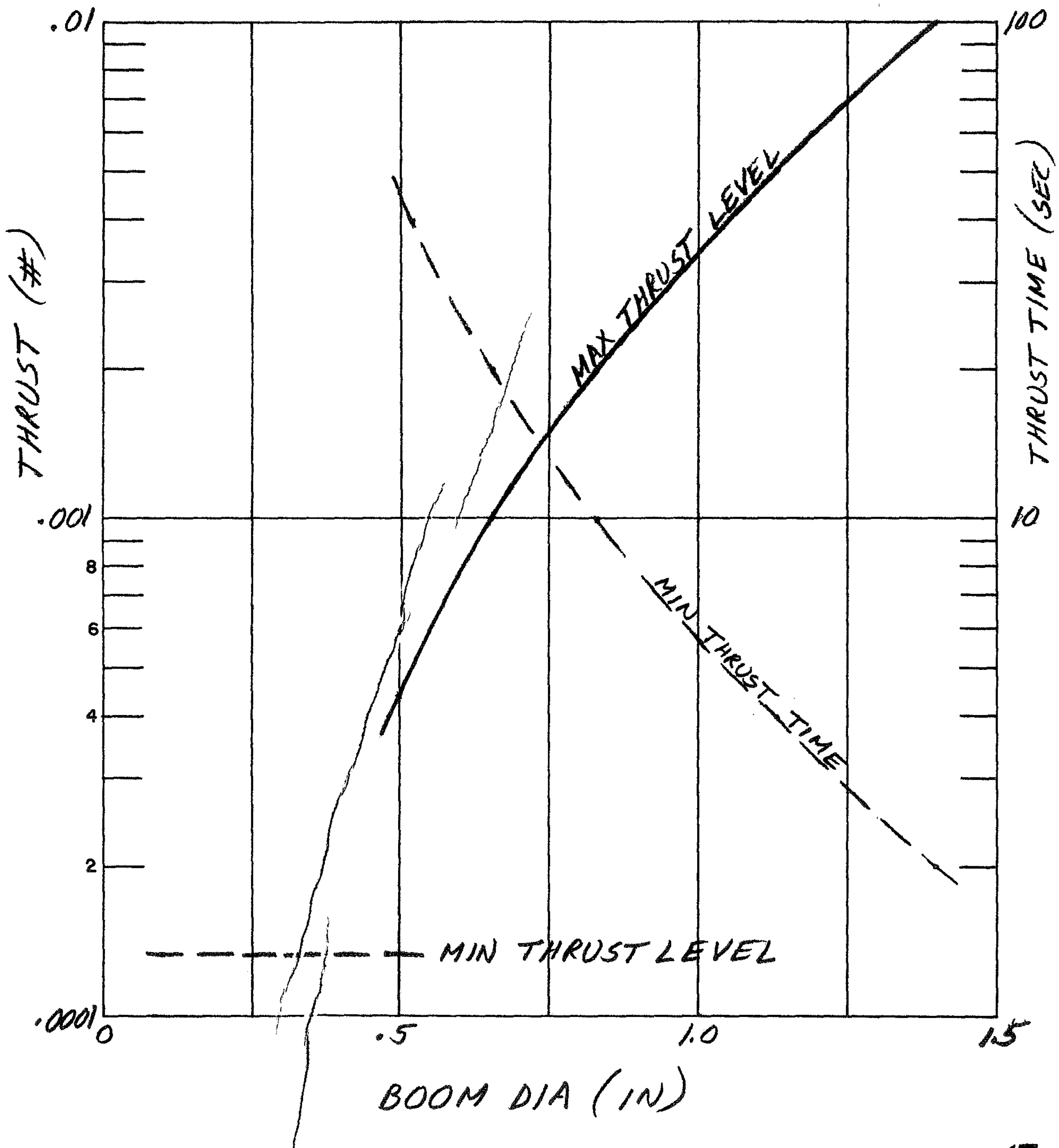

.01
.001
8
6
4
2
.0001
100
10
THRUST (#)
THRUST TIME (SEC)
MAX THRUST LEVEL
MIN THRUST TIME
MIN THRUST LEVEL
0
.5
1.0
1.5
BOOM DIA (IN)

FIGURE VI

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒ F220-EL-70-61

TO	DEPT.	FROM	DEPT.	DATE
J. E. Mozzicato	F220	E. Lawlor	F220	8/14/70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-F220-033		

SUBJECT

Stability of Gravitational Systems

DISTRIBUTION	SIGNED / APPROVED
S. Brzeski, K. Arnesen, D. Fields, W. Hailey, R. Litte, G. Kunkel, G. Pfeiffer, F. Scammell, M. Weinberger, Files	E. C. Lawlor Jr / J. E. Mozzicato

INFORMATION REQUESTED / RELEASED

The behavior of a gravitationally stabilized satellite can be studied for a restricted assumed state by the use of the relative Hamiltonian (see References 1 and 2). The analysis is limited by the following assumptions. The orbit is circular and the central body (i.e., the Earth) is assumed to be perfectly spherical. No perturbative torques (solar, aero or magnetic) are acting. Under these conditions, the relative Hamiltonian can be used to construct a Liapunov function.

The performance of a gravitationally stabilized satellite is measured in terms of the alignment of the principal inertia axes of the satellite with the local vertical reference frame. The local vertical reference frame is defined as follows. The three axis of the local vertical reference frame lies along the radius vector to the satellite, positive outward from the central body (i.e., the Earth). The two axis of the local vertical reference frame is normal to the orbit plane and has the direction of the vector R $\otimes$ V where R is the position vector to the satellite, and V is the velocity vector of the satellite. The one axis of the local vertical reference frame is chosen to complete a right handed rectangular cartesian coordinate frame, Figure 1. For a circular orbit, the inertial angular velocity of the local vertical reference frame, measured in the local vertical reference frame, is a constant vector Ω_o along the two axis of the local vertical reference frame.

The orientation of the principal inertia axes of the satellite with respect to the local vertical reference frame is defined in terms of an ordered set of rotations θ, α, ψ; pitch, roll, yaw; from the local vertical reference frame to the satellite fixed set of principal axes, Figure 2. A vector measured in the satellite fixed reference frame $\{y_{is}\}$ is transformed into a vector $\{y_{iL}\}$ measured in the local vertical reference frame by the transformation matrix $[A]_R$.

$$\{y_{iL}\} = [A]_R \{y_{is}\}$$

$$[A]_R = [A]_\theta [A]_\alpha [A]_\psi$$

$$[A]_R = \begin{bmatrix} c\theta & 0 & s\theta \\ 0 & 1 & 0 \\ -s\theta & 0 & c\theta \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & c\alpha & -s\alpha \\ 0 & s\alpha & c\alpha \end{bmatrix} \begin{bmatrix} c\psi & -s\psi & 0 \\ s\psi & c\psi & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

$$= \begin{bmatrix} c\theta & 0 & s\theta \\ 0 & 1 & 0 \\ -s\theta & 0 & c\theta \end{bmatrix} \begin{bmatrix} c\psi & -s\psi & 0 \\ c\alpha s\psi & c\alpha c\psi & -s\alpha \\ s\alpha s\psi & s\alpha c\psi & c\alpha \end{bmatrix}$$

$$= \begin{bmatrix} c\theta c\psi + s\theta s\alpha s\psi & -c\theta s\psi + s\theta s\alpha c\psi & s\theta c\alpha \\ c\alpha s\psi & c\alpha c\psi & -s\alpha \\ -s\theta c\psi + c\theta s\alpha s\psi & s\theta s\psi + c\theta s\alpha c\psi & c\theta c\alpha \end{bmatrix}$$

The elements of the transformation matrix are designated by a_{ij}.

The relative Hamiltonian is constructed from the relative kinetic energy and the relative potential energy which includes both gravitational and centripetal potentials, Reference 1. The relative kinetic energy is given by the following

$$2T_R = I_1\Omega_{1R}^2 + I_2\Omega_{2R}^2 + I_3\Omega_{3R}^2$$

where:

Ω_{iR} are the components of the angular velocity of the satellite relative to the local vertical reference frame, measured in the satellite fixed principal axes frame.

I_i are the principal mass moments of inertia of the satellite.

The relative potential energy of the satellite, adjusted to zero at the equilibrium point $\theta = \alpha = \psi = 0$ is obtained from Reference 3.

$$2V_R = 3\Omega_o^2\left[a_{31}^2(I_1 - I_3) + a_{32}^2(I_2 - I_3)\right] + \Omega_o^2\left[a_{21}^2(I_2 - I_1) + a_{23}^2(I_2 - I_3)\right]$$

The Hamiltonian is obtained from the sum of the kinetic and potential energies

$$2H_R = 2T_R + 2V_R$$

Since both $2T_R$ and $2V_R$ are zero at the origin; $\theta = \alpha = \psi = 0$, $\Omega_{iR} = 0$, this function is a candidate Liapunov function

$$L = L_T + L_V = \frac{2T_R}{I_2\Omega_o^2} + \frac{2V_R}{I_2\Omega_o^2}$$

$$L_T = \bar{I}_1\bar{\Omega}_{1R}^2 + \bar{\Omega}_{2R}^2 + \bar{I}_3\bar{\Omega}_{3R}^2$$

$$L_V = 3\left[a_{31}^2(\bar{I}_1 - \bar{I}_3) + a_{32}^2(1 - \bar{I}_3)\right] + \left[a_{21}^2(1 - \bar{I}_1) + a_{23}^2(1 - \bar{I}_3)\right]$$

$$\bar{I}_1 = \frac{I_1}{I_2} \qquad \bar{I}_3 = \frac{I_3}{I_2} \qquad \bar{\Omega}_{iR} = \frac{\Omega_{iR}}{\Omega_o}$$

L is positive definite if $\bar{I}_1 \leq 1$; $\bar{I} \geq \bar{I}_3$

For the simple case of a dumbell (i. e., $I_1 = I_2$) the potential term reduces to the following form

$$\bar{I}_1 = 1$$

$$L_V = (1 - \bar{I}_3)\left[3a_{31}^2 + 3a_{32}^2 + a_{23}^2\right]$$

Introducing the expressions for the elements of the transformation matrix, the potential expression becomes

$$L_V = 3(1-\bar{I}_3)\left[s^2\theta + \frac{4}{3}s^2\alpha - s^2\theta\, s^2\alpha\right]$$

The potential surface described by this function is shown in Figure 3 by plotting α vs θ for constant contours of $\bar{L}_V$ where $\bar{L}_V$ is defined as follows

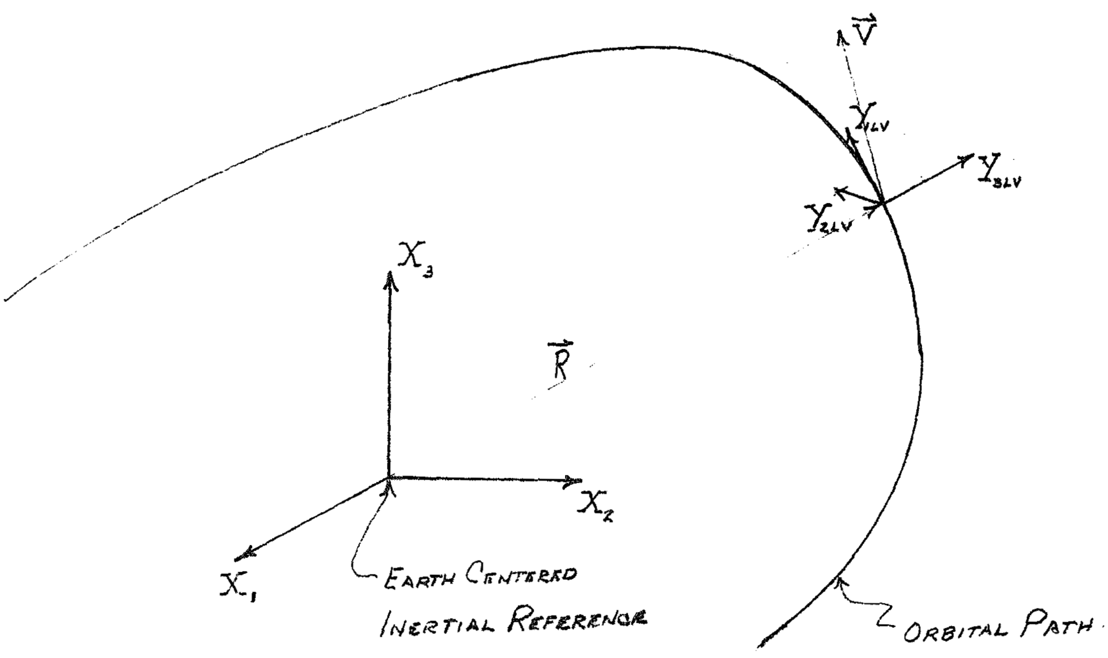
$$\bar{L}_V = \frac{L_V}{1-\bar{I}_3} = \frac{2V_R}{I_2\Omega_o^2(1-\bar{I}_3)} \qquad \text{(Figure 3)}$$

It should be noted that for the dumbell configuration, the yaw angle, ψ, is undefined by the potential function L_V.

The relative kinetic energy L_T for the dumbell configuration has the following form

$$L_T = \bar{\Omega}_{1R}^2 + \bar{\Omega}_{2R}^2 + \bar{I}_3\,\bar{\Omega}_{3R}^2$$

The magnitude of the yaw axis component of the relative angular velocity vector, Ω_{3R}, must be limited in order to guarantee the alignment of the satellite with the local vertical to within a specified tolerance. For instance, for $\bar{I}_3 = 0.01$ and requiring a quaranteed 1^o tolerance on pitch at the 5^o pitch energy surface, the yaw rate Ω_{3R} must be limited to orbital rate Ω_o.



$\vec{V}$
Y_{1LV}
Y_{3LV}
Y_{2LV}
X_3
$\vec{R}$
X_2
X_1
EARTH CENTERED
INERTIAL REFERENCE
ORBITAL PATH

FIGURE 1

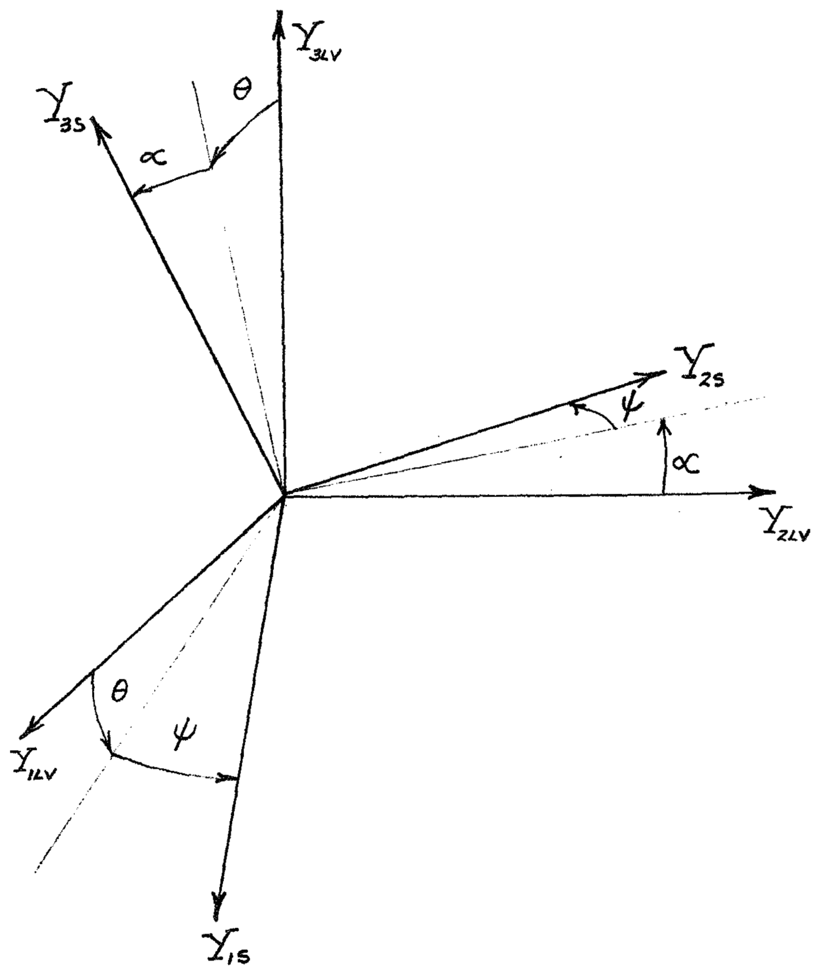
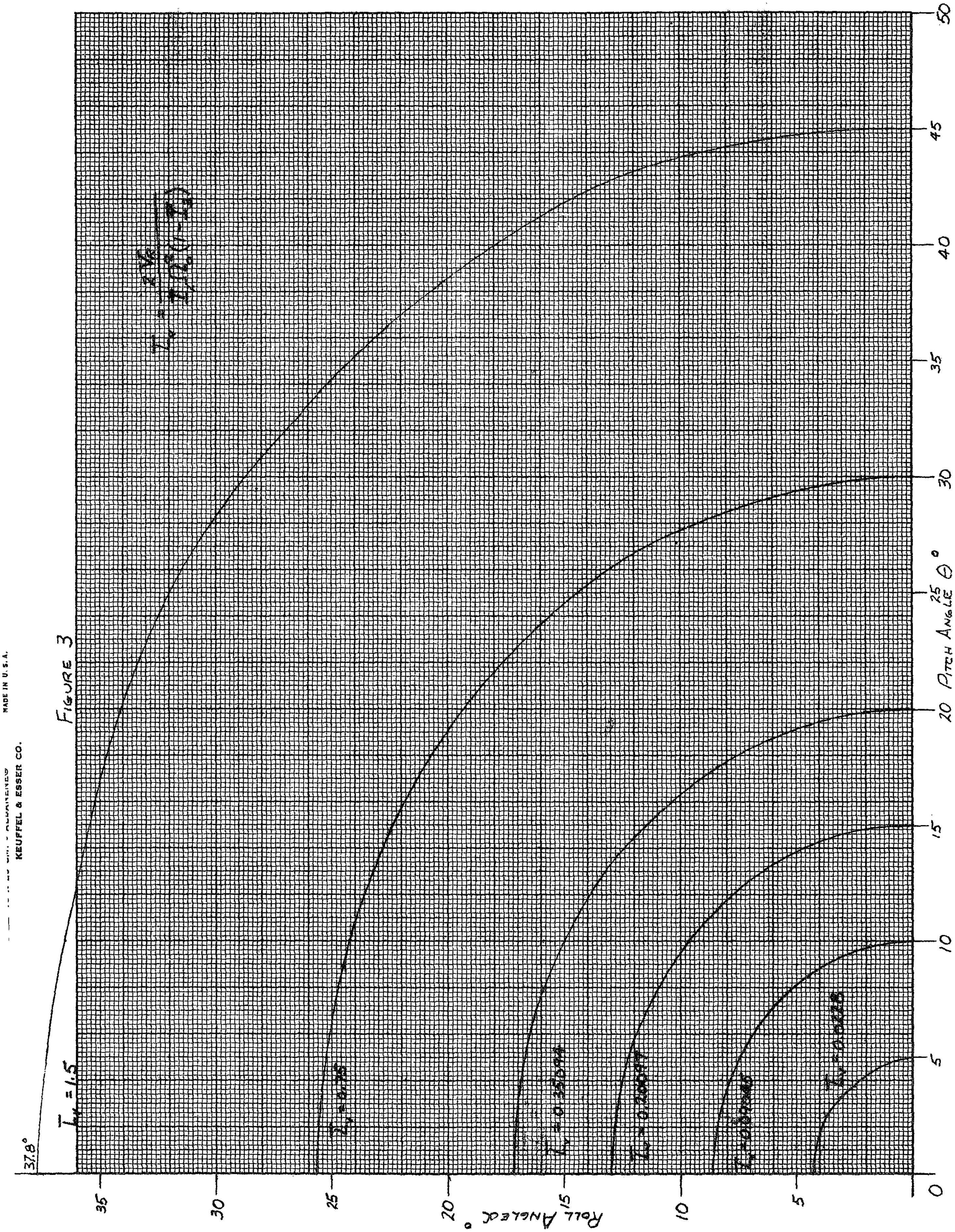


FIGURE 2


KEUFFEL & ESSER CO.
MADE IN U.S.A.
FIGURE 3
PITCH ANGLE
ROLL ANGLE
$I_v = 1.5$
$I_v = 0.75$
$I_v = 0.35594$
$I_v = 0.20097$
37.8°
0
5
10
15
20
25
30
35
40
45
50

Frequencies:

The "small" motion frequencies of the system are determined by writing the linearized set of equations of motion. For the case of a single boom along the yaw axis of the satellite, the following system of linear equations results (Reference 3).

$$I_1 = I_2 = I$$

ROLL EQUATION

$$I\dot{\Omega}_1 + \int_s (\ddot{y}_3 y_2 - y_3 \ddot{y}_2)\,dm + 3\Omega_0^2 (I - I_3)\alpha - \Omega_0^2 \int_s y_2 y_3\,dm - \Omega_2 \Omega_3 (I - I_3) = 0$$

PITCH EQUATION

$$I\dot{\Omega}_2 + \int_s (\ddot{y}_1 y_3 - y_1 \ddot{y}_3)\,dm + 3\Omega_0^2 (I - I_3)\theta = 0$$

YAW EQUATION

$$I_3 \dot{\Omega}_3 = 0$$

BOOM EQUATION q = GENERALISED COORDINATE (A OR B TIP DISPLACEMENTS)

$$\dot{\Omega}_1 \int_s \left(y_2 \frac{\partial y_3}{\partial q} - y_3 \frac{\partial y_2}{\partial q}\right) dm + \dot{\Omega}_2 \int_s \left(y_3 \frac{\partial y_1}{\partial q} - y_1 \frac{\partial y_3}{\partial q}\right) dm + \int_s \left(\ddot{y}_1 \frac{\partial y_1}{\partial q} + \ddot{y}_2 \frac{\partial y_2}{\partial q} + \ddot{y}_3 \frac{\partial y_3}{\partial q}\right) dm$$

$$- \Omega_2^2 \int_s \left(y_1 \frac{\partial y_1}{\partial q} + y_3 \frac{\partial y_3}{\partial q}\right) dm + \Omega_2 \Omega_3 \int_s \left(y_3 \frac{\partial y_2}{\partial q} + y_2 \frac{\partial y_3}{\partial q}\right) dm + \frac{\partial V_s}{\partial q}$$

$$= -\Omega_0^2 \int_s \left(y_1 \frac{\partial y_1}{\partial q} + y_2 \frac{\partial y_2}{\partial q} - 2 y_3 \frac{\partial y_3}{\partial q}\right) dm$$

$$- 3\Omega_0^2 \theta \int_s \left(y_1 \frac{\partial y_3}{\partial q} + y_3 \frac{\partial y_1}{\partial q}\right) dm + 3\Omega_0^2 \alpha \int_s \left(y_2 \frac{\partial y_3}{\partial q} + y_3 \frac{\partial y_2}{\partial q}\right) dm$$

ANGULAR VELOCITY	ANGULAR ACCELERATION (LINEAR)
$\Omega_1 = \Omega_{1R} + \Omega_0 c\alpha\, s\psi$	$\dot{\Omega}_1 = \dot{\Omega}_{1R} + \Omega_0 \dot{\psi}$
$\Omega_2 = \Omega_{2R} + \Omega_0 c\alpha\, c\psi$	$\dot{\Omega}_2 = \dot{\Omega}_{2R}$
$\Omega_3 = \Omega_{3R} - \Omega_0 s\alpha$	$\dot{\Omega}_3 = \dot{\Omega}_{3R} - \Omega_0 \dot{\alpha}$

INTEGRATING YAW EQUATION $\Omega_3 = 0 \quad \therefore \quad \dot{\psi} = \Omega_0 \alpha$

Rewriting this set of equations gives two pairs of coupled equations, one pair for roll motion and one pair for pitch motion. The position components for a boom along the 3 axis of the body frame can be written as follows.

$$Y_1 = -B\chi(z_1)$$
$$Y_2 = A\chi(z_1) \qquad A \text{ AND } B \text{ ARE BOOM TIP DISPLACEMENTS}$$
$$Y_3 = z_1 - \tfrac{1}{2}(A^2+B^2)Z_1(z_1)$$

LINEAR EQUATIONS OF MOTION

ROLL EQUATION
$$I\ddot{\alpha} - \ddot{A}\int_s z_1\chi(z_1)dm + \Omega_o^2(4I-3I_3)\alpha - \Omega_o^2\int_s z_1\chi(z_1)dm = 0$$

BOOM EQUATION $q = A$
$$-\ddot{\alpha}\int_s z_1\chi(z_1)dm + \ddot{A}\int_s \chi^2(z_1)dm + \Omega_o^2 A\int z_1 Z_1(z_1)dm - \Omega_o^2\alpha\int_s z_1\chi(z_1)dm + \frac{3.2EI_A}{\ell^3}A$$
$$= -\Omega_o^2 A\left[\int_s \chi^2(z_1)dm + 2\int z_1 Z_1(z_1)dm\right] + 3\Omega_o^2\alpha\int z_1\chi(z_1)dm$$

THE CHARACTERISTIC EQUATION FOR THIS PAIR OF EQUATIONS IS AS FOLLOWS.

$$\bar{\lambda}^4\left\{1 - \frac{\left[\int_s z_1\chi(z_1)dm\right]^2}{I\int_s\chi^2(z_1)dm}\right\}$$
$$-\bar{\lambda}^2\left\{1 + \frac{3\int_s z_1 Z_1(z_1)dm}{\int_s\chi^2(z_1)dm} + \frac{\omega_n^2}{\Omega_o^2} + (4-3\bar{I}_3) - 2\frac{\left[\int_s z_1\chi(z_1)dm\right]^2}{I\int_s\chi^2(z_1)dm}\right\}$$
$$+\left\{(4-3\bar{I}_3)\left[1 + 3\frac{\int_s z_1 Z_1(z_1)dm}{\int_s\chi^2(z_1)dm} + \frac{\omega_n^2}{\Omega_o^2}\right] - \frac{\left[\int_s z_1\chi(z_1)dm\right]^2}{I\int_s\chi^2(z_1)dm}\right\} = 0$$

WHERE $\bar{\lambda} = \dfrac{\omega}{\Omega_o}$

$$\omega_n^2 = \frac{3.2EI_A}{\ell^3\int_s\chi^2(z_1)dm}$$

PITCH EQUATION $\qquad I\ddot{\theta} - \ddot{B}\int_S z_1 \chi(z_1)dm + 3\Omega_o^2(I - I_3)\theta = 0$

BOOM EQUATION $\quad q = B \quad -\ddot{\theta}\int_S z_1\chi(z_1)dm + \ddot{B}\int \chi^2(z_1)dm - 3\Omega_o^2\theta\int_S z_1\chi(z_1)dm + 3\Omega_o^2 B\int_S z_1 {}'Z_1(z_1)dm + \frac{3.2EI_A}{\ell^3}B$
$= 0$

THE CHARACTERISTIC EQUATION FOR THIS PAIR OF EQUATIONS AS FOLLOWS.

$$\lambda^4\left\{1 - \frac{\left[\int_S z_1\chi(z_1)dm\right]^2}{I\int \chi^2(z_1)dm}\right\}$$

$$-\lambda^2\left\{3\frac{\int_S z_1 {}'Z_1(z_1)dm}{\int \chi^2(z_1)dm} + \frac{\omega_n^2}{\Omega_o^2} + 3(1-\bar{I}_3) - 3\frac{\left[\int z_1\chi(z_1)dm\right]^2}{I\int_S \chi^2(z_1)dm}\right\}$$

$$+\left\{3(1-\bar{I}_3)\left[3\frac{\int_S z_1 {}'Z_1(z_1)dm}{\int_S \chi^2(z_1)dm} + \frac{\omega_n^2}{\Omega_o^2}\right]\right\} = 0$$

NUMERICAL EXAMPLE

BOOM PROPERTIES $\quad \ell = 60$ FT. $\quad$ NOMINAL $\frac{1}{2}$ IN DIAMETER BOOM

MASS PER UNIT LENGTH $\rho = 0.0005$ SLUG/FT $\quad$ TIP MASS $= \frac{5}{32.2}$ SLUGS $= M_T$

BENDING RIGIDITY $\quad EI_A = 2000$ LB-IN^2

NATURAL FREQUENCY $\quad \omega_n^2 = 0.00126\ (\text{RAD}/\text{SEC})^2$

SYSTEM INERTIA PROPERTIES

$$I_1 = I_2 = I_{1C} + \tfrac{1}{3}\rho\ell^3 + M_T\ell^2 = 600 \text{ SLUG-FT}^2$$

$$I_3 = I_{1C} = 6 \text{ SLUG-FT}^2 \qquad \therefore\ \bar{I}_3 = 0.01$$

SHAPE FUNCTION MASS INTEGRALS

$$\int_S \chi^2(z_1)dm = \rho\ell(0.25679) + M_T = 0.162983 \text{ SLUGS}$$

$$\int_S z_1\chi(z_1)dm = \rho\ell^2(0.28889) + M_T\ell = 9.83677 \text{ SLUG-FT}$$

$$\int_S z_1 {}'Z_1(z_1)dm = \rho\ell(0.30123) + M_T(1.14286) = 0.1865 \text{ SLUG}$$

INTRODUCING THE NUMERICAL VALUES GIVES THE FOLLOWING SET OF FREQUENCIES FOR $\Omega_o = 0.00114$ RAD/SEC

(200 MILE ORBITAL ALTITUDE)

ROLL FREQUENCY $\omega_R = 1.99\,\Omega_o = 0.00226$ RAD/SEC

PITCH FREQUENCY $\omega_P = 1.721\,\Omega_o = 0.00196$ RAD/SEC

ROLL VIBRATION FREQUENCY $\omega_{RV} = 284\,\Omega_o = 0.323$ RAD/SEC

PITCH VIBRATION FREQUENCY $\omega_{RP} = 284\,\Omega_o = 0.323$ RAD/SEC

Temperature Distortion

For a relatively stiff boom (i. e., high natural frequency WRT orbital frequency) the affect of gravitational and centrifugal stiffening can be neglected.

FROM REFERENCE 3 OR 4 AT NORMAL INCIDENCE

$$\frac{3.2 E I_A}{l^3} A - E \alpha \Delta T_o \frac{\pi r^2 t}{2 l} \cdot \frac{4}{3} = 0$$

INTRODUCING $I_A = \pi r^3 t$ AND REWRITING

$$\frac{A}{l} = \frac{1}{4.8} \alpha \Delta T_o \frac{l}{r}$$

FOR $\alpha = 1 \times 10^{-5} / {}^{o}F$ AND A 60 FT BOOM LENGTH

$$\frac{A}{l} \sim 0.006 \Delta T_o$$

IF $\Delta T_o = 5^{o}F$ $\frac{A}{l} \sim 0.03$

Notation

A	=	Tip displacement l = boom length
ΔT_o	=	Temperature difference across the boom
r	=	Tube radius t = tube wall thickness
α	=	Coefficient of thermal expansion

Note $\Delta T_o = 5^{o}$ is the high end of temperature gradients for a 1/2 in diameter (t=0.002") boom with little attention paid to controlling temperature

References

1) "Hamiltonian Calculation & Viscous Libration Damper", AVSSD-0027-69-RR Contract NAS 5-9461-10, Feb., 1969

2) Bainum, P. M., "On the Motion and Stability of a Multiple Connected Gravity-Gradient Satellite with Passive Damping", TG-872, January 1967, Johns Hopkins University

3) "Interim Report for the Investigation of the Dynamic Characteristics of a V Antenna for the RAE Satellite", Final Report Phases B & C, AVSSD-0103-67-RR, Contract No. NAS 5-9179, Octob er 1966.

4) "Users Manual for IMP Dynamics Computer Program", Volume 1, Contract NAS 5-11149, June 1970

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒ F220-JEM-70-60

TO	DEPT.	FROM	DEPT.	DATE
S. Brzeski	L910	J. E. Mozzicato	F220	8/19/70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-F220-033		page 4

SUBJECT: Sizing of Nutation Damper and Yo-Yo Despin Device

DISTRIBUTION: K. Arnesen, D. Fields, W. C. Hailey, E. J. Lawlor, R. Litte, G. Kunkel, G. Pfeiffer, F. Scammell, M. Weinberger, Files

SIGNED:

APPROVED: F. H. Scammell

INFORMATION REQUESTED / RELEASED

This technical release presents calculations of the size and weight of a Nutation Damper and of a Yo-Yo despin device for the vehicle under study in the "Attitude Control for Small Satellites and Related Systems" program.

Conclusions

The previous weight estimate of 2 pounds for a Yo-Yo and one pound for a nutation damper can be easily achieved.

I. Viscous Fluid Ring Damper Design

The ring damper configuration (Figure I) consists of a metal toroid filled with a viscous fluid.

The following dimensions will be used for this calculation:

b = toroid radius (ft)
a = tube radius (ft)
t = wall thickness (ft)
ρ_F = fluid density (#/ft^3)
ρ_M = metal density (#/ft^3)
M_F = fluid mass (slugs)
M_M = metal mass (slugs)
I_s = satellite spin inertia (slug ft^2)
I_T = satellite transverse inertia (slug ft^2)
$\lambda = \dfrac{I_s}{I_T} - 1$
ω_s = satellite spin rate (rad/sec)
v = fluid kinematic viscosity (ft^2/sec)
τ = damper time constant (sec)

It can be shown that (Ref. 1):

$$\tau = \frac{1}{\left(\frac{M_F b^2}{I_s}\right)\left(1+\lambda\right)^2 \omega_s \Gamma} \quad \text{(sec)}$$

where :

$$\Gamma = f\left(\frac{a^2 \lambda \omega_s}{v}\right)$$

$$M_F = \rho_F \, 2\pi^2 \, b \, a^2/32.174 \text{ (slugs)}$$

$$M_M = \rho_M \, 2\pi^2 \, a \, b \, t/32.174 \text{ (slugs)}$$

$$W = \text{Total Weight (\#)} = 2\pi^2 a \, b \, (\rho_m t + \rho_F a)$$

$$D = \text{Max. Dia. (in)} = 24 (a + b + t)$$

If we assume that the fluid has the density of water and any desired viscosity, we can set

$$\frac{a^2 \lambda \omega_s}{v} = .63.$$

This yields the maximum value of Γ ($\Gamma = .19$).

If we also assume the tube is aluminum with a wall thickness of .084 (inches), that ω_s = 18 rad/sec, $\lambda + 1 \approx 1.0$, and I_s = 6 slug ft^2, we can establish the time constant (τ) as a function of total damper weight (W) and max. diameter (D) (see Figure II).

If we establish the design criteria, the coning angle must be damped to 2 percent of its initial value within 1/4 of an orbit at a 200 nm altitude. This yields a time constant requirement of 5.8 minutes. Figure II shows that this can be accomplished with a 1 pound damper which is 13 inches in diameter.

II. Yo-Yo Despin Device

The formulas for the Yo-Yo despin device will not be developed here but can be found in Reference 2 and several other places.

The formulas of interest are:

$$L = R\sqrt{C\,\frac{\omega_0 - \omega_f}{\omega_0 + \omega_f}}$$

$$C = \frac{I}{mR^2} + 1$$

$$\omega = \omega_0\left(\frac{C - \omega_0^2 t^2}{C + \omega_0^2 t^2}\right)$$

where:

- L = length of the cord (ft)
- R = radius of cord winding (ft)
- ω_0 = initial spin rate (rad/sec)
- ω_f = final spin rate (rad/sec)
- I = vehicle inertia (slug ft^2)
- m = total Yo-Yo mass (slugs)
- ω = spin rate (rad/sec)
- t = time (sec)

If $\omega_f = 0$

$$L = \sqrt{\frac{I}{m} + R^2}$$

Figure III shows m vs. L for I = 6 slug ft^2 and R = 1.25 ft.

As a baseline design, we can select two weights of 1 pound each on cords 9.9 ft. long.

By differentiating ω we can obtain a formula for the tensile force in each cord.

$$F = \frac{2\,C\,I\,\omega_o^{3}\,t}{R\,(C + \omega_o^{2}\,t^{2})^{2}} \quad \text{(POUNDS/CORD)}$$

Figure IV shows the tensile force vs. time and also shows that the despin requires only .44 sec.

<u>References</u>

1. G&CD-Z350-176, "Analysis of Viscous Fluid Ring Damper", by H. Dudler, dated 9/18/67.

2. "Guidance and Control of Aerospace Vehicles", edited by Cornelius T. Leondes, McGraw-Hill, 1963.

NUTATION DAMPER

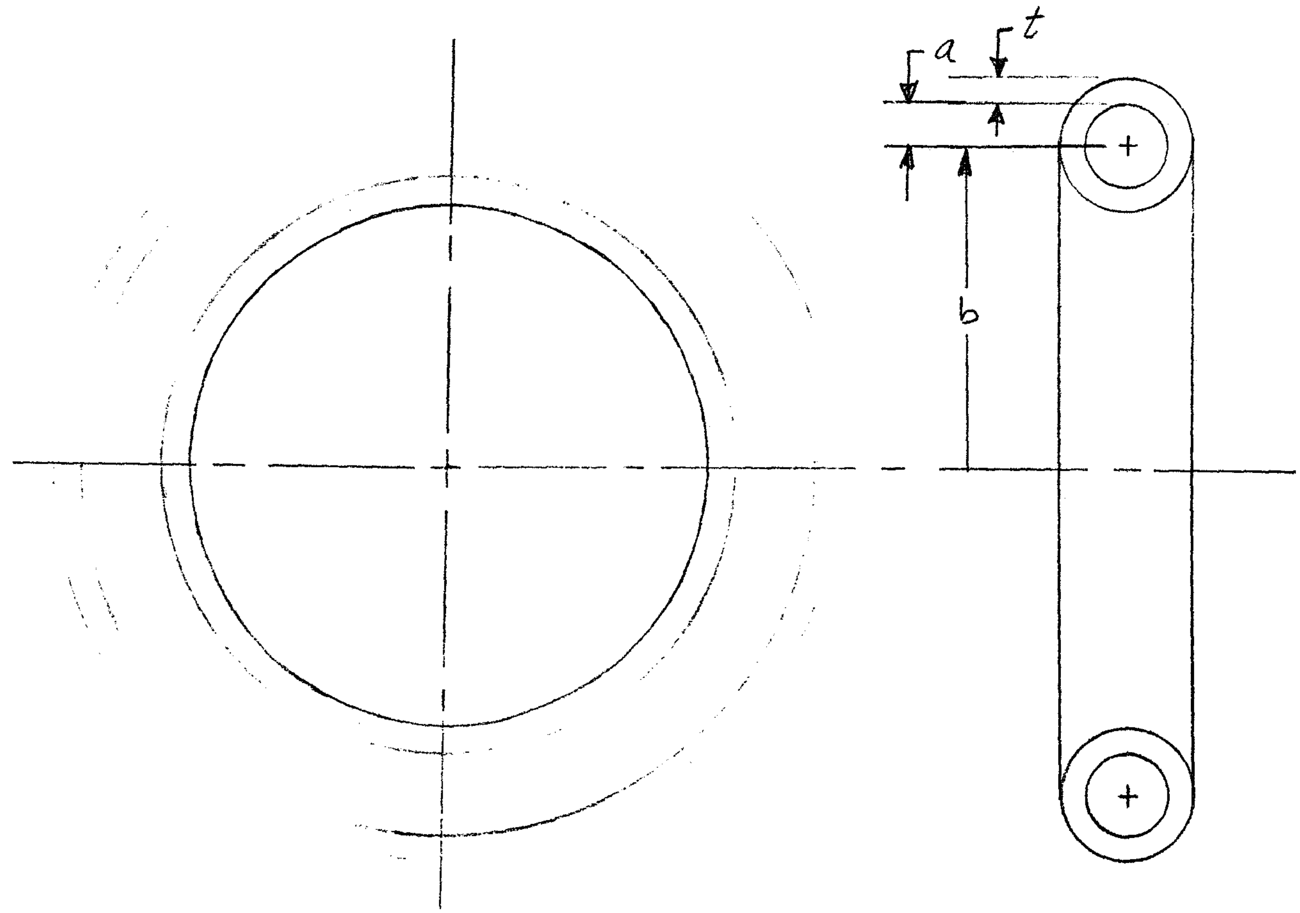


FIG I.

DAMPER TIME CONST VS WEIGHT & DIA.

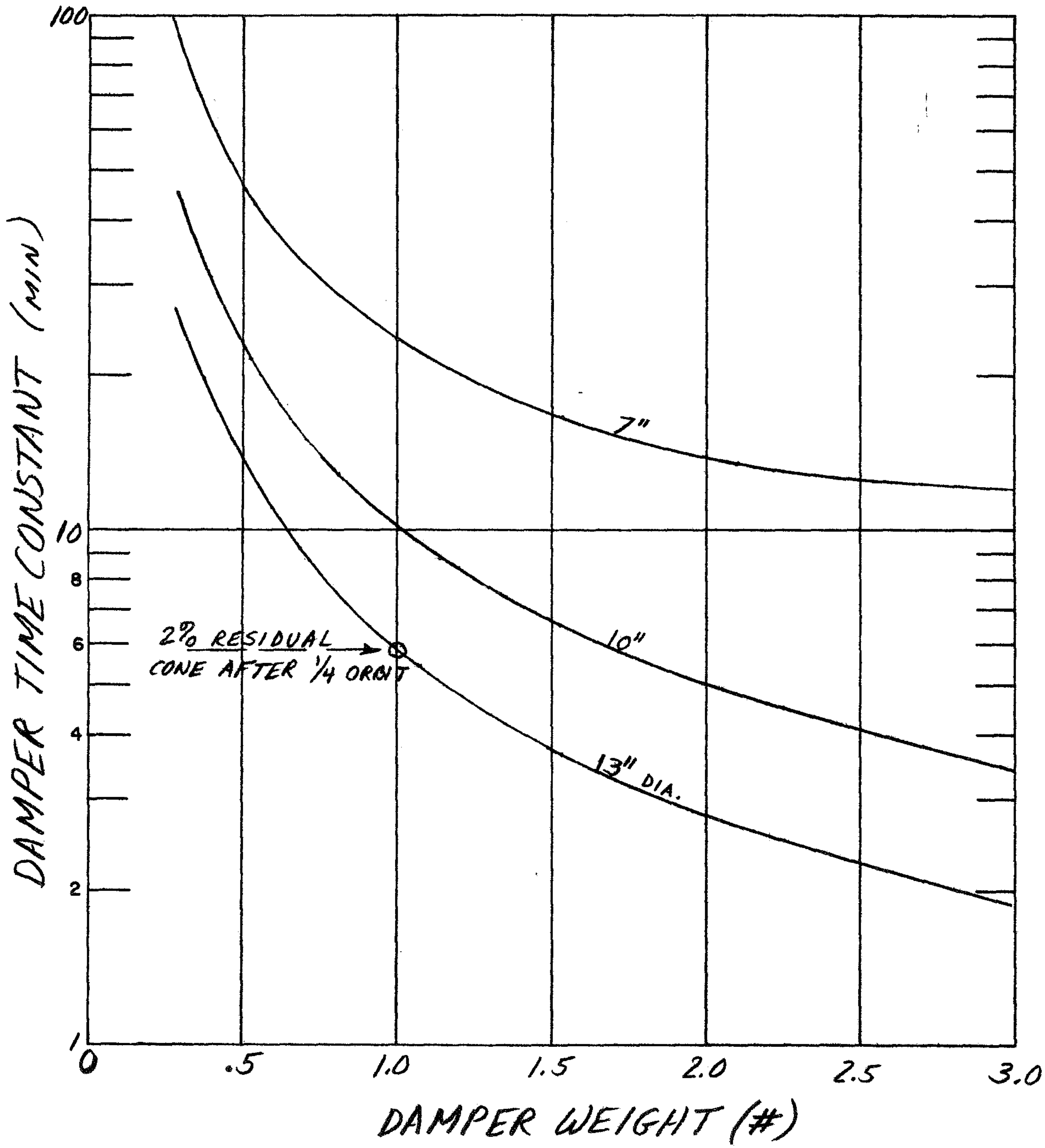


FIG. II

YO-YO LENGTH VS. MASS

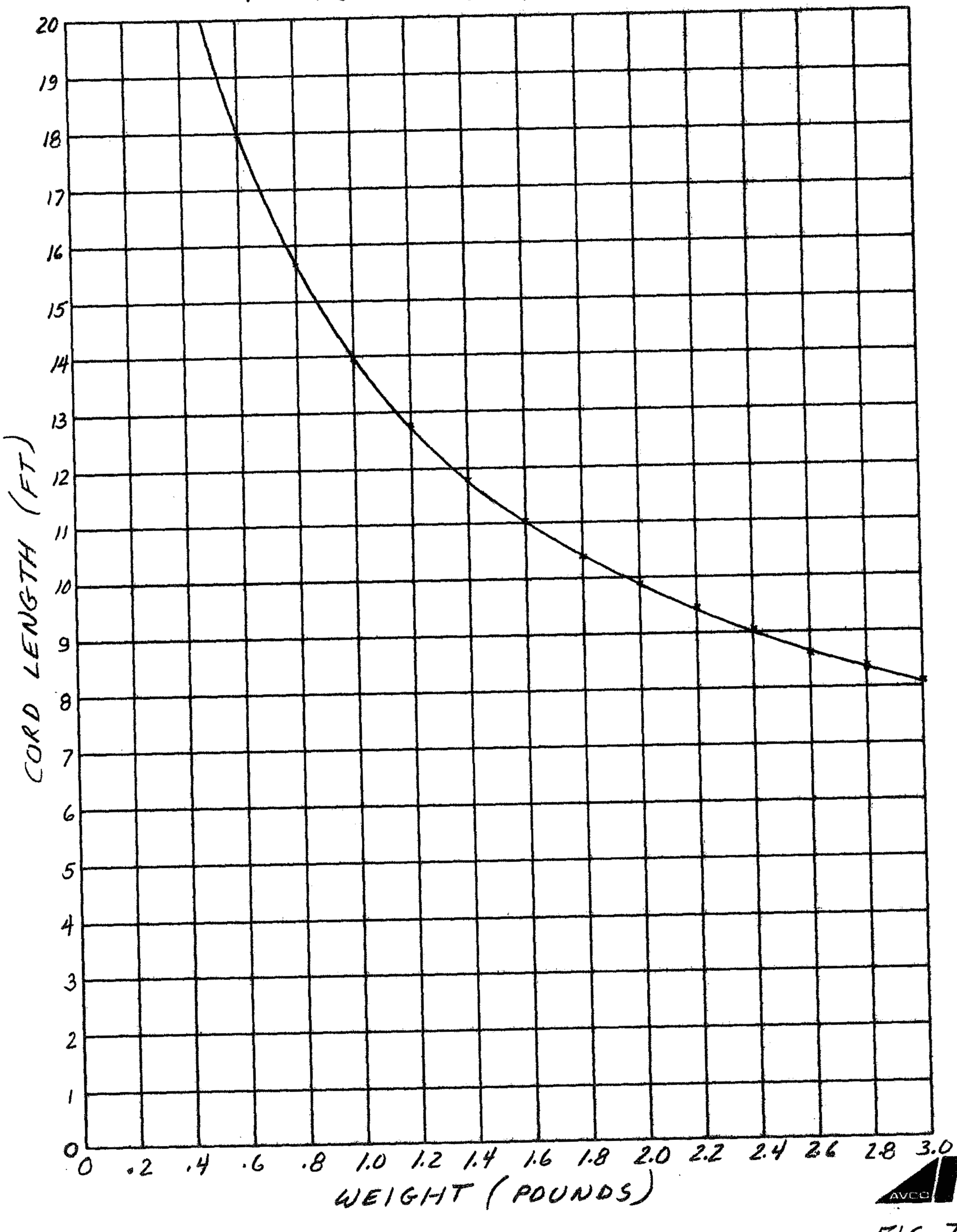


FIG. III

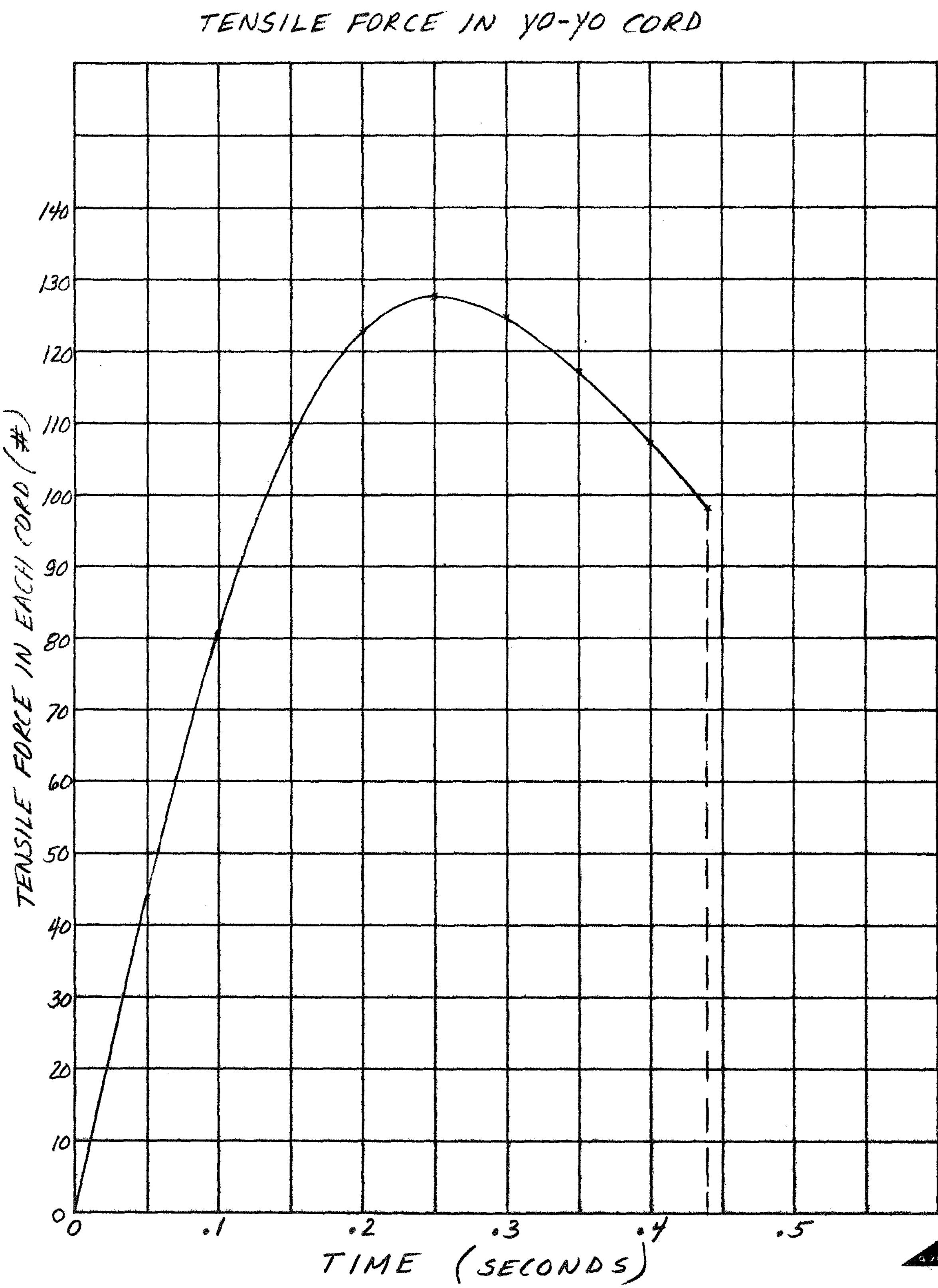

TENSILE FORCE IN YO-YO CORD
TENSILE FORCE IN EACH CORD (#)
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
0
.1
.2
.3
.4
.5
TIME (SECONDS)

FIG. IV

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X]

F200-RL-70-115

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	R. Litte	F200	28 Aug 70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control of Small Satellites and Related Subsystems	W159-F200-031	28 Aug 70	

SUBJECT

Attitude Determination Subsystem Analyses

DISTRIBUTION	SIGNED / APPROVED
List B	R. Litte / J. E. M.
	James E. Moggie

INFORMATION REQUESTED / RELEASED

This Technical Release presents the component and sensor investigations as background information utilized in the selection of hardware for the implementation of a recommended attitude determination subsystem for the subject program.

1. Ion Sensors

The ion attitude sensor indicates attitude by measuring the ion currents intercepted by apertures moving relative to the medium. Thus, an aperture of Area A moving with a velocity W through the ionosphere at an angle B with respect to the normal of the aperture, will intercept a current due to ions of

$$I = N_o \, A \, Q \, W \cos B \left[\frac{1}{2} + \frac{\operatorname{erf} \gamma}{2} + \frac{e^{-\gamma^2}}{2\gamma\sqrt{\pi}} \right]$$

N_o = total number of velocity points to be considered in some volume

A = aperture area

Q = is the ionic electrical charge

γ is given by: $\dfrac{2W \cos B}{V \sqrt{\pi}}$

V is the average particle velocity

$$V = \left(\frac{8 \, k \, T}{\pi \, m} \right)^{\frac{1}{2}}$$

k is Boltzmann's constant = 1.38×10^{-23} Joules $\text{mol}^{-1}\ {}^{o}\text{K}^{-1}$

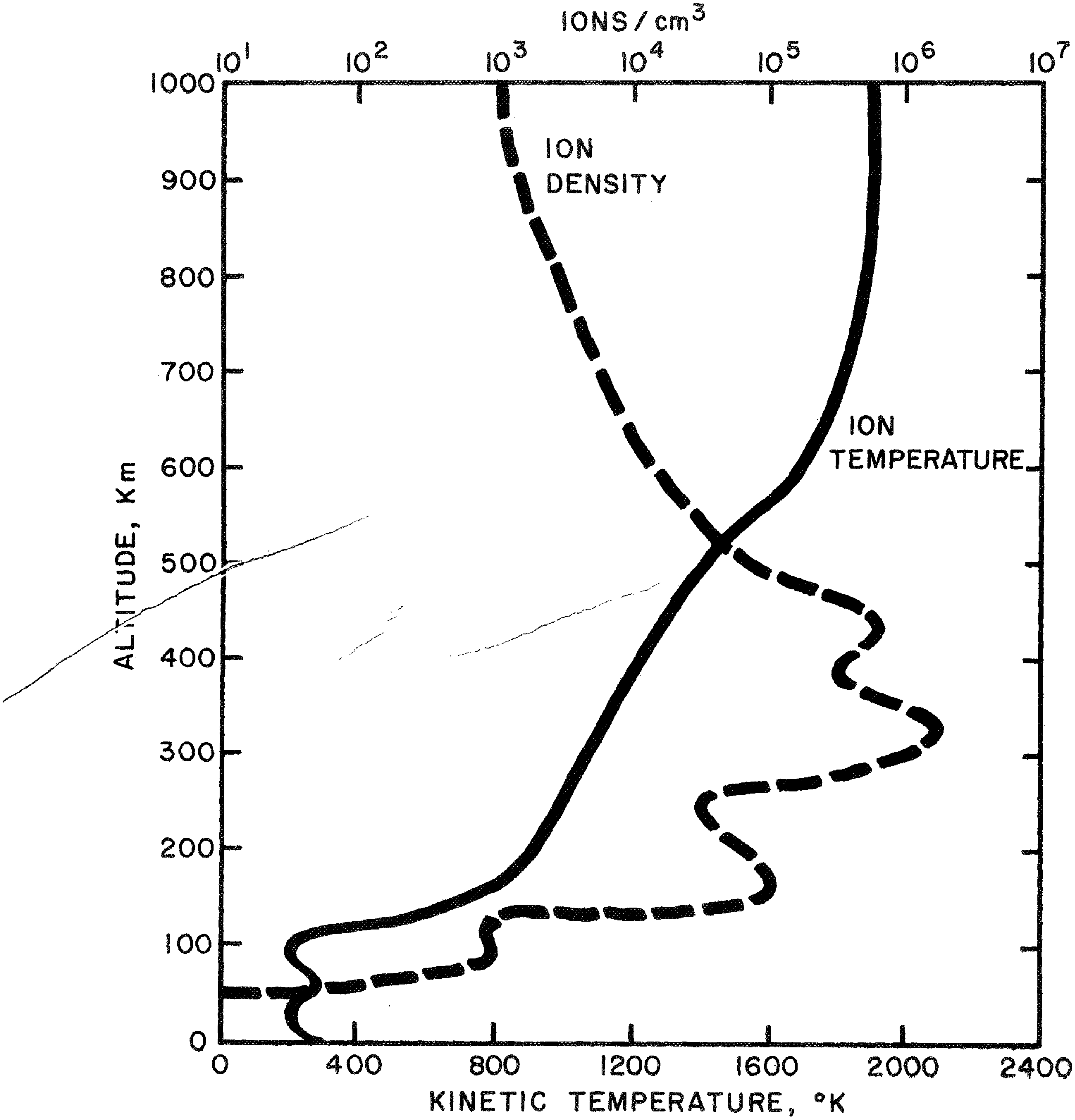
T is the absolute temperature, ^{o}K

m is the particle mass

The ion temperature and density as a function of altitude is shown in Figure 1 while the latitude dependence is shown in Figure 2.

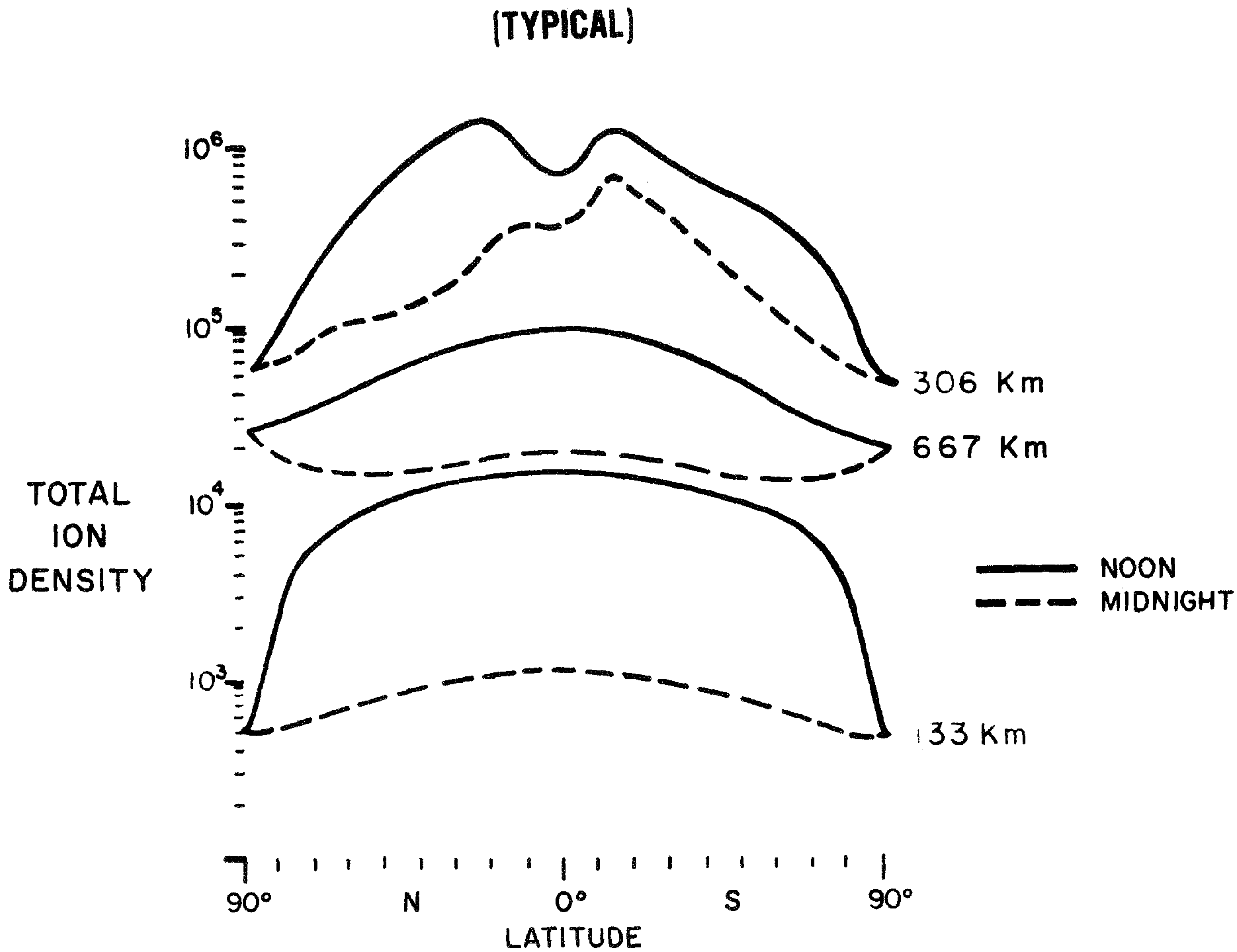
For small angles of B and since the ratio of vehicle velocity to the average particle velocity is greater than four the current is given closely by

$$I = N_o \, A \, Q \, V \cos B$$



UPPER ATMOSPHERIC REGIONS

Figure 1.



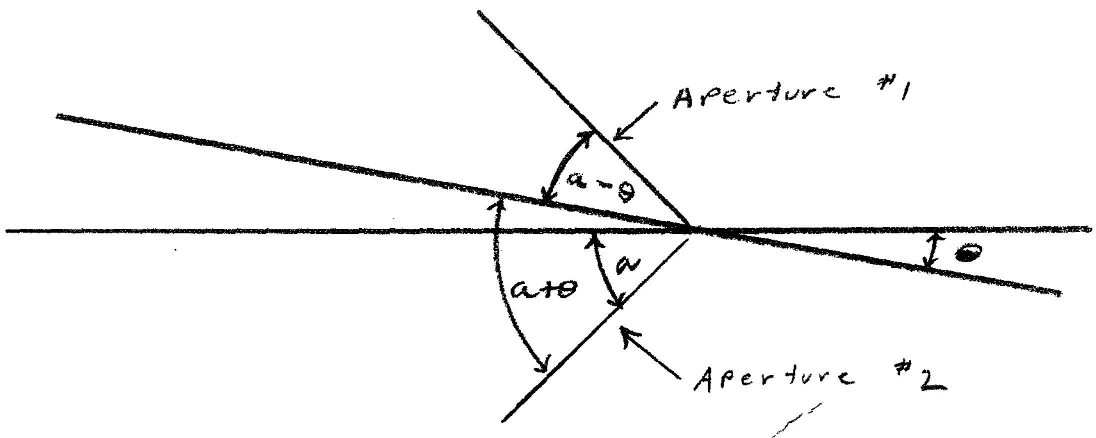
CONSTANT ALTITUDE IONIZATION VARIATIONS

Figure 2.

For a null indication it would be sufficient to simply use the algebraic difference between two apertures each at some angle a with respect to the velocity vector.

$$I_1 = N_o \, A \, Q \, W \, \mathrm{Sin} \, (a - \theta)$$

$$I_2 = N_o \, A \, Q \, W \, \mathrm{Sin} \, (a + \theta)$$



The gain dependence of altitude can be taken into account by taking the ratio of the current differences to the current sum.

It can be shown that $\dfrac{\pm I_1 + I_2}{I_1 + I_2} = \dfrac{\tan \theta}{\tan a}$

and since term a is a constant

$$\theta = \tan^{-1} k \, \frac{\pm I_1 + I_2}{I_1 + I_2}$$

The tangent function is very nearly linear for small angles; therefore, the output signal is very nearly proportional to the attitude angle throughout a range that is certainly great enough for attitude control functions. See Figure 3.

1 - 1690

Ion Attitude Sensor

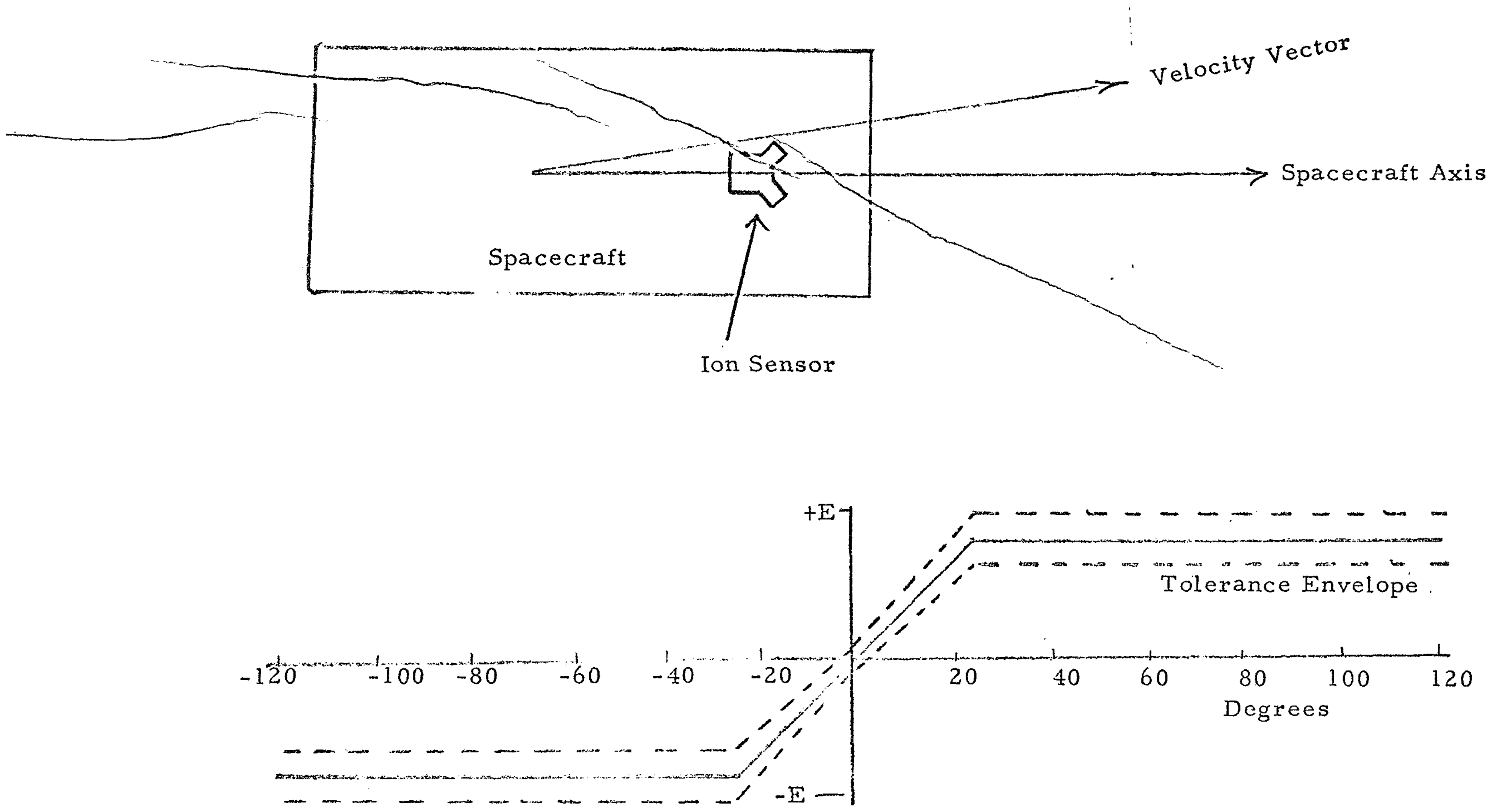


Figure 3.

The ionosphere rotates at the same angular velocity as the Earth. Therefore, a vehicle in orbit will experience a cross wind dependent on the orbital inclination as well as latitude and altitude.

The indicated yaw angle induced by the cross wind can be shown to be

$$\alpha = \tan^{-1} \frac{w\,(r+h)\,\cos\Phi\,\sin\Omega}{W - w(r+h)\,\cos\Phi\,\cos\Omega}$$

where

w is the angular velocity of the Earth

r is the radius of the Earth

h is the altitude

Φ is the latitude

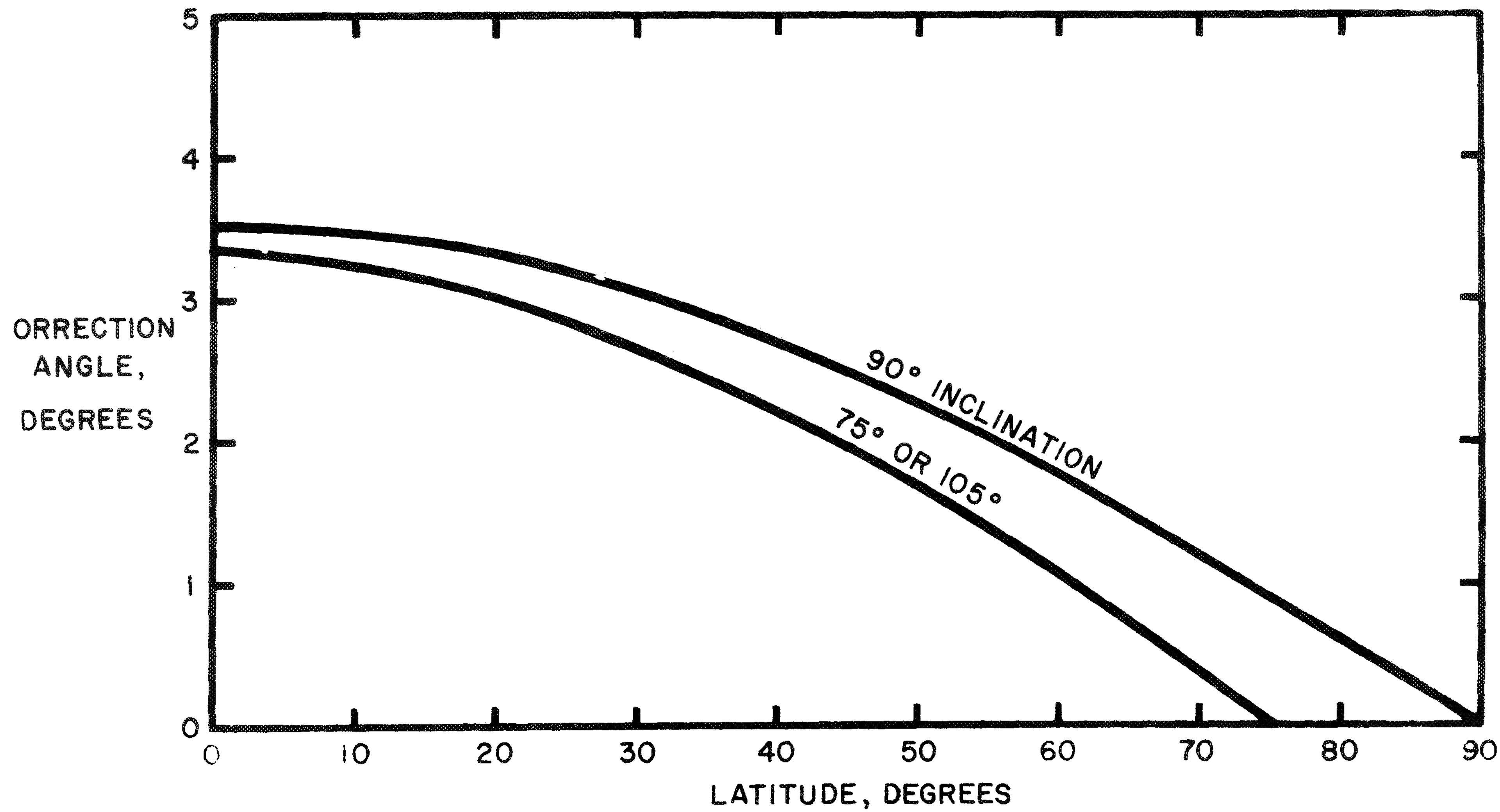
Ω is the orbital inclination

W is the tangential velocity in the non-rotating inertial system

Figure 4. indicates the resultant indicated yaw due to such a cross wind.

As data is compiled from the various spacecraft experiments from past, present and future, knowledge of the ionosphere is growing. It is reasonable to expect that the ionosphere model will get more complex but more precise thereby permitting more accurate ion attitude determination to be performed. This has occurred in the case of the terrestrial magnetic field as well as in the case of the infra red horizon where attitude determination by these means has improved because of improved models.

$$a = \mathrm{Tan}^{-1}\left[\frac{\omega(r+h)\,\mathrm{Cos}\,\phi\,\mathrm{Sin}\,\Omega}{W-\omega(r+h)\,\mathrm{Cos}\,\phi\,\mathrm{Cos}\,\Omega}\right] \qquad (10)$$



ATMOSPHERIC - WIND CORRECTION FACTOR

Figure 4.

Several prototype feasibility models of ion sensors have been flown on Air Force satellites to verify the concept of ionic attitude sensing. Some data from experiment # 10 of the Gemini series is shown in Figure 5. , where the ion sensor performance is compared with the on-board inertial information. This is a direct comparison of the inertial measurement unit and the ion sensor with no wind or dynamic correction factors applied.

Figure 6 illustrates the basic functional aspects of the ion sensor.

The pertinent characteristics of an ion sensor developed by Avco are tabulated in Table I.

TABLE I

Ion Sensor Characteristics

Linear Range	$\pm 25^{\circ}$
Saturated Range	$\pm 120^{\circ}$
Scale Factor	.25 volts degree^{-1}
Performance	$\pm 1^{\circ}$ to $\pm 3^{\circ}$
	$\pm 5^{\circ}$ to $\pm 25^{\circ}$
Response Time	0.1 seconds
Power	3 watts
Input Voltage	28 volts
Weight	5 lbs.
Temperature	-20°F to 160°F
Shock	200 g
Vibration	60 g
MTB	25, 000 hours

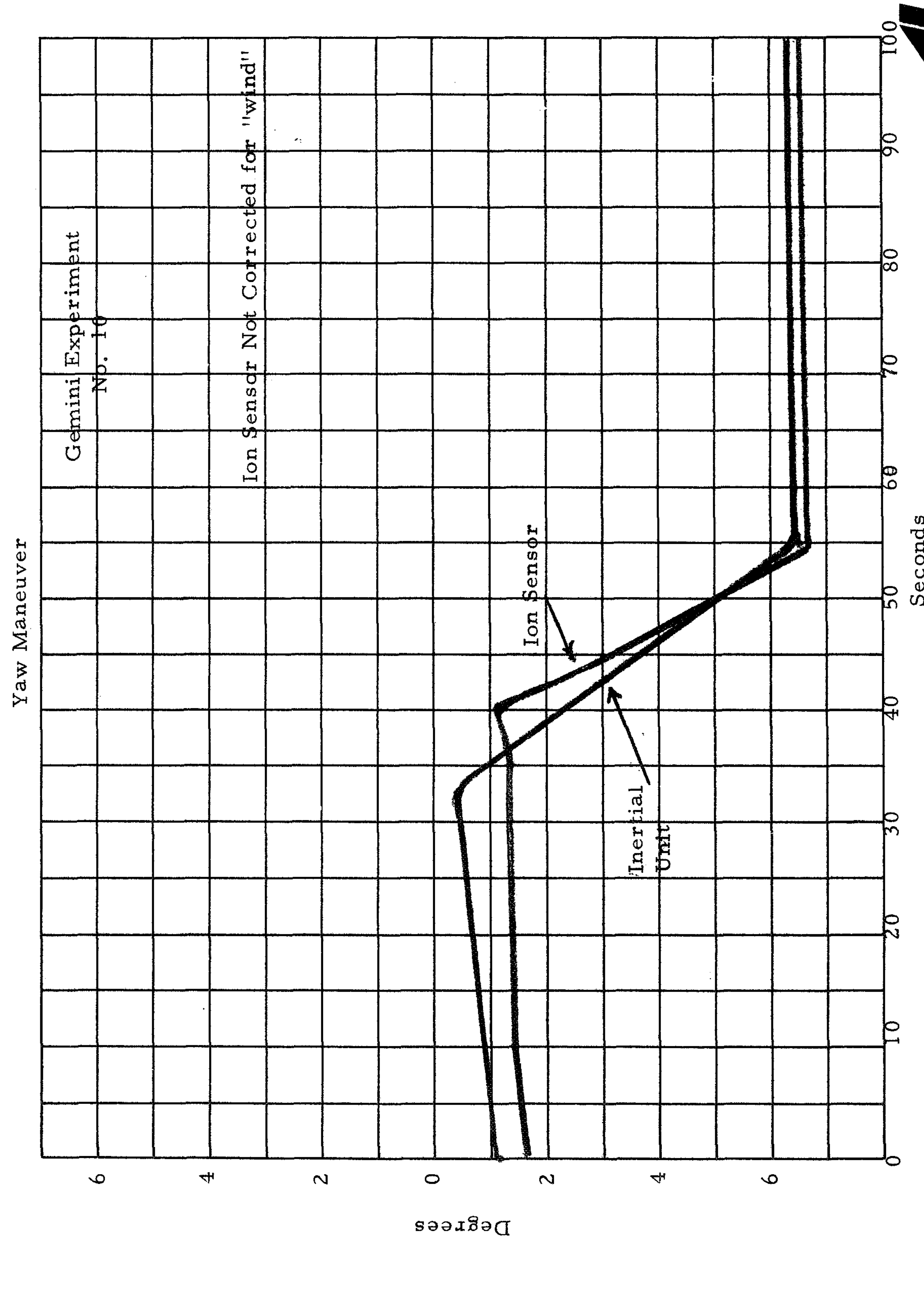

Yaw Maneuver
Gemini Experiment
No. 10
Ion Sensor Not Corrected for "wind"
Ion Sensor
Inertial
Unit
Degrees
6
4
2
0
2
4
6
Seconds
0
10
20
30
40
50
60
70
80
90
100

Figure 5.

SENSOR FUNCTIONAL DIAGRAM

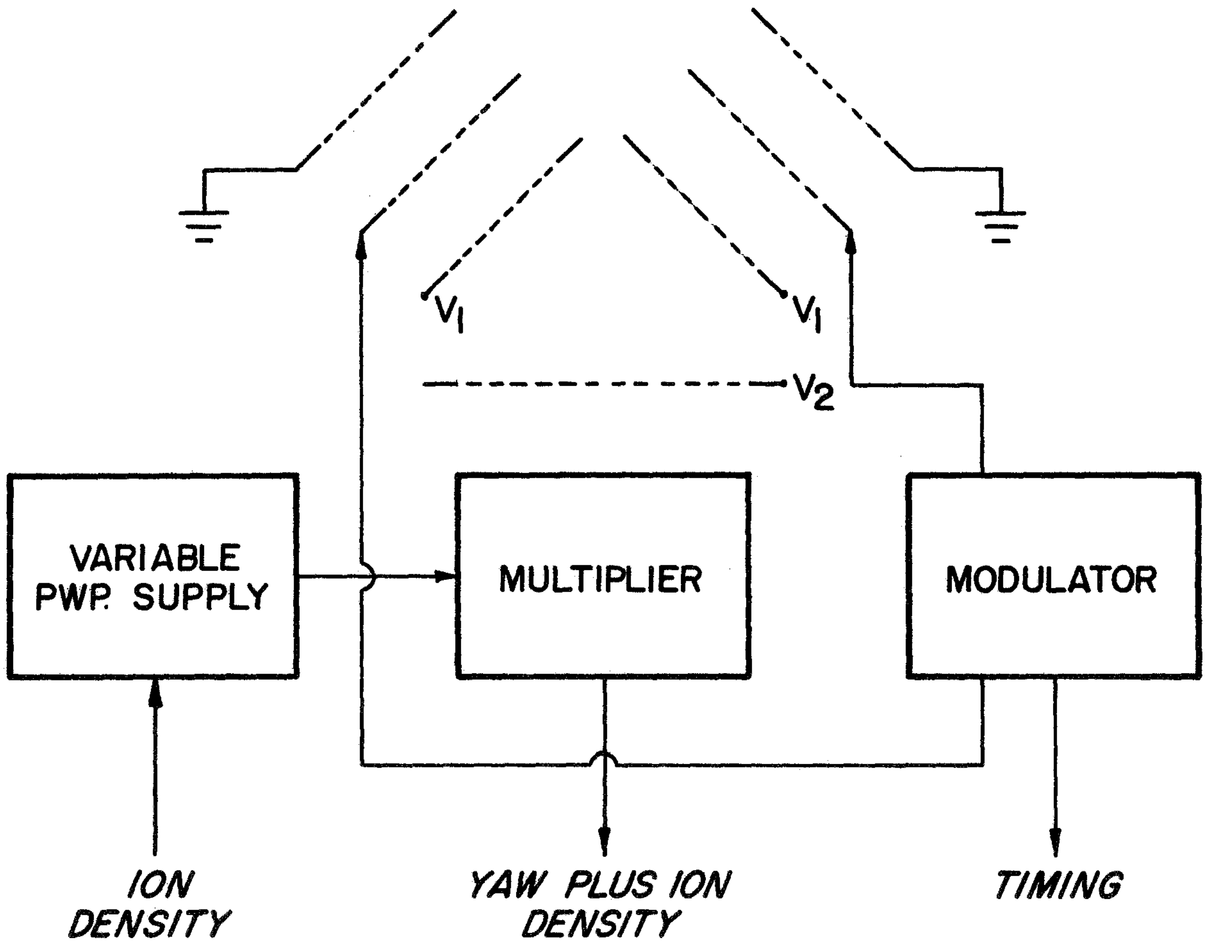


Figure 6.

2. Horizon Sensors

The purpose of a horizon sensor is to determine the direction of the local vertical. This is accomplished indirectly by locating horizon-space intercepts then deducing the intersection of two orthogonal diameters defined by these intercepts.

Horizon sensors operate more effectively in the infrared portion of the spectrum than they do with visual radiation. In the infrared thermally emitted radiance from the Earth is sensed and there is little contribution due to solar reflection. This permits the sensor to operate on a dark horizon as well as in the daylight. The development of the IR horizon sensor has led to operation in a narrow spectral band centered at 15 microns. In this spectral region the atmosphere is opaque; therefore, the high temperature contrasts as between a hot desert and the polar regions for example are minimized. The sensor therefore detects the discontinuity between the upper atmosphere and space rather than at the true horizon. Table II illustrates the above points with some quantitative examples.

TABLE II

Black Body Temperature $^{\circ}$Kelvin	Radiant Emittance Watts cm^{-2}		Solar Reflection watts cm^{-2} 14 - 16u
	14-60u	.4-1.4u	
300	$2x10^{-3}$	10^{-17}	$2x10^{-5}$
250	$1x10^{-3}$	10^{-20}	$2x10^{-5}$

It can be seen from Table II that in the visible part of the spectrum, the radiation is dominated by solar reflection with a completely negligible contribution from thermal emission. On the other hand, in the 14-16 micron region reflected sunlight contributes little. Further, the solar reflection depends on the reflection coefficient which varies significantly while the thermal emission is relatively constant in the upper atmosphere.

Horizon sensors may be classified into three types, each having its own advantage. The three types are: Radiation Balance, Edge Tracking and Scanner.

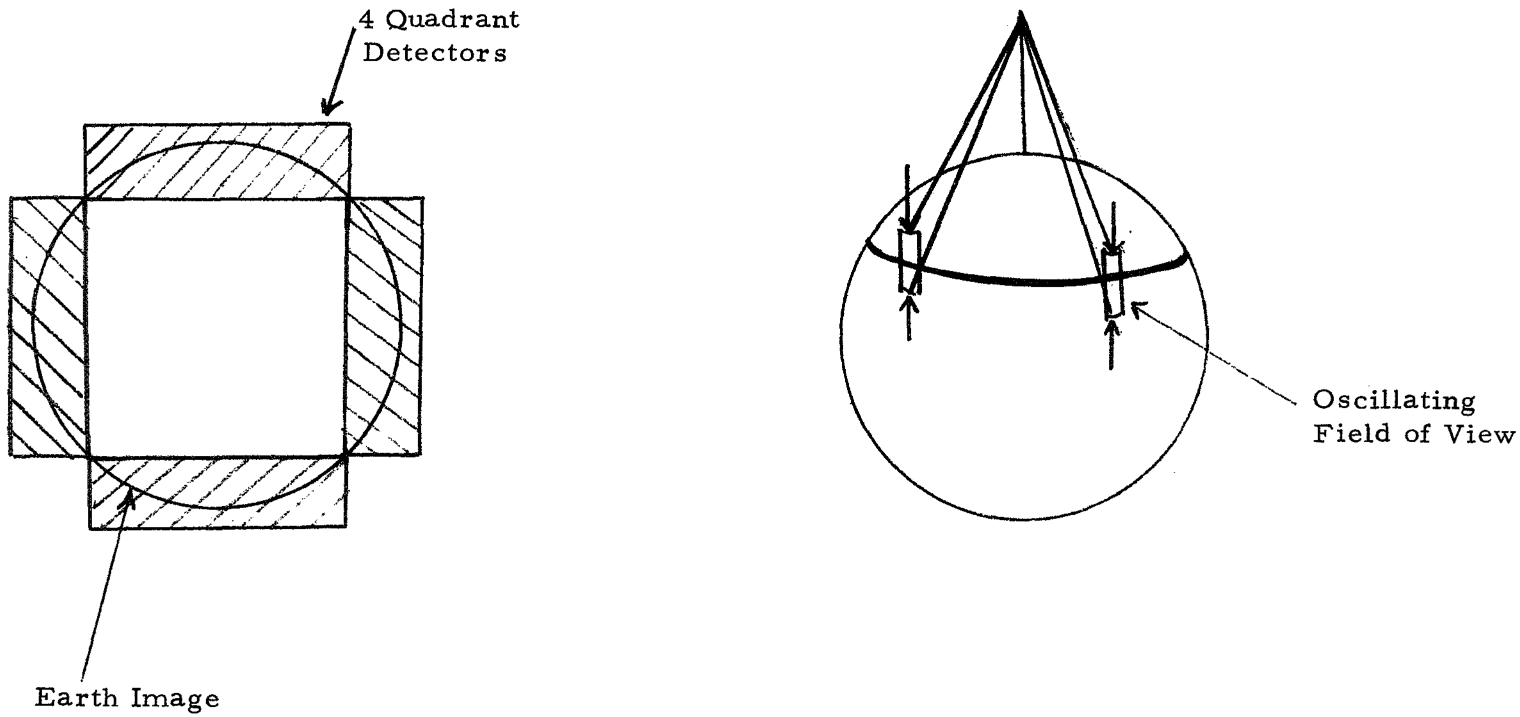
Radiation Balance

The radiation balance type of sensor has no moving parts and functions by comparing radiation from opposite portions of the Earth. Consider an IR image of the Earth falling on a quadrant of sensors as shown in Figure 7-A. When the image is symmetrically on the four detector elements, null signals are developed. Thus, two axis information can be obtained. The principle problem in this type of sensor is that the null is the difference between two large quantities thus leading to instability of null indication as well as poor resolution. This type of sensor is simple, rugged, reliable and uses very little power; therefore, it is useful where limited performance is adequate.

Edge Tracker

The basic principle of the edge tracker horizon sensor is shown in Figure 7-B. A small field of view is caused to oscillate about the horizon space discontinuity. Either three oscillating fields of view are implemented or one is caused to trace the horizon.

Horizon Sensor Schemes



Radiation Balance

A

Edge Tracker

B

Figure 7.

In either case the horizon location is determined by measuring the center of the oscillating field with respect to the spacecraft axes. This type of system is probably the most accurate but also the most complex.

Conical Scanner

The scanning type of horizon sensor consists of a field of view that is caused to move continuously with a conical motion thus intercepting the horizon in two places. A pair of these units mounted on right angles will develop two axes signals. This type of sensor has a good flight history and is sufficiently simple to permit good reliability. The scan geometry and method of developing signals is shown in Figure 8.

These two sensors are installed so that one is looking normal to the orbital direction while the second is looking 180 degrees away. Each field of view moves in a cone to inersect the Earth during a portion of each scan. Each scan therefore develops two pulses, one at each Earth-space interface. The time difference between the two pulses is bisected by a reference pulse to develop an error signal about the spacecraft "pitch" axis. When the spacecraft has the proper attitude, the time difference between the two pulses is the same for both sensors. Any difference is a measure of the "pitch" error. The pulse marked t_o is a reference pulse developed by the scan mechanism. When the roll error is zero, then the duty cycle of each sensor head is the same. Differences in the duty cycle is the roll signal.

Horizon Sensor Errors

The sources of error in the horizon sensor are:

1. Temperature gradients at the horizon
2. Instrument errors
3. Installation errors

Horizon Sensor Scans

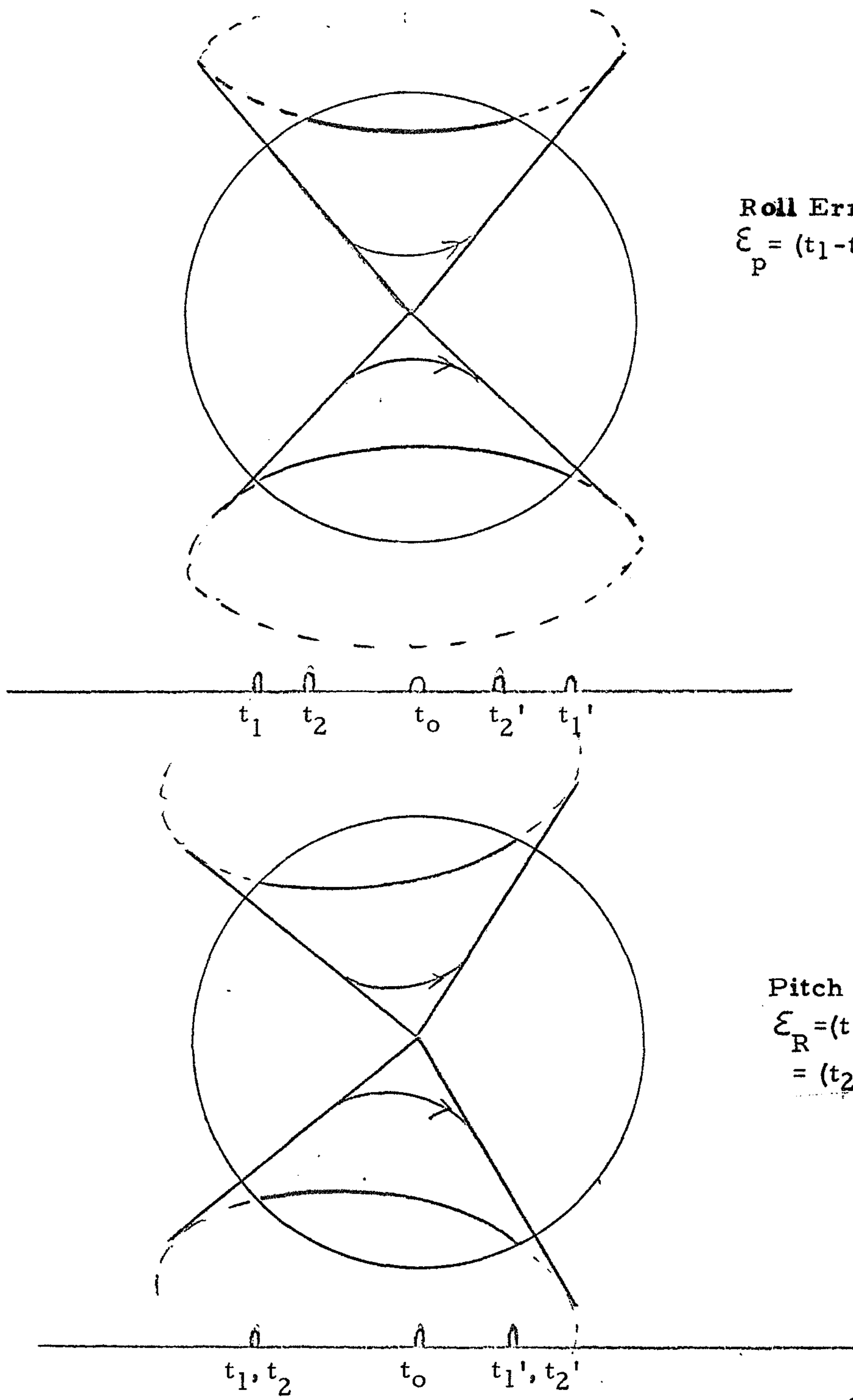


Figure 8.

Horizon Variance

Considerable effort has been expended to categorize the IR horizon for all seasonal and climatological conditions. Some of the noteworthy efforts were the NASA-LRC programs with the X-15 Project Scanner and the horizon definition measurement program. The results of these programs are discussed in the section on signal processing.

Instrument Errors

The instrument errors will depend on the complexity, size and cost of the sensor unit. A representative group of developed sensors is shown in Table III. From the table it can be seen that instrument errors will range between 0.1 to 1.5 degrees.

Installation Errors

The purpose of the horizon sensor is to determine the orientation of the spacecraft axes with respect to local vertical. However, more explicitely the purpose is to orient some other sensor which in general is part of the experiment with respect to some point on the Earth. Therefore, it is advantageous to align the horizon sensor with respect to the experiment sensor. A feasible technique for doing this is to have an optical reference cube on both units. If requirements are severe, on-board alignment optics can be implemented to continuously keep track of this alignment. If the latter is done, alignment within .01 degrees is

TABLE III
Horizon Sensor Development
Barnes

Model Year	Accuracy	Cone Angle (1/2)	Weight	Power	Freq. Response
133-1966	1.5°	55	6.8	9	.25Hz
150-1966	0.3	37	15	14	.1 Hz
151-1966	0.1	12.5	17.5	10	.1 Hz
156-1968	.25	20	22	22	.25
155-1968	.25-.07	55	17	20	.25-.06
157-1969	.11	90	6	6.5	—
159-1969	.25-.07	55	10.1	7.5	—

This data has been taken from a current Barnes catalog and might be subject to some interpretation.

achievable. If not, rigidity and stability of the spacecraft have to be considered and depending on the nature of the experiment sensor the alignment will range between .05 to 0.2 degrees. Horizon sensors are unique in that determination of their true axes must be determined with a horizon simulator. Therefore, a potential bias error exists to the degree that the simulator is correct.

Error Summary

1. Horizon Variance	
Full processing	$.01^{\circ}$
Simple processing	$.2^{\circ}$
No processing	1.0°
2. Instrument Error	
Best	$.05^{\circ}$
Normal	$.2^{\circ}$
3. Installation Error	
Best	$.05^{\circ}$
Normal	$.1^{\circ}$
Total Error	
Best	$.07^{\circ}$
Normal	$.3^{\circ}$

The above error summary is a gross simplification but is indicative of typical cases.

Signal Processing

The use of the 15 micron spectral band reduces but does not eliminate the errors due to unsymmetrical temperature gradients at the horizon intercepts. These gradients will generate sensor outputs that have different slopes and amplitudes at the diametrically opposite intercepts. A great deal of effort has been expended in learning how to process the sensor output signals to reduce the above errors.

The horizon definition measurement program (HDMP) study, supported by NASA/Langley Research Center, studied horizon sensor problems in great detail. In this work many detection schemes were evaluates and compared. Some of the schemes that were investigated are shown in Figure 9.

Scheme 1: A small field of view scans the horizon gradient and develops a pulse when a preset threshold is reached.

Scheme 2: Here the radiance is normalized to the peak value. Otherwise it is identical to scheme 1.

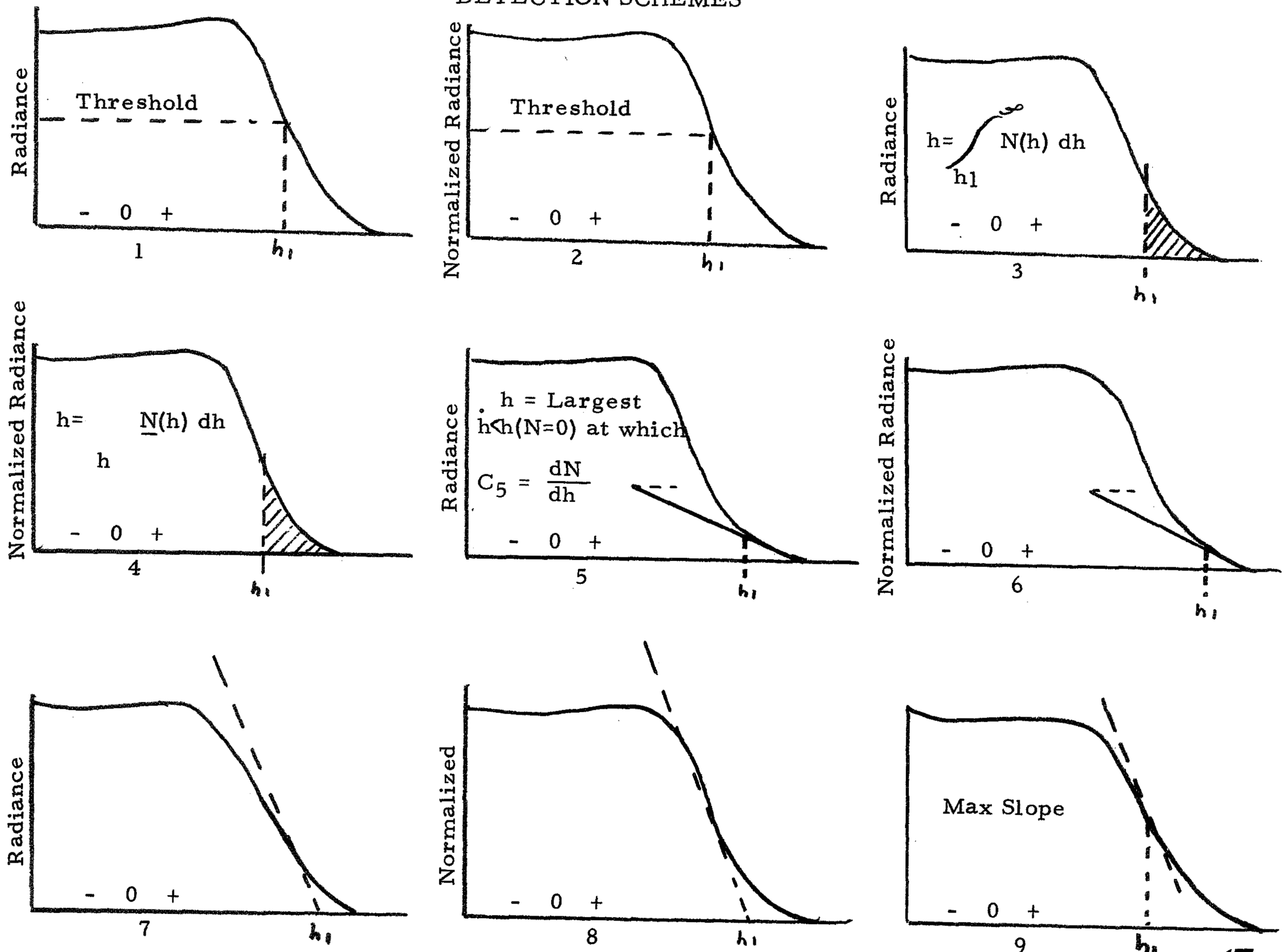
Scheme 3: Here the scan must be in a space to Earth direction. The detector output is integrated and when the integral reaches a preset threshold a pulse is generated.

Scheme 4: This is the same as scheme 3 but normalized to the peak radiance value.

Scheme 5: When the detector output has a preset rate of change, a pulse is generated.

Scheme 6: Same as scheme 5 but normalized to the peak radiance.

DETECTION SCHEMES



Tangent Ht. is the Abscissa in Each Case

Figure 9.

Scheme 7: Two threshold values are sensed and the slope of the resulting line is extrapolated back to the zero radiance level. This latter intersection is defined as the horizon intercept.

Scheme 8: Normalized version of scheme 7.

Scheme 9: The inflection of the radiance curve is detected.

The comparisons were made for a wide range of latitude, geographical, seasonal and diurnal conditions. Some of the information from HDMP is shown in Table IV.

The standard deviations shown are for one intercept; therefore, the local vertical determination is less accurate by a factor of $\sqrt{2}$.

The result of the HDMP study along with other studies permits a model to be made that can be employed to enhance the accuracy of attitude determination. The nature of the inputs to the model is shown in Figure 10. The improvement in horizon determination that is so achieved is shown in Figure 11.

TABLE IV

Horizon Detection Schemes and Standard Deviation

500 KM Altitude

	Detection Scheme	σ Degrees
1.	Radiation magnitude threshold	.022
2.	Normalized radiance threshold	.018
3.	Integrated radiance threshold	.015
4.	Integrated normalized radiance threshold	.011
5.	Slope threshold	.022
6.	Normalized radiance slope threshold	.021
7.	Slope extrapolation	.014
8.	Slope extrapolation on normalized radiance	.017
9.	Inflection point detection	.035

DATA PROCESSING

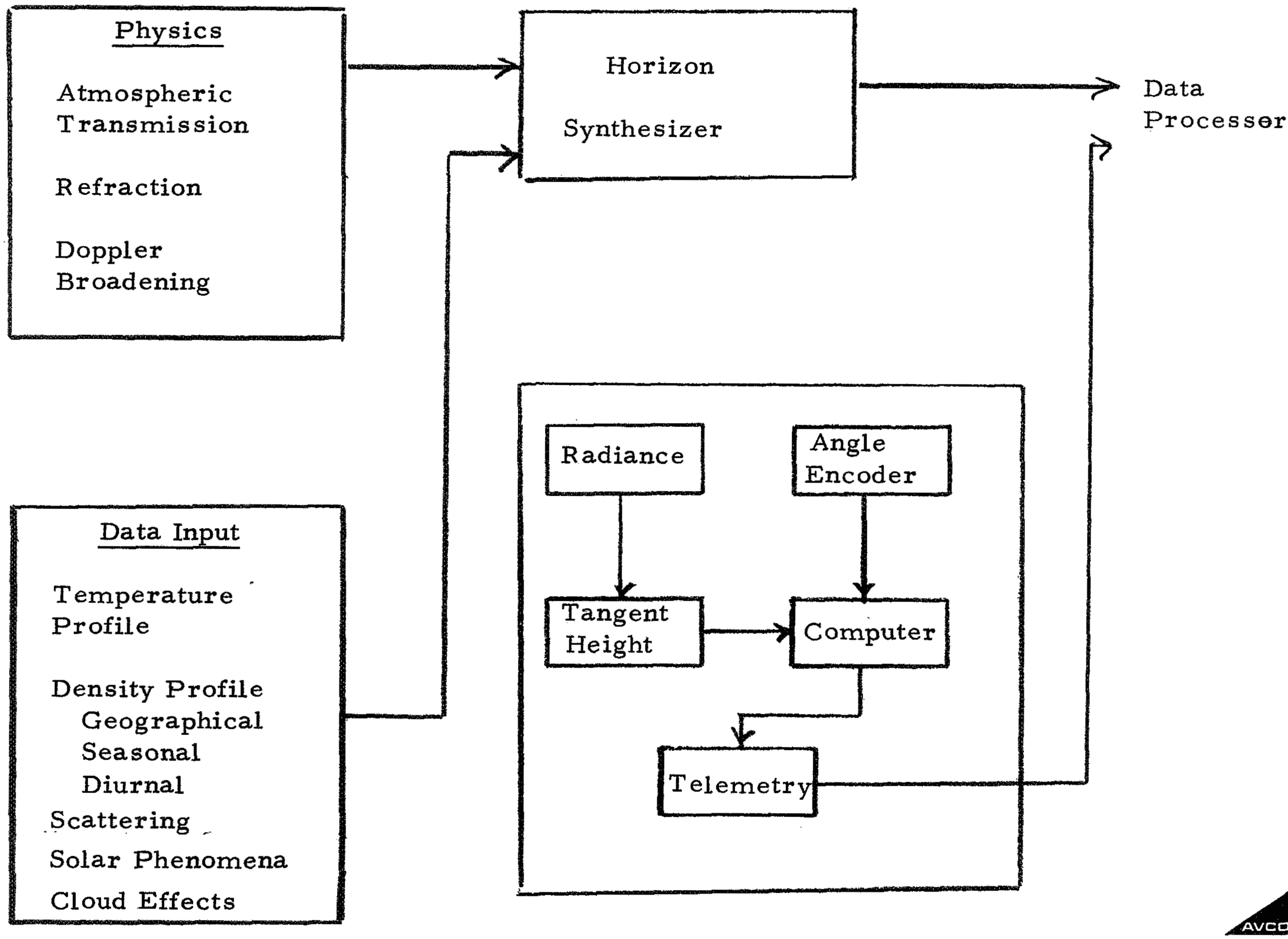


Figure 10.

Reduction in Residual Std. Dev. of Variances

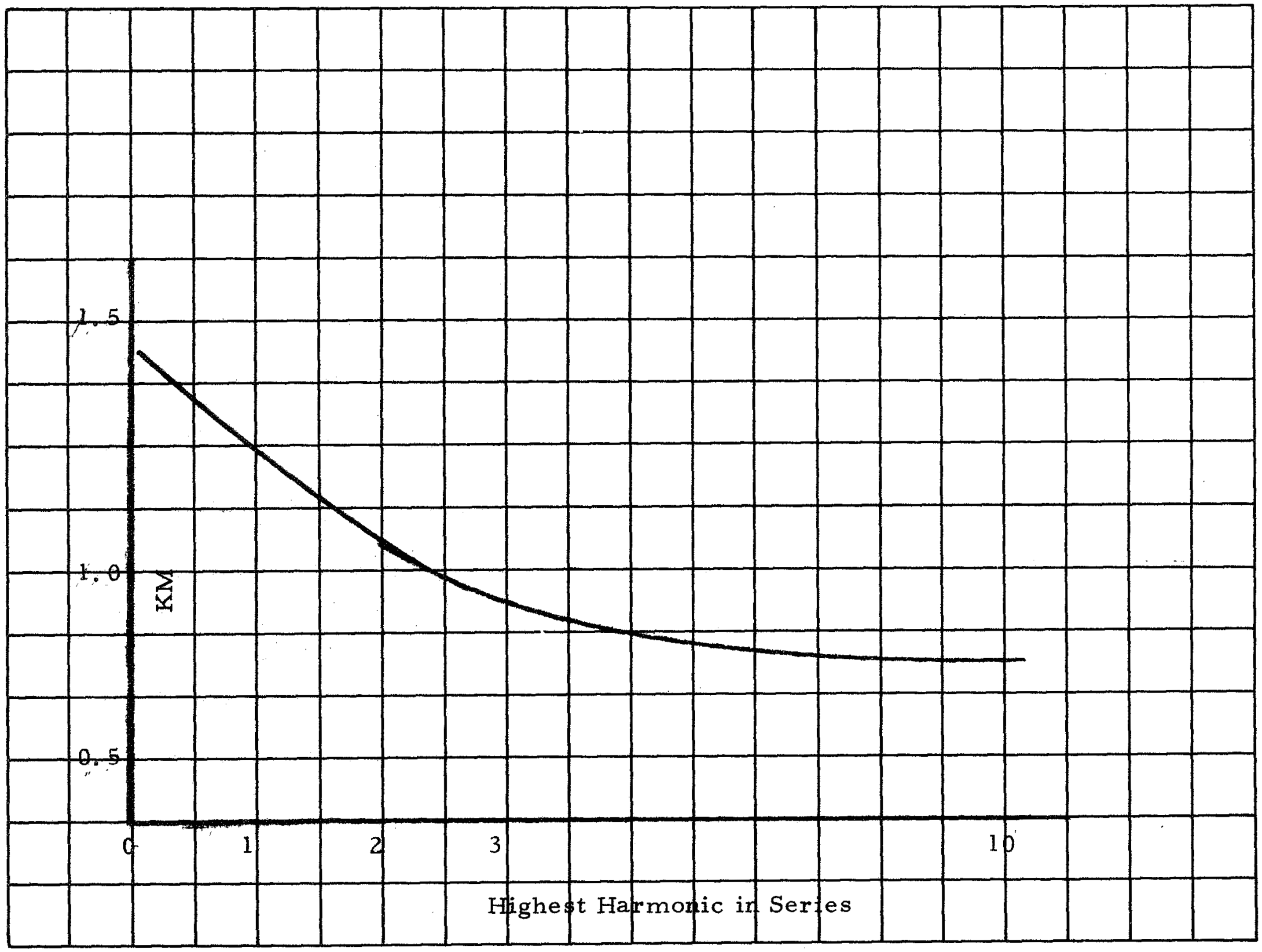


Figure 11.

3. Sun Sensors

The purpose of the coarse sun sensors is to generate approximate solar aspect angle information. This information is useful during orbit injection and while orienting the spacecraft into the mission attitude.

Coarse sun sensors are generally silicon cells used in a photovoltaic mode. In this mode no biasing is necessary and the circuits used to implement the sensor are simple and reliable. The circuit impedances are low, thus minimizing interference and pickup problems. The cell characteristics are shown in Figures 12-A and -B. The most effective way to use the cells is with an output load that approaches a short circuit. This mode of operation will result in good null stability and linearity. A feasible method of approaching the short circuit condition is to sum the two cells that compose a sensor element by connecting them in parallel opposition and feeding the output through an appropriate resistor directly into the summing point of an operational amplifier as shown in Figure 13. The summing point of the amplifier is effectively at ground; therefore, the temperature compensation criteria is satisfied.

Peak current is in the order of 200 ua; therefore, to obtain a typical signal voltage of 5 volts, the feedback resistor will be 2.5×10^4 ohms which is quite a reasonable value.

The basic element is a small flat plate which has a field of view of 2π steradians. The output function is nearly $T_o = k \cos \theta$ where θ is the angle between the sun line and the normal to the cell element. The relative response is shown in Figure 14-A, where the close approximation to a cosine function can be seen.

Solar Cell Characteristics

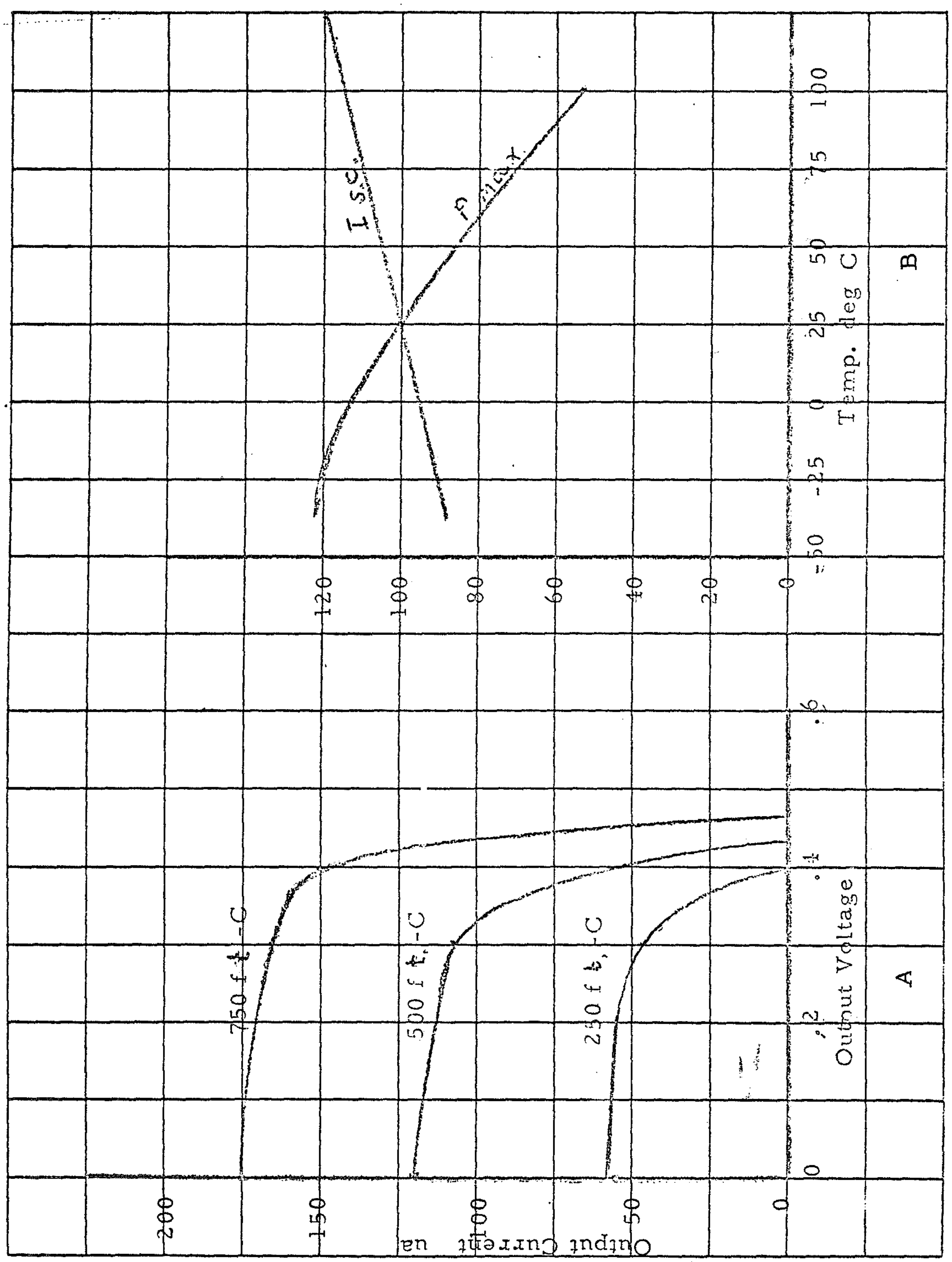
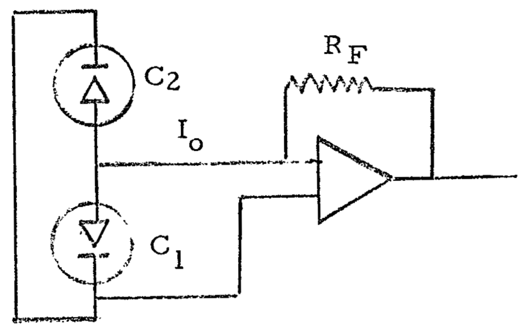


Figure 12.

Solar Sensor Circuit



$$E_o \doteq I_o R_F .$$

$$I_o = k \sin \theta$$

$$E_o = k' \sin \theta$$

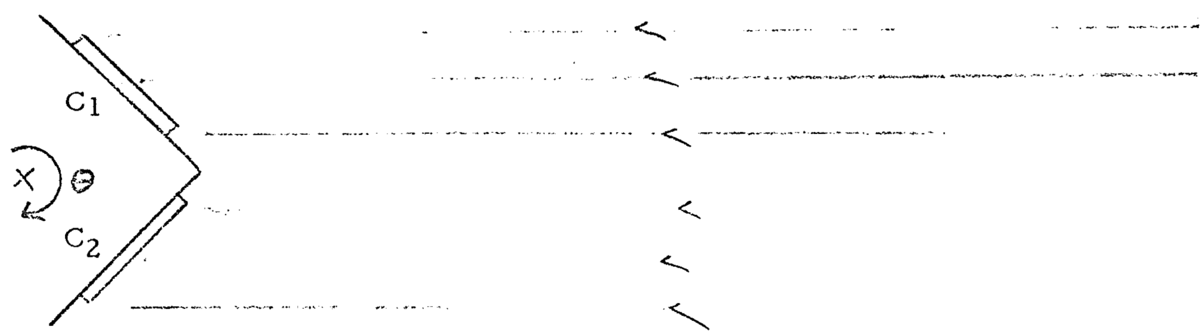


Figure 13.

Coarse Solar Sensor

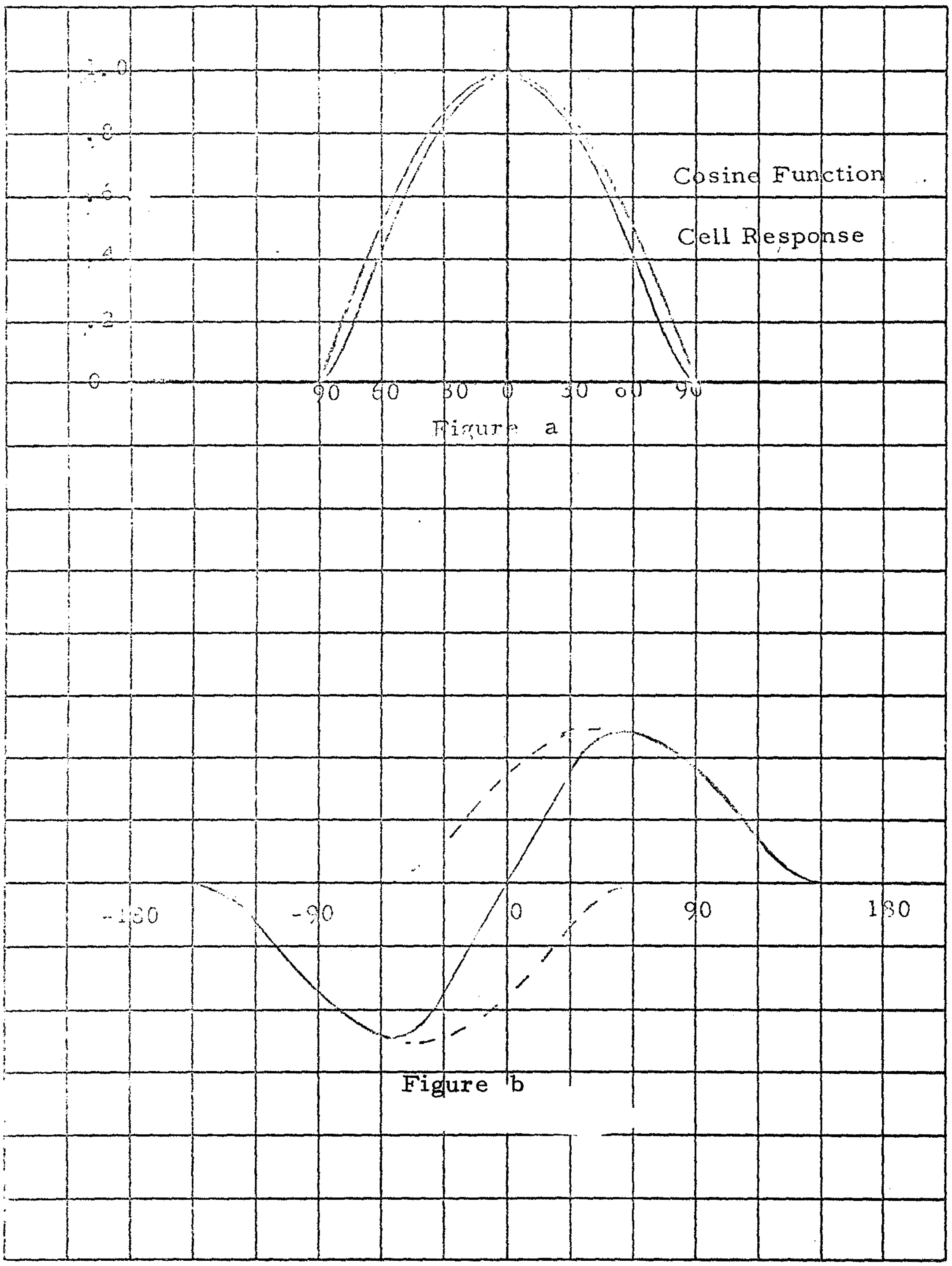


Figure 14.

In most applications, two cell elements are combined with their surfaces normal to each other to form the solar sensor. The outputs are electrically combined by simply connecting the cells in parallel opposition to generate the function shown in Figure 14-B.

Two sensor units can be combined to produce a full 360 degree coverage as shown in Figure 15-a. There are, of course, possible ambiguities in this function which may not be resolvable if the spacecraft spin direction is not known or if the attitude history is confused.

These ambiguities can be eliminated by using two complements of the above sensors that are spacially phased by 90 degrees. The output functions shown are sine and cosine functions very much as in the familiar synchro resolvers. The two functions produce a ratio that is unique for any possible angle about an axis. In this manner if a sensor is placed on each corner of a cube, it is possible to implement simple logic to permit solar aspect determination about any axis as shown in Figures 15-b and 16. Of course, for any attitude, rotation about the sun line is not sensed.

Fine Sun Sensors

The coarse sensor can be implemented with baffles to cut the field of view. This technique reduces errors due to albedo or reflections from some part of the spacecraft although the angular range of operation is reduced. A typical function is shown in Figure 17-a. The addition of a simple lens can increase the slope of the function as shown in Figure 17-b. . In the above configurations the units are often referred to as fine solar sensors. Typically, the sensors weigh less than an ounce apiece and have a volume less than one cubic inch.

Solar Sensor Configurations

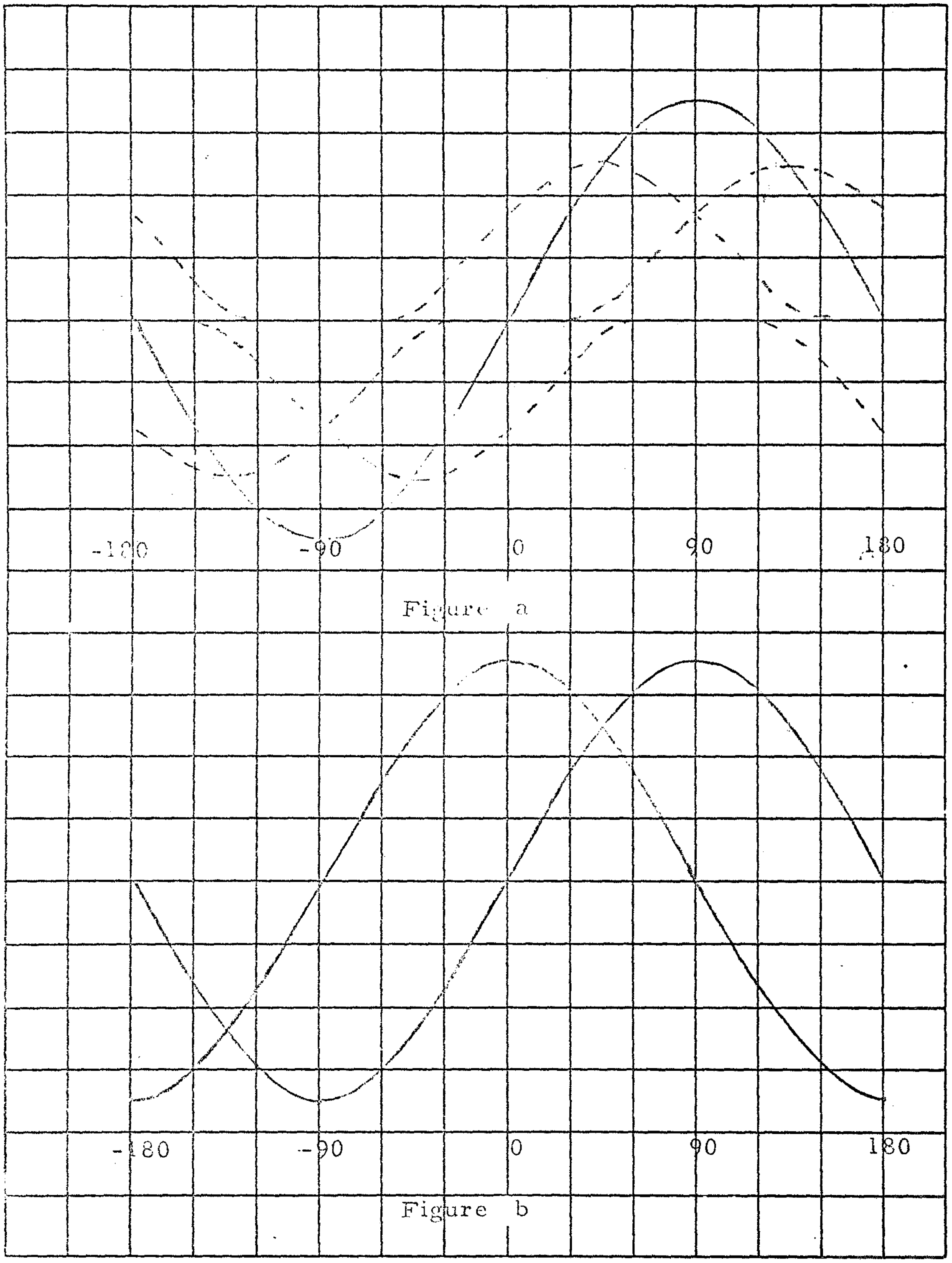
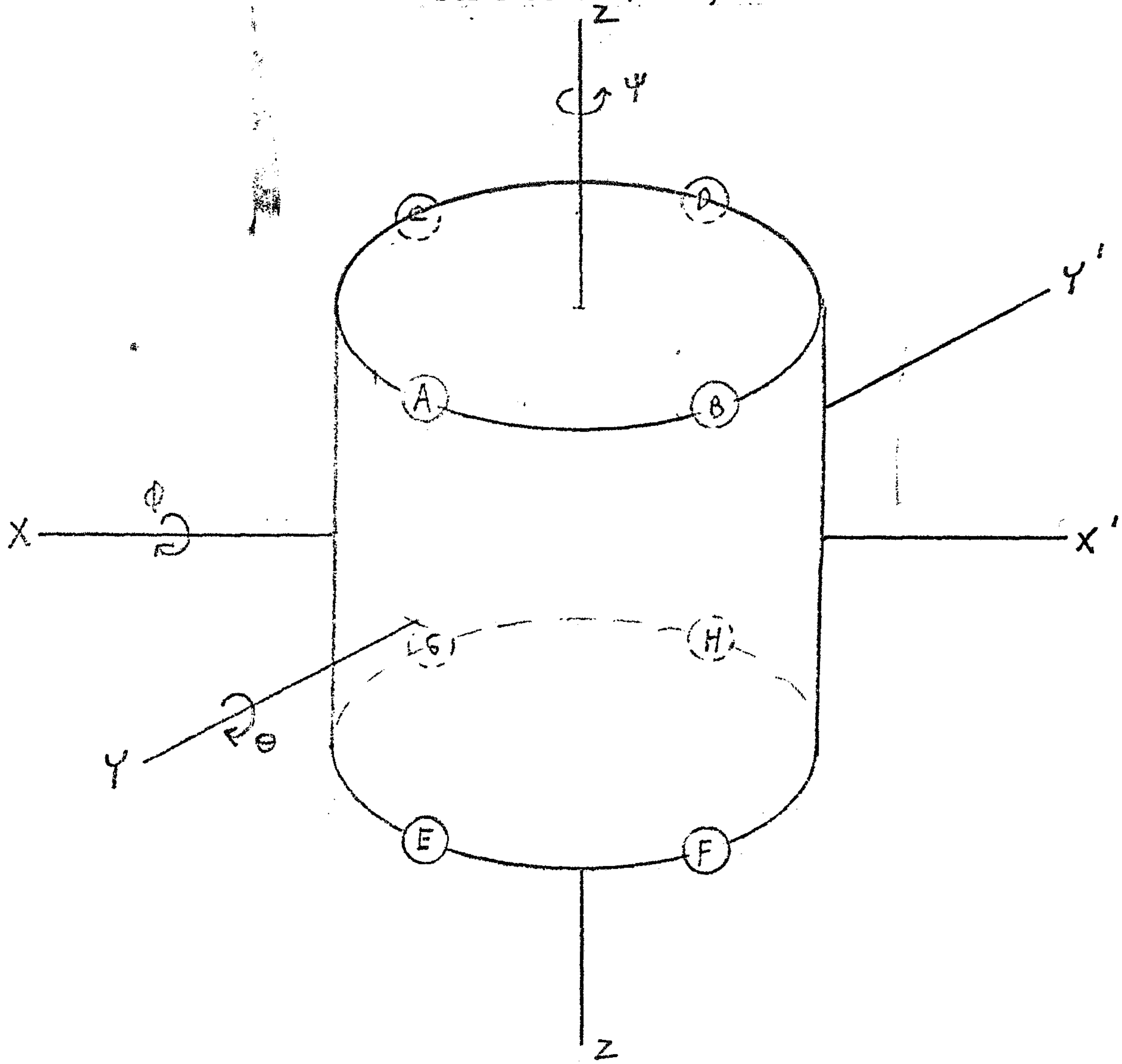


Figure 15.

Solar Sensor Array



When sun is perpendicular to the Z axis

$A - B + C = D + E - F + G - H = \sin \Phi$

$A + B - C - D + E + F - G - H = \cos \Phi$

When sun is perpendicular to the Y axis

$A + B + C + D - E - F - G - H = \sin \theta$

$A + B - C \_ D + E + F - G - H = \cos \theta$

When the sun is perpendicular to the X axis

$A - B + C - D + E - F + G - H = \cos \psi$

$A + B + C + D - E - F - G - H = \sin \psi$

AVCO

Figure 16.

Fine Sun Sensors

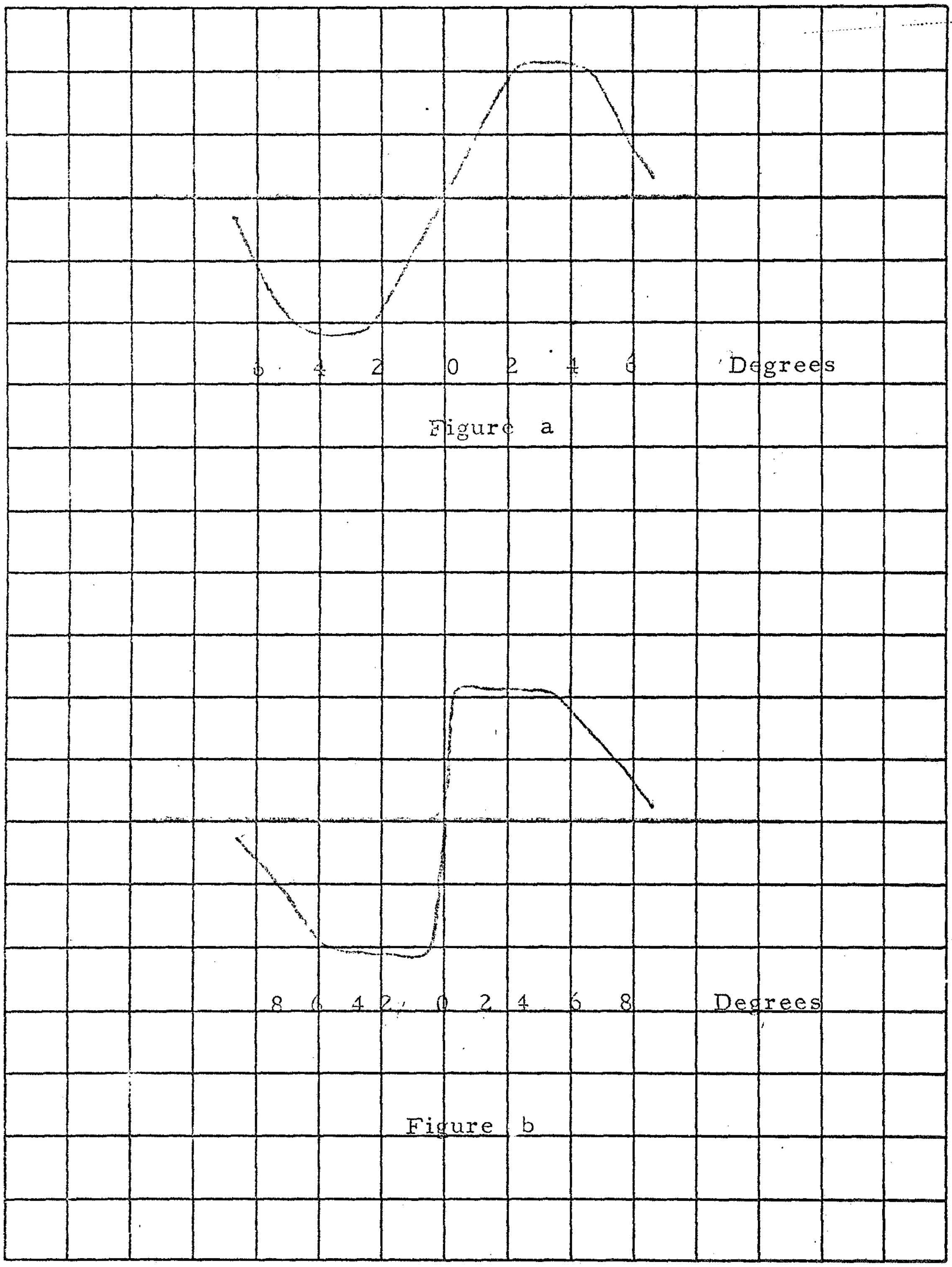


Figure 17.

While this type of sun sensor may not be capable of generating accurate aspect information, they do have high resolution capability; therefore, when used as revolution timers, a high degree of precision is possible.

Digital Units

The slit or digital solar sensors utilize a detector element similar to the ones used in the coarse sensors. The output of a single cell is compared to a preset threshold value. When this threshold level is equalled a pulse is generated. The optical system is sized so that typically the output of the cell is in the order of 25 to 50 microamperes to develop the threshold level.

Figure 18 illustrates the block diagram of a digital solar sensor. The read and readout commands come from logic elements that are appropriate for a particular mission.

The digital sensor shown in Figure 19 is a basic unit made by Adcole that has an extensive history. There is a more sophisticated version of this sensor under development. In this latter case there are four sets of cells and corresponding reticles. The sets are identical but spacially staggered by 1/4 of the quantum or least significant bit. This scheme permits an electrical "vernier" to read in between the least significant bits which in the case of the 7 bit unit with a range of $\pm$ 64 degrees is one degree ($2^7 = 128$). Thus, there will be a transition every 0.25 degrees instead of every one degree. Since the sun is a finite size source, the cell outputs approach a sinusoidal function. Therefore, the cell outputs can be ratioed to permit extrapolation of a smaller quantum. The design goal is in the order of 1 $\widehat{\text{min}}$.

1 - 1690

Digital Solar Sensor Electronics

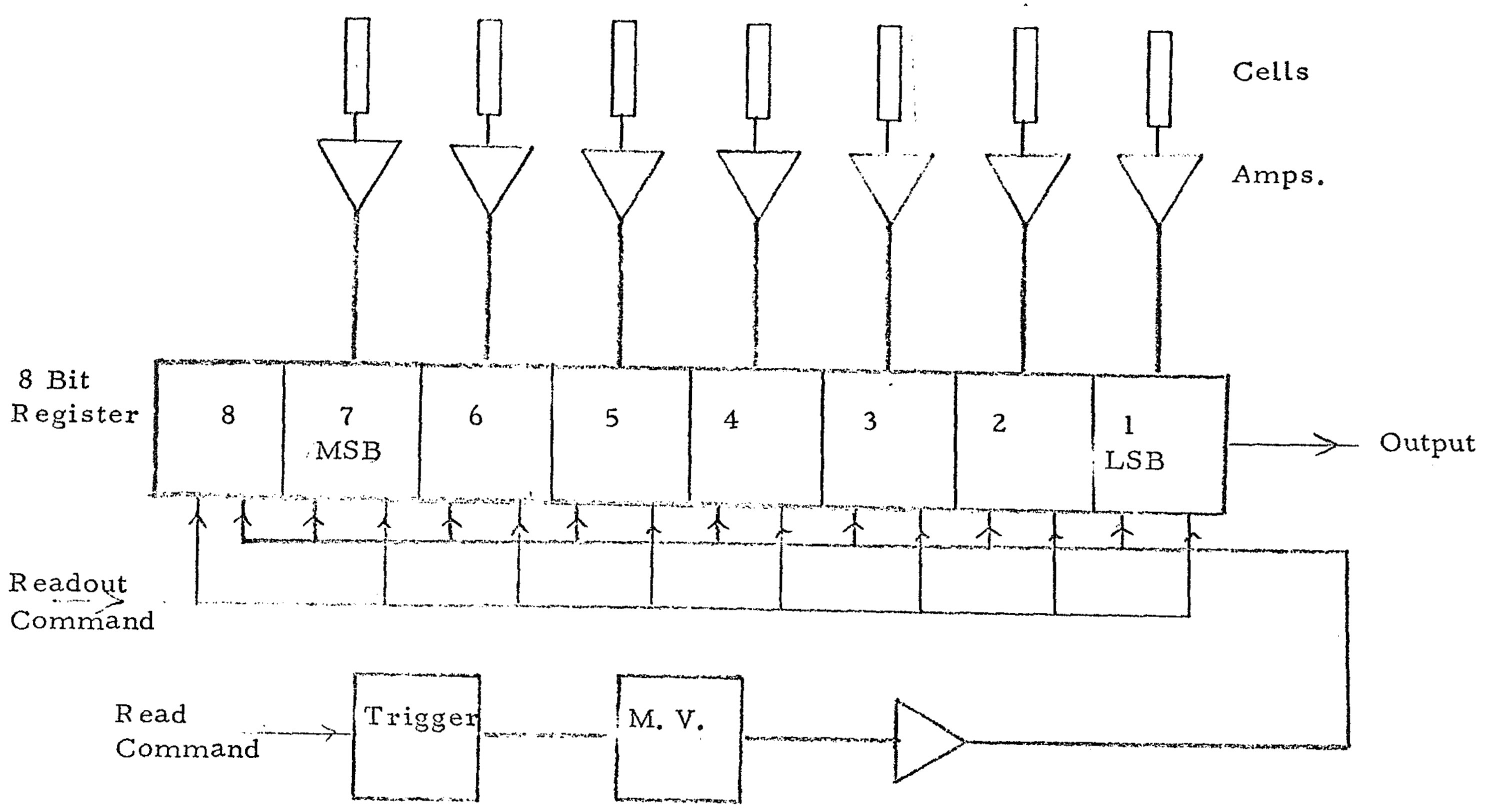


Figure 18.

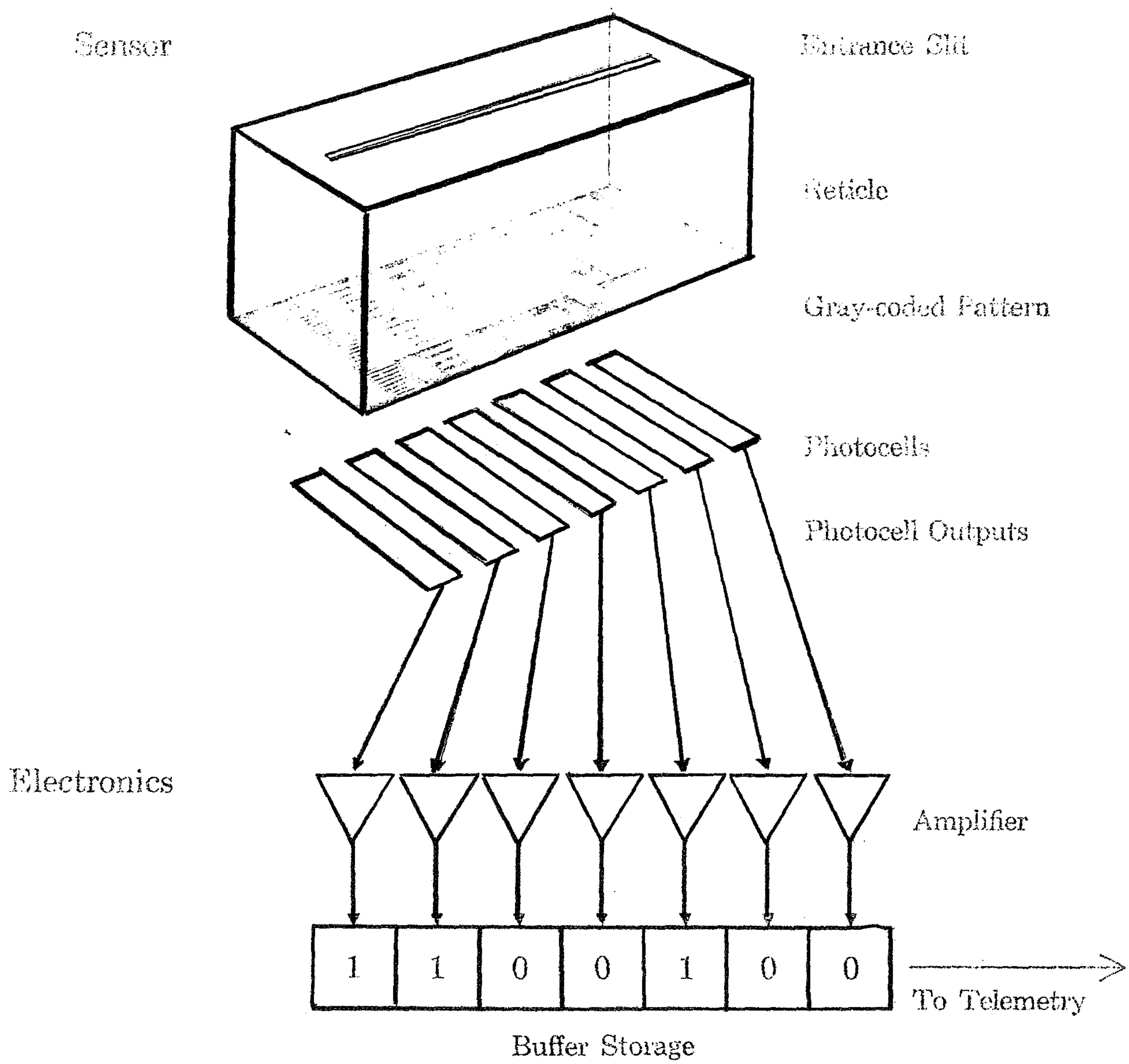

Sensor
Entrance Slit
Reticle
Gray-coded Pattern
Photocells
Photocell Outputs
Electronics
Amplifier
1
1
0
0
1
0
0
To Telemetry
Buffer Storage

Figure 19.

Solar Sensor Errors

Temperature

The temperature coefficient of a silicon cell is k = 0.13% per degree centigrade. Since the differential cells are mutually attached to a common heat sink the temperature difference between the cells can be small. A maximum difference of 0.5 degrees C is a reasonable value to expect. The null error in the latter case is:

$$\varepsilon_{null} = k \times \Delta T \times 2 = .0013 \text{ radians} = 4 \overset{\frown}{\text{min}}.$$

Stability

The long period stability of silicon cells is high and there is a tendency for the two elements to age symmetrically. Typically, the null stability is quoted by vendors as 5 $\overset{\frown}{\text{min}}$. Tests at Avco confirm this value of stability as a reasonable number to use.

Albedo Error

The coarse solar sensors are radiation balance type with wide fields of view. There will be times when this field of view will include the sun lit Earth which in general will cause unsymmetrical illumination on the two elements of the sensor thus causing an error. The following analysis is used to determine the magnitude of this error.

Mean Albedo of Earth	=	.35
Mean Radius of Earth	=	6378 KM.
Solar Constant	=	1390 $W.M.^{-2}$

The output function of the solar sensor is given by the following expression with reference to Figure 20.

$$\frac{K_a}{\pi} \int \frac{\cos \beta \ \cos \gamma \ \sin \lambda}{D^2} \, ds - K \sin \epsilon = 0$$

K = solar constant

a = Albedo

D = distance between a surface element of Earth d_s and the sensor

B = angle between sun line and normal to d_s

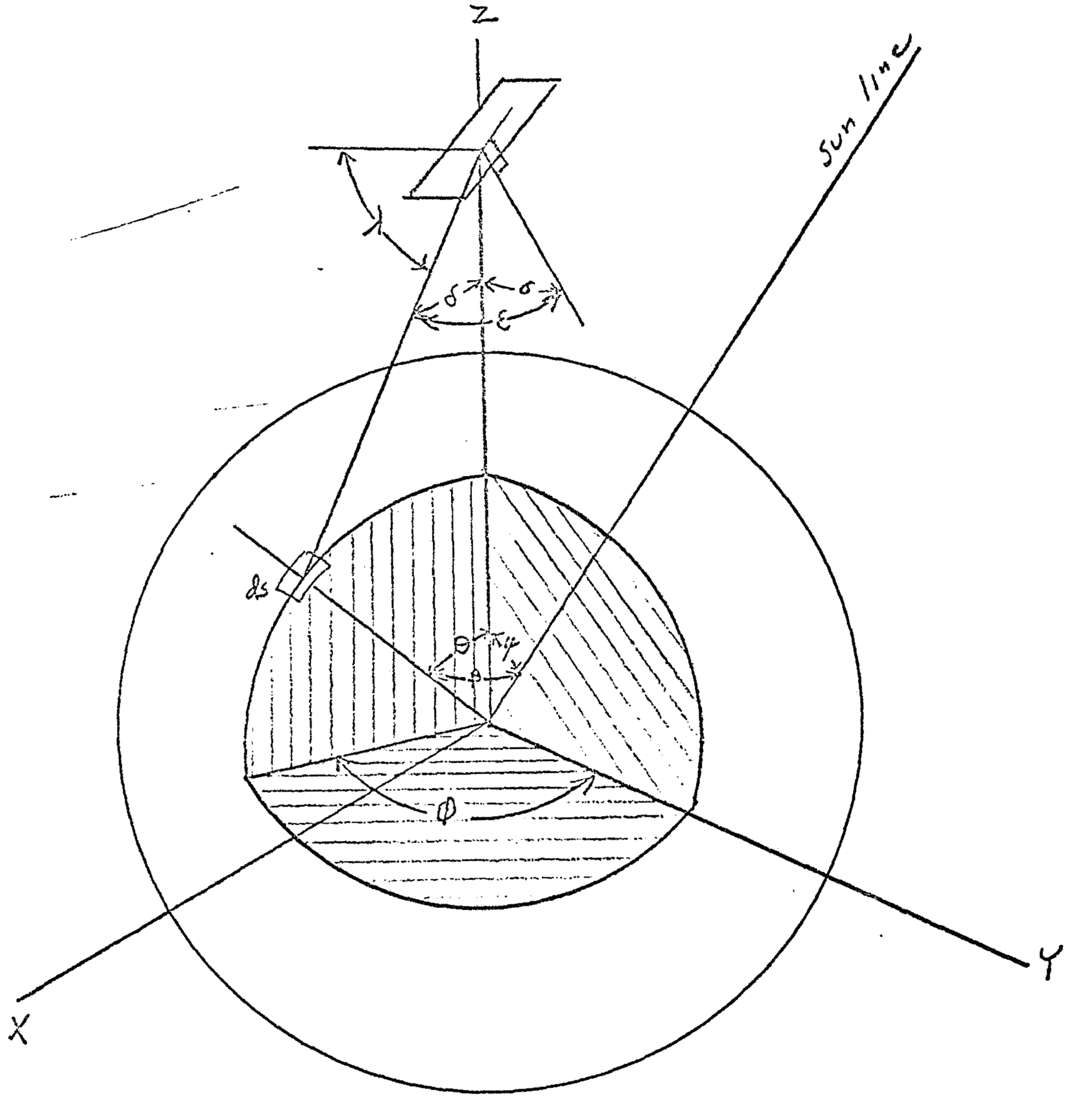
γ = angle between normal to d_s and d_s - sensor line

λ = angle between sensor plane d and d_s - sensor line

ψ = angle between Z axis and sun line

The left hand term is the Albedo contribution while the other is the direct solar radiation.

The relative solar orientation error as a function of the geocentric angle is shown in Figure 21. The mathematical development **is shown in** summary No. 1.



$H = h + R$

$ds = R^2 \sin\theta \, d\theta \, d\phi$

$D^2 = H^2 + R^2 - 2RH\cos\theta$

$\cos\gamma = \dfrac{H\cos\theta - R}{D}$

$\cos\beta = \cos\theta\cos\psi + \sin\theta\sin\psi\cos\phi$

$\sin\lambda = \cos\varepsilon = \cos\delta\cos\sigma + \sin\delta\sin\sigma\cos\phi$

$\sin\delta = \dfrac{R\sin\theta}{D}$

$\cos\delta = \dfrac{H - R\cos\theta}{D}$

$\sigma = \left| \dfrac{\pi}{2} - \psi - \varepsilon \right|$

AVCO

Figure 20.

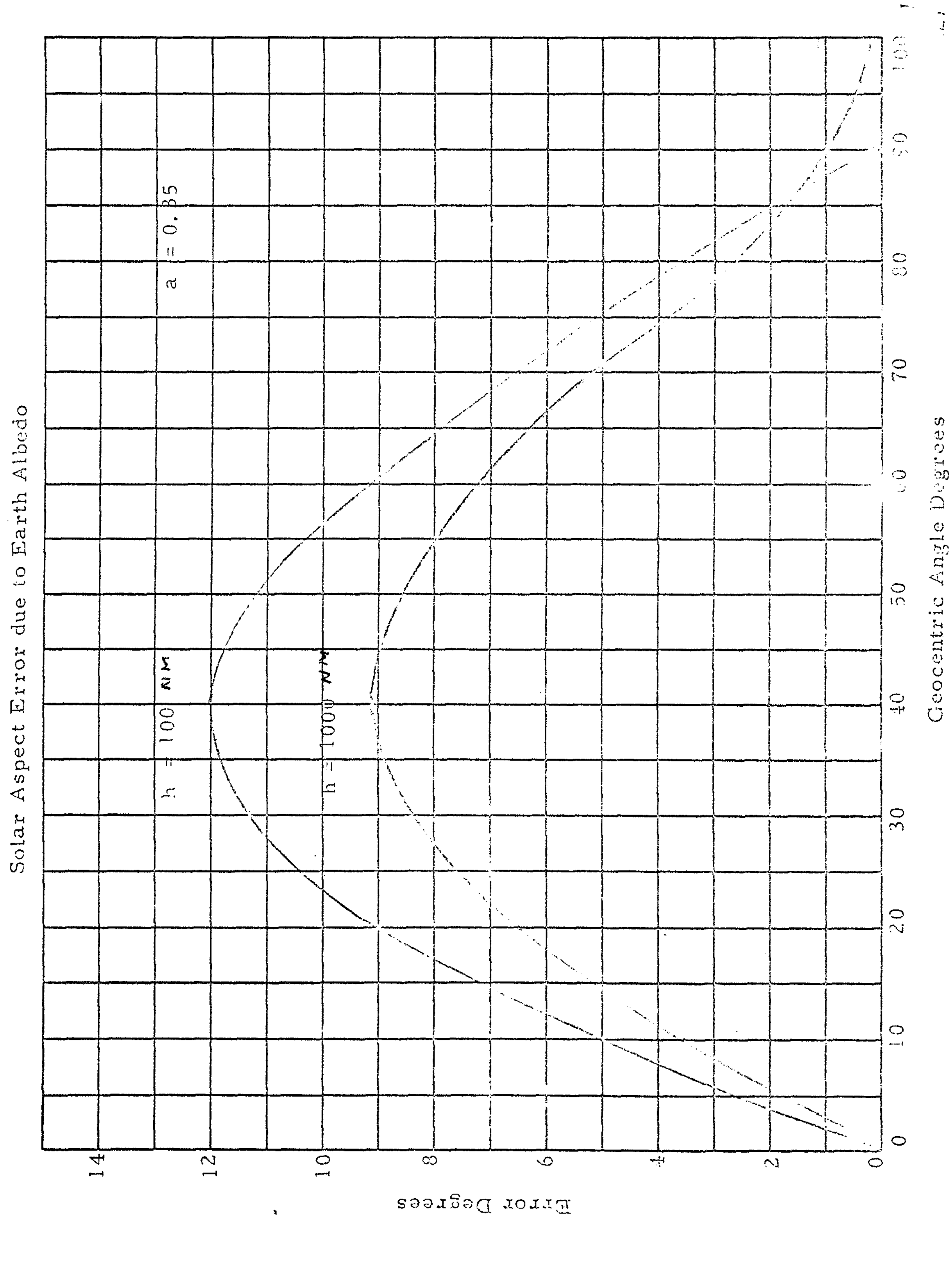

Solar Aspect Error due to Earth Albedo
h = 100 NM
h = 1000 NM
a = 0.35
Error Degrees
0
2
4
6
8
10
12
14
Geocentric Angle Degrees
0
10
20
30
40
50
60
70
80
90
100

Figure 21.

1 - 1690

SUMMARY # 1

$$\frac{K\,a}{\pi}\int \frac{\cos\beta\cos\gamma\sin\lambda}{D^2}\,ds - K\sin\mathcal{E} = 0$$

$$\frac{2\,aR^2}{\pi}\int_0^{\theta}\int_0^{\Phi}\frac{(cs\theta cs\psi+sn\theta sn\psi cs\Phi)(Hcs\theta-R)\left[(H-Rcs\theta)\left(cs\left|\frac{\pi}{2}-\psi-\mathcal{E}\right|\right)+R\,sn\theta\left(sn\left|\frac{\pi}{2}-\psi-\mathcal{E}\right|\right)cs\,\Phi\right]sn\theta\,d\Phi\,d\theta}{(H^2+R^2-2RH\,cs\,\theta)^2} - \sin\mathcal{E} = 0$$

In Simplified Notation

$$C\int_0^{\theta}\int_0^{\Phi} F(\theta, \Phi, \mathcal{E}, \psi, H)\,d\Phi\,d\theta - \sin\mathcal{E} = 0$$

Installation Error

Precise alignment of wide angle sensors is very difficult because it is so difficult to apply a correct input. Meteorological phenomena are troublesome when the actual sun is used and spurious reflections are a problem when a simulated sun sensor is being used. Experience has shown that alignment with a simulator can be done to within 12 min and a little better with the actual sun but the practical difficulties in the latter case are quite severe.

Coarse Sensor Error Summary

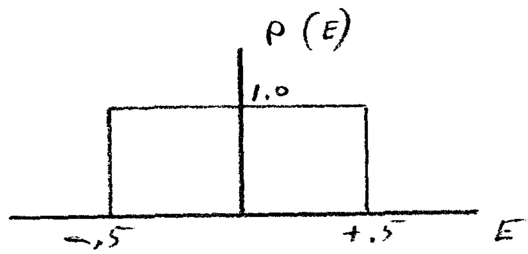
Null Stability	5 min	
Temperature	4 min	
Installation	12 min	
R. S. S.	14 min (Null)	Scale Factor 2 Degrees

This error can be negligible compared to the Albedo error. Since the Albedo varies over different parts of the Earth depending on cloud distribution, it is difficult to compute the albedo error; and it is much simpler to depend on additional sensors with restricted fields of view.

Digital Sensor Error

The quantization error is developed in the following manner:

The correct angle for a given readout is taken to be the midpoint of the angular excursion over which the readout remains unchanged. Thus, for a given readout the sun angle may be changed by ± 1/2 quantum before the output word is changed. Since a readout may be required at any particular position within a quantum, the probability distribution function for quantization error is simply a constant. There is zero probability that the error will exceed ± .5 bits.



$$\sigma_Q^2 = \int_{-\infty}^{+\infty} E^2 p(E)\, dE$$

The quantization error = .289 bits, since the bit size is one degree the quantization error is .289 degrees.

Code error	= .25 degrees
Electronic error	= .1 degree
Installation error	= .1 degree
Total RSS error	= .41 degrees

4. Magnetometers

Improvements in magnetometers and increased knowledge of the Earth's magnetic field have combined to offer a useful method of attitude sensing. There are two fundamental types of magnetometers that are being used in spacecraft. One measures the magnitude of the magnetic vector while the second measures the total vector. The former is typically the molecular resonant type while the latter is typically the flux gate type. The flux gate magnetometer also known as the second harmonic or saturable reactor type is rugged, accurate and has an extensive spacecraft flight history.

The principle of operation is based on the non-linear characteristics of the magnetization function of ferromagnetic materials. Consider a high permeability core with a triad of windings. Let one of the coils be excited with an A. C. current of sufficient magnitude to insure that the core is saturated during a part of the excitation cycle. Equal voltages will be induced into the other two coils and if the two voltages are differenced the result will be zero at each instant even though the excitation waveshape is complex. When the Earth's field is in a particular direction, the D. C. flux will add to the A. C. excitation flux to increase the instantaneous flux in one coil and reduce it in the other. This will cause the summation to be non zero and have a second harmonic value that is a function of the Earth field vector. The second harmonic can be phase discriminated and rectified to produce the desired output signal.

The performance and stability of the flux gate magnetometer is remarkably good. Units designed to measure full Earth's field demonstrate resolution in the order of a gamma and accuracies of 0.5% of full scale. Null stabilities in the order of a few gamma can be expected for substantial periods of time.

A good example of this type of performance is the NASA/Ames Torroidal magnetometer. Some of the measured performance of this magnetometer is shown in Figures 22 and 23. The basic sensor element is shown in Figure 24. The sensor element which is a two-axes unit is in the order of 2 in^3 and the electronics unit can be packaged in about 4 in^3 per axis. The electronics is quite straightforward as shown in block diagram form in Figure 25.

The feedback circuit shown is a sophistication that improves linearly and reduces the effect of the sensor on the field.

Stability of Toroidal Magnetometer

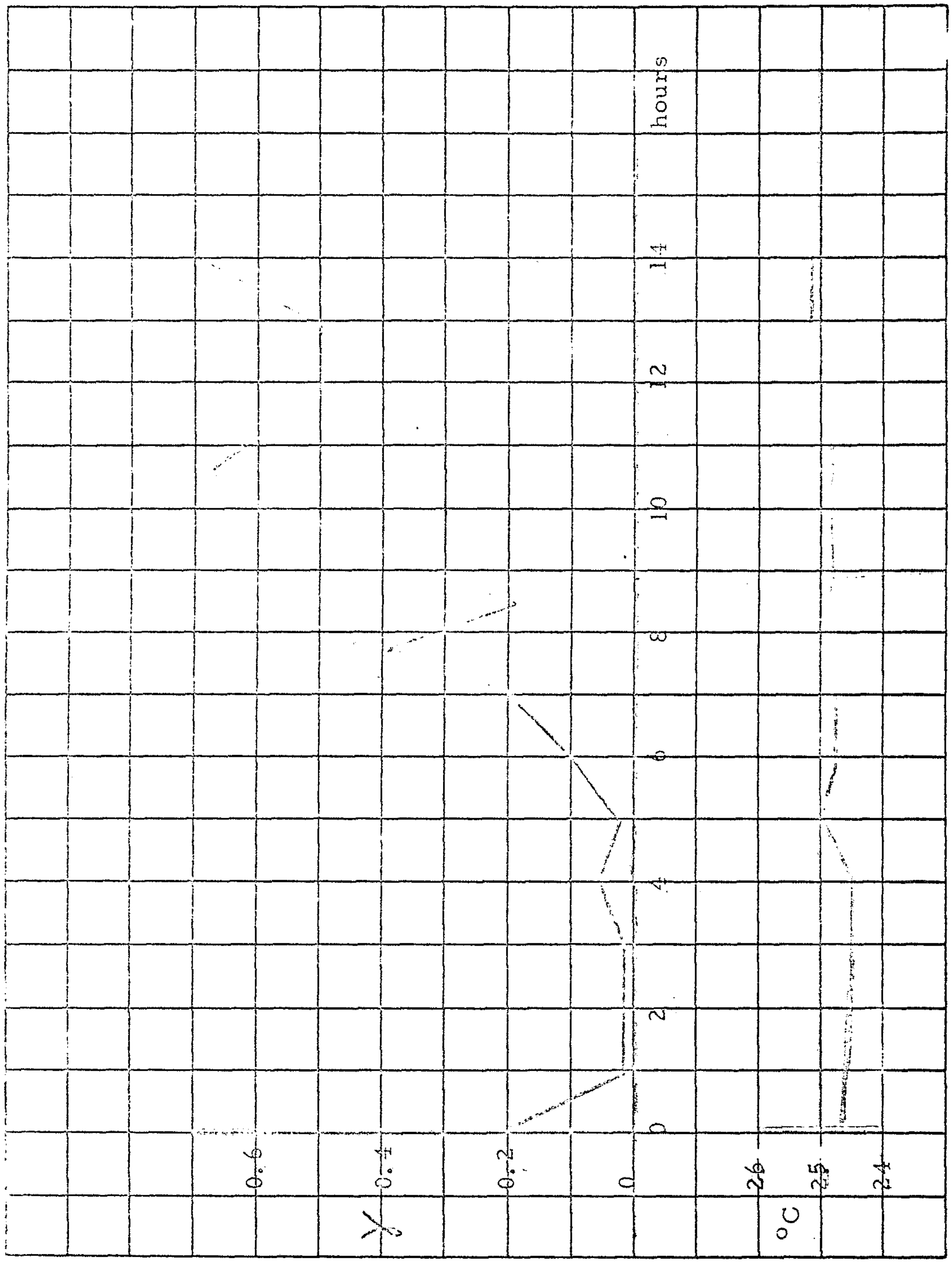


Figure 22.

SATURABLE REACTOR MAGNETOMETER
TOROIDAL COIL

	γ	
FLUCTUATION AT CONSTANT TEMPERATURE	$\pm$ 0.1	24 hrs.
FLUCTUATION DURING TEMPERATURE CHANGE	$\pm$ 0.15	-30, +60°C
OUTPUT SHIFT NULL POINT HIGH-LOW TEMP.	0.35	-30, +60°C
OUTPUT CHANGE START-END OF TESTS	0.10	

NOISE 0.2 γ 10H BANDWIDTH

REFERENCE N69 33967

Figure 23..

1 - 1691

2 AXES MAGNETOMETER

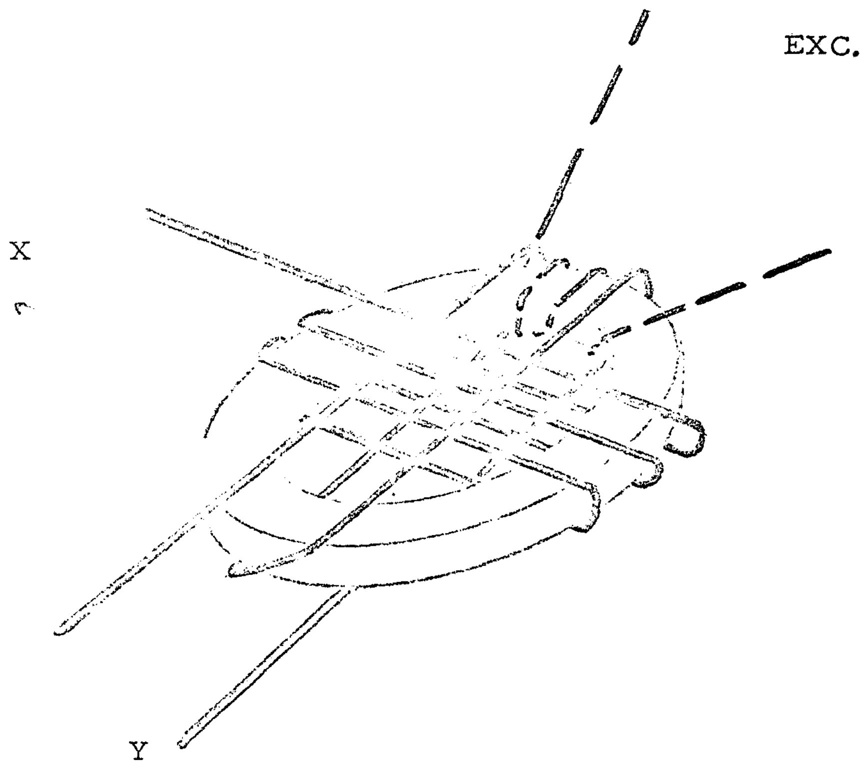
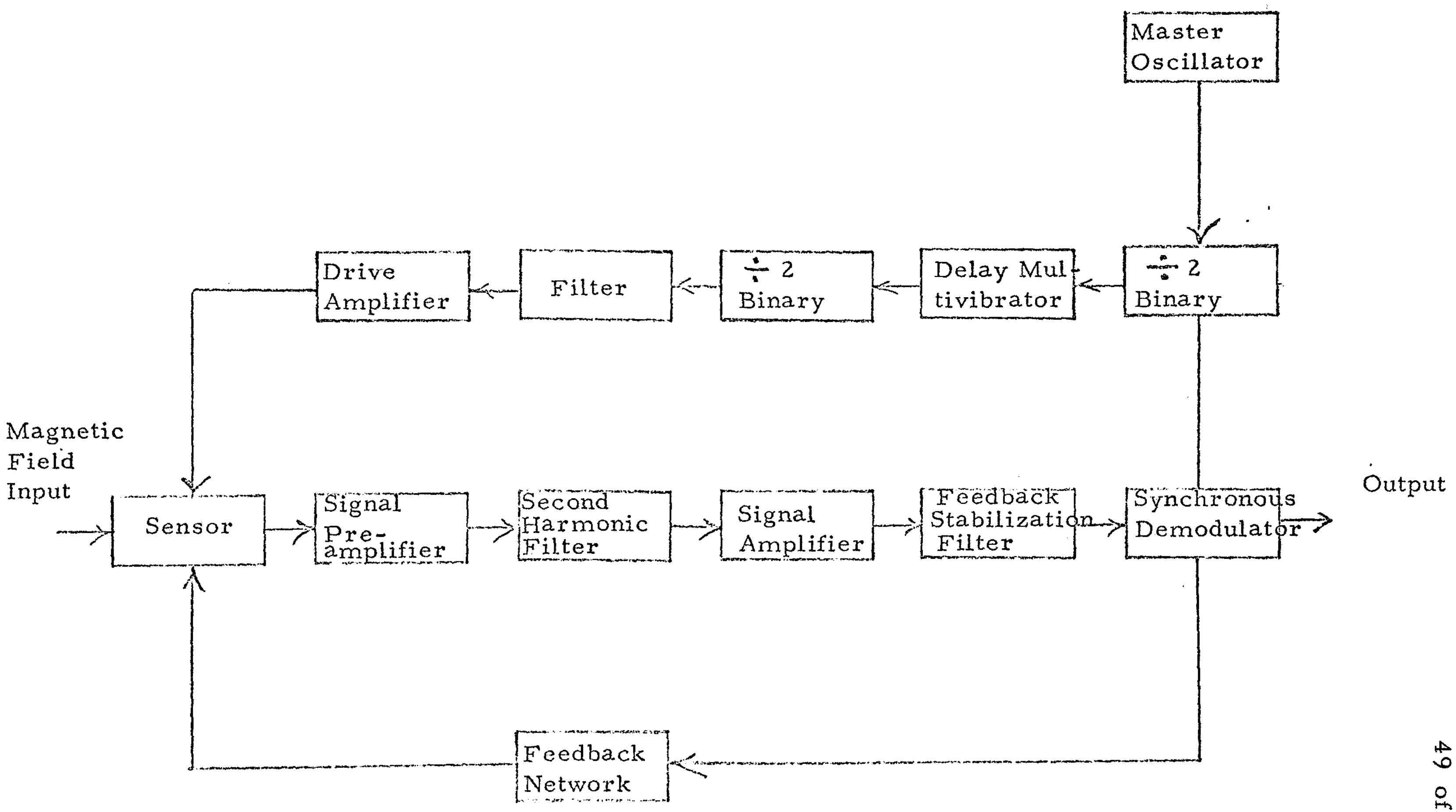


Figure 24.



Functional Block Diagram of the Single Axis Sensor
Electronics Subsystem

Figure 25.

Terrestrial Magnetic Field

Extensive observations of the magnetic field on and above the Earth's surface have been made for more than a century. Recent data from aircraft and space exploration have changed the concept of the geomagnetic field from one of a simple dipolar field to one that is described by a spherical harmonic expansion model that requires 120 coefficients.

The model becomes quite effective at normal orbital altitudes since the local disturbances such as ore deposits become less of a disturbance. The effectiveness of the model can be seen in Figure 26. where the difference between observed and computed values is shown as a function of the number of terms in the model. Since the total magnetic vector has a magnitude of $>$ 30,000 gamma, the residuals in Figure 26 could represent an angular uncertainty in the order of $\frac{30}{30,000}$ radians = 1 milliradian $\cong$ 3 $\widehat{\text{min}}$.

The improvement with model complexity is listed in Table V.

There are temporal variations in the field both short and long term. There is a great deal of detailed knowledge of these variations in the listed references.

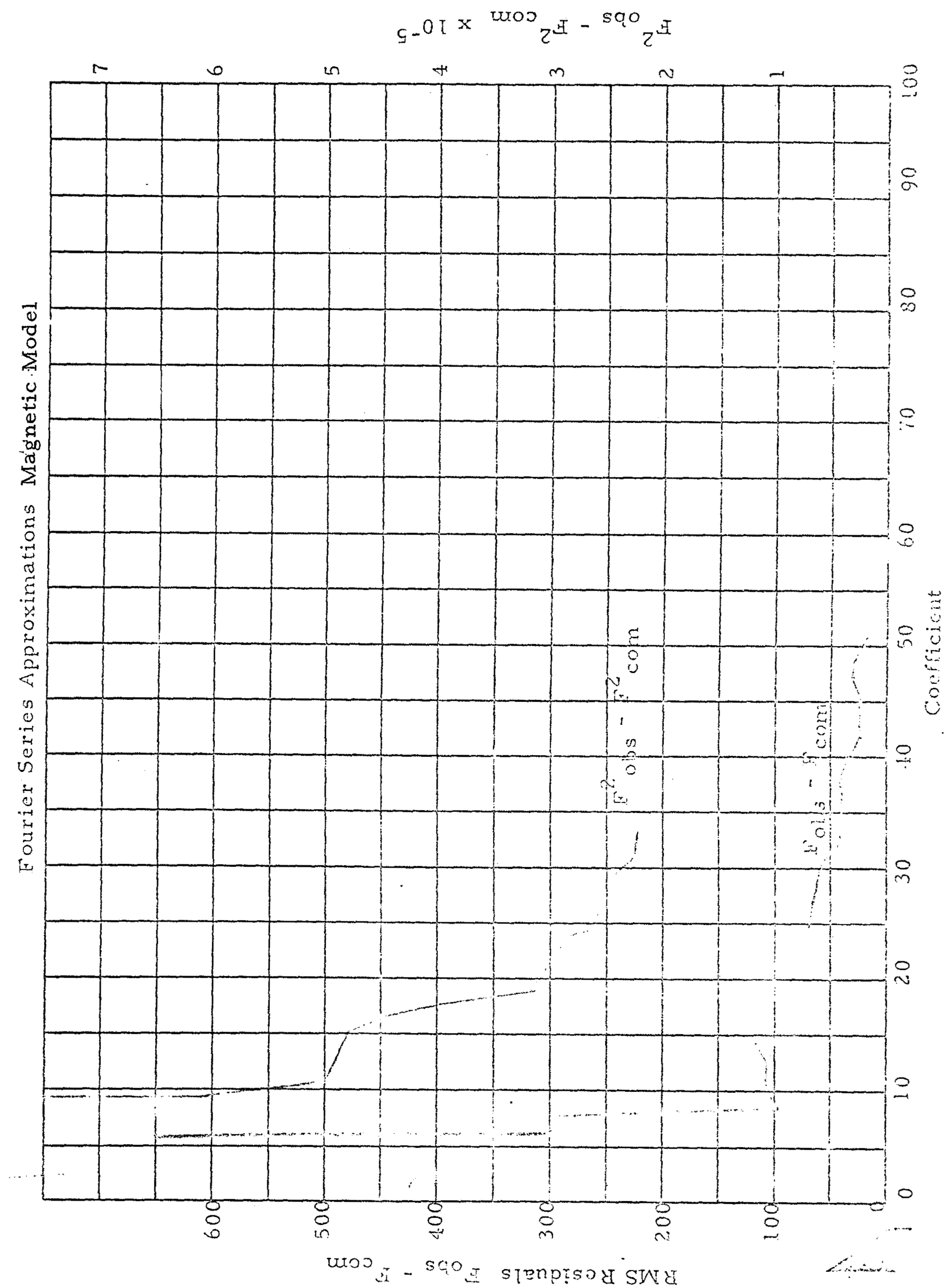

Fourier Series Approximations Magnetic Model
$F^2_{obs} - F^2_{com} \times 10^{-5}$
1
2
3
4
5
6
7
RMS Residuals $F_{obs} - F_{com}$
0
100
200
300
400
500
600
Coefficient
0
10
20
30
40
50
60
70
80
90
100
$F^2_{obs} - F^2_{com}$
$F_{obs} - F_{com}$

Figure 26.

GEOMAGNETIC FIELD MODELS

DIPOLE MODEL (SPIN AXIS) LEAST ACCURATE	20 degrees
DIPOLE MODEL (TILTED 11.4°)	7 degrees
SPHERICAL HARMONIC 120 COEFFICIENTS	0.1 degrees

TABLE V

References

Stability Measurements of Fluxgate Magnetometers

N69-33967

A Highly Stable Fluxgate Magnetometer for Space Exploration

N69-33968

Magnetic Fields - Earth and Extraterrestrial

NASA SP-8017
March 1969

The World Magnetic Survey, Goddard Space Flight Center

X-611-63-107
May 1963

5. Rate Integrating Gyro

The operation of a single degree of freedom gyro is expressed by Torque = Input Rate x Angular Momentum. Thus, if an input rate is applied to an unrestrained gyro, the output axis will accelerate. In the rate integrating gyro the output axis is restrained by a fluid which generates a restraining torque proportional to the velocity of the output axis. This effectively performs an integration; therefore, for a rate input to the gyro, the output axis will develop a proportional rate.

It is not feasible to make such a gyro without elements that produce anomolous torques that are time independent, elastic in nature or proportional to case acceleration and in fact each is present to some extent in any gyro. These torques can be measured and the gyro calibrated. The quality of the gyro is not necessarily specified by the magnitude of these torques but rather to their stability.

In the more exotic class of gyros, these torques are stable enough to permit performance limits in the order of 5×10^{-9} rad sec^{-1}. However, the same gyro after being subjected to the boost environment will degrade by an order of magnitude and after a couple of months of operation can degrade further by another order of magnitude.

Some of this performance capability can be recovered by simple on-board calibration. Some of the possible on-board calibration techniques are:

1. Turn of wheel power and measure output when wheels are stopped.

2. Reverse the rotation of the wheels. Since the torque due to orbital rate will reverse but most of the anomolous torque will not, they can be separated.

3. Reverse the direction of the gyro input axis. This requires a motorized mechanism but will perform the best calibration.

However, performance in the order of 5×10^{-7} rad sec^{-1} (3×10^{-5} degrees sec^{-1}) is probably adequate for the majority of the spacecraft experiments.

It may be advantageous to utilize a lower class gyro for the sake of size, cost and power and improve performance with one of the on-board calibration schemes listed above.

Gyro Compassing

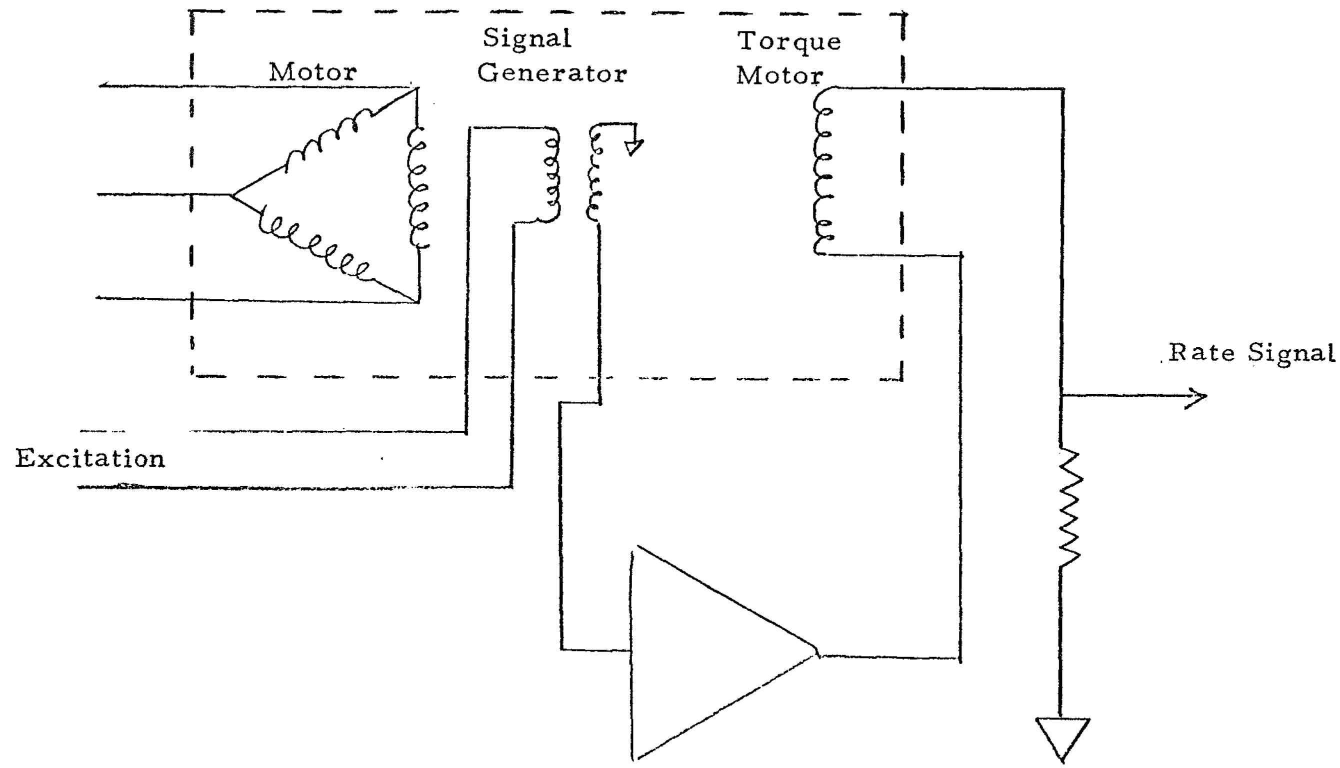
A rate gyro or a rate integrating gyro configured in a rate mode (Figure 27) can be implemented so that its input axis is along the spacecraft roll axis. This mode of operation is often referred to as gyro compassing because of the similarity to this form of use on the Earth.

When the roll axis is along the velocity vector, this gyro will sense no component of orbital rate. If the yaw error is 90 degrees, then the gyro will sense full orbital rate which is in the order of 0.05 degrees sec^{-1}.

$$E_y = w_{orbit} \sin \psi$$

A simple spring restrained rate gyro has an operational threshold at best of .01 deg sec^{-1}; therefore, the indicated yaw would be uncertain by 12 degrees.

Rate Integrating Gyro
in Rate Mode



$$W_n = \sqrt{\frac{S.G. \times Amp. \times Torque}{Gimbal\ Inertia}} > 100\ Rad\ Sec^{-1}$$

Figure 27.

A miniature rate integrating gyro operated in a rate measuring mode has an operational threshold of less than 3×10^{-4} deg sec^{-1}; therefore, the yaw uncertainty will be less than 0.3 degrees. It is possible to implement on-board calibration techniques that could improve this performance by more than two orders of magnitude.

Angular Increment Command

A rate integrating gyro is fitted with a torque generator on its output shaft. Thus, an electric current applied to the torque motor will develop a proportional torque on the output shaft. This torque is equivalent to a precessional torque that is developed by an input rate to the gyro. Torque on output shaft equals the product of the gyro angular momentum and the angular input rate.

$$T = W \times H$$

Thus, the integral of the gyro torque motor current is proportional to an angular increment input.

If the output of the gyro is used in the ACS in a closed loop manner, then a very precise angular increment or angular rate can be commanded by an appropriate input to the gyro torque generator. This is shown for one axis of the ACS in Figure 28.

If some arbitrary angular increment is desired, then it can be accomplished by commanding the three gyros in sequence. This technique will avoid the complexity of Euler relationships and resulting transformation difficulties.

Gyro Control

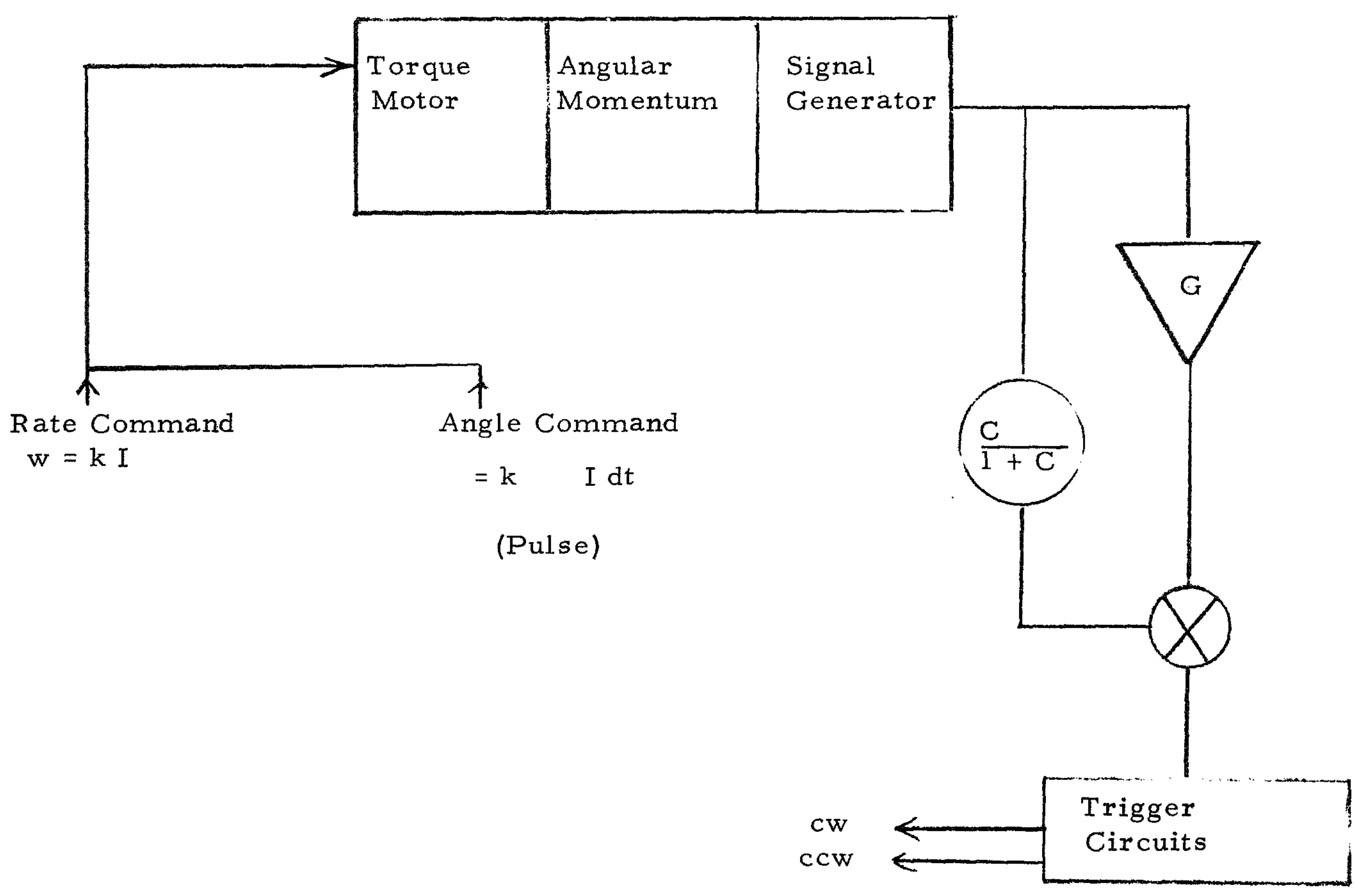


Figure 28.

Gyro Availability

There are a host of proven gyros made by several vendors that are suitable for this application. Therefore, a single reasonable candidate is specified. There is a wide range of sophistication available in rate integrating gyros with a corresponding range of performance. The more exotic gyros are temperature controlled which may be an unreasonable power drain for some spacecraft missions. Therefore, the illustrated gyro is one not designed to be operated at a fixed temperature.

Random Drift	2.7×10^{-4} deg sec^{-1}
Absolute Drift	1.4×10^{-3} deg sec^{-1}
Angular Momentum	5×10^{4} gram cm^{2} sec^{-1}
Gain	.5
Torquer Scale Factor	.4 deg sec^{-1} ma^{-1}
Torquer Linearity	0.2%

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X]

KA-70-22

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	K. Arnesen		14 July 1970

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-L91-0041		

SUBJECT

Cost Considerations of Data Handling Options

DISTRIBUTION

List "B"

SIGNED K. Arnesen

APPROVED E. Donadio

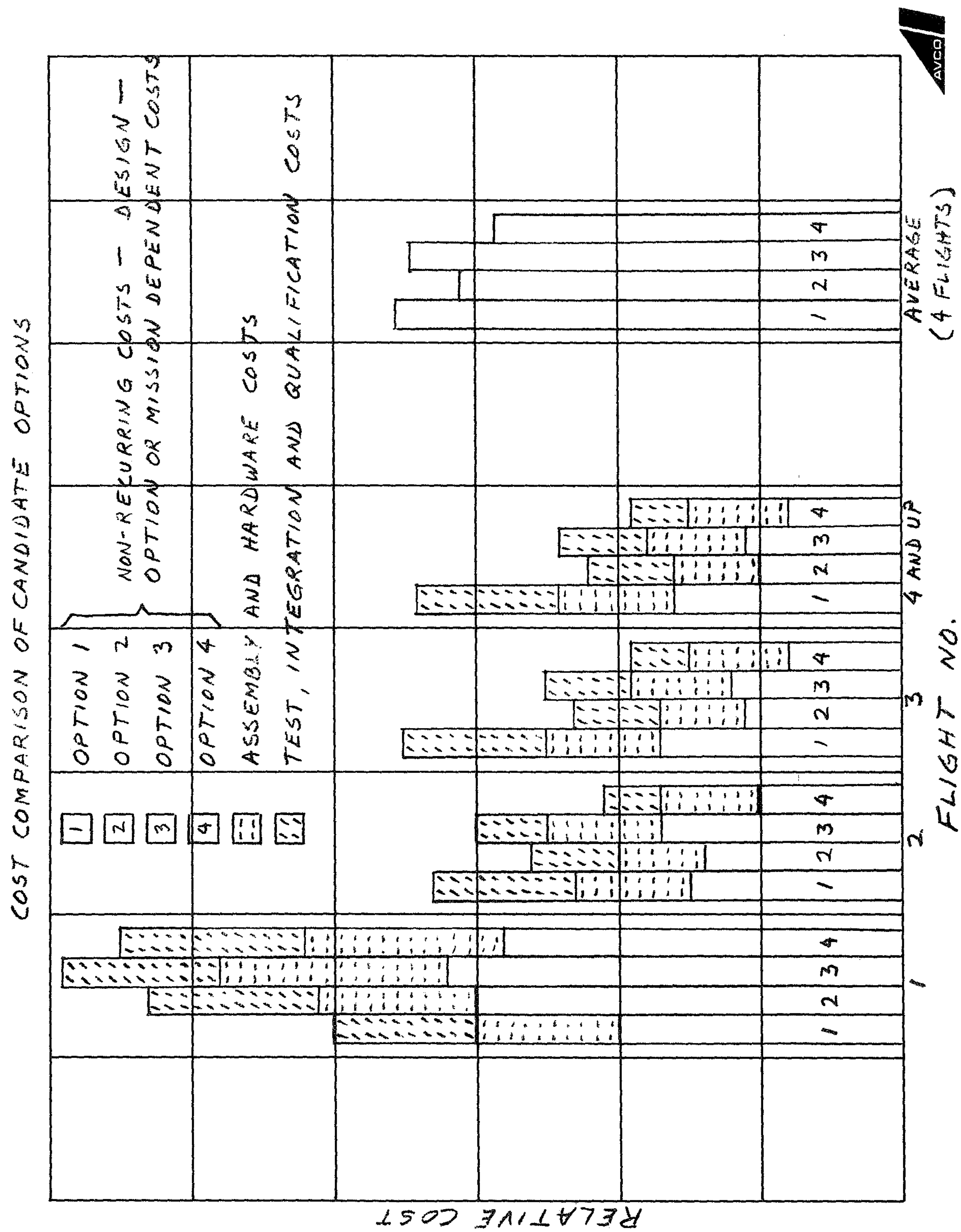
NFORMATION REQUESTED / RELEASED

Reference: KA-70-20, 7 July 1970 - Data Handling Options

The following TR considers the various factors contributing to the final cost of a fully qualified data handling unit installed and tested on a spacecraft. The four data handling options outlined in the referenced TR have been compared up through flight number 4. Costs for flights beyond 4 will remain essentially constant, except for option number 1 since that particular option always calls for a mission peculiar design.

The first flight for all four options will require at least two complete units to be built. From the second flight on, only option number 1 requires two units, while the other options generally will require one unit to be built, while the flight spare from the previous flight can be retrofitted as a spare.

The cost comparisons assume that the experiment data to be handled are not drastically different from mission to mission. Furthermore, it is assumed that the experiment data will consist mostly of a limited number of analog channels to be sampled at a relatively low rate (10^4 samples/second total or less) with resolution of 10 bits or less (0.1%).


COST COMPARISON OF CANDIDATE OPTIONS
1 OPTION 1
2 OPTION 2
3 OPTION 3
4 OPTION 4
NON-RECURRING COSTS — DESIGN — OPTION OR MISSION DEPENDENT COSTS
ASSEMBLY AND HARDWARE COSTS
TEST, INTEGRATION AND QUALIFICATION COSTS
1 2 3 4
1
1 2 3 4
2
1 2 3 4
3
1 2 3 4
4 AND UP
1 2 3 4
AVERAGE (4 FLIGHTS)
FLIGHT NO.
RELATIVE COST

In addition, attitude determination and housekeeping data will be handled with the same unit.

The typical data handling unit will have a memory. Storage may or may not be required for the experiment data, but "snapshots" of attitude determination data may turn out to be highly desirable through the orbit.

Important Cost Considerations

Option Number 1

Two complete units must be built every time.

There will be a learning curve involved, but new design and a new set of drawings will be required for each flight.

After the first couple of flights, the required lead time might be a factor of two higher than for option 4. Lead time reflects in higher costs if any changes are required, and may severely affect costs in other areas of the spacecraft.

Note: It is reasonable to assume that the typical mission will require storage of one kind or another. For that reason, the spread in cost (initially) between the various options may actually turn out to be less than that shown.

Option Number 2

This option might typically require 2 full systems to be built initially. Later on, this can probably be reduced to perhaps 1 1/4 or less.

The relatively large wiring harness with associated connectors will probably affect system reliability more than cost, but does increase the required lead time and perhaps some costs not directly related to the data handling unit.

The modular design would probably appear more attractive if memory storage were not a requirement for many of the missions.

Hardware and assembly costs will be relatively high because of the circuitry which handles nothing but interfaces between modules. Integration costs will tend to go up because several modules will have to be considered rather than just one unit.

Option Number 3

This particular option is probably the hardest one to justify because it lacks flexibility just like option number 1, and would tend to be large and heavy like a system built with option number 2 approach. It is also larger than option number 4, although it is equally easy to integrate mechanically.

6-0446
9-68

TECHNICAL REQUEST/RELEASE	FROM K. Arnesen	Page 4 of 4 DATE 14 July 1970

Option Number 3 (Cont'd)

It is possible to make an interesting observation at this point:

Option number 2 will, if the level of modularity becomes sufficiently low, become option number 1. If the level of modularity changes in the more complex direction, and the "modular" approach finally becomes a single module, the result will be option number 3.

Costs associated with option number 3 will tend to be high because the design time remains relatively high while the assembly and hardware costs are higher than for all the other options.

Indirect costs associated with the relatively long lead time requirement will also be fairly high.

Option Number 4

The initial definition and design of a data handling subsystem based on this option will be slightly easier than the designs based on options 2 and 3, but will be more complex than for option number 1. Similarly, assembly and hardware costs as well as test, integration and qualification costs will follow the same pattern.

It is assumed that two complete units have to be built for the first flight. Follow-on flights, however, require only one unit per flight since the left-over unit from the first flight can be retrofitted as a flight spare for any number of flights. Hardware changes, if any, will be insignificant.

This option is expected to represent the lowest cost in the long run, and for follow-on flights it also represents the shortest lead time. Since each unit will be mechanically identical to the previous unit, mechanical integration will be simplified.

In the case of option number 4, it is very likely that add-on modules may be desirable in the following areas:

1. Additional memory capacity (plated wire or core)

2. Input/output circuits

The circuitry to handle these additional modules will be contained in the basic data handling unit. This added circuitry does not represent a significant portion of the total cost.

Other types of storage, such as tape, can, of course, be used with any of the four options.

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒

KA-70-23

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	K. Arnesen		8-3-70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-L91-0041		

SUBJECT
Data Handling - Scope of Study, Accomplishments, Summary of Results and Key Parametric Data

DISTRIBUTION	
List "B"	SIGNED K. Arnesen APPROVED S. J. Brzeski

INFORMATION REQUESTED / RELEASED

References: TR KA-70-20; 7 July and TR KA-70-22; 14 July 1970

The following TR is intended to outline the scope of the study in the area of data handling and programming. It also summarizes the accomplishments to date.

This is followed by a brief description of the recommended design approach and some of the conclusions that were made as a result of the study.

Finally, this TR contains a summary of key parametric data that led to the recommended approach.

Scope of Study

1. Analysis of data handling requirements

   a) Experiment requirements (from list)
   b) Attitude determination
   c) Housekeeping

2. Identify major data handling options

   a) Mission-peculiar designs
   b) Modular-universal designs
   c) Single-unit universal design with mission-peculiar features handled in hardware
   d) Single-unit universal design with mission-peculiar features handled in software

3. Evaluate the major data handling options

   a) Cost considerations
   b) Hardware considerations
   c) Lead-time considerations
   d) Effect on other subsystems

4. Select the recommended design approach

   a) Define the design concept
   b) Synthesize a design to the point where size, weight and power can be estimated with reasonable confidence

5. Generate a preliminary interface "specification" between the data handling subsystem and experiment (as well as other subsystems).

6. Generate design data sheets for the data handling unit as well as for major components of the subsystem (i. e. ADC, Memory).

7. Outline programming requirements

   a) Pyrotechnics
   b) Deployment
   c) ACS
   d) Experiments

<u>Basic Assumptions in the Data Handling Area:</u>

1. The "universal" data handling unit does not necessarily have to be capable of handling all experiments on the list furnished by LRC.

2. The reference concept will be STADAN compatible.

3. Experiments not compatible with the basic concept can be "flagged" as requiring special handling.

6-0446
9-58

TECHNICAL REQUEST / RELEASE	FROM K. Arnesen	 DATE 8-3-70

Work Accomplished to Date:

Items 1, 2 and 3 are essentially complete.

Item 4 is in process. A design approach has been selected, and a preliminary design definition has been made. Preliminary size and weight estimates also exist. More work needs to be done on the synthesis of the design.

Items 5, 6 and 7 have to be done.

Summary and Conclusions:

Analysis of the data handling requirements shows that the "common denomination" for various missions will be sufficiently pronounced so that a single design can be synthesized. The recommended design approach for the data handling system is based on a "universal" concept where changes from mission to mission can be accomplished in software rather than in hardware.

A memory will be used for program storage as well as for data storage. The reference design will make use of a relatively "small" memory of perhaps 1K X 16, while missions where additional storage is required can add on memory as desired.

The subsystem will also include a 10-bit successive approximation ADC. The analog multiplexer will have add-on capability to handle the more complex experiments.

This design approach is expected to lead to the lowest cost and the shortest lead times after the first mission, while permitting a standard interface between experiment and data handling subsystem.

Key Parametric Data Leading to the Recommended Approach

1. The data handling requirements do not vary greatly from mission to mission for most of the flights.

2. The attitude determination requirements make a solid state memory highly desirable. Consequently, it makes sense to base the design on the use of a stored program, since additional memory comes relatively cheap.

3. The lead-time required for a given mission will be short because only simple software changes are required.

4. The cost of a "universal" data handling system with changes taking place in software will be lower in the long run.

5. Test, qualification and integration after the first unit will become more or less routine.

6. It will be possible to standardize the interface between experiment and data handling system with a few exceptions.

7. Compliance with STADAN and special program requirements can be assured only by having the spacecraft integration contractor responsible for the data handling subsystem specification. The experimenter is generally the one who is the least equipped to specify and design his own data handling unit.

8. If desired, re-programming in orbit can be accomplished. This may turn out to be a highly desirable feature in the case where an experiment encounters unpredictable phenomena.

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒

KA-70-20

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	K. Arnesen		7 July 1970

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-L91-0041		

SUBJECT

Data Handling Options

DISTRIBUTION

List "B"

SIGNED K. Arnesen

APPROVED S J Brzeski

INFORMATION REQUESTED / RELEASED

The following TR summarizes in outline form, the basic data handling options that can be considered for the study of the Attitude Control of Small Satellites and Related Subsystems. The brief description of each of the four basic alternatives is followed by a listing of advantages and disadvantages for each option.

The next TR will consider projected costs for each of the four options. It is expected that a recommendation can be made to select one of the four options, perhaps in modified form, based on cost as well as other considerations.

Data Handling Basic Options:

1. Design from scratch for each different experiment. This will more than likely result in minimum size and weight on a particular mission.

2. Modular design where a given data handling subsystem can be configured using "standard" modules.

3. "Universal" design capable of handling a variety of missions. Changes from one mission to the next will be accomplished by changing interconnections, etc. (Hardware Changes).

4. "Universal" design capable of handling a variety of missions. Changes from one mission to the next will be accomplished primarily in software. Some modularity may be desirable.

Option No. 1

Advantages:

1. Lowest size, weight, power.
2. Low non-recurring costs initially.
3. Easy to specify.

Disadvantages:

1. Separate designs for each flight.
2. Each unit must go through full qualification cycle.
3. Different test procedures for each unit.
4. Longer lead-times for follow-on units.
5. More units may have to be built of each kind.
6. No cost savings in the long run (probably).

Option No. 2 - Modular Design

Advantages:

1. The major part of the design is done only once.
2. Each type of module will have to go through full qualification tests only once.
3. Quick turn-around time after the initial design.
4. Only one module of each type required as spare for each flight.

Disadvantages:

1. To be truly universal, the design must be modular at a very low level of complexity.
2. Size and weight will tend to be high in most cases.
3. A substantial inter-module wiring harness is required increasing number of connectors as well as number of connections.
4. The interface between the various modules will be difficult to specify, in particular, if a given interface has to accommodate more than one type of module combination.
5. The recurring costs will be high, in particular, for the more complex combinations of modules.
6. A given task will require more electronic hardware than that required with Option No. 1, primarily because some electronics handle nothing but the interface between modules.

Option No. 3 - "Universal", - Changes in Hardware

Advantages:

1. The major portion of the design is done only once.
2. The unit will have to go through complete qualification tests only once.
3. Mechanical integration will be drastically simplified for follow-on units.
4. Electrical test and integration will be simpler and easier with each successive unit.
5. The flight spare can be retrofitted for succeeding missions with relative ease.

Disadvantages:

1. High initial as well as recurring costs.
2. Costly in terms of weight and volume.
3. Very difficult to specify and design, - requires long lead-time initially.
4. Retrofitting from one mission to the next requires design time as well as hardware changes.
5. The design will be paced by the most complex mission. Much of the hardware will not be required on a majority of flights.

Option No. 4 - "Universal", - Changes in Software

Advantages:

1. The design effort will be nearly complete with the first unit.
2. The unit will have to go through complete qualification tests only once.
3. Mechanical integration will be drastically simplified on follow-on units.
4. Electrical test and integration will be simpler and easier with each successive unit.
5. Retrofitting for different missions require only software changes which can be accomplished in a very short time period.
6. Since the concept is based on the use of a memory, additional storage for buffer and/or data applications will come cheap!
7. If desired, reprogramming in orbit can be accomplished at a low cost in terms of heardware, complexity and cost.

Disadvantages:

1. High initial and recurring hardware costs.
2. Relatively costly in terms of weight and volume for the simpler missions.
3. Requires a relatively long initial definition and design time period.

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X]

KA-70-24

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	K. Arnesen		8-14-70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control of Small Satellites and Related Subsystems	W159-L91-0042		

SUBJECT

STADAN Compatibility - Data Handling

DISTRIBUTION

List "B"

SIGNED K. Arnesen

APPROVED SJ Brzeski

INFORMATION REQUESTED / RELEASED

Reference: W. J. Kubicki, Trip Report - To: S. J. Brzeski, 12 August 1970 - F440-WJK-4093

The following TR will list questions concerning the data handling area which were asked in the meeting at GSFC on 10 August 1970. The answers to the various questions are outlined along with comments about features that are not necessarily spelled out in documents. In addition, changes that are anticipated in the STADAN capability have been described.

It can be concluded that STADAN permits a wide range of variations in rates as well as format, so that compatibility generally should pose no problem.

STADAN COMPATIBILITY

Questions were asked in the following areas:

1. Modulation

Q: What types of modulation are acceptable?

A: The following types can be handled with present equipment:

1. NRZ-C
2. NRZ-M
3. Split Phase

Q: What are the requirements on waveform symmetry?

A: The PCM telemetry standard specifies that the symmetry shall be maintained within 2% of the nominal bit period as measured at the telemetry receiver output. In practice, symmetry is generally measured at the transmitting end. Actually, the waveform symmetry is not very critical, and would be of importance only under very marginal conditions.

Comments: The waveform symmetry may become a problem when transmitting NRZ. If the first bit in the transmitted word appears as a result of a parallel transfer, the symmetry spec may be violated if the transfer pulse has been sufficiently delayed from the previous shift pulse. A way of getting around the problem is to provide one additional bit of delay, in other words, "de-glitching" the output.

There is no problem with a split phase transmission, since the split phase encoder generally operates clock-synchronous.

2. Bit Rates

Q: What range of bit rates can be accepted?

A: The PCM telemetry standard specifies a range from 1 bps to 200, 000 bps. This is still the limitation at VHF, but is not necessarily true at higher frequencies (400 MHz, 1700 MHz or S-band). The present equipment is fully capable of handling bit rates up to 1 MHz. By 1972-73 it is probable that rates in the 1-5 MHz range can be handled at the site.

Comment: With one or two exceptions, this means that all experiments on the LRC list can be handled without any problems. It may be desirable to bring the nominal maximum rate capability of the data handling and telemetry equipment up from 200 KHz to 1 MHz.

Q: What is the tolerance on the transmitted bit rate?

A: The long term stability requirement (1 year) is ± 5% of bit rate. Short term (5 minutes) the requirement is for less than ± 1/2% of bit rate. There is also a requirement that instantaneous (i.e. peak-to-peak tape recorder flutter) has to be less than 3% in a bandwidth wide enough to include all significant components (nominally 600 Hz).

Comment: The bit rate stability requirements are primarily based on what the bit synchronizer can tolerate. In general, the requirements are such that a tape recorder can be used. Whenever there are problems establishing bit sync, it is generally because the tape recorder is out of spec. This implies that tape recorders should be avoided if at all possible. Without a tape recorder, the bit rate stability requirement is very easily met.

Q: Can the bit rate be varied by means of ground commands?

A: Yes. This is quite commonly done.

Q: Can the bit rate be varied by means of on-board programming with proper flags in the telemetry format?

A: STADAN does not have such facilities at the present time, and none are planned. At the present time, the bit rate switching is done manually. It is certainly possible to do the switching automatically, but this has generally been considered undesirable.

Comment: It is highly unlikely that automatic bit rate switching will be required for any of the experiments on the LRC list. If, however, adaptive data handling ever becomes a necessity, bit rate changing by automatic means would be highly desirable.

3. Format

Q: What is the maximum acceptable word length?

A: The maximum word length is 32 bits, and there are no plans to change this.

Comment: 32 bits is more than sufficient for any small satellite data handling requirement.

Q: Can words be of variable length within a given frame format?

A: The equipment can currently handle only one word length in a given format, and there are no plans for any changes in that part of the PCM equipment.

Comment: Variable word length can, if required, be implemented using variable number of fixed bytes at slightly lower efficiency.

Q: Can a word consist of several syllables, and can these syllables have variable length?

A: Yes, to both parts of the question.

Comment: This is another way of implementing variable word length. The decommutating operation may eventually become a bit messy, however, it is expected that a limited amount of variable-syllable usage will be required or desired on a typical small satellite.

Q: What is the maximum acceptable frame length?

A: The maximum length of a minor frame is 8192 bits. A major frame consists of a maximum of 256 minor frames.

Comment: In general, this is more than sufficient to meet small satellite requirements. In the case of the HIRS experiment, however, it became necessary to stop the scan mirror momentarily in order to insert a sync code before the frame exceeded 8192 bits. If sufficient buffer storage had been available for the experiment, this momentary halt would have been unnecessary. HIRS on a small satellite with a truly versatile data processor would not have required any "hiccups" in the scan program.

Q: Can the frame format be changed by means of ground commands?

A: Yes.

Q: Can the frame format be changed by means of automatic on-board programming?

A: The equipment is capable of handling variable formats when there are proper flags in the telemetry format. However, such formats are not handled as a matter of routine, and a waiver will be required.

Comment: Variable formats under automatic on-board control may be highly desirable at times, such as when attitude determination data at regular intervals around the orbit alternates with occasional experiment data "snapshots".

Q: How many different subcommutators can be handled in any one given format?

A: The maximum number of subcommutators in a given format has not been specified. Actually, any reasonable number of subcommutators can be used.

Q: If there are two or more subcommutators, can they be of different length?

A: Yes, but they have to be integrally related. They must also be synchronized together.

Comment: The PCM specifications say that when two or more sub-multiplexers are used, they must have an equal number of channels or binary multiple. However, it was clearly stated that the ratio of the two does not have to be a binary number. If, for instance, the longest subcommutator is 45 frames long, other subcommutators could very well have lengths 3, 5 or 15. Subcommutators of different lengths will be used on MSS, and may very well be useful on other small satellites.

Q: Is a list of preferred synchronization codes available?

A: Yes. Appendix A to the PCM telemetry standards has a list of codes ranging from 7 to 30 bits.

Q: Do synchronization codes have to be an integral number of telemetry words?

A: No. The code can have any length. The left-over bits in the partial word would be marked with a "don't care" tag in the sync detector.

Comment: It may be desirable to use left-over bits to insert various flags into the telemetry format.

Q: Is there any standard for frame numbering, or can any convenient numbering system be used?

A: Any numbering system can be used.

Q: Can the frame number be placed anywhere in the format?

A: Yes.

Comment: Freedom of choice of numbering system as well as the positioning of the time code in the telemetry frame can very often lead to substantial on-board hardware savings and greater ease of data reduction on the ground.

Q: What must the capacity of the clock be to insure unambiguous labelling of the data?

A: The required capacity of the clock relative to the duration of the mission must be determined by the project office and not by STADAN.

4. Special Features

A number of the features of the STADAN system are not necessarily treated in the STADAN manual. If it is, it is quite often difficult to find the required information about the desired features.

a) Pre-detection Recording

STADAN does not use pre-detection recording, and it is not considered available. It was used several years ago on one satellite project, and the equipment is still around. After being idle for several years, the condition of the equipment is rather questionable.

The STADAN reasoning is, briefly, that if the signal from a spacecraft is so noisy that it is difficult to detect and establish phase lock and/or bit sync, it will be even more difficult to dig the signal out after recording, since the tape recorder adds even more noise.

b) Recording

When data is recorded, it is done after the detector but before the bit synchronizer.

c) Coding

STADAN will add convolutional coding equipment, primarily to handle data from IMP-I (Eye). The decoding will not be done in real time. Also, the decoding will be done on the detected signal.

d) Real-time Data

Up to 20 words can be selected from the telemetry format for real-time look, but STADAN will probably be unwilling to assign all 20, wanting to keep some spares.

The real-time data can be transmitted anywhere at a maximum transmission line bit rate of 2400 bps including the required message headers which must be inserted in the format. This feature is desirable when the spacecraft is controlled from a location other than the tracking station.

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [X]

KA-70-25

TO	DEPT.	FROM	DEPT.	DATE
S. J. Brzeski	L910	K. Arnesen		8-19-70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control for Small Satellites and Related Subsystems	W159-L91-0042		

SUBJECT

Recommended Design Approach, Data Handling and Control

DISTRIBUTION	
List "B"	SIGNED S J Brzeski for K. Arnesen
	APPROVED S J Brzeski

INFORMATION REQUESTED / RELEASED

References: TR KA-70-20, TR KA-70-22, TR KA-70-23, TR KA70-24

The following TR discusses the recommended design approach for the data handling and control subsystem.

6-0446
9-58

TECHNICAL REQUEST / RELEASE	FROM K. Arnesen	Page 2 of 21 / DATE 8-19-70

TABLE OF CONTENTS

RECOMMENDED DESIGN APPROACH
DATA HANDLING AND PROGRAMMING

1.0 Introduction and Summary

2.0 Rationale and Key Parametric Data

- 2.1 Basic Choice: Experimenter or Spacecraft Contractor
- 2.2 Design Options
- 2.3 Cost Considerations
- 2.4 Other Considerations

3.0 Data Handling Requirements

- 3.1 Experiment Data
- 3.2 Attitude Determination
- 3.3 Housekeeping

4.0 Programmer Requirements

- 4.1 Pyrotechnic Programmer
- 4.2 ACS and Command
- 4.3 Systems Programmer

5.0 Conclusions

6.0 Data Handling and Programming Subsystem

- 6.1 General
- 6.2 Memory
- 6.3 CPU
- 6.4 ADC
- 6.5 Multiplexers
- 6.6 I/O
- 6.7 Programmer
- 6.8 Add-on Features
- 6.9 Arithmetic Unit

7.0 Hardware Considerations

- 7.1 Technology
- 7.2 Size, Weight and Power

RECOMMENDED DESIGN APPROACH DATA HANDLING AND CONTROL

1.0 Introduction and Summary

Previous technical releases have considered the various data handling options that can be selected for a small satellite designed to carry a single experiment (which can be selected from a large group of experiments). In addition, cost factors have been considered along with the effect the data handling choice has in other areas, such as test, qualification and integration costs as well as the way in which the data handling choice affects the cost of other subsystems.

It has been concluded that the most cost effective means of meeting the overall design goal will be to design a versatile data processor where changes from one mission to the next takes place primarily in software. Such a design can easily be made STADAN compatible, and can be designed such that nearly all experiments on the LRC list can be accommodated. In addition, a single data handling unit, capable of handling experiment data, will also be capable of handling attitude determination and housekeeping data.

The recommended design is based on the use of a solid state memory and a successive approximation analog to digital converter. Both of these components are available nearly off-the-shelf. The rest of the subsystem, although a substantial design effort is involved, does not require any technological breakthrough. As a matter of fact, once the data processor has been clearly specified, the design is very nearly a routine one.

This TR will give a general description of the recommended design, and will also discuss key parametric data and the rationale leading to its selection.

2.0 Rationale and Key Parametric Data

2.1 Basic Choice: Experimenter or Spacecraft Contractor

Before attempting to decide what a small satellite data handling unit should look like, there is another question that must be answered, namely: Who should have responsibility for the experiment data processor? The alternatives are the following:

2.1.1 Let the experimenter specify, design and build the data handling unit in-house or on subcontract. The spacecraft contractor will specify the following interfaces:

a) The interface between the experiment and the telemetry subsystem (transmitter/modulator) including the requirements imposed by STADAN as well as special data handling equipment specifically assigned to that particular spacecraft.

b) The interface between the experiment and the command subsystem.

c) Power interface

d) Mechanical interface

2.1.2 The second alternative will let the spacecraft contractor specify, design and build the data handling unit in-house or on subcontract. He would impose clearly defined interface specifications on the experimenter as follows:

a) The interface between the experiment electronics and the data processor. STADAN and special data handling requirements will be solely the spacecraft contractor's responsibility in this case.

b) The interface between the experiment and the programmer, particularly the command distribution unit.

c) Power interface

d) Mechanical interface

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

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

Many experimenters are capable of specifying and designing sophisticated data handling units. Most of them, however, do not have the necessary experience. In general, the experimenter is the one <u>least</u> equipped to specify and build sophisticated data processing hardware. Even if the experimenter processes all the required capabilities, the situation might very well arise where he does not have enough funding available to build the desired flexibility and sophistication into his equipment, and must be satisfied with a compromise. Such compromises may not be necessary if a flexible data handling unit is available as part of the spacecraft.

The integration and qualification of a spacecraft will be simpler and faster when the spacecraft contractor has responsibility for data processing. It is recommended, therefore, that the spacecraft contractor be given responsibility for experiment data handling up to a simple, well-defined interface.

2.2 Options

There are four basic data handling options that can be selected for a small satellite. These options were discussed in a previous technical release (KA-70-20 Data Handling Options) where the advantages and disadvantages of each option were discussed.

The first option, which amounts to a mission-optimized design, is very similar to the alternative discussed in Section 2.2.1. The result would be a different design for each flight.

The other three options represent three different ways of designing a "universal" data handling unit, capable of handling nearly all experiments on the LRC list. Of these three options, the last one undoubtedly results in the most flexible design. Before a selection can be made from the four options, cost and other considerations must be taken into account.

2.3 Cost Considerations

When attempting to compare the relative costs of the four data handling options, the following ground rules must be kept in mind:

a) The primary goal is to have a high mission cost effectiveness.

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

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

The comparative costs discussed in a previous TR (KA-70-22 "Cost Considerations") show that, although the initial costs are high, the costs for follow-on flights drop off very rapidly. The average cost for the first block of four flights shows the most flexible design to be the least costly. The second block, if shown, would have indicated even more dramatically the cost advantage of this option. Thus, cost considerations lead to the conclusion that the most cost effective data handling option is the most sophisticated and flexible one, assuming that a memory is required in any event. This assumption is based on the attitude determination data requirements which will be discussed in Section 3.2.

2.4 Other Considerations

A number of considerations other than those discussed already, tend to strengthen the conclusion that the fourth option, a "universal" design where changes from mission to mission take place in software, will result in the maximum return per dollar of investment.

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

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

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

3.0 Data Handling Requirements

The sources of data to be handled by the subsystem, falls into three categories. Each of these categories will be briefly described in the following paragraphs.

3.1 Experiment Data

The vast majority of the data to be transmitted by a satellite generally originates with the on-board experiments. This will generally be true also of a small test-bed satellite. For this study, the list of candidate experiments supplied by LRC was used as a guide.

Despite the wide variety of experiments on the list, it is possible to make a few observations that characterize nearly all of them from a data handling point of view:

a) Practically all experiments on the list have analog outputs from their sensors, and will require analog to digital conversion.

b) Although there is no information on the required amplitude resolution, it is expected to lie in the range from 6 bits for some imagery experiments to perhaps 12 bits or better for an IR sounder. A resolution of 10 bits is expected to satisfy nearly all experiment requirements.

c) The speed of the analog to digital conversion process will be generally compatible with a successive approximation converter. In the few cases where much higher speeds are required, analog-to-grey code or "cyclic" converters can be used.

d) Pulse counting requirements are the exception rather than the rule. On many satellites, pulse counting circuits in either the spacecraft or experiment data processors constitute a substantial portion of the overall data handling circuits. IMP-I (Eye), S^3 and MSS are examples of this. On MSS, the pulse counting circuits account for very nearly 50% of the total data handling and programming circuits.

e) Total experiment bit rate requirements are with few exceptions well below 100,000 bits per second.

f) Other than in the number of outputs to be sampled and the frequency of the sampling, the experiment data requirements do not vary greatly. It is, therefore, possible to conceive of a flexible data handling unit which is capable of handling nearly all experiments on the list. The few experiments that cannot be handled with standard hardware can be "flagged" as requiring special treatment, or in some cases, perhaps, as being incompatible with a small satellite concept.

g) The small number of experiment data "classes" makes it possible to conceive of a standard, simple and straight forward experiment interface, capable of handling nearly all experiments on the LRC list.

3.2 Attitude Determination Data

The attitude determination data will be obtained from a set of instruments which, in the case of the reference design, consists of:

1. Horizon sensor
2. Ion sensor
3. Sun sensors
4. Gyro (optional)

The primary instruments will be the horizon sensor and the ion sensor.

The output from the horizon sensor will consist of pulses whose timing (relative to a reference) can be used to determine the attitude of the spacecraft about the pitch and roll axis. The data handling unit must be capable of determining the time of occurrence of these pulses to the required precision. The information will be telemetered, but may also be used by the attitude control portion of the programmer.

The ion sensor will have an analog output whose magnitude and polarity will indicate how well the spacecraft is aligned along the velocity vector. This sensor, in other words, will give information about the yaw axis attitude. The sensor output will be digitized and transmitted by the data handling unit, and will be used in either analog or digital form by the attitude control portion of the programmer for yaw axis control.

There will also be a total of 8 coarse sun sensors whose output will be analog. These are used for coarse attitude determination only, and will be telemetered. The sun sensors, however, will not be used by the programmer for attitude control.

The output from the rate gyro will consist of analog voltages giving rate information for each of the three axes. The outputs will be digitized and telemetered, and will also (in the rate mode) be used by the programmer for attitude control.

3.3 Housekeeping Data

Most of the housekeeping data is expected to be very similar to the engineering and diagnostic data which traditionally has been transmitted from satellites. The data will mostly be in analog form, but some signals, such as switch positions and status flags may be bi-level. In addition, it will probably be desirable to keep a running check on the program which has been stored in memory.

4.0 Programmer Requirements

The spacecraft programmer, which should not be confused with that portion of the data handling unit acting on the stored program, will be designed to provide control signals for the spacecraft in the following areas:

1. Pyrotechnic sequencing and control
2. Attitude control
3. Command distribution
4. Systems programming

Each of these areas will be briefly described in the following paragraphs.

4.1 Pyrotechnic Sequencing and Control

The pyrotechnic control circuits will generally be armed upon separation from the booster. The circuits must incorporate fail-safe design features to guard against premature firing of explosive devices, and must, at the same time, have a probability near unity of actuating these devices at the proper time. The timing of the actuator may, in some cases, be controlled by a separate sequencer which is part of the pyrotechnic control unit. In other cases direct ground commands may be used, or delayed commands, stored with the proper time tag in memory, may be used.

Pyrotechnic events may include deployment of solar paddles, deployment of experiment booms, yo-yo despin, blowing off experiment covers, etc. In addition, the attitude control subsystem will include some squib-operated valves.

4.2 Attitude Control Electronics

The attitude control electronics portion of the programmer will include the circuits necessary to:

a) Control the gravity gradient boom, either manually or automatically

b) Control the thrusters which are part of the cold gas attitude control system in three different operating modes:

1. Coarse pointing automatic operating mode
2. Fine rate controlled automatic operating mode
3. Manual operating mode

It is expected that the electronic "trigger circuits" which actually operate the ACS valves will be located near these valves in a separate package. The interface between the programmer and the trigger circuits will consist of low-level on/off signals.

4.3 Command Distribution

The command distribution unit will receive commands from that command decoder in real time or from the data handling unit in the case of stored commands. The commands will be standardized, depending on the type of command required, to either pulses or levels, and be distributed around the spacecraft to the experiment or to the various subsystems requiring commands.

In the case of commands to be stored in memory, however, the proper time tags will have to be included, and these commands will go direct to the data handling unit.

4.4 Systems Programmer

Into the systems programmer will be incorporated all the odds and ends that don't belong in any of the other programmer components. Included will be subsystem mode switching, power switching and various power control override functions, as well as the generation of various experiment control signals.

The interface between the systems programmer and the other subsystems on the spacecraft (including the experiment) will be either a power command, obtained from the power bus through a switch or latching relay, or it will be a low-level control signal.

5.0 Conclusions

As a result of the rationale which has been developed, and the parametric work that has been done, the following major conclusions can be drawn:

1. The responsibility for the specification and design of the data handling unit must lie with the spacecraft contractor and not with the experimenter.

2. A versatile data handling unit, based on the use of a solid state memory, and where changes from one mission to the next takes place primarily in software, has been recommended for the following reasons:

a) It is the most cost effective design approach
b) It is the only approach which is compatible with the concept of a versatile, yet cost effective "test bed" spacecraft
c) The data handling requirements show a substantial "common denominator" despite the wide range of experiments which were considered.
d) Such a data processor can be designed with current technology. No technological breakthrough is required.
e) Major components, such as memory and analog to digital converters, are available essentially "off-the-shelf".

3. A versatile design approach will permit add-on capabilities, primarily in the multiplexer and memory areas.

4. The recommended approach will permit the definition of a simple and straight forward interface between experiment and data handling unit.

5. In-orbit reprogramming may, in the case of many experiments, turn out to be highly desirable, particularly where an experiment encounters unpredictable phenomena while in orbit.

6. Reprogramming, as well as spacecraft control, can be performed using an off-the-shelf 70-command tone-digital command decoder. If it is anticipated that reprogramming will be performed very often, or if a very large number of commands will regularly be transmitted during a pass, a much more expensive PCM command system could be contemplated.

7. Compliance with STADAN as well as with special program-dedicated ground data handling equipment can be assured.

8. It makes sense to consider the spacecraft programmer as part of the data handling subsystems, since many programming tasks are similar in nature to data handling chores.

9. As far as the single experiment is concerned, a versatile data processor can, by means of software changes, be made to look like the data handling unit on any operational spacecraft. Slight hardware modifications might be required in the input/output unit.

6.0 Data Handling and Programming Subsystem Description

In the following paragraphs a general description will be given of the recommended data handling and programming design concept. It is recommended that in a future study, this concept be used to develop a set of specifications from which hardware development can take place. In this study, only a functional description will be given of the subsystem and its components.

6.1 General

The recommended data handling concept shown in Figure 6.1 is in reality a flexible data manipulator whose operation is determined by a program stored in a high speed, solid state memory. If optional arithmetic capability is used, the unit becomes a fairly simple computer. Computational capability would be limited to fairly simple operations such as averaging or digital integration. Multiplication capability might also be considered.

The unit would contain an analog to digital converter with 10 bit resolution, having an analog multiplexer with sufficient capacity to handle all spacecraft engineering and diagnostic data as well as a limited number of experiment data channels.

The solid state memory would nominally have a capacity of 1024 words X 16 bits, and be capable of storing a program controlling sampling and formatting. In addition, it would provide buffer storage for data as well as storing commands or command sequences to be executed at a later time.

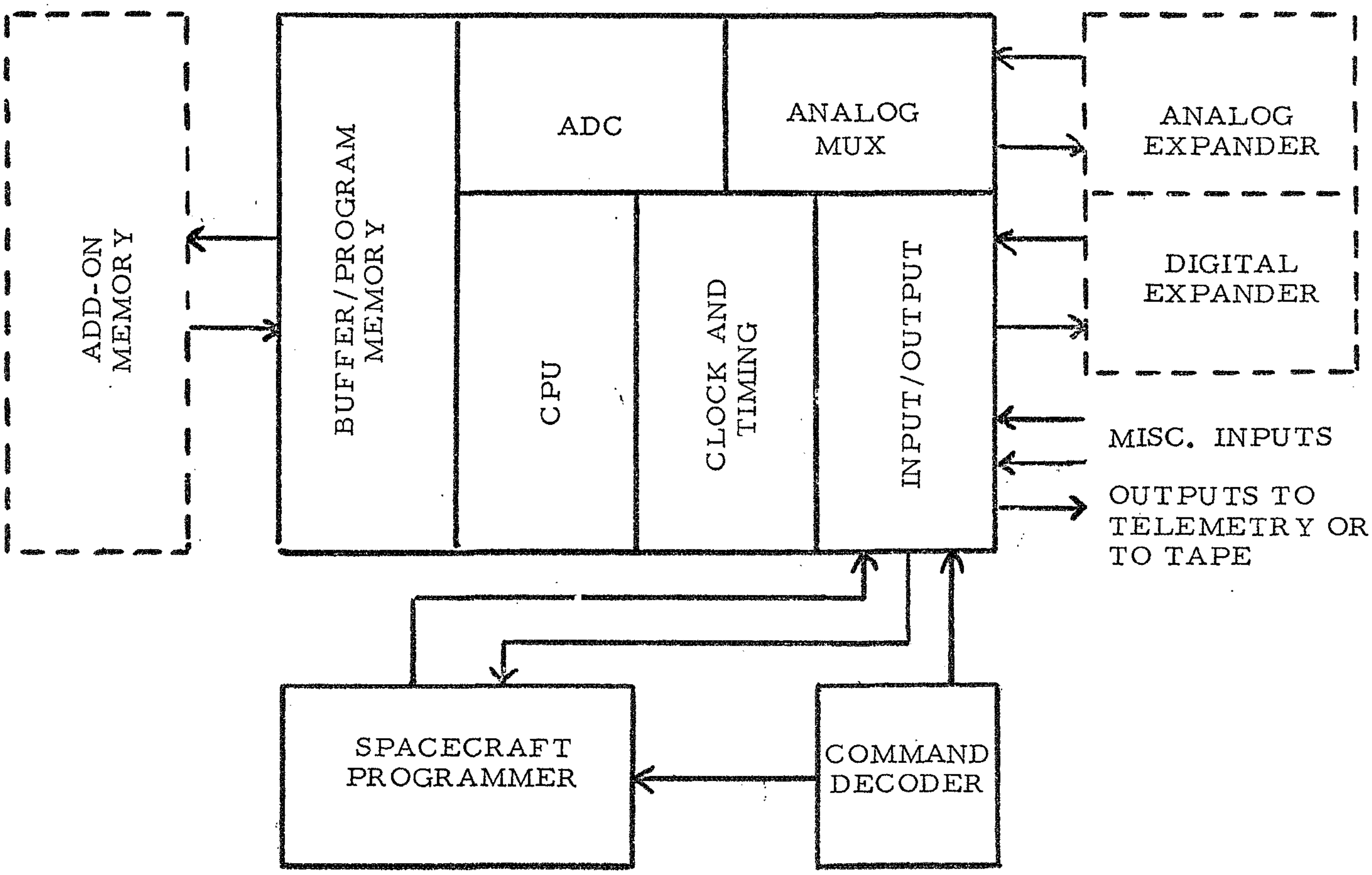
The digital multiplexer will be capable of accepting a limited number of serial or parallel inputs. In addition, there will be circuits capable of determining the time of occurrence of random digital pulses.

There will be at least two output serializers with buffer circuits to feed one or two transmitters and/or tape recorders. These outputs may carry the same or different formats.

It will also be possible to expand the unit, particularly in the memory and multiplexer areas, to accommodate special requirements. Reprogramming in orbit will be feasible, using a standard tone-digital command system, or, if special requirements warrant the higher cost, a PCM command system.

FIGURE 6.1

DATA HANDLING CONCEPT



Spacecraft control functions will be handled through the spacecraft programmer.

6.2 Memory

The memory will be required for:

a) Program storage
b) Buffer storage of data
c) Command storage

In addition, when it is required, add-on memory can be used to provide storage of selected data for periods of an orbit or more.

There are three basic types of memories that can be considered for use with the data handling unit. These types are:

1. Ferrite core (low power design)
2. Plated wire
3. Semiconductor

Table 6.1 summarizes the status of these memories with respect to flight history, development status, power and special characteristics. Some principal sources of hardware have also been mentioned.

Because of its desirable NDRO (non-destructive readout) coupled with the fact that it is also electrically alterable, the plated wire memory would be the primary choice for this application. Its low dynamic power consumption is also a very desirable feature.

As a backup, the ferrite core memory is recommended. Although it is basically a DRO (destructive readout) device, it can be operated in a read/restore mode, so that it effectively behaves like an NDRO memory. It shares the non-volatile characteristic with the plated wire memory, meaning that is retains its contents indefinitely when the power is shut off.

Semiconductor memories are not recommended for this application. These memories are volatile, meaning their contents are lost when power is turned off. An exception is the ROM (read-only memory) which has limited usefulness. In addition, some semiconductor memories are dynamic, which means that even with power continuously on, their contents must be periodically refreshed.

Since the memory periodically must operate at high burst rates, it is recommended that parallel random addressing be used along with parallel input/output of data.

TABLE 6.1

MEMORY STATUS

Type	Flight History	Development Status	Power	Special Characteristics	Principal Sources
Ferrite Core (Low Power Designs)	Several Have Flown *	Fully Qualified Routine Design	Low Standby High Dynamic	Non-Volatile DRO	SPACETAC EMI
Plated Wire	None Have Flown **	Expected to be Qualified in Time; Routine Electrical Design; Less Routine Packaging	Low Standby Low Dynamic Elec. Alterable	Non-Volatile NDRO	Honeywell SPACETAC Spacecraft, Inc. Motorola and Others
Semiconductor	None Have Flown	May be Qualified in Time	Medium (Bi-Polar) Low (MOS-COSMOS)	Volatile NDRO Elec. Alterable	Numerous Semiconductor and Systems Manufacturers

Recommendations: Ferrite Core or Plated Wire, with plated wire as the primary choice and core as backup.

*On four NRL Gravity Gradient Satellites (4K x 8). A 4K x 16 core memory is scheduled for IMP-I.

**Qualified for Poseidon; flown on Tital III-C but not satellites.

At the present time it is difficult to assess the requirement for memory capacity. It is expected, however, that a memory consisting of 1024 words of 16 bits each will cover the majority of applications.

6.3 Central Processing Unit (CPU)

The central processing unit (CPU) provides the executive functions for the data handling unit. As an option it may be desirable to provide an arithmetic unit as part of the CPU. The principal tasks performed by the CPU are as follows:

1. Addressing the memory
2. Execute stored program instructions
3. Format control
4. Input/output control

Except for optional arithmetic operations, the CPU essentially performs bookkeeping functions, and will generally correspond to the "programmer" portion of a computer.

6.4 Analog to Digital Converter

It is recommended that the analog to digital converter (ADC) be placed under the direct control of the CPU just like the memory has been. This will reduce the timing problems sometimes encountered in high-speed operations.

The recommended design is based on the use of a 10-bit successive approximation ADC. Table 6.2 shows three basic types of analog to digital converters, and how they compare with respect to flight history, development status, power, aperture time, resolution and special characteristics. Principal sources of the various types are also listed.

Since some of the experiments on the LRC list may require resolution in excess of 10 bits, it is recommended that the successive approximation type, which can be designed for up to 14 bits of resolution be selected. This design will satisfy practically all experiments. The exceptions are those experiments requiring extremely high-speed analog to digital conversion. None of the converters shown in Table 6.2 will satisfy such requirements. In that case, it is recommended that the analog-to-grey code or "cyclic" converter be considered. These converters have been run at bit rates to 40 MHz, and further development might very well bring the bit rate capability even higher.

For the nominal case, however, it is expected that a 10-bit successive approximation ADC will be satisfactory, and any requirement in excess of that must be flagged as requiring special handling.

TABLE 6.2

ANALOG TO DIGITAL CONVERTER STATUS

Type	Flight History	Development Status	Power	Aperture Time	Resolution	Special Characteristics	Principal Sources
Voltage-Controlled Oscillator	Several IMP-Series Spacecraft	Fully Qualified	Low	Several Milliseconds (fixed)	0.5%	Integrating Over Aperture	GSFC In-House Design
Dual Ramp Integrating	Several 8-Bit ADC's Have Flown *	Fully Qualified	Low	256 μs (8-Bit) to 1024 μs (10-Bit)	To 0.1%	Integrating Over Aperture	SPACETAC
Successive Approximation	8-Bit Designs Flew on OGO and Other Spacecraft	Fully Qualified (8-Bits)	High	15 μs/Bit	0.25%	Instantaneous Value at Aperture Time	TRW and Others
	Will Fly in Near Future **	Passed all Pre-flight Tests	Low	10 μs/Bit	14-Bits with Correction for Temp. 12-Bits Absolute	Instantaneous Value at Aperture Time	SPACETAC

*On four NRL Gravity Gradient Satellites, a 10-Bit Design has been flown on AFCRL Sounding Rocket.

**14-Bit Design scheduled for IMP-I. Pre-flight tests have been performed. A 10-Bit Design will fly on Magnetic Storm Spacecraft.

6.5 Multiplexers

The data handling unit will require multiplexers capable of handling analog signals as well as digital inputs. The output from the analog multiplexer will be a single line leading to the input of the analog to digital converter. Most of the spacecraft engineering, diagnostic and attitude determination data in analog form, can be handled by a relatively slow sub-multiplexer with 64 or perhaps 128 inputs. Some of the more slowly varying experiment data or diagnostic outputs could also very well be handled by a sub-multiplexer. This multiplexer could be of fixed format, and not be programmable, except for the position or positions of the submultiplexer in the main telemetry format.

It is expected, however, that most of the analog experiment data will be assigned to a high speed multiplexer whose inputs can be sampled under stored program control. This multiplexer is expected to have a limited number of inputs. Few experiments seem to require more than 8 or at the most 16 high-speed sampling channels. It is recommended, therefore, that the basic design be capable of handling 16 channels whose sampling can be programmed. Experiments requiring more than 16 "high-speed" analog channels may require one or more analog expanders containing 16 channels each.

The experiment could also routinely be provided with 16 or perhaps 32 channels on the submultiplexer (non-programmable) where diagnostic data could be inserted. If fewer channels are needed, several inputs could be "strapped" to provide multiple readouts during the submultiplexer cycle.

All analog outputs to be sampled by the multiplexer must be standardized with respect to voltage as well as impedance levels, providing for a well-defined interface. Minimum time between samples must also be carefully specified. In those cases where an experiment requires simultaneous or more closely spaced samples, the experimenter must furnish his own sample-and-hold circuits with the experiment, while the data processing unit will supply the necessary flags and timing pulses to the experiment.

The digital multiplexer must be capable of handling various types of digital inputs, such as:

1. Parallel digital data
2. Serial digital data
3. Bi level signals
4. "Random" pulses whose occurrence must be accurately timed.

It is not expected that the data handling unit normally will be equipped with pulse counting circuits, and digital data from the experiments will, in many cases, be limited to various status "flags". Such flags, along with other types of bi level signals, can be transferred to the buffer memory or direct to the output serializer by strobing the parallel data bus at the appropriate time. Parallel digital data will be gated onto a bus under program control, and after the bus has settled, it will be sampled by the data handling unit. The interface specification for the parallel bus can be a very simple one, specifying voltage and impedance levels for binary "1" and for binary "0". At the point where the bus is sampled by the data handling unit, a sufficient amount of filtering will be used to guarantee a clean signal.

Serial digital data will require an interface resembling that which any one bit on the parallel interface will see. The settling time, however, will be faster than for the parallel case. Generally, the serial input will carry binary information which is shifted serially, bit by bit, at a rate which may or may not be the same as the telemetry bit rate. The operation will be under program control.

"Random" pulses, whose time of occurrence must be precisely determined, are among the more likely signals which must be handled by the data handling unit. The horizon sensor is a typical example of a source of such pulses. It is expected that such inputs will be handled on an interrupt basis by the CPU. Depending on the required precision of the timing, it is also possible that special buffer circuits will be assigned in the input/output unit.

6.6 Input/Output

The input/output (I/O) circuits generally handles the communication between the data handling unit and the "outside world". The necessary input and output buffer circuits are contained here, including those handling the parallel and serial digital inputs and output, as well as the random pulses.

Pulses and levels (discretes) used to control other subsystems are generated here under program control. The output serializers and such things as split-phase encoders are also part of the I/O circuits.

6.7 Spacecraft Programmer

The requirements for the spacecraft programmer were discussed in Section 4.0. The design of the unit will, in many respects, be an extension of the I/O circuits. The main reasons for wanting to physically separate this unit from the data handling unit are the following:

1. Parts of the unit, such as the pyrotechnic sequencer and control circuits require special considerations to accomplish satisfactory arming and safing.

2. Relatively high power levels are being handled in some areas, such as the power switching unit. It is desirable to keep high-level switching away from low-level circuits wherever possible.

3. Sections of the programmer may be turned on or off independent of each other or the data handling unit.

6.8 Add-on Features

If it is desired, the data handling unit can be designed to accommodate various expansion features. These include:

1. The addition of arithmetic capability to the CPU. This could conceivably be accomplished by adding another electronics "deck" to the unit. The arithmetic unit in its simplest form may have add capability only. In this manner, it will be possible to integrate or average several readings digitally before transmitting a single number.

   A more complex arithmetic unit would be capable of add, subtract, multiply and divide as well as, perhaps, floating point compression operations.

2. Analog multiplexer expansion may be required by some experiments.

3. Digital multiplexer expansion may also be required by some experiments, in particular when considering the somewhat limited capability for handling digital data.

4. Add-on memory may be desirable to provide data storage. With a 1K memory the data handling unit is still pretty much a real-time device, although some storage of principally attitude determination data can be provided.

   Storage capacity up to perhaps as much as 32K words may be desirable for some applications.

7.0 Hardware Considerations

When trying to determine which of the many available circuit technologies to recommend for this satellite, the following should be kept in mind:

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

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

   It is, therefore, reasonable to set the following ground rules for the data handling unit:

   a) Size and weight should be kept reasonably low.

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

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

The technologies which can be considered for the data handling and programmer subsystem are the following:

1. Monolithic bipolar integrated circuits.

2. Thick or thin film hybrid circuits.

3. P-channel or complementary MOS integrated circuits.

Each of these technologies have been used onboard spacecraft, and each has its own peculiar characteristics that must be evaluated before a choice can be made.

A data handling unit built with bipolar integrated circuits will generally require power a factor of 10 to 50 higher than that of a corresponding unit built with either hybrid circuits or MOS circuits. As far as size and weight is concerned, a unit built with bipolar integrated circuits will at best come out about equal to the hybrid unit, and larger and heavier than the MOS unit (neglecting shielding requirements). It would, therefore, seem reasonable to advise against the use of bipolar integrated circuits for this application.

Consequently, it boils down to a choice between hybrid electronics and MOS circuits. Both methods lead to low power consumption, when properly done, and the difference between the two will be in the noise level, as far as available power is concerned.

Size and weight would, for any application but a typical space application, come out in favor of MOS circuits. MOS circuits are, however, more than two orders of magnitude more susceptible to damage from ionizing radiation than the bipolar circuits are. By the time the MOS circuits have been protected against radiation, their size and weight will exceed that of the hybrid circuits.

In addition, MOS has not been used very much in space applications. A notable exception is the IMP program. The IMP spacecraft, however, have orbits that are extremely favorable from the point of view of radiation damage. Most of their orbits lie well outside the radiation belts.

It is recommended, therefore, that hybrid electronics be considered for the data handling unit because a proper design combines low power and high-speed capability with small size and low weight. In addition, these circuits have proven highly reliable in space, and have shown excellent resistance to radiation damage.

The following estimates, then, are based on the use of thick film hybrid electronics in the three "black boxes" constituting the Data Handling and Programming Subsystem:

1. Data Processor

   a) CPU
   b) I/O
   c) ADC
   d) Multiplexers

   Size: 5.5" x 4.5" x 4.5"
   Weight: 4.5 lbs.
   Power: 1.0 watts

2. Memory

   Size: 5.5" x 4.5" x 3"
   Weight: 3.0 lbs.
   Power: 0.5 watts (standby)

3. Spacecraft Programmer

   Size: 5.5" x 4.5" x 1.75"
   Weight: 1.6 lbs.
   Power: 0.5 watts (nominal)

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒ ESDM-F440-4086

TO	DEPT.	FROM	DEPT.	DATE
S.J. Brzeski (5 copies)	L910	W.J. Kubicki	F440	5 Aug. 1970

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control Study for Small Satellites and Related Subsystems	W159-F440-041		

SUBJECT

SOLAR POWER AVAILABILITY

DISTRIBUTION

H. Burke, W. Fitzgerald, BC, CF, RF

SIGNED W.J. Kubicki

APPROVED W.J. Kubicki

INFORMATION REQUESTED / RELEASED

This T/R presents a cursory review of the factors affecting solar power availability. A detailed tradeoff is included for the solar power available from the reference right circular cylinder design for various orbits originating from launches at Wallops Island, San Marco and the Western Test Range (WTR).

Specific solar power profiles are included for selected orbits.

INTRODUCTION

Two basic parameters that influence the electrical output characteristics of any given solar cell or array of cells in a solar panel configuration are:

a.) Solar cell temperature

b.) Illumination incidence angle

The effect of temperature variations upon the electrical characteristics for normal incidence is illustrated in Figure 1. It should be noted that the cell output voltage is significantly dependent upon cell temperature when operated at or left of the maximum power point while the output current is relatively unaffected. Maximum power point is normally located in the "knee" region of the I-V curve. It is therefore essential to maintain solar cell panels at the lowest practicable operating temperature. The variation in open circuit voltage is approximately 2.2 millivolts per ^{o}C while the variation in short circuit current is approximately 0.1% per ^{o}C.

One of the most significant parameters to be considered is the effect of variations in solar incidence angle upon a solar cell or array of cells. Figure 2 best illustrates this phenomenon. It should be noted that in contrast to the temperature variation phenomenon, the current is significantly dependent upon solar incidence angle when operated at or left of the maximum power point. The cell output voltage is relatively insensitive to solar incidence angle variations. Open circuit voltage falls rapidly to zero, however, as the solar incidence angle approaches 90^{o} these effects may be more readily observed in Figure 3.

The output current from a cell or an array of cells, as a function of incidence angle, will depend upon the cell loading or where on the curve we operate the cell. At cell voltages other than zero volts, the current output will drop to zero prior to reaching an incidence angle of 90°. For cells at a temperature of 25°C, the cutoff angle will generally lie between 85° and 88° when the cell is loaded at a fixed voltage ranging from 0.30 volts to approximately 0.45 volts.

Since maximum power output from a solar panel array is usually desired, the array should be loaded a conservative voltage as close to the knee as practicable. Careful determination of maximum panel temperature, either by analysis or experiment, is essential prior to a detailed design of a solar array.

REFERENCE DESIGN POWER CALCULATIONS

Typically the temperature of a body mounted insulated solar cell array on a spacecraft operating in near earth orbit in a non spinning attitude will be $\approx$ 105°C. If the same array is removed from the spacecraft permitting the back side to be exposed to space, the temperature drops to $\approx$ 45°C. A right circular cylinder 30" diameter x 40" long with body mounted cells spinning at 3 RPM or greater will have an array temperature of 8 - 25 °C.

The power predictions are based upon an array temperature of +105°C on a spacecraft whose configuration is a 29.4" diameter x 36.6" long right circular cylinder. Figure 4 represents the I-V curves from which solar cell performance is obtained.

At +105°C, each cell produces 0.32 volts at the maximum power point. To obtain a +28 VDC bus voltage, 88 cells in series are required. A layout of a single 88 cell string on the cylinder circumference requires two rows (once down once up) and) and an equivalent 6° surface arc. Refer to Figure 5. A 180° segment can accommodate 30 strings for a total power output of 56 watts at 28 VDC for the body mounted array at normal incidence sun illumination angle.

From orbit injection to initiation of de-spin, the array is operating at a cooler temperature and providing more power. The excess power could be dumped into a shunt regulator of a Direct Energy Transfer (DET) Power Subsystem.

The effect of sun aspect angle variation within the orbit and as a function of orbit is to vary the actual solar power available from a fixed body mounted array. The ground rules which guided the orbit selections in order to obtain an insight into mission tradeoffs were:

1. Launches from Wallops Island at inclination angles of 38° and 52°; San Marco at 3° inclination angle; and Western Test Range (WTR), sun synchronous at 99° inclination angle.
2. Launch Times: 12 Midnight
   6 AM
   12 NOON
   6 PM
3. Reference orbit 486 nautical miles circular. This is about midway between the altitude limits set for the study and the orbit essentially repeats itself after one day for ground station viewing.
4. Launch date of 21 May 1974 places the sun in the northern hemisphere and minimizing sun occultation in the northern hemisphere.

5. Mission life of 90 days where one orbit per day every 10 days is investigated.

Table 1.0 lists the sun angle referenced to the body mounted solar array at spacecraft injection for the orbits selected. This angle remains essentially constant except for earth motion about the sun ($\approx 1^\circ$/day) as long as the spacecraft remains spin stabilized. Also listed is the array power available during this period while in sunlight. In general, launch times of 12 Noon provide more power at injection.

For all but WTR launches, the oblateness of the earth causes the angle between the sun line and the orbit plane to change in the mission. This also means that unless some technique is employed to continually re-orient the solar array, the power available will vary accordingly. Tables 2, 3, 4 and 5 represent the sun line orbit plane angle variations for a 90 day mission where the observation is made for one orbit every 10 days. Referring to Table 3, Wallops Launch @ I = 38°, the sun varies in position from 58° to the orbit plane slowly drifting through it to an angle of -14° and back for the 90 day mission. The slow angle drift for the WTR launches presented in Table 5 can be attributed to the small latitude motion of the sun.

Sun line/orbit plane angle variations and its effect on solar power availability for the fixed body mounted array on a satellite in gravity gradient attitude can best be presented in Figure 6. The "central angle traversed by vehicle" defines one full orbit about the earth. "Sun Angle" defines the angle between the sunline and the array (90° indicates normal incidence to the array). The simplest relationship exists when the angle between the sunline and orbit plane is 90° which results in no sun angle variation throughout the orbit. For a 60° sunline/orbit plane angle, the sun will vary $\pm 30^\circ$ from normal incidence throughout the orbit reducing the solar power available by cosine 30° (.866) at the angular extremes. Worst case exists with the sun in the orbit plane or 0° sunline/orbit plane angle.

The effect of sunline/orbit plane and sun angle variations within a 90 day mission for specific representative orbits is presented in Figures 7 through 11 inclusive. The solar power available throughout the mission is also included.

Figure 7 represents the sun angle variations and resultant solar power available from a fixed body mounted array for a San Marco launch, 21 May 1974, 12 Mid in a 486 n.mi. circular orbit. The right circular cylinder is in its gravity gradient mode of operation. Maximum sun angle does not occur at the same point in the orbit, however, the figure shows them at the same point for clarity in presentation. Sun angle variations for the 1st, 20th, 30th, and 90th day are shown to indicate the effect of sunline. orbit plane variation. A sun angle of -28° represents exiting sun occultation and a solar power value of $\approx 49^+$ watts. The value will be somewhat higher since the solar array temperature is much cooler than the steady state value of + 105°C. As the array heats up, the solar power will follow the curves.

Figure 8 represents similar information for a Wallops Island launch @ 38° inclination. The large sunline/orbit plane variations listed in Table 3 are responsible for the difference in power levels indicated for the 1st, 10th, 20th and 50th orbits.

Figures 9 and 10 are for launches from Wallops Island at 52° and WTR at 99° respectively. They also represent an unfavorable power profile throughout an orbit.

An example of a favorable profile is presented in Figure 11 which is for a similar WTR launch however, at a launch time of 6 pm, instead of the 12 Mid time presented earlier. The reason for the change is evident by referring to Table 5 which lists a nominal 78° sunline orbit plane angle and no sun occultation. In reference to sun occultation, Figure 12 is included to permit a convenient method of determining length of sun occultation as a function circular orbit altitude and sunline/orbit plane angle. The orbit period in minutes is also included for reference. Angular values which do not intersect the orbit altitude in question indicate no sun occultation. Example: For a sunline orbit plane angle of 60°, no sun occultation occurs for circular orbits higher than ≈ 550 nautical miles. At 500 nautical miles, the maximum sun occultation is 6.5 minutes, 22.5 minutes at 250 n.mi. and 30 minutes at 100 nautical miles. The complement of the sunline/orbit plane angle represents the sun angle variation experienced in the orbit. Therefore, the same 60° sunline/orbit plane angle results in a maximum sun angle of 30°.

A review of the parametric data just presented indicate that a fixed solar array on a right circular cylinder in a gravity gradient mode does not provide an attractive solar power profile. This statement is applicable for all but a select few sun synchronous orbits. Solar paddles, either sun seeking or fixed, are required to achieve the desired power. Fixed paddles are inherently simpler and more reliable however, their location and quantity are a function of final orbit selection when sun angles can be investigated.

Figure 13 is a replot of the solar power profile for the San Marco launch originally presented in Figure 7 (1st day only). The solar power available varies from a peak of 56 watts to a minimum of 16 watts. At the circular orbit altitude of 486 n.mi., the sun occultation exit and enter angle remains constant at 28° from broadside. Locating a single fixed 29" diameter paddle at 30° as shown, the solar power available is increased to a peak of 89 watts and a minimum of 42 watts. This is based upon placing 12 strings of solar cells, 64 cells per string, on the paddle.

Specific mission and orbit requirements are mandatory prior to an accurate assessment of solar power availability and solar array configuration can be defined.

THE JOHNS HOPKINS UNIVERSITY
APPLIED PHYSICS LABORATORY
SILVER SPRING, MARYLAND

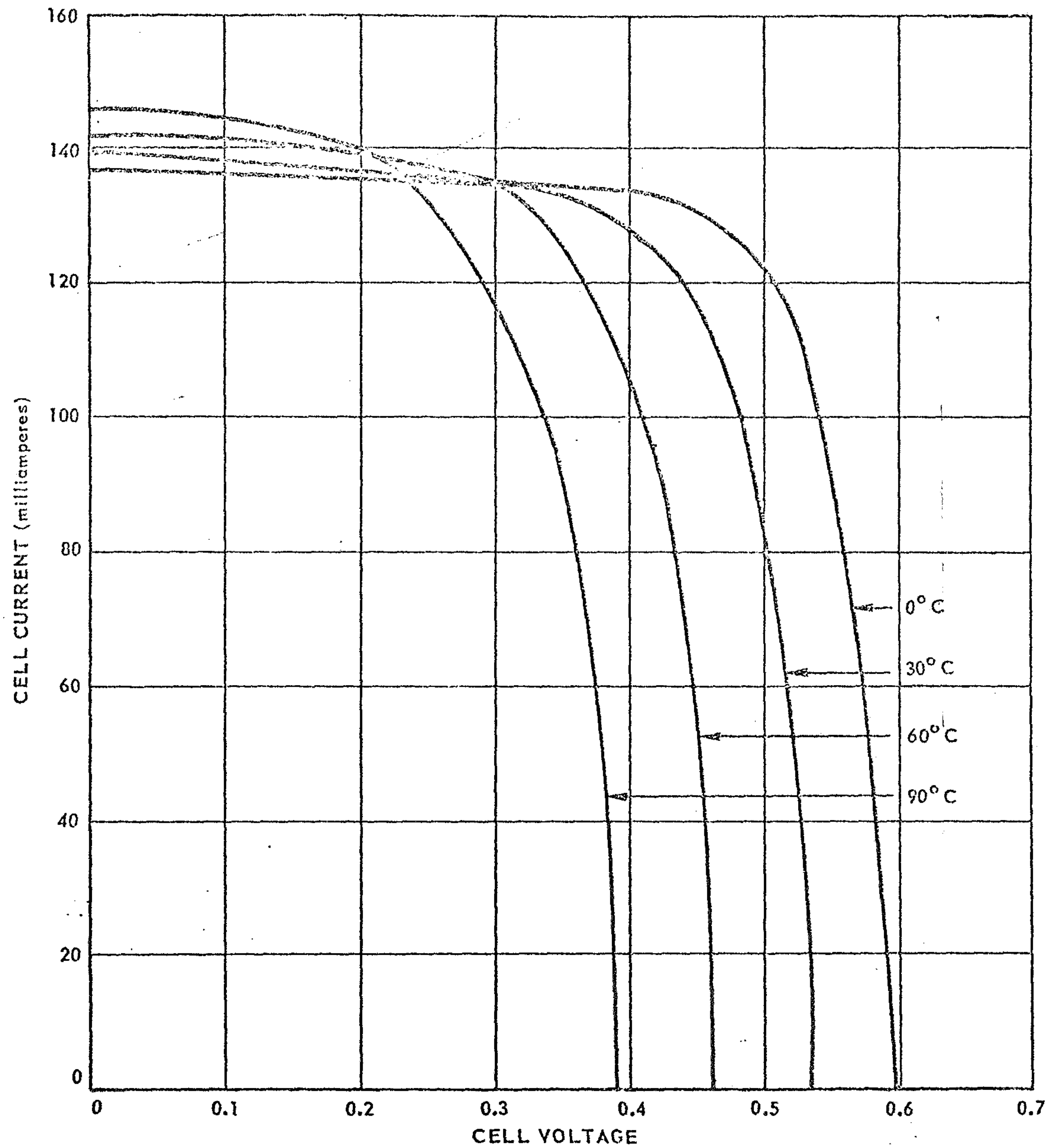


Fig. 1 EFFECTS OF TEMPERATURE UPON SOLAR CELL PERFORMANCE FOR A 2 BY 2 CM, 10 OHM-CM CELL

THE JOHNS HOPKINS UNIVERSITY
APPLIED PHYSICS LABORATORY
SILVER SPRING, MARYLAND

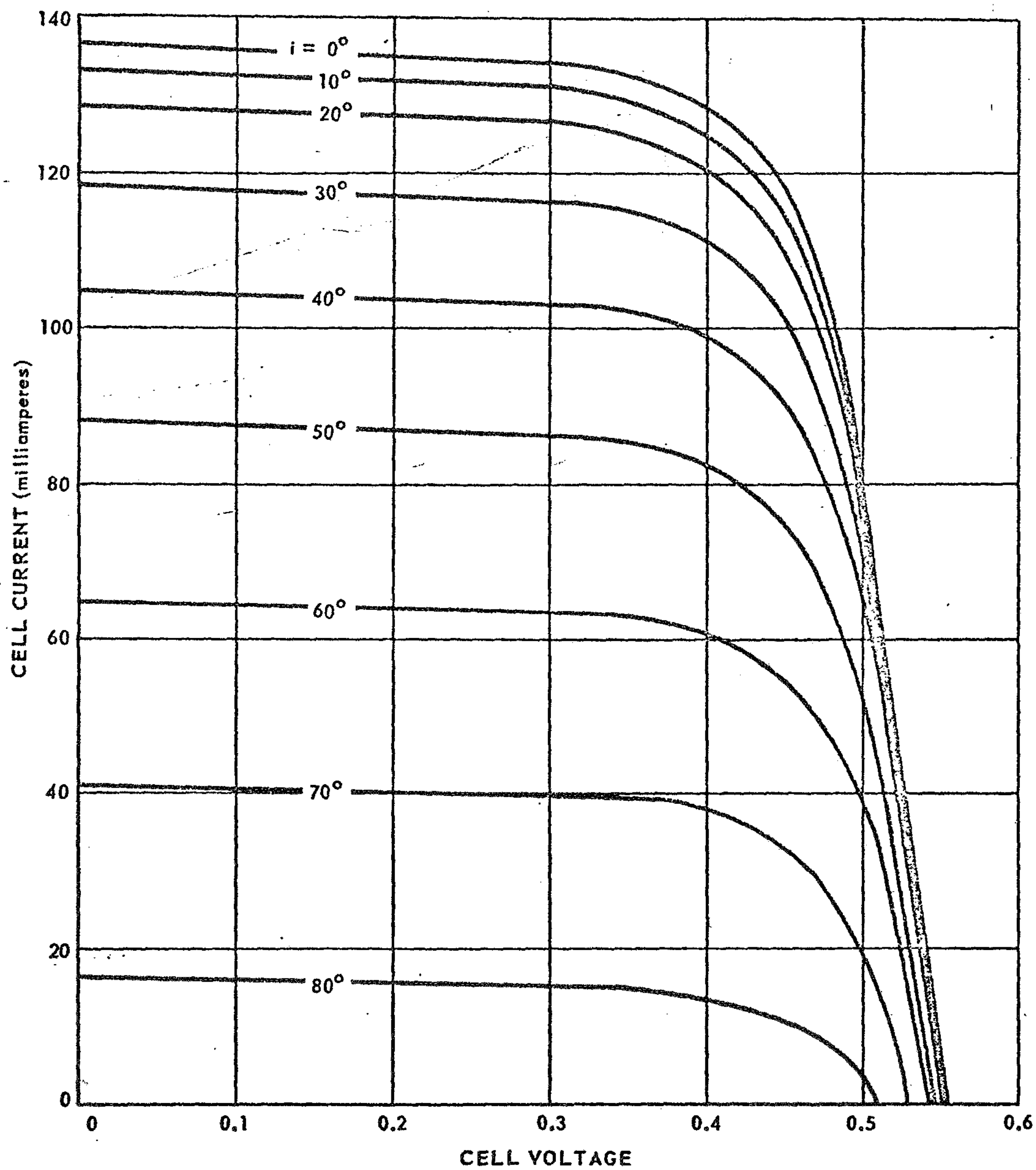


Fig. 2 TYPICAL SOLAR CELL CHARACTERISTICS AS A FUNCTION OF INCIDENCE ANGLE, CELL TEMPERATURE 25° C

THE JOHNS HOPKINS UNIVERSITY
APPLIED PHYSICS LABORATORY
SILVER SPRING, MARYLAND

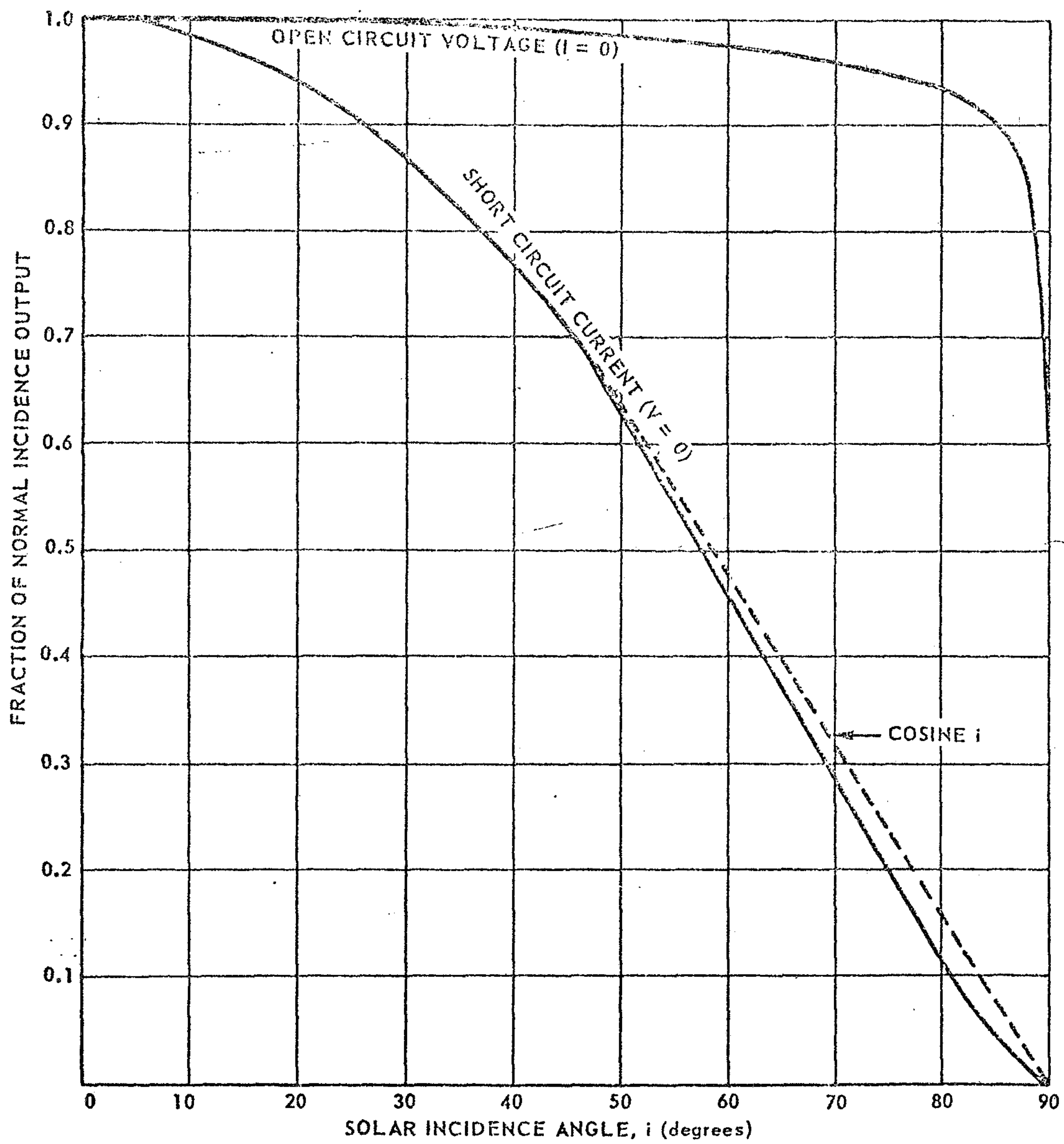


Fig. 3 NORMALIZED SOLAR CELL ELECTRICAL OUTPUT AS A FUNCTION OF SOLAR INCIDENCE ANGLE
(Corning 0211 Microsheet – 0.006 Inch OCLI Blue Filter and Antireflective Coating)

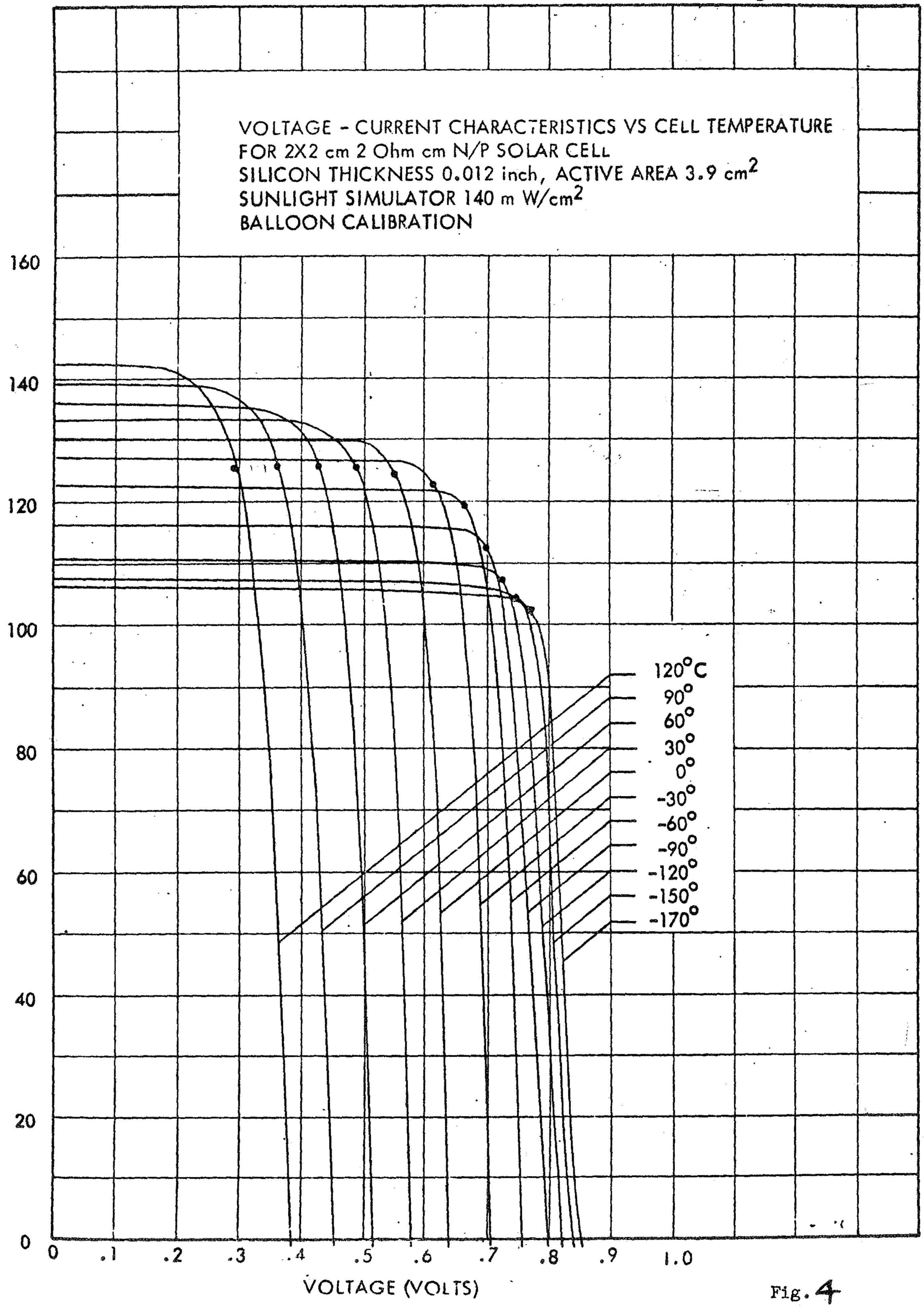

VOLTAGE - CURRENT CHARACTERISTICS VS CELL TEMPERATURE
FOR 2X2 cm 2 Ohm cm N/P SOLAR CELL
SILICON THICKNESS 0.012 inch, ACTIVE AREA 3.9 cm²
SUNLIGHT SIMULATOR 140 m W/cm²
BALLOON CALIBRATION
160
140
120
100
80
60
40
20
0
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
120°C
90°
60°
30°
0°
-30°
-60°
-90°
-120°
-150°
-170°
VOLTAGE (VOLTS)

Fig. 4

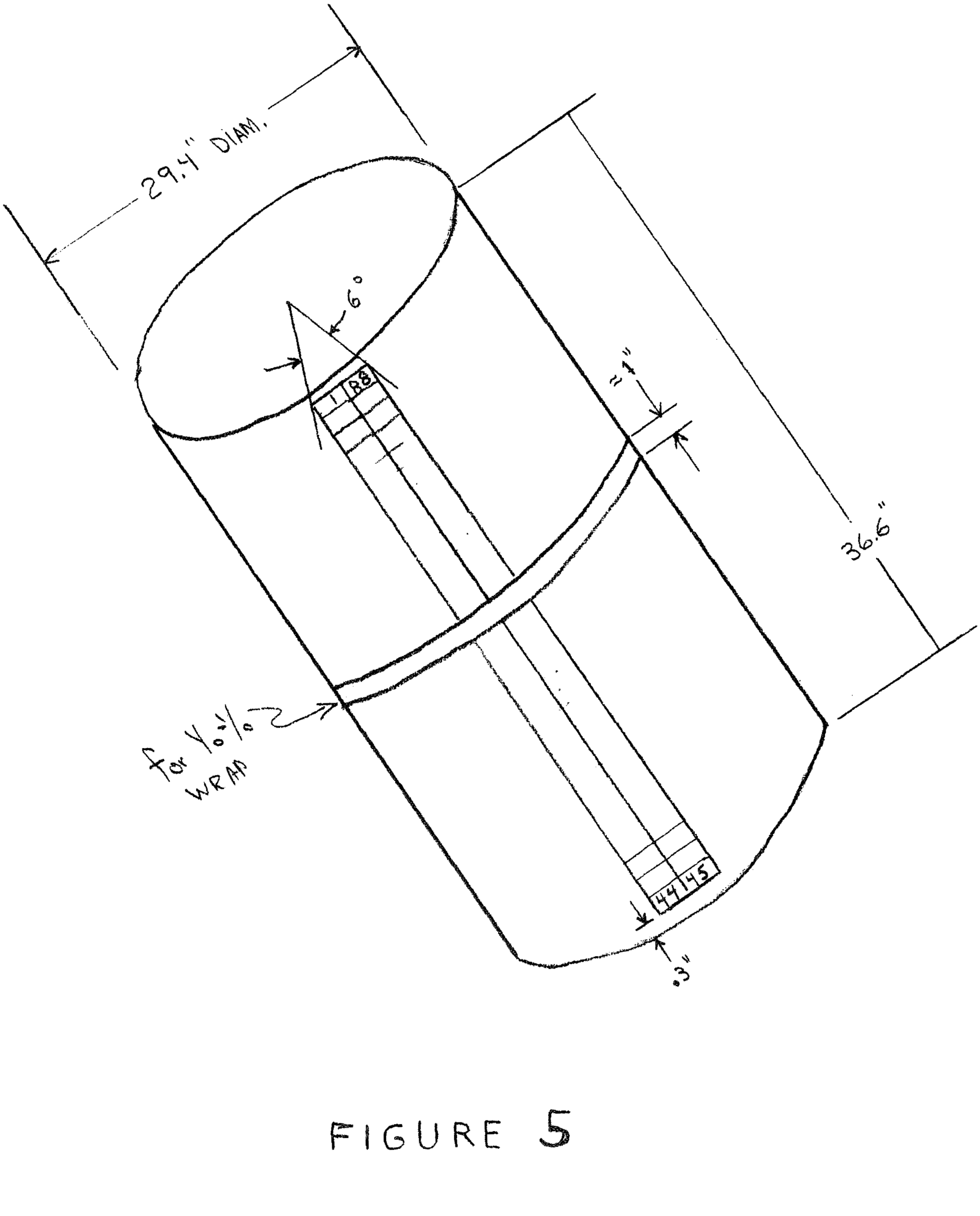
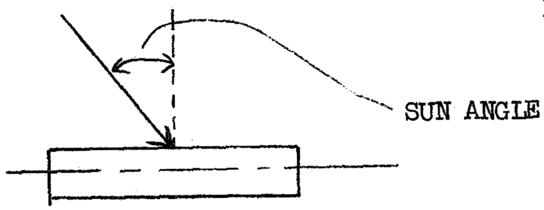


FIGURE 5

TABLE 1

LAUNCH SITE	INCLINATION	TIME	INSERTION SUN ANGLE [1]	SOLAR POWER AVAILABLE [2]
SAN MARCO	3°	12 MID	20°	52.5 WATTS
		6 AM	62°	26.2
		12 NOON	19°	53
		6 PM	60°	28
WALLOPS	38°	12 MID	18°	53.5
		6 AM	51°	35.2
		12 NOON	26°	50
		6 PM	66°	22.8
WALLOPS	52	12 MID	15°	54
		6 AM	25°	51
		12 NOON	13°	54.5
		6 PM	65°	23.8
WTR	99°	12 MID	32°	47.5
		6 AM	30°	48.5
		12 NOON	7°	55.5
		6 PM	9°	55.4

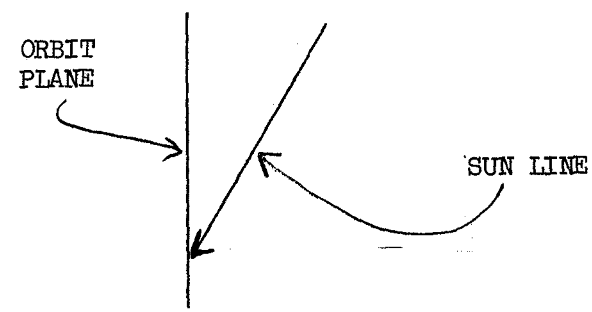
1 

2 BASED UPON 56 WATTS @ 0° SUN ANGLE

TABLE 2

SAN MARCO LAUNCH
I = 3°
DATE 21 MAY 1974
ORBIT ALTITUDE 486 N.M.I. CIRC.

DAY \ LAUNCH TIME	12 MID	6 AM	12 NOON	6 PM
	SUN LINE/ORBIT PLANE ANGLE			
1	17°	20°	23°	20°
10	21	19	22	25
20	25	22	20	24
30	26	25	21	21
40	22	26	24	21
50	19	22	25	23
60	20	18	21	24
70	21	17	16	20
80	18	18	14	14
90	12	16	14	12



1 - 1690

TABLE 3

WALLOPS ISLAND LAUNCH
I = 38°
DATE 21 MAY 1974
ORBIT ALTITUDE 486 NM CIRC.

DATE \ LAUNCH TIME	12 MID	6 AM	12 NOON	6 PM
	SUN LINE / ORBIT PLANE ANGLE			
1	58	10	-17	21
10	42	46	-6	-6
20	7	58	28	-14
30	-14	26	60	10
40	-3	-6	45	45
50	29	-14	8	58
60	58	9	-16	26
70	40	43	-9	-9
80	2	52	22	-21
90	-2	19	49	0

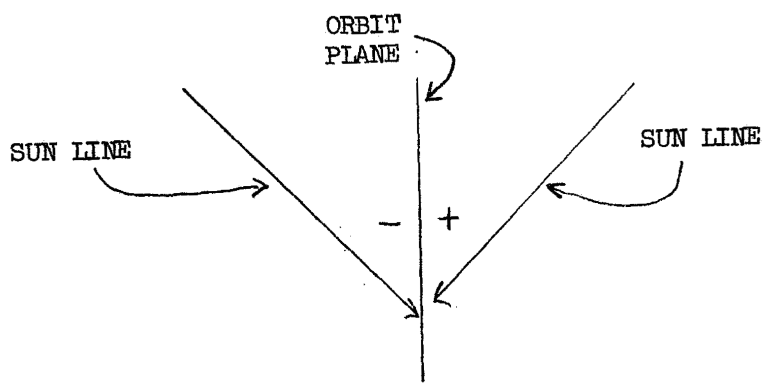


TABLE 4

WALLOPS ISLAND LAUNCH
I = 52°
DATE 21 May 1974
ORBIT ALTITUDE 486 N.M.I. CIRC.

SUN LINE / ORBIT PLANE ANGLE

DAY \ LAUNCH TIME	12 MID	6 AM	12 NOON	6 PM
1	42	53	-14	-22
10	6	72	19	-29
20	-22	39	57	-10
30	-26	2	70	25
40	-2	-25	33	63
50	34	-25	-4	65
60	69	1	-29	27
70	55	37	-26	-10
80	16	67	2	-35
90	-21	47	38	-21

TABLE 5

WTR LAUNCH
I = 99°
DATE 21 MAY 1974
ORBIT ALTITUDE 486 N.M. CIRC.

SUN LINE / ORBIT PLANE ANGLE

DAY \ LAUNCH TIME	12 MID	6 AM	12 NOON	6 PM
1	-8	60	2	-78
10				
20				
30				
40				-74
50		57		
60	-10			
70				
80				
90	-9	66	5	-81

FIGURE 6

ANGLE BETWEEN VEHICLE AXIS AND SUNLINE
(VEHICLE AXIS ALIGNED WITH LOCAL VERTICAL)

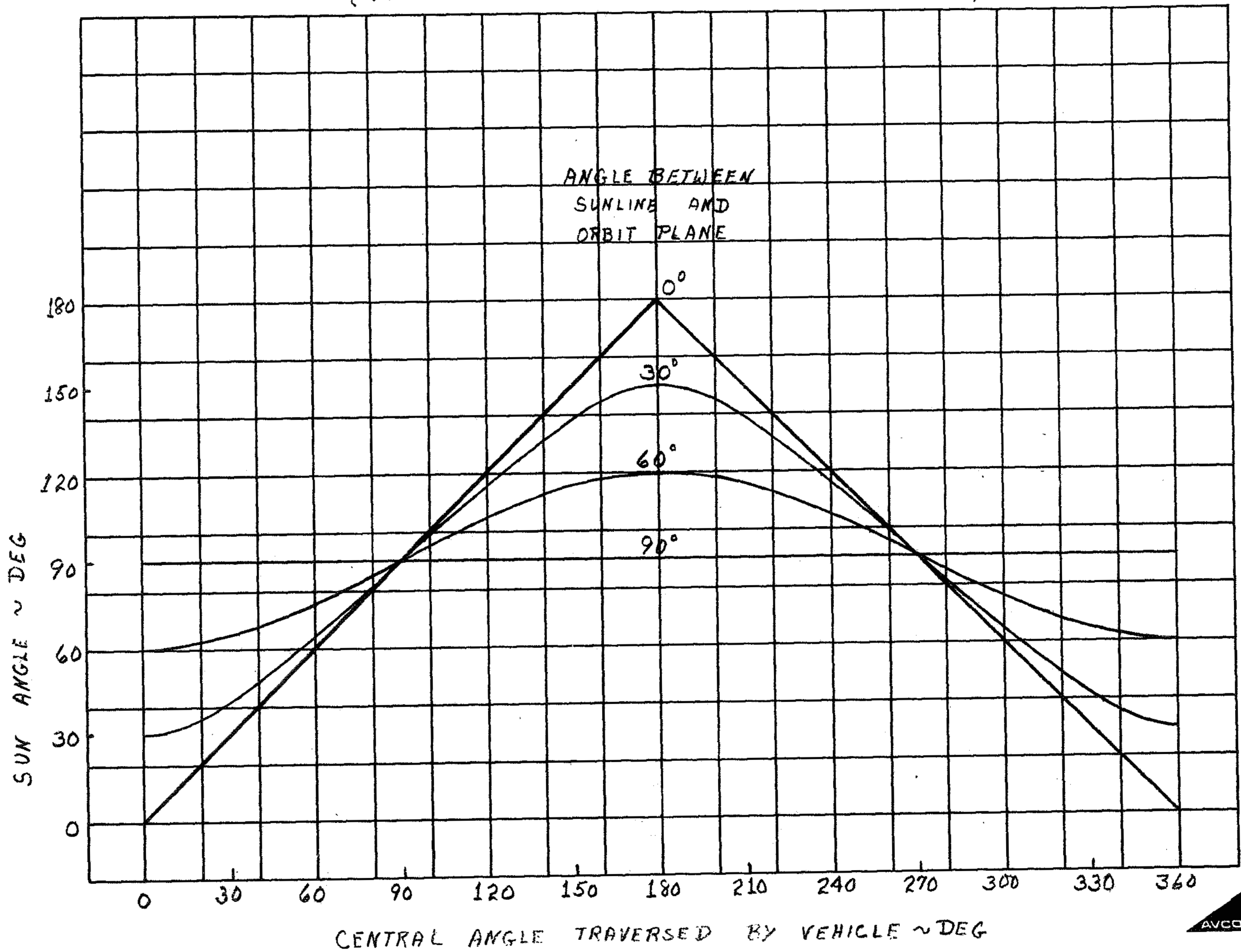
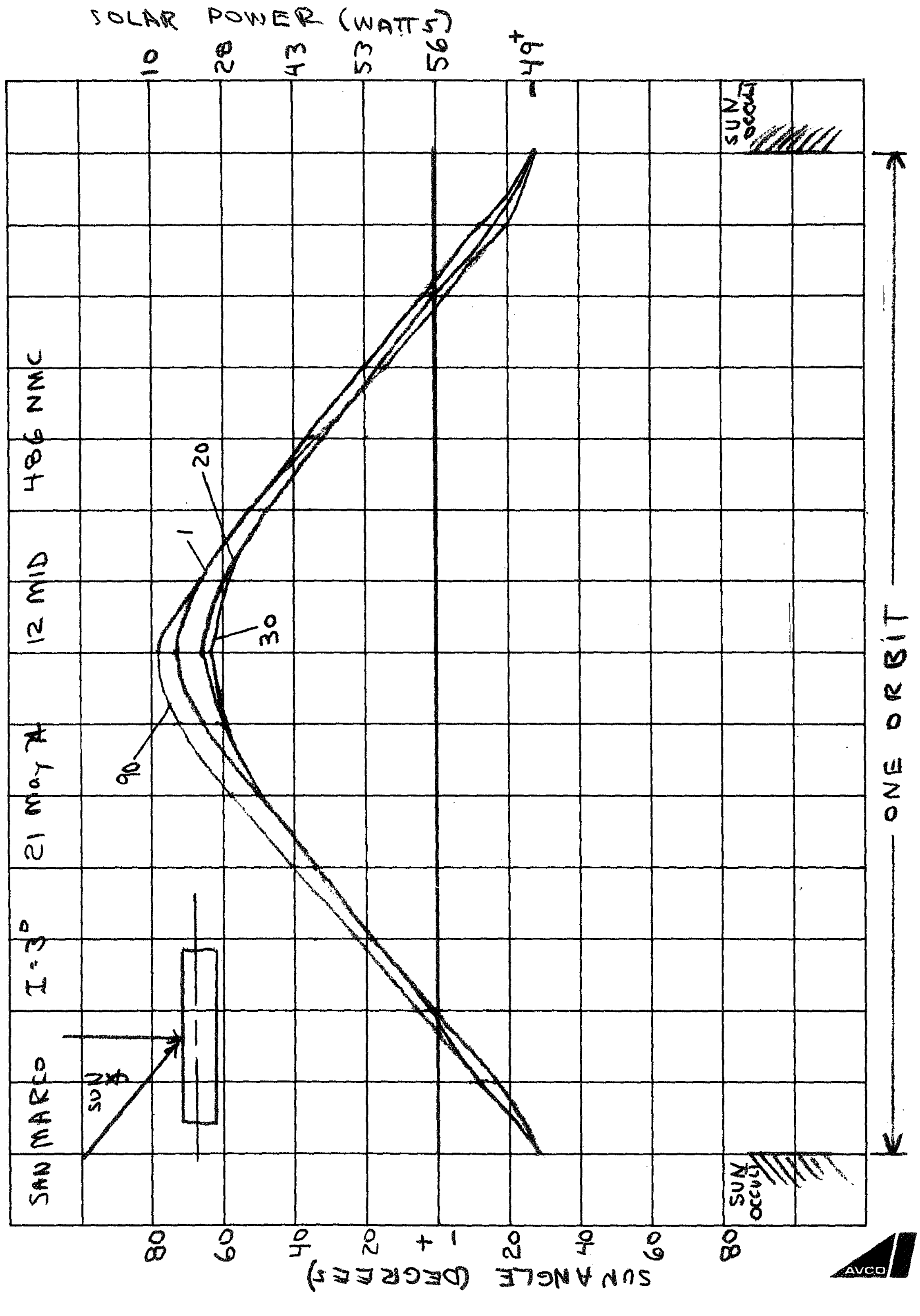


FIGURE 7


SAN MARCO
I = 3°
21 May
12 MID
486 NMC
SOLAR POWER (WATTS)
10
28
43
53
56
49+
SUN
1
20
30
90
SUN OCCULT
SUN OCCULT
ONE ORBIT
80
60
40
20
+ 1
20
40
60
80
SUN ANGLE (DEGREES)

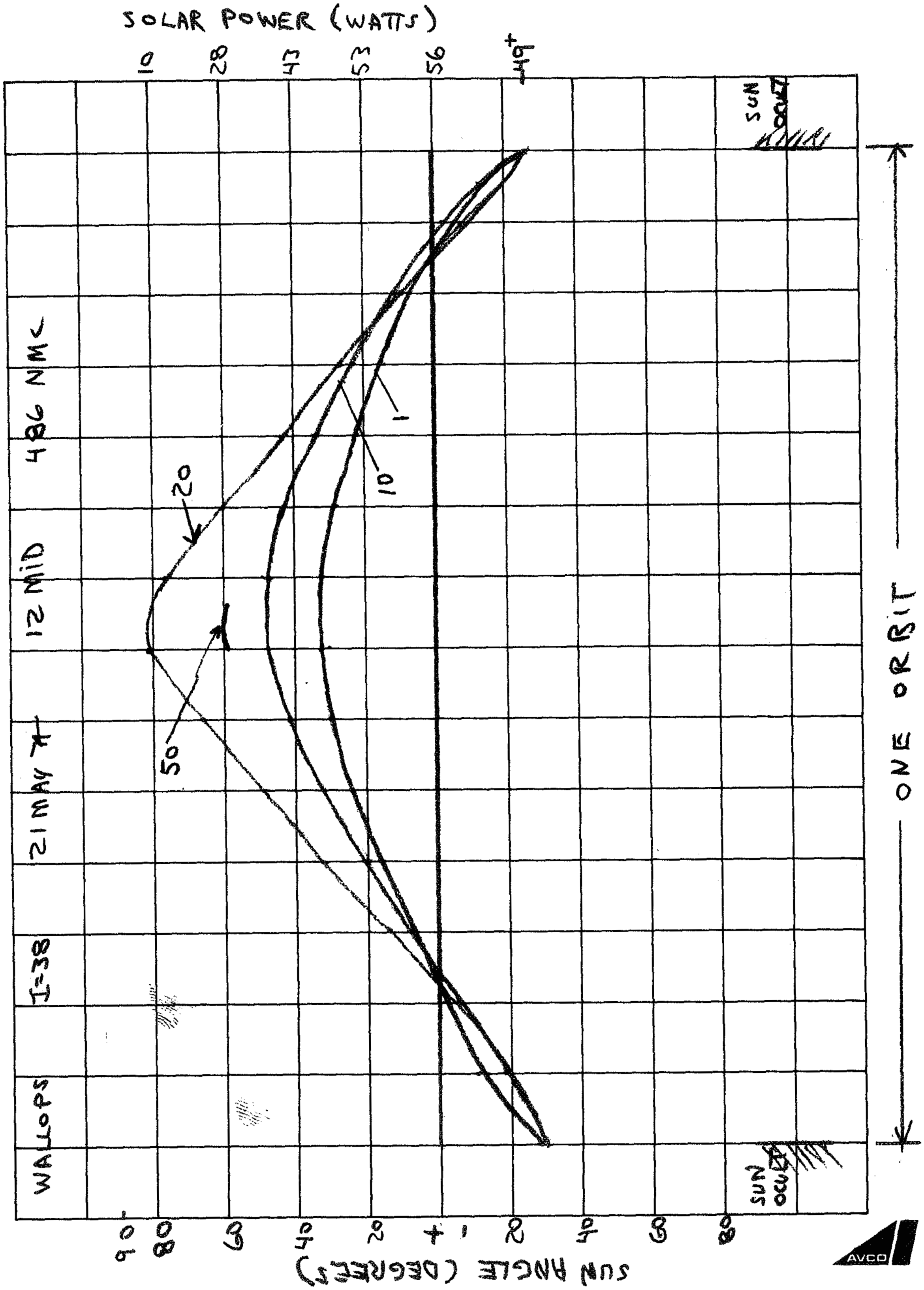

SOLAR POWER (WATTS)
10
28
43
53
56
49+
SUN ANGLE (DEGREES)
90
80
60
40
20
1
20
40
60
80
SUN OCULT
ONE ORBIT
WALLOPS
I=38
21 MAY 71
12 MID
486 NMC
1
10
20
50

FIGURE 8

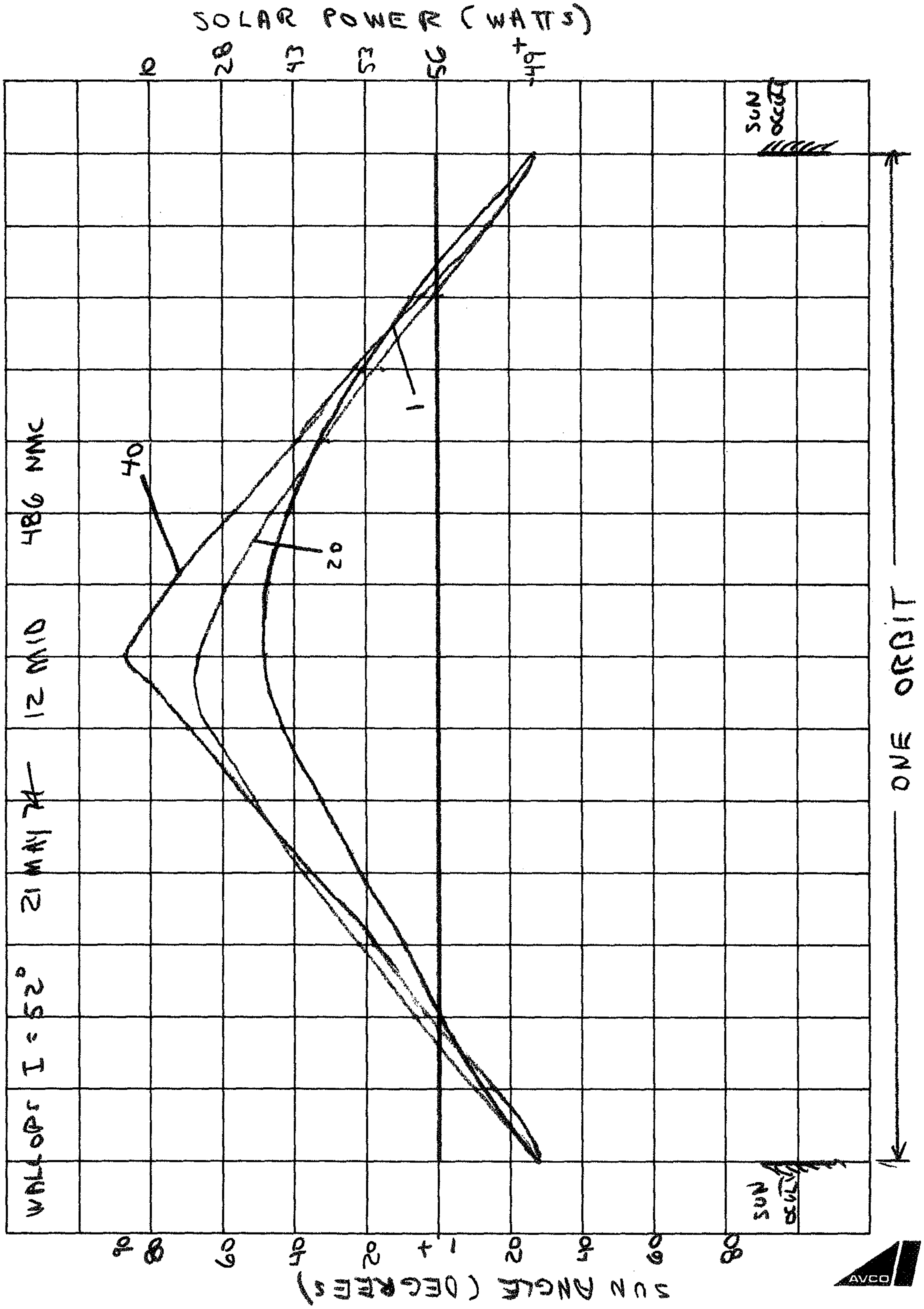

SOLAR POWER (WATTS)
SUN ANGLE (DEGREES)
WALLOPS I = 52° 21 MAY 74 12 MID 486 NMC
ONE ORBIT
SUN OCCULT

FIGURE 9

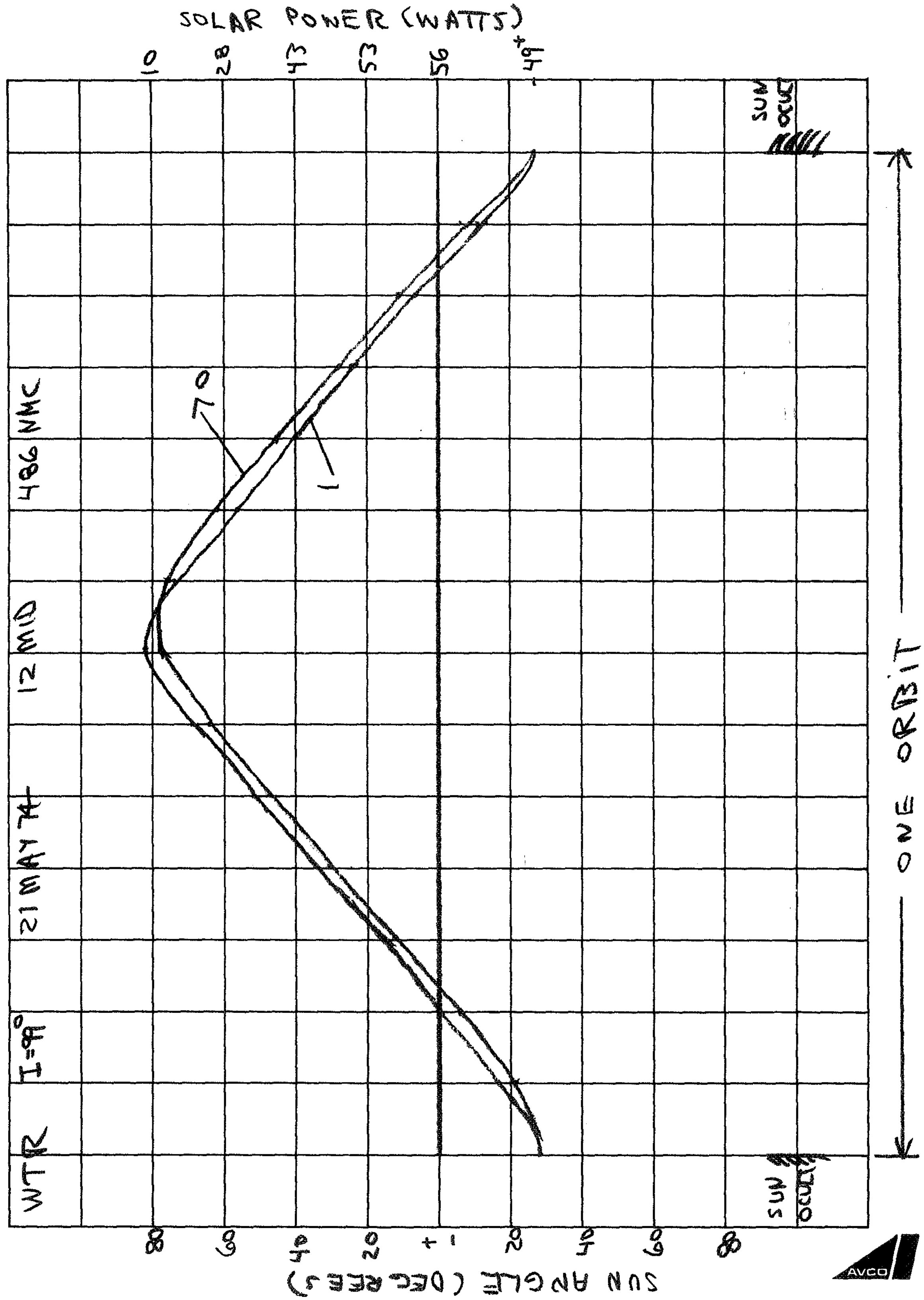

SOLAR POWER (WATTS)
SUN ANGLE (DEGREES)
ONE ORBIT
WTR
21 MAY 74
12 MID
486 NMC
SUN OCULT

FIGURE 10

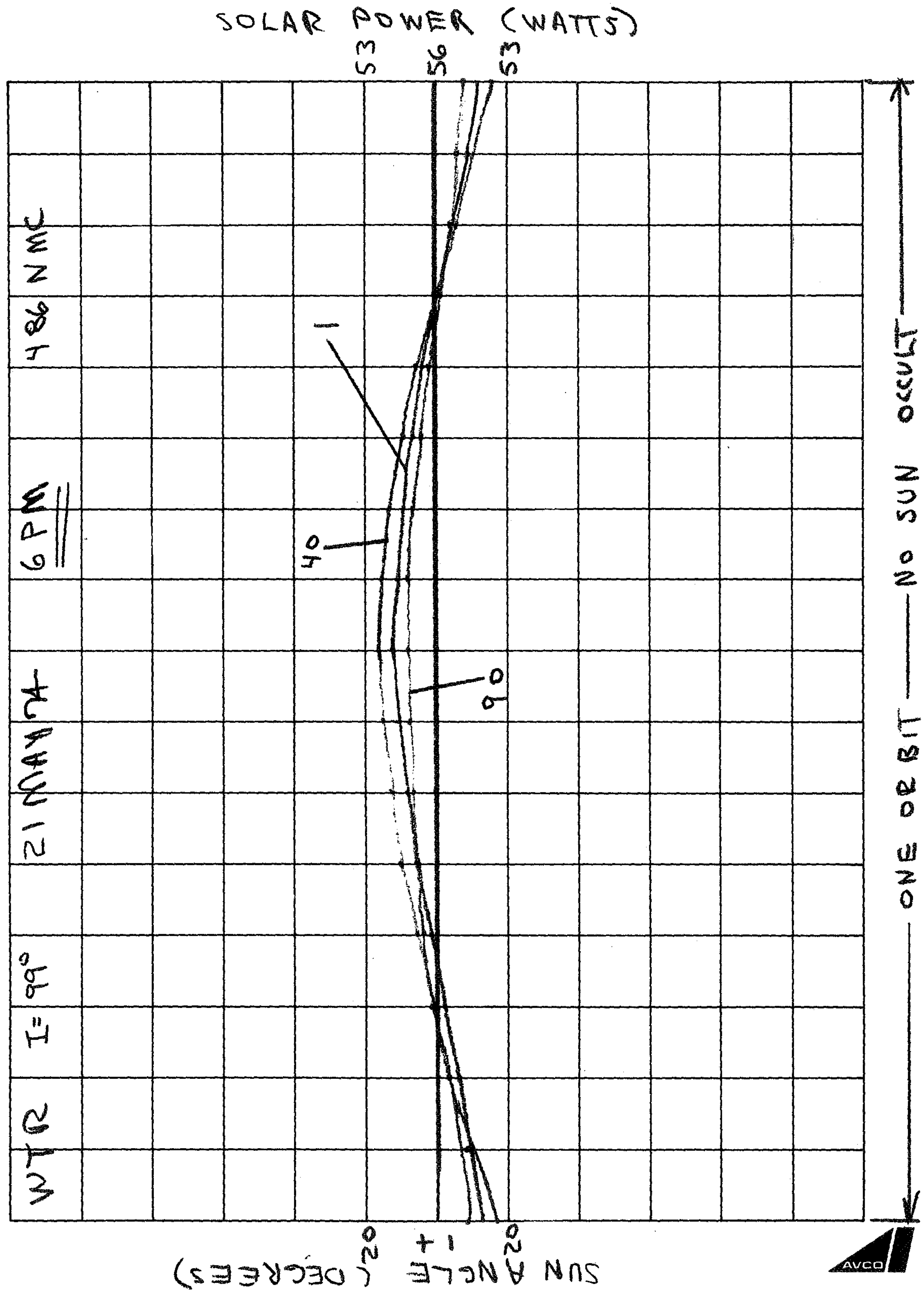
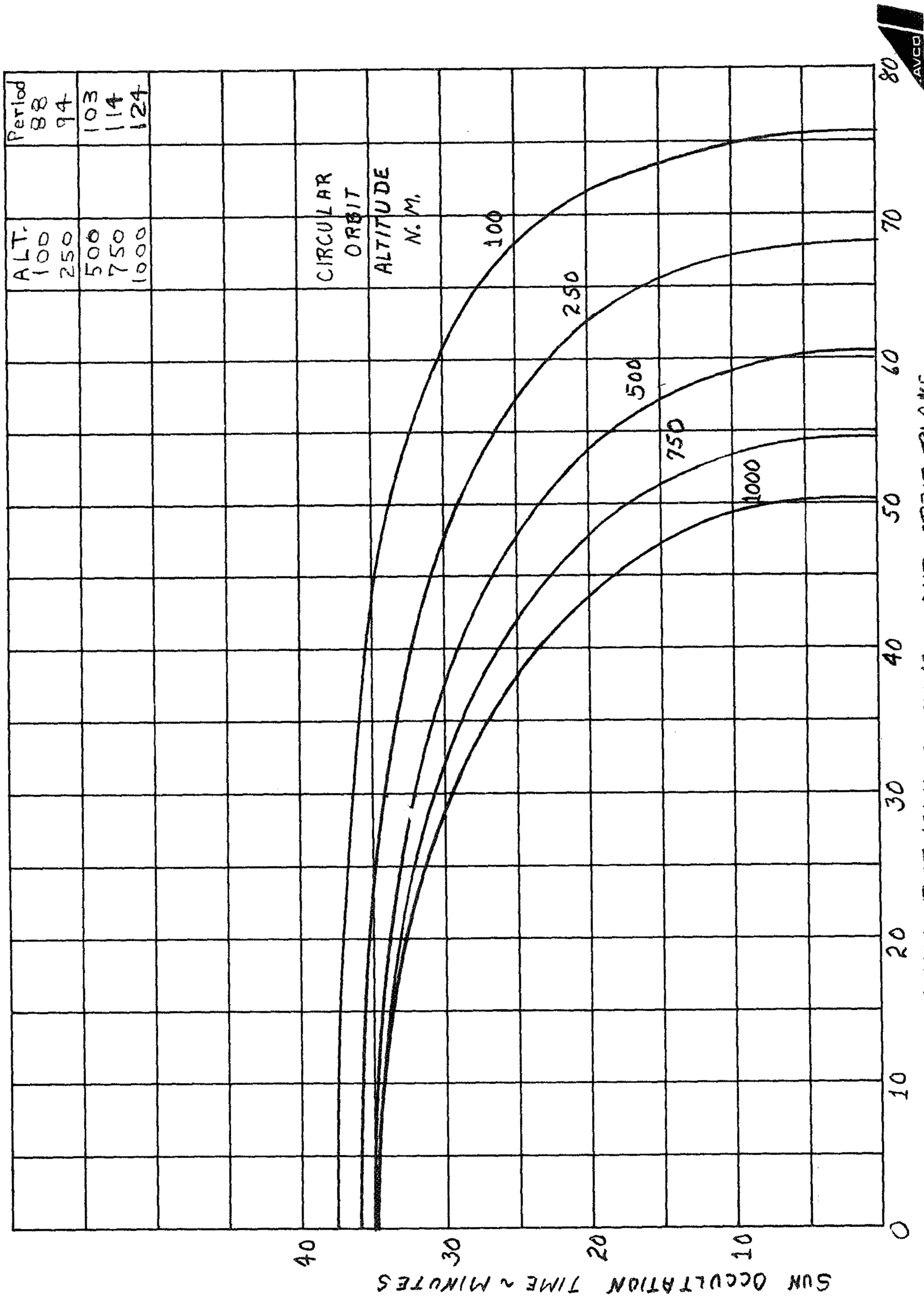

SOLAR POWER (WATTS)
53
56
53
486 NMC
6PM
I = 99°
WTR
1
40
90
ONE ORBIT —— NO SUN OCCULT
SUN ANGLE (DEGREES)

FIGURE 11

FIGURE 12

SUN OCCULTATION TIME



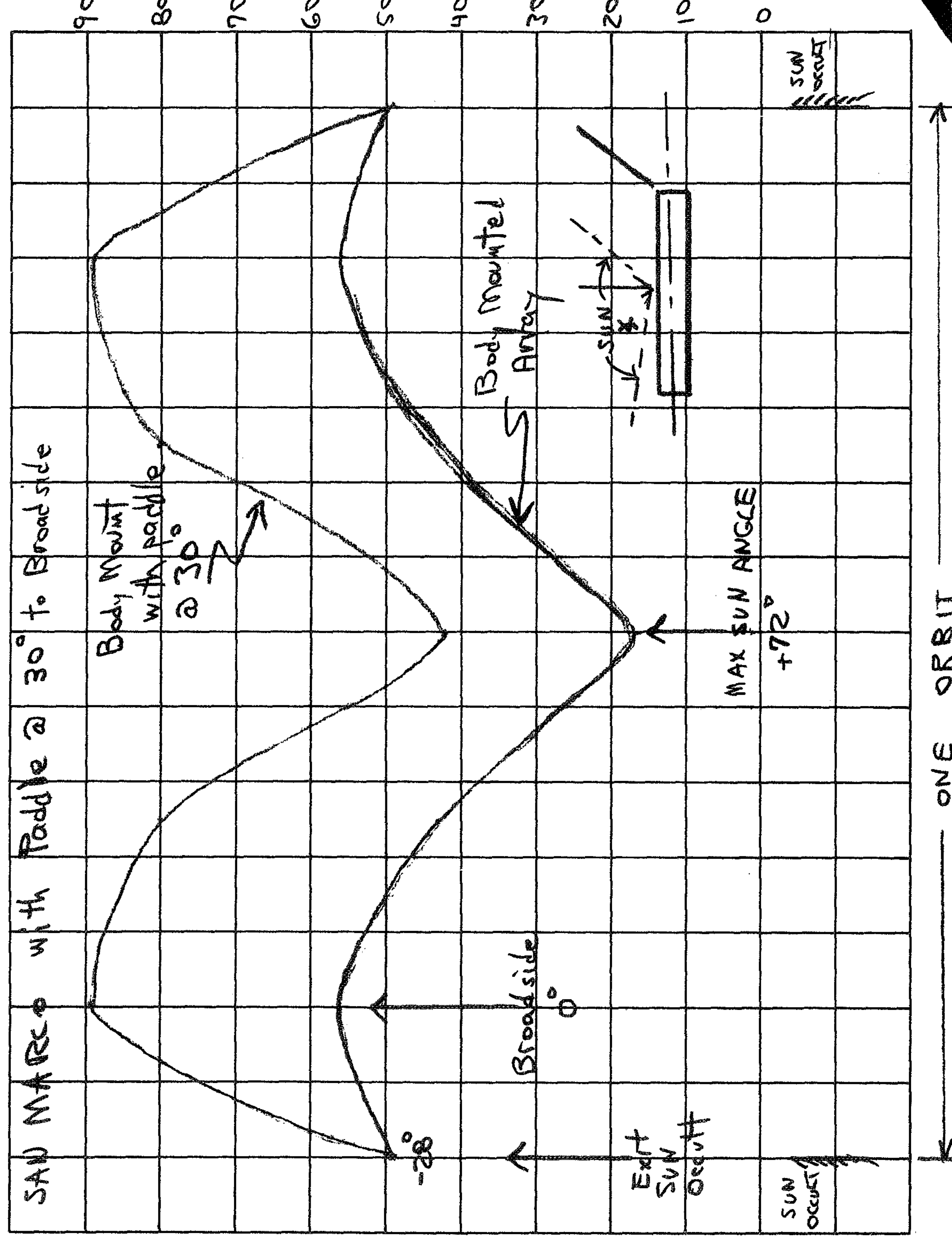

SAN MARCO with Paddle @ 30° to Broadside
90
80
70
60
50
40
30
20
10
0
Body Mount with paddle @ 30°
Body Mounted Array
SUN
MAX SUN ANGLE +72°
Broadside 0°
-28°
Exit SUN Occult
SUN OCCULT
SUN OCCULT
ONE ORBIT

FIGURE 13

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST ☐ RELEASE ☒ ESDT/R-F440-4097

TO	DEPT.	FROM	DEPT.	DATE
S.J. Brzeski (5 copies)	L910	W.J. Kubicki	F440	14 August 70

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control Study for Small Sattelites and Related Subsystems	W159-F440-041		

SUBJECT

COMMUNICATIONS

DISTRIBUTION	SIGNED
H. Burke, W. Fitzgerald, BC, CF, RF	W. J. Kubicki
	APPROVED

INFORMATION REQUESTED / RELEASED

This T/R represents the communication analysis conducted in support of the program. The analysis included a review of the telemetry, tracking and command links together with antenna definitions for VHF tracking, VHF Command and S-Band telemetry. Communication times with STADAN ground stations are also presented.

Telemetry

The communication analysis was conducted from the standpoint of determining the Effective Radiated Power (ERP) required to achieve a link. ERP is defined as the algebraic sum of transmitter power and antenna gain expressed in DB. The performance of the links was investigated at maximum range from ground station to satellite established by a 1000 nautical mile circular orbit and minimum elevation angle of 10° above the horizon.

A review of the parameters listed in Table 1, S-Band Telemetry, indicate a required ERP of -22 dbw. A frequency of 2200 MHz was used to evaluate space loss while the receiving antenna gain and system noise spectral density values are predicted for mid 70's STADAN capability.

A candidate S-Band downlink antenna design and predicted antenna pattern is presented in Figure 1. The half-beamwidth required for full earth coverage at a satellite altitude of 100 n.mi. is approximately 75° where the antenna gain is -4db. Using this gain value and utilizing a readily available 0.1 watt (-10 dbw) transmitter, adequate margin is still available.

Satellite tracking for the circular orbits in question will in all probability still be accomplished by the STADAN Minitrack. While tracking can be accomplished below a horizon elevation angle of 45°, it is preferred to make the measurements above this value where the antenna pattern characteristics are better defined. Table 2 lists the parameters considered in the tracking link resulting in an ERP of -22.5 dbw.

A candidate VHF antenna configuration and predicted pattern coverage is illustrated in Figure.2. The turnstile design featuring four quarter-wavelength whips at 136 MHz provide the coverage required. Utilizing a readily available 0.1 watt (-10 dbw) beacon transmitter requires a minimum antenna gain of -12.5 db. At $\theta = 45^{\circ}$ which corresponds to the tracking elevation angle, the antenna gain is -3 db which again provides ample margin in the tracking link. The beacon transmitter could be modulated with housekeeping data or scientific data in a back-up mode if desired.

Command reception in the 148 - 150 MHz band can be accomplished with the same VHF turnstile antenna and diplexer. Table 3 lists the parameters considered in the command link which result in a minimum spacecraft antenna gain requirement of -29 db. Referring to Figure 2 indicates the requirement can be satisfied.

In order to obtain an insight in the STADAN ground coverage available, several circular orbits were investigated in the 100 to 1000 nautical mile range. The orbits included 911, 685, 486, 310 and 152 n.mi. These altitudes are peculiar in that they essentially repeat themselves in ground station coverage after a 24 hour period or 1 day in orbit. The associated number of orbits/day for these altitudes are 12, 13, 14, 15 and 16 (16 orbits/day for 152 n.mi).

The STADAN manual, X-530-69-109 dated August 1969, lists 19 operational ground stations in the network. Some are mobile van installations assigned to specific satellite programs, at least one has been phased out since the publication data and one is used as a test and training facility leaving 10 currently considered available. Table 4 lists the STADAN ground stations considered in reviewing view times. The stations identified by the asterik will have a predicted S-Band telemetry capability by mid '74.

Figure 3 represents the average view time available for each STADAN ground station for a Wallops Island launch at an inclination angle of 38°. The vertical bars shown for each ground station are representative of the 5 reference orbits previously mentioned where the first bar located on the left side of each group is for the lowest altitude of 152 n.mi. The numerical value located a top the bar represents the number of times the view period is available per day. Example: Station 3, Johannesberg, has an average view time of 250 seconds occuring 5 times a day for a 152 n.mi. orbit; 450 seconds occuring 7 times a day for a 310 n.mi. orbit; 620 seconds occuring 8 times a day for a 486 n.mi. orbit , etc. A "dot" located at the "0" view time indicates that station cannot view the satellite for that particular orbit altitude. Station 2, Fairbanks, can view the satellite only at a 911 n.mi. orbit twice a day for 300 seconds each view. View time is established as long as the satellite remains equal to or greater than 10° above the horizon for the particular ground station.

Figure 4 represents essentially the same information for satellite tracking by the Minitrack Interferometer System. The horizon elevation angle restriction of 50° reduces the overall view times and viewing opportunities. Stations 2 and 8, Fairbanks and Winkfield, provide no tracking information for the orbits reviewed.

Figures 5 and 6 refer to a Wallops Island launch at 52° while Figures 7 and 8 refer to a WTR launch at 99° sun synchronous.

Figures 9 and 10 refer to a San Marco launch at 3° where only station 5, Quito, provides tracking information. Telemetry data transfer is not as restricted but still considerably less than previously reviewed.

Table 5 presents a concise view time/day list for a 486 n.mi. orbit from the four launch points considered. Similar tables could be prepared from the bar graphs presented and will provide the user with data useful in establishing on board storage requirements, data dump time over ground stations and bit rate, opportunity and length of real time data transfer, etc.

TABLE 1

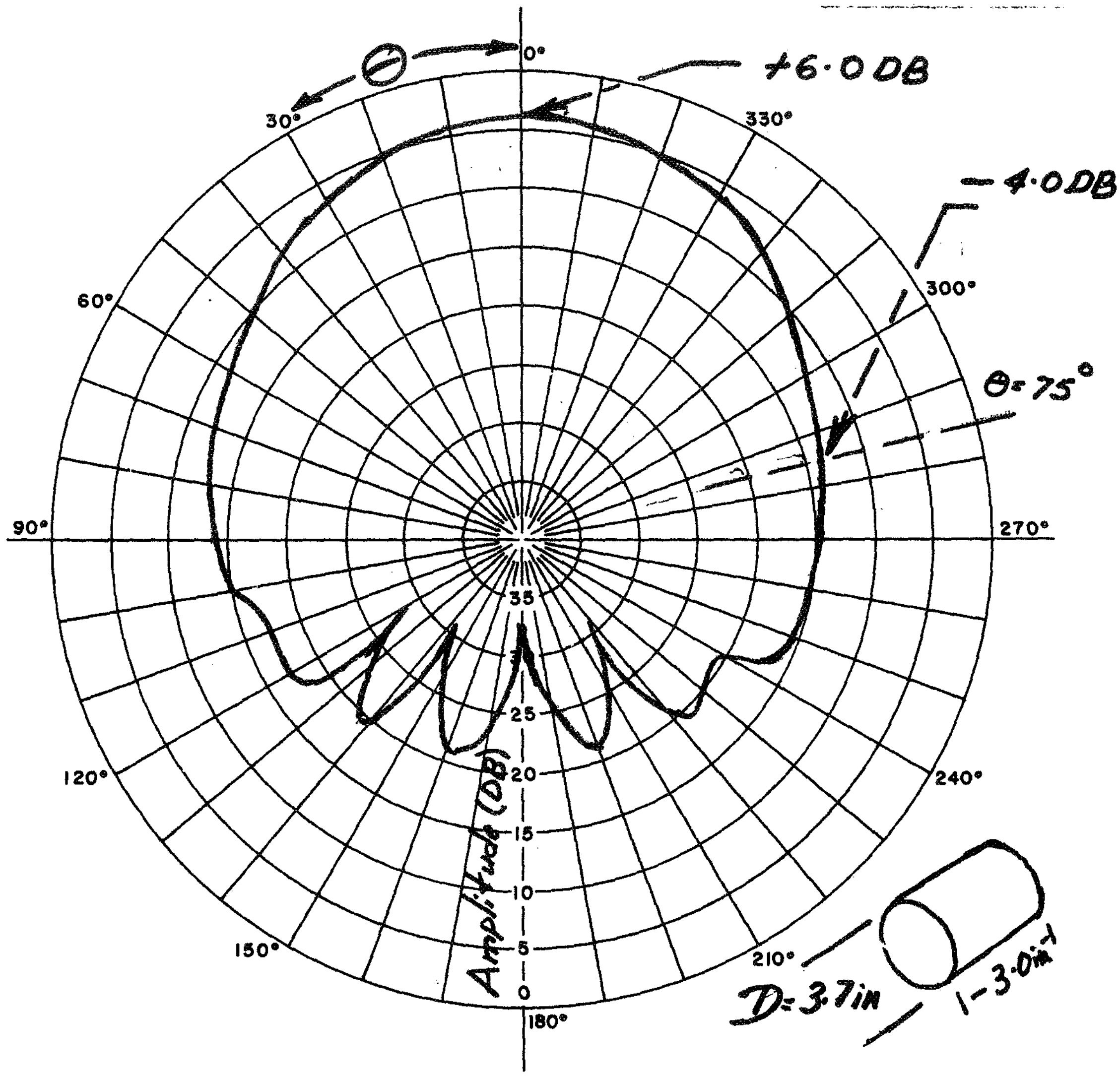
S-BAND TELEMETRY

Parameter	Nominal Value (db)	Note
1. Transmitting circuit loss	-2.0	Assumed hardware loss
2. Space loss at maximum slant range, 10^{o} elevation $\measuredangle$ 1000 N.Mi circular orbit	-161.9	37.85 + 20 log (frequency) + 20 log (range) frequence = 2200 MH_z range = 2272 N.Mi
3. Polarization loss	-.3	Polarization diversity combiner recovers all but 2.7 db of signal
4. Receiving antenna gain	45.0	Predicted gain for 40' dish at 2200 MH_z
5. Total link loss	-119.2	Σ 1 through 4
6. Boltzmann constant	-228.6 dbw	1.37×10^{-23}
7. Reference Noise Temperature	24.6	$290^{o}K$
8. System noise spectral density	-204.0	6 + 7
9. Bit rate (100 Kc)	50.0	Assumed
10. Received $E/_{No}$	34.8	5 + 8 + 9
11. Required $E/_{No}$ $P_{be} = 10^{-3}$ *	6.8	Theoretical requirement
12. Desired range equipment margin	6.0	STADAN personnel request
13. ERP required **	-22.0 dbw	Effective radiated power.

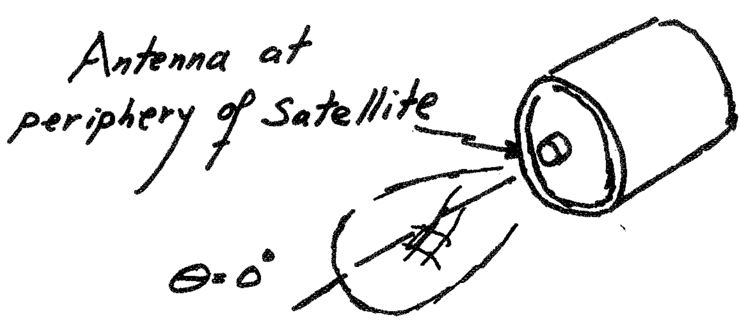
* A more conservative required $E/_{No} = P_{be} = 10^{-5}$ would give a 9.5 db

** For a 1 MH_z data rate, the ERP would be = -12.0 dbw

1-1689



Open ended waveguide Radiation Pattern



S-Band Downlink Antenna

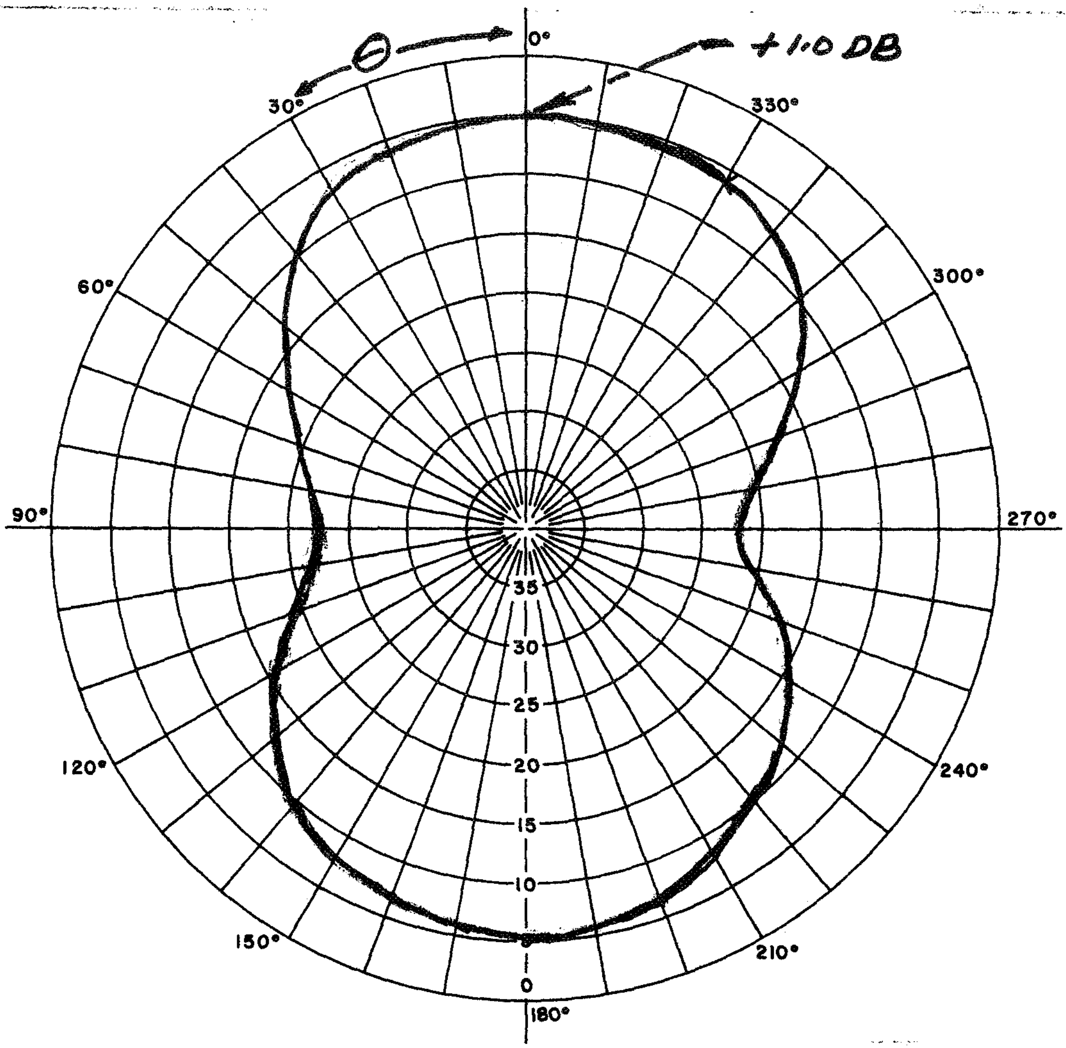
FIGURE 1

TABLE 2

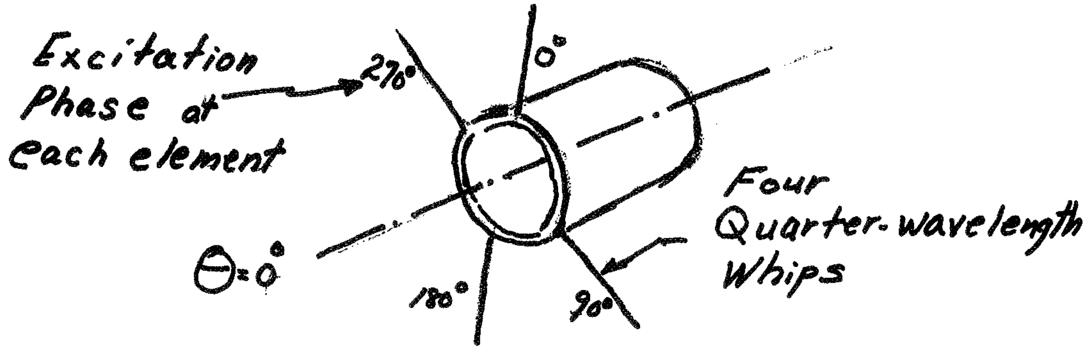
VHF TRACKING

PARAMETER	NOMINAL VALUE (DB)
1. TRANSMITTING CIRCUIT LOSS	- 1.0
2. SPACE LOSS @ MAX SLANT RANGE, 50° ELEVATION ∡ , 1000 N.MI. CIRCULAR ORBIT	-141.5
3. POLARIZATION LOSS	- 3.0
4. MINITRACK ANTENNA GAIN @ 50° ELEVATION ∡	3.0
5. TOTAL LINK LOSS	-142.5
6. REQUIRED RECEIVED POWER LEVEL	-165.0 DBW
7. ERP REQUIRED	- 22.5 DBW

1-1689



Turnstile - Radiation Pattern



VHF - Uplink Antenna

FIGURE 2

TABLE 3

COMMAND LINK

	PARAMETER	NOMINAL VALUE (DB)
1.	TRANSMITTER POWER	33.9 DBW (2.5 KW)
2.	COMMAND ANTENNA GAIN	22.2
3.	RECEIVER CIRCUIT LOSS	-2.0
4.	SPACE LOSS @ MAX SLANT RANGE 10° ELEVATION & 1000 NMI CIRCULAR ORBIT	-143.0
5.	TOTAL RECEIVED POWER	-88.9 DBW
6.	RECEIVER SENSITIVITY	-128.0 DBW
7.	PERFORMANCE MARGIN	+39.1 DB
8.	DESIRABLE RECEIVER INPUT SNR	+10 DB
9.	MINIMUM SPACECRAFT ANTENNA GAIN REQUIRED	-29.1 DB

1 - 1691

TABLE 4

STADAN GROUND STATION IDENTIFICATION

#	STATION	EAST LONGITUDE	LATITUDE
1	FT. MYERS	278-08-03	N-26-32-53
2*	FAIRBANKS	212-29-05	N-64-58-36
3*	JOHANNESBURG	027-06-02	S-25-52-58
4	MOJAVE	243-06-02	N-35-19-48
5*	QUITO	281-25-14	S-00-37-21
6*	ROSMAN	277-07-40	N-35-12-00
7*	SANTIAGO	289-19-51	S-33-08-58
8	WINKFIELD	359-18-14	N-51-26-44
9*	ORRORAL	148-57-20	S-35-37-52
10*	TANANARIVE	47-18-10	S-19-01-17

* PREDICTED S-BAND CAPABILITY BY MID '74

AVCO

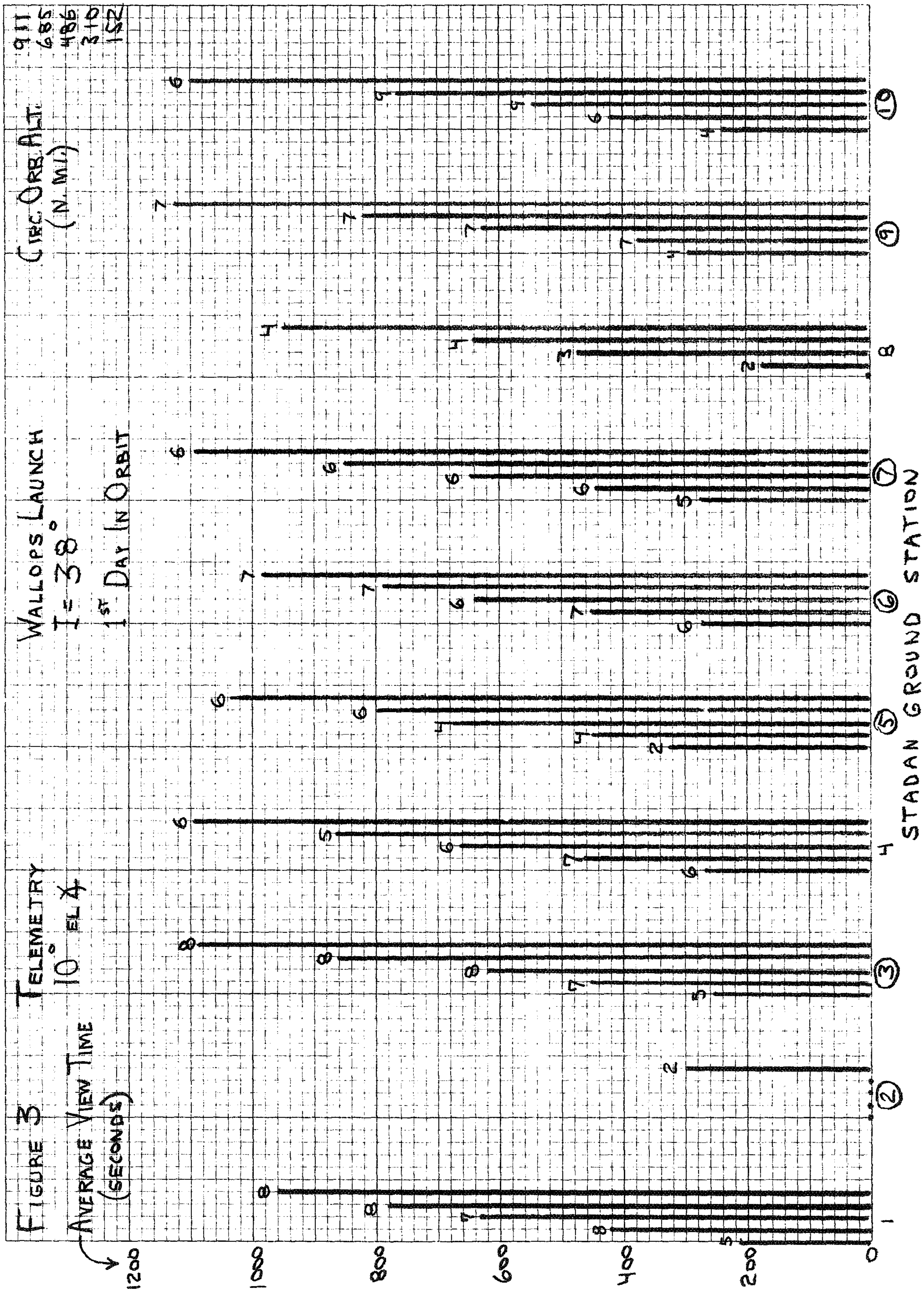


Figure 3
Telemetry
10° EL ∡
Wallops Launch
I = 38°
1st Day In Orbit
Circ. Orb. Alt. (N. Mi.)
911
685
486
310
152
Average View Time (Seconds)
0
200
400
600
800
1000
1200
STADAN Ground Station

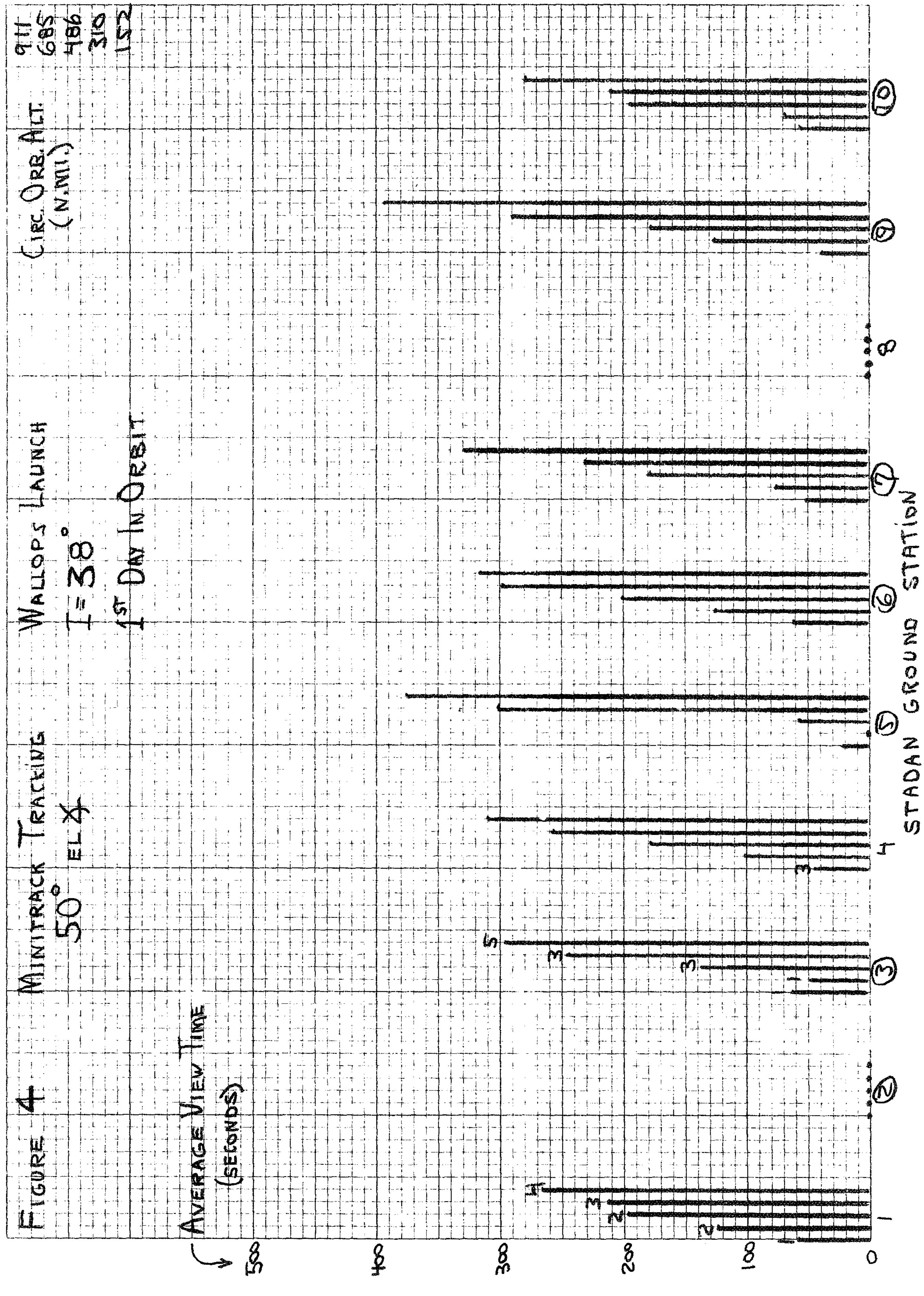


FIGURE 4
AVERAGE VIEW TIME
(SECONDS)
MINITRACK TRACKING
50° EL ∡
WALLOPS LAUNCH
I = 38°
1ST DAY IN ORBIT
CIRC. ORB. ALT.
(N.MI.)
911
685
486
310
152
STADAN GROUND STATION
0
100
200
300
400
500
1
②
③
4
⑤
⑥
⑦
8
⑨
⑩

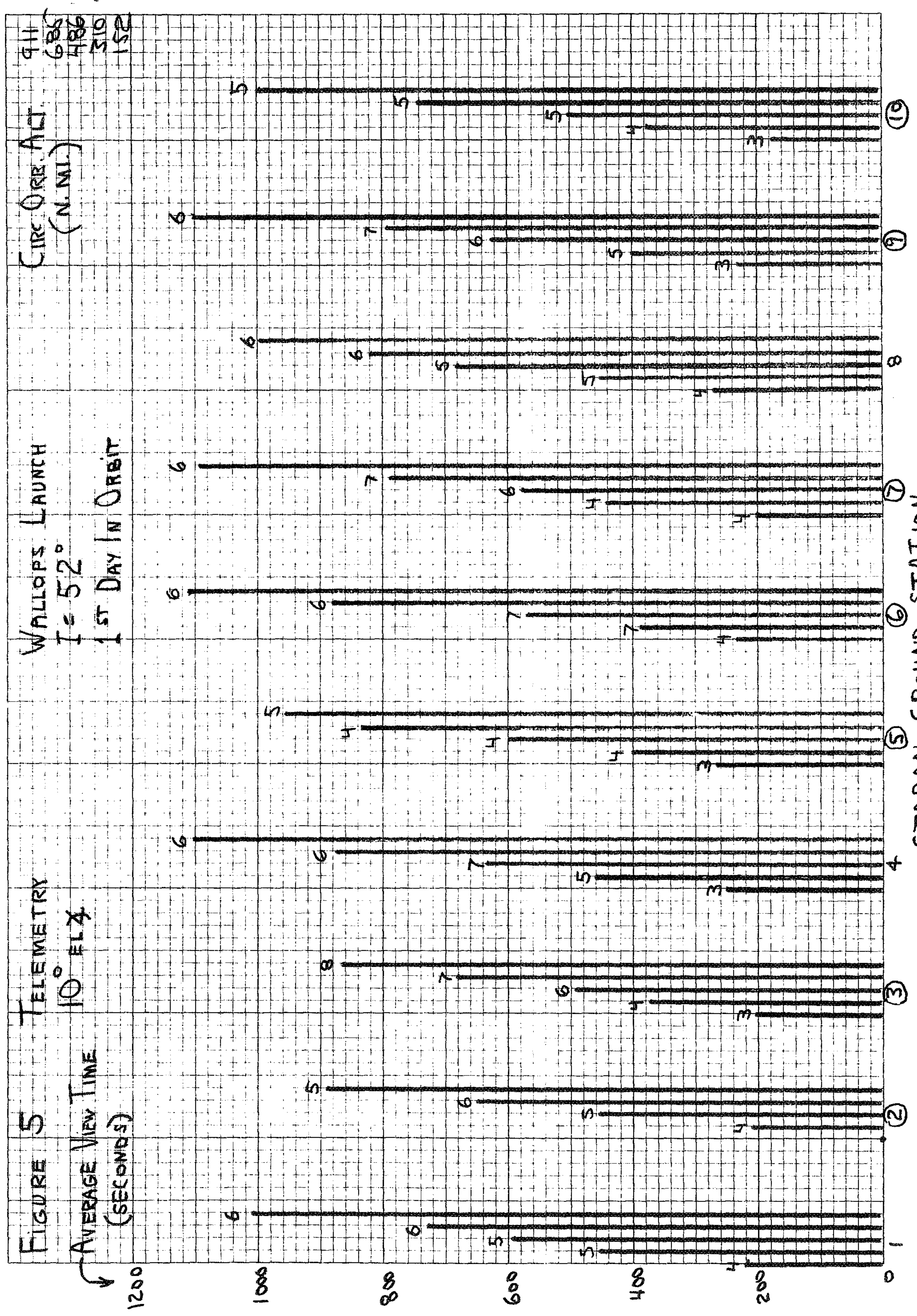


FIGURE 5
TELEMETRY
10° EL∡
WALLOPS LAUNCH
I = 52°
1ST DAY IN ORBIT
CIRC ORB ALT.
(N. MI.)
911
686
486
310
152
AVERAGE VIEW TIME
(SECONDS)
1200
1000
800
600
400
200
0
1
2
3
4
5
6
7
8
9
10
STADAN GROUND STATION

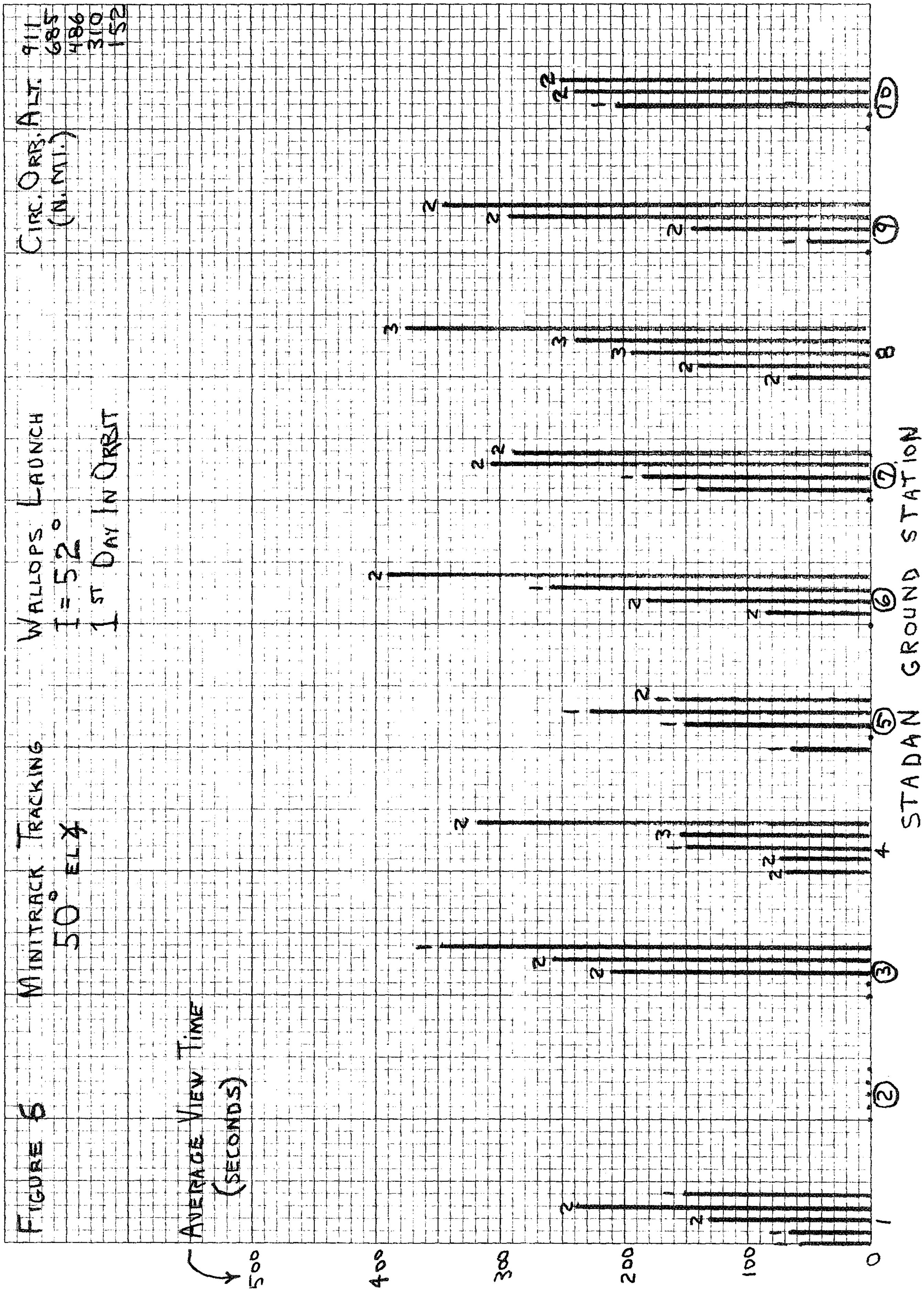


FIGURE 6
MINITRACK TRACKING
50° EL ∡
WALLOPS LAUNCH
I = 52°
1ST DAY IN ORBIT
CIRC. ORB. ALT. (N. MI.)
911
685
486
310
152
AVERAGE VIEW TIME (SECONDS)
500
400
300
200
100
0
STADAN GROUND STATION
1
2
3
4
5
6
7
8
9
10

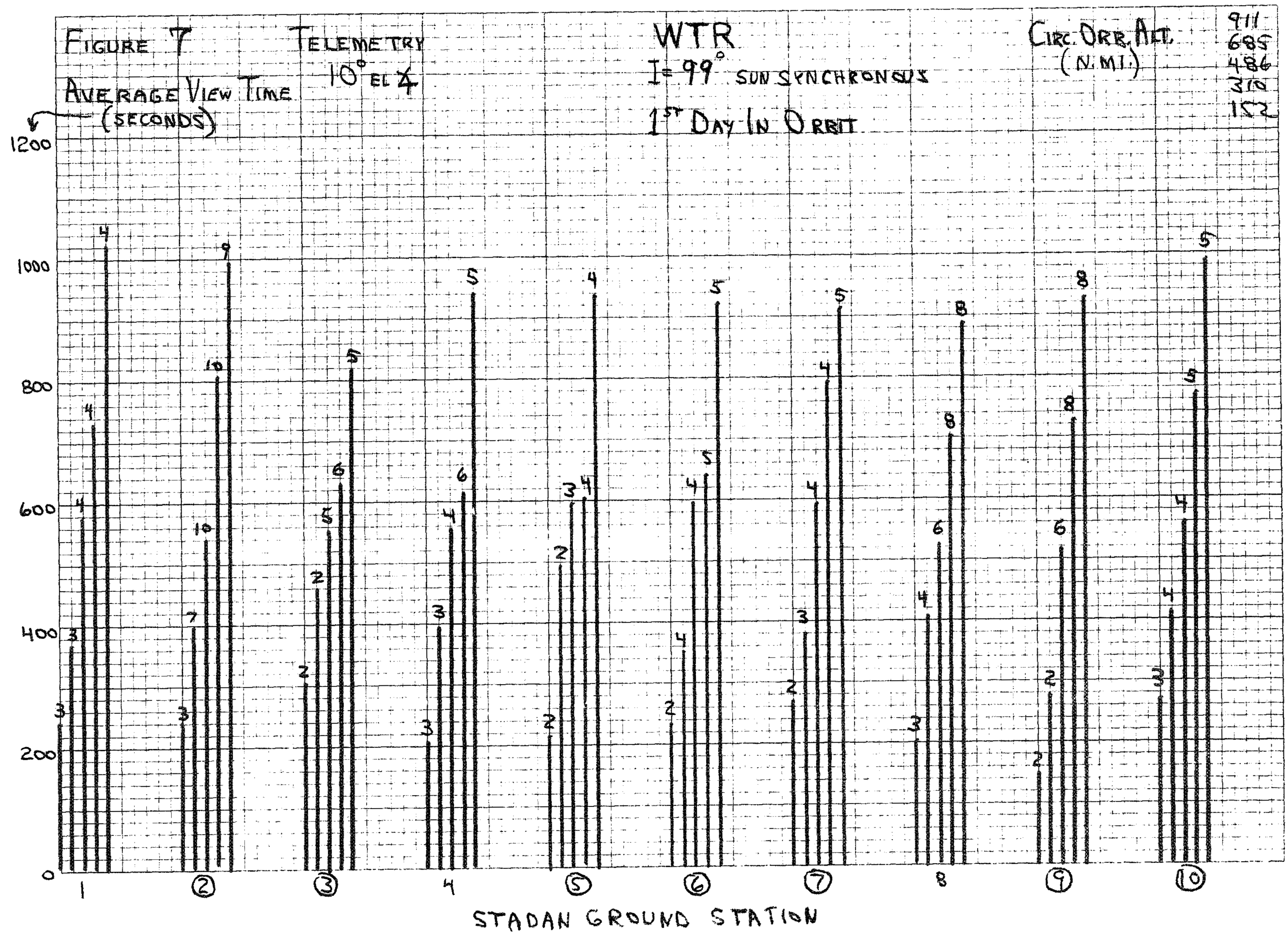


FIGURE 7
AVERAGE VIEW TIME
(SECONDS)
TELEMETRY
10° EL ∡
WTR
I = 99° SUN SYNCHRONOUS
1ST DAY IN ORBIT
CIRC. ORB. ALT.
(N.MI.)
911
685
486
310
152
1200
1000
800
600
400
200
0
1
2
3
4
5
6
7
8
9
10
STADAN GROUND STATION

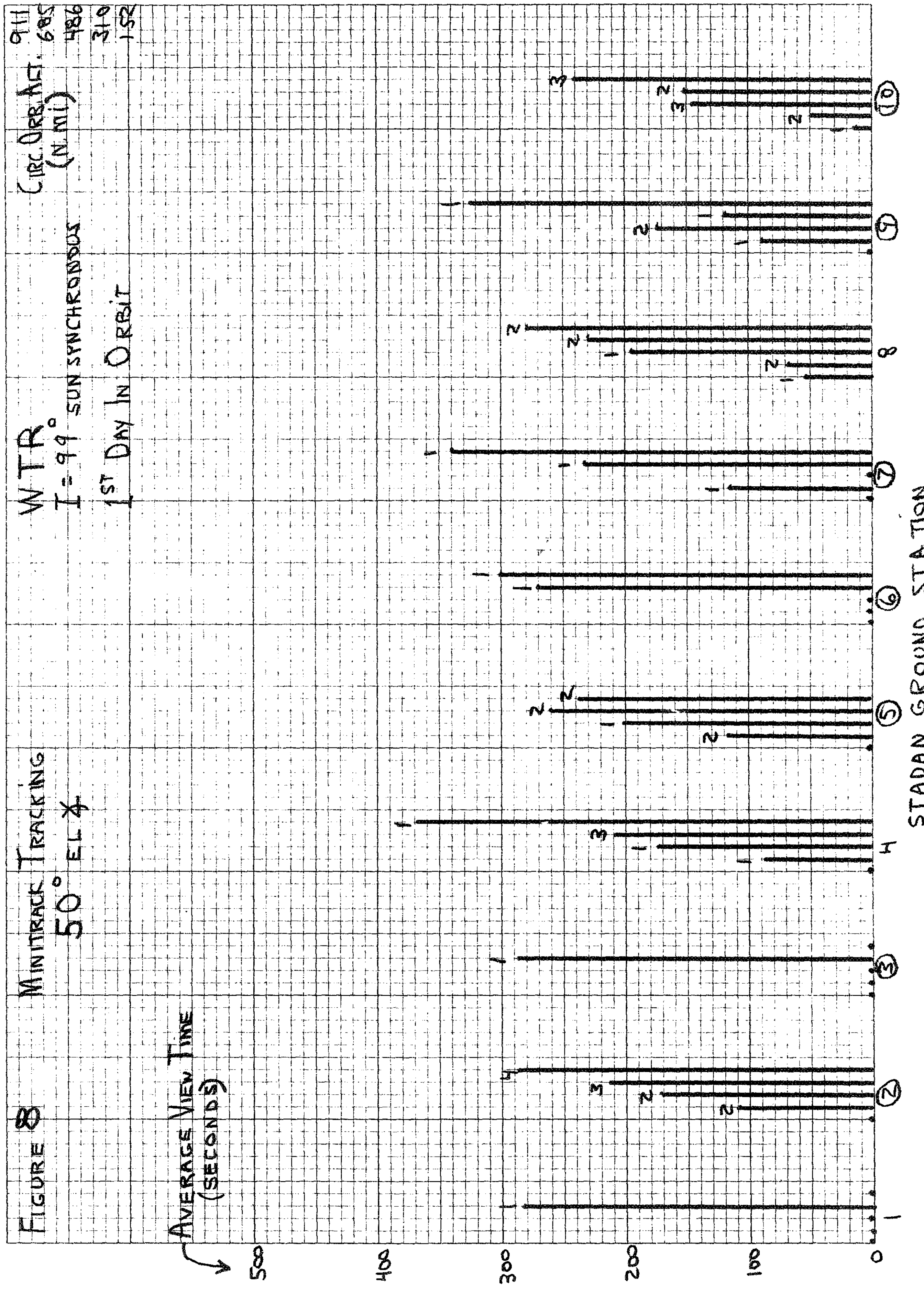


FIGURE 8
MINITRACK TRACKING
50° EL ∢
WTR
I = 99° SUN SYNCHRONOUS
1ST DAY IN ORBIT
CIRC. ORB. ALT. (N. MI.)
911
685
486
310
152
AVERAGE VIEW TIME (SECONDS)
500
400
300
200
100
0
STADAN GROUND STATION
1
2
3
4
5
6
7
8
9
10

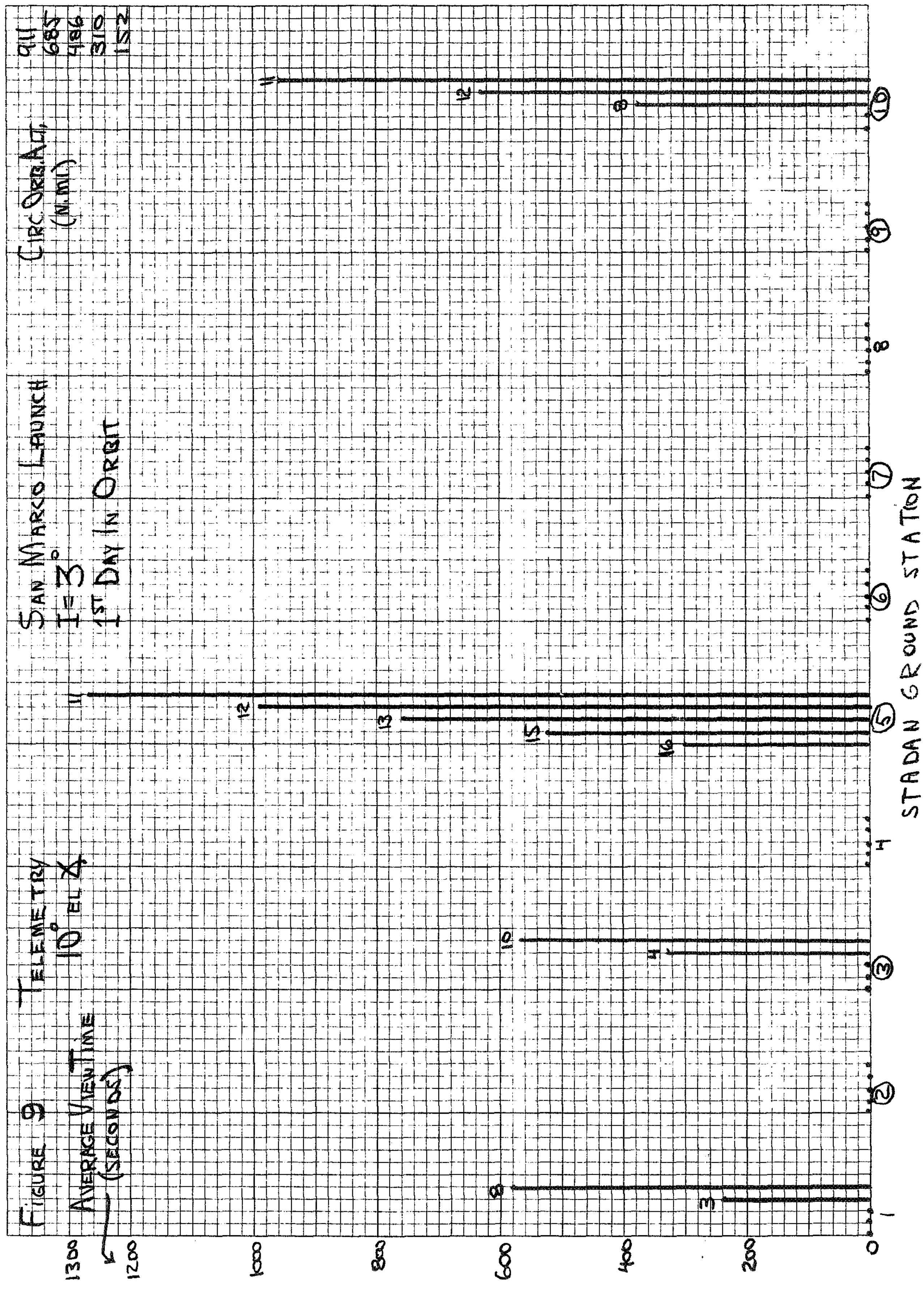


FIGURE 9
AVERAGE VIEW TIME
(SECONDS)
TELEMETRY
10° EL ∡
SAN MARCO LAUNCH
I=3°
1ST DAY IN ORBIT
CIRC ORB ALT.
(N.MI.)
911
685
486
310
152
STADAN GROUND STATION
0
200
400
600
800
1000
1200
1300

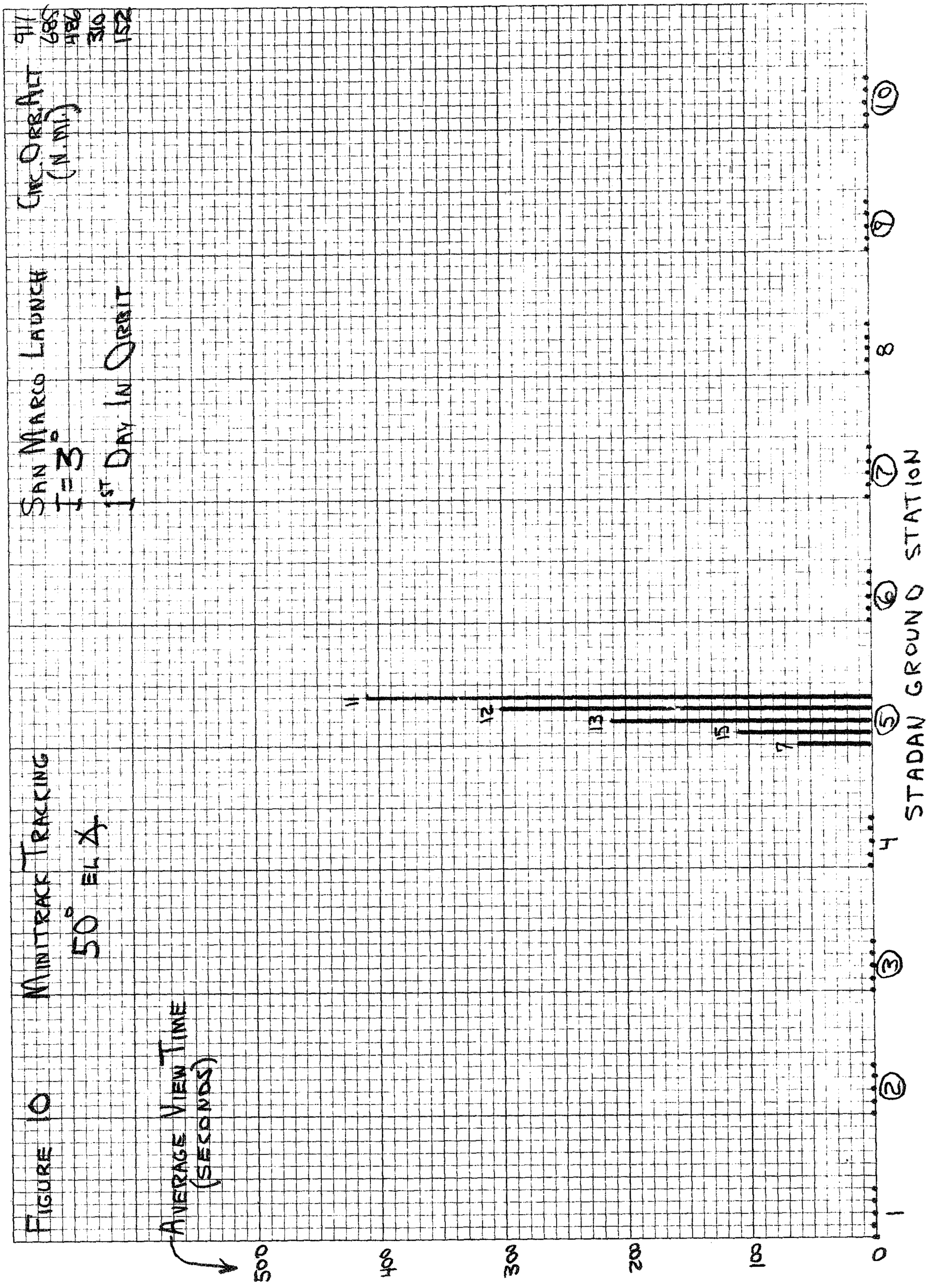


FIGURE 10
MINITRACK TRACKING
50° EL ∡
SAN MARCO LAUNCH
I = 3°
1ST DAY IN ORBIT
CIRC. ORBIT ALT (N.MI.)
685
486
310
152
AVERAGE VIEW TIME (SECONDS)
500
400
300
200
100
0
11
12
13
15
7
1
2
3
4
5
6
7
8
9
10
STADAN GROUND STATION

1 - 1690

TELEMETRY
VIEW TIMES / DAY
486 N.MI. CIRC.

STADAN STATION / LAUNCH POINT	1	2	3*	4	5*	6*	7*	8	9*	10*
SAN MARCO I ≐ 3°			13/12							
WALLOPS I = 38°	7/10		8/10	6/11	4/11	6/11	6/11	3/8	7/10	9/9
WALLOPS I = 52°	5/10q	5/8	6/8	7/11	4/10	7/9	6/10	5/11	6/10	5/8
WTR I = 99°	4/10	10/9	5/9	4/9	3/10	4/10	4/10	6/9	4/9	4/9

OF VIEWS/AVERAGE TIME (MINS)

* PREDICTED S-BAND CAPABILITY BY MID '74

TABLE 5

6-0445
6-59

AVCO CORPORATION

AVCO SYSTEMS DIVISION

201 LOWELL STREET, WILMINGTON, MASSACHUSETTS 01887

TECHNICAL REQUEST [ ] RELEASE [xx] ESDT/R-F440-4002

TO	DEPT.	FROM	DEPT.	DATE
S.J. Brzeski	L910	W.J. Kubicki	F440	25 Aug. 1970

PROGRAM	WORK ORDER NO.	DATE INFO. NEEDED	REFERENCES
Attitude Control Study for Small Satellites and Related Subsystems	W159-F440-041		See Below

SUBJECT

STADAN COMPATIBILITY: PRESENT AND PREDICTED BY MID 1974

DISTRIBUTION

H. Burke, W. Fitzgerald, List B, BC, CF, RF

SIGNED W.J. Kubicki

APPROVED S.J. Brzeski

INFORMATION REQUESTED / RELEASED

References: K. Arnesen to S.J. Brzeski, KA-70-24, 14 August 1970, STADAN Compatibility Data Handling

W.J. Kubicki to S.J. Brzeski, F440-WJK-4093, 12 August 1970, STADAN Capabilities Review

The following T/R and referenced trip reports summarize the STADAN compatibility survey conducted in support of the program. Additional and detailed information in the areas of data handling, telemetry and command are available in the GSFC STADAN Manual X-530-69-109, August 1969 and associated manuals obtained at the GSFC Compatibility meeting attended by Kjell and myself.

The general compatibility in the area of data handling concluded by Kjell can also be expected in the telemetry and command areas.

INTR' JUCTION

The STADAN is one of three NASA-supported networks. The other two NASA networks are the Deep Space Network (DSN), which supports deep-space missions such as the Mariner and Pioneer programs, and the Manned Space Flight Network (MSFN), which supports manned spacecraft programs such as the Apollo flights.

NASA provides support for a fourth tracking network, not a part of the three previously described, which is the Smithsonian Astrophysical Observatory (SAO) network. The SAO supplies an optical tracking capability using Baker-Nunn telescopic cameras. Other tracking networks are maintained by France, Italy, and European Space Research Organization (ESRO). A constant interchange of information exists between all these networks on scientific projects.

The STADAN is a National Aeronautics and Space Administration (NASA) ground support network designed and equipped to handle multiple spacecraft missions simultaneously and continuously. The name STADAN is an acronym derived from Space Tracking and Data Acquisition Network. This network is a worldwide arrangement of multi-function ground instrumentation stations linked to a central operating control facility by a data/teletypewriter (TTY) and voice communications network. This communications network is, in part, the NASA Communications requirements. However, the STADAN is composed only of strategically positioned field stations and technical support activities.

The primary mission of the STADAN is to provide ground-operated instrumentation support to programs utilizing unmanned scientific and applications spacecraft. These spacecraft programs are efforts of both U.S. and international origin intended for the peaceful exploration and utilization of space.

'he secondary mission of the STADAN stations is to support manned spaceflight missions requiring additional instrumentation capabilities when requested by the MSFN. Reciprocally, the MSFN stations support unmanned scientific and application spacecraft requiring additional instrumentation capabilities when requested by the STADAN.

STADAN GROWTH

Within recent years of the national space program, science has made rapid advances in the spacecraft data-gathering process. The results obtained from spacecraft probes have provoked the need for additional space-and atmospheric-related information to aid the progress of space research. This demand for more information became a stimulus for producing a greater number of more complex and sophisticated spacecraft, which in turn increased the space-tracking loads at the stations. To accommodate these increasing tracking requirements and to ensure adequate data gathering ground instrumentation support, additional data-gathering equipments (telemetry links) were added periodically, as required, to the existing STADAN. Each added data link increased the STADAN capabilities, along with increasing its complexity.

With the advent of the Orbiting Geophysical Observatory (OGO) in 1964, a special transportable van containing telemetry, command, data read-out, and recording capabilities was established at the Darwin, Australia, STADAN station. After supporting one OGO mission, this transportable van was transferred to Winkfield, England.. OGO project-unique equipment was also installed at the Johannesburg, Canberra, Quito, Rosman, Santiago, and Fairbanks stations.

Project-unique equipment is used by the STADAN stations in support of specific spacecraft, from launch through the normal scientific lifetime of the space vehicle. These spacecraft require for their space research programs distinctive support equipment ·hich serve particular functions. This project-unique equipment is located at sele ted stations throughout the STADAN.

Another transportable van was instrumented by the Special Projects Section of the STADAN Engineering Division, GSFC, during October 1964. In 1965, the van was placed on the Pacific Missile Range ship USNS Wheeling for a polar orbit OGO mission (POGO). It was located at Darwin, Australia, for an eccenteic orbit OGO (EGO) mission during 1966. Following this period, the transportable station was again installed on the USNS Wheeling in support of POGO missions. Subsequently, in September of 1967, it was situated at Fort Churchill, Canada, to support POGO missions before reinstallation at Darwin in January of 1968. In the autumn of 1968, the van was returned to GSFC for renovation and has undergone further modifications to increase its versatilliy.

Other project-unique capabilities were added to the STADAN as requirement arose. Nimbus spacecraft, for example, required project equipment at the Rosman and Fairbanks stations and a control center at GSFC to fulfill support capabilities.

The OAO program required the installation of project-unique equipment at the Rosman, Quito, Tananarive, Canberra and Santiago stations. This equipment consisted of a ground operating equipment (GOE) console, command console, and a general purpose computer. The latest equipment addition to support the OAO program is the general-purpose modified POGO transportable station. The van has been relocated at the Kauai, Hawaii, site with the primary objective of supporting OAO missions and other STADAN supported missions. This van is now referred to a s the Mobile Command Data Acquisition (CDA) station. Tiros project-unique equipment, consisting of video, telemetry, programming, and photo-processing equipment, was installed at the Fairbanks station nd then later transferred to the Gilmor building. The Gilmor building and its equipment responsibility were transferred to the Environmental Science Service Administration (ESS) in 1966. This facility is now operated and maintained for the National Environmental Satellite Center/Communications Data Acquisition (NESC/CDA) of the ESSA.

In 1964, NASA spacecraft began operating in multifrequency bands: 136, 400, and 1700 MHz. Network-wide station equipment was reconfigured to operate at these multifrequency bands.

The installation of the 85-foot and 40-foot wideband, data-acquisition antennas did not diminish the need for better telemetry antennas to handle the larger class spacecraft. The development of the satellite automatic tracking antennas (SATAN), a widebeam yagi type, began at GSFC soon after the Center was created. The purpose of the SATAN antennas is to complement the data-acquisition and command functions of the large dishes and replace the small, manually controlled, yagi antennas of Minitrack vintage. The SATAN antennas also perform as acquisition aids for the narrow-beam-width dish antennas.

The design and implementation of an integrated control and link verification was initiated during 1964 with the engineering of video-switching equipment and installation of it at most operating STADAN stations. This video-switching subsystem permits the coupling of any telemetry receiver output with any tape recorder input in the station data link. With this subsystem, station equipment configuration control was changed from manual operation to semiautomatic by use of prepatched plug-in boards peculiar to each spacecraft. The equipment operators must manually set status switches which provides the operator with the capability of remotely switching receivers and tape recorders into and out of a data link, and of borrowing receivers and recorders from another station data link, if required. In addition, other signals, such as timing, may be selected for presentation to a tape recorder. The video-switching equipment provides the any-receiver-to-any-tape-recorder capability, referred to as the flexible telemetry link concept.

In 1966, the Canberra station became capable of transmitting a single channel of digital data to GSFC in real-time. This was accomplished by means of a basic transmission subsystem. The same transmission subsystem is presently installed at nine STADAN stations to provide them a real-time data relationship with GSFC.

To meet the necessary requirements for the Applications Technology Satellite (ATS) project, the STADAN also installed, in 1966, a transportable ground station near Toowoomba, Australia. ATS project-unique equipment is also presently installed at the Barstow and Rosman stations.

Commencing in 1968 and continuing through 1969, Goddard Range and Range Rate system was reconfigured by design modifications to provide enhanced operational capability, compatibility with other systems, and growth potential. Three primary features of the modifications directly affecting our program are:

1) Improved system signal thresholds by the addition of an advanced design receiver providing polarization diversity combining.
2) Increased data capability by adding telemetry demodulators to process telemetry, range, and range-rate data simultaneously.
3) Inversion of the present GRARR up-link and down-link S-Band frequencies as a step toward full compatibility with Unified S-Band (USB) systems.

Maximum possible growth potential is incorporated for expansion to a full S-Band integrated system which includes all elements of the existing STADAN, Deep Space Incrumentation Facility (DSIF), Spacecraft Ground Location System (SGLS), and USB networks. The following salient growth features have been considered:

1) Future expansion to provide full tracking, command, and telemetry compatibility with the Apollo USB System.
2) Future expansion to provide full tracking, command, and telemetry compatibility with DSIF and SGLS.
3) Provisions to incorporate specific additional receiving frequency bands (400-410 MHZ, 1435-1535 MHz, and 1700-1710 MHz) by adding the appropriate preamplifiers, mixers, and local oscillators.
4) Provisions to extend the telemtry and command capability by adding the appropriate baseband demodulators and data processing equipment.
5) Consideration of possible expansion, with minimum changes, to include a tracking capability with generalized ratio-type transponders which do not presently exist.
6) Provisions to expand to computer-programmed operating sequences.

PRESENT STATUS

Since its formation, the STADAN has varied in its number of operational stations from the original 11, to a maximum of 22 during 1965 and back to its status 18 operational as of August 1969. Refer to Table 1 for a list of the stations and locations. Although 18 are considered operational, 8 stations have peculiar restrictions as to their use and should not be considered available for support at this time. The restrictions may no longer be applicable by mid '74 therefore are included for reference.

The stations are divided into two groups: STADAN sites, and STADAN stations located on Manned Space Flight Network (MSFN) sites. The division is based on the source of management control of the site on which the station is situated. STADAN sites are those directly under GSFC/STADAN management and which contain STADAN facilities and project-unique equipment. STADAN stations located on MSFN sites are those primarily containing MSFN facilities directly under GSFC/MSFN management, with some STADAN facilities under STADAN operations scheduling and control. All STADAN field sites are directly under the administration of GSFC/STADAN management, except for two, which are Carnarvon, Australia (CARVON) and Kauai, Hawaii (KAUAIH); these two sites are managed by the GSFC/MSFN. Telemetry links in the 136 MHz and 400 MHz range and command link in the 148 MHz range are available at KAUAIH while a VHF command link and S-Band Range and Range Rate link exist at CARVON.

Station 24 at Greenbelt Maryland is used as a Network Test and Training Facility and is not available for supporting any missions.

Station 49 GILMOR is operated and maintained for the National Environmental Satellite Center/Communications Data Acquisition (NESC/CDA) of the Environmental Sciences Service Administration (ESSA).

Cooby Creek, Australia (TOOMBA) is used solely in conjunction with the Applications Technology Satellite (ATS) program, the main purpose of which is to acquire experimental information for the development of satellite communications subsystems and techniques.

Singapore and Solant, two collateral (associated) stations, are operated by the United Kingdom which support STADAN related missions when requested.

Kashima Machi, a collateral station operated by Japan, supports Applications Techr ɔlogy Satellite missions.

TRACKING

Spacecraft tracking is accomplished through the use of four systems presently available in the STADAN; they are:

a. Minitrack Interferometer Tracking System (M/T)

b. Minitrack Optical Tracking System (MOTS)

c. Goddard Range and Range Rate (GRARR) System

d. Compact All Purpose Range Instrument (CAPRI) Radar Tracking System

The Minitrack interferometer tracking system is a radio angle measuring system. The purpose of this system is to supply information for accurately determining the angular position and orbital path of a spacecraft with respect to two orthogonal baselines (north-south and east-west) at a known fixed location.

The Minitrack optical tracking system is a photographic system for precisely determining the position of a spacecraft with respect to time as it passes over a ground station. It is also used for periodic aircraft calibration of the Minitrack interferometer tracking system.

The GRARR system is a sidetone-ranging, doppler-measuring, angle-encoding tracking system. The primary mission of the GRARR system is to track spacecraft from near-earth to lunar distances and beyond. The system provides precise range, range rate, and angular data with respect to a particular spacecraft from as many as three accurately positioned ground stations simultaneously.

The CA RI radar tracking system consists primarily of an AN/FPS-16 (V) C-Band radar set. This system is a high-accuracy, long-range, amplitude-comparison monopulse radar which is capable of manual or automatic acquisition and tracking of objects in flight or orbit.

In addition to the four preceding tracking systems, the 85- and 40-foot telemetry dish antennas also provide antenna X-Y position data.

The Minitrack is available at most stations (refer to Table 1) and should be considered the candidate tracking system for mid '74 operation for the range of orbits considered. The system is VHF and no plans have been formulated to date for converting to S-Band. This results in a dual antenna system on board the satellite. The beam tracking signal can be modulated with telemetry information within a band width and modulation components consistent with GSFC standards.

The GRARR system is presently available at only 5 stations with predicted mid '74 operation at 2 others (Quito and Joburg).

There are two Goddard range and range rate (GRARR) systems presently used: one built by Motorola, Inc. (GRARR-1), and one built by the General Electric Co. (GRARR-2). Both systems have been modified by the General Dynamics/Electronics Corp. Each is capable of operating at either VHF of S-band frequencies. The CRARR-1 system is used at Carnarvon, Rosman, and Tananarive. The GRARR-2 system is used at Fairbanks and Santiago.

The GRARR system, when operated in conjunction with suitable spacecraft transponders, is a high-precision spacecraft-tracking system used for accurately determining the range, range rate, and angular data of spacecraft in near-earth orbits out to lunar distances and beyond. In addition, the system is capable of transmitting commands, and of receiving and demodulating telemetry data.

Two separate RF bands can be used by the tracking stations, S-band and VHF. The S-band frequencies allow more precise angle tracking and avoid the propagation anomalies associated with lower frequencies. In the S-band mode, as many as three ground stations can be operated simultaneously with the same spacecraft transponder. In the VHF mode, only one station at a time can operate with a given transponder. If desired, the VHF receiver subsystem can be used as an acquisition aid when operating with the S-band subsystem. The GRARR system operates in the following frequency bands:

Link	VHF	S-band
Up-link	148 to 150 MHz	1750 to 1850 MHz
Down-link	136 to 138 MHz	2200 to 2300 MHz

The system multifunctional receiver includes the capability for demodulating various types of telemetry modulation. Synchronous and nonsynchronous demodulation of AM and PM is provided, as well as nonsynchronous demodulation of FM. The optimal ratio-combining of the telemetry signals ensures that the best possible signal-to-noise ratio is obtained in the data demodulators. The demodulator output filter bandwidths are selectable in steps from 1.5 MHz to 10 MHz for the nonsynchronous demodulator channels and from 1.5 MHz to 5 MHz for the synchronous demodulator channels.

The remaining tracking alternatives are not considered serious candidates due to their limited use and/or lower tracking accuracy.

TELEMETRY

Telemetry reception within the STADAN is presently available in the 136-MHz, 400-MHz and 1700-MHz with a major portion of the data return at 136-MHz. Reception is also possible on a smaller scale in the S-band (2200-2300 MHz). S-Band telemetry reception is recent operational enhancement capability mode available by adding telemetry demodulators to the GRARR system.

The PCM telemetry standard presently specifies a maximum bit rate capability of 200 Kbps. The limitation exists at VHF, but is not necessarily true at the higher frequencies. The present equipment is fully capable of handling up to 1 MHz bit rates. By mid '74, it is probable that bit rates in the 1-5 MHz range can be handled within the STADAN.

Antenna gains at the S-band installations are predicted to be approximately +45db for the 40' dishes and +50db for the 85' dishes.

A predicted System Noise Spectral Density value of about -204 DBM/Hz can be used to determine telemetry link performance.

Assuming the satellite orbit has been determined to a reasonable accuracy, it is normally detected as it crossed the local horizon. In some cases land masks and physical antenna stops prevent acquisition below 10 degrees. Multipath problems are generally minimized if not eliminated above this limit which is fairly universal throughout the STADAN. Using the 40' S-Band dish, a 6° elevation may be possible assuming no local land masks. For low altitude orbits where the time over a ground station is minimized, the additional time gained would be desirable especially if the data dump is considerable.

COMMAND

The overall capabilities of the two STADAN general-purpose encoders are presented in the following paragraphs.

The Consolidated Systems Corporation (CSC) tone-digital encoder consists of a control panel, Tally paper tape reader, tone encoder, and tone or digital encoder. The control panel allows selection of mode of operation and format of input code during manual operation. The panel displays the code transmitted and houses all other controls and indicators.

The Tally paper tape reader controls the subsystem in the automatic mode. Punched paper tape containing tone on digital commands is fed into the reader, which senses the punched code. Coded data are used to control subsystem operations.

The tone encoder generates 30 tone bursts over the frequency range between 1025 Hz and 11,024 Hz. The length of each tone burst can be determined by manually setting a thumbwheel switch. These tones are produced by 30 separate tuning-fork oscillators that feed a decoder matrix. The decoder matrix selects one of the 30 tone gates that pass the desired frequency to the selected transmitter.

The tone-digital encoder produces a total of 90 digital commands, comprising 70 eight-bit commands and 20 six-bit commands. The selected command lights indicators on the control panel and is applied to the selected transmitter. The tone-digital commands are PCM coded and are used either to modulate the transmitter carrier to produce PCM/AM/AM signals or to key the transmitter on during pulses and off between pulses, resulting in PCM/AM signals.

The OGO command encoder was designed to provide the commands required to control the OGO. The command to be transmitted to a spacecraft may originate at peripheral equipment (such as a computer or paper tape reader) or from the control center by communication lines to a remote station. It also has a built-in control panel from which commands may be initiated. Commands are initiated manually, by digital switches or by a preprogrammed patchboard that is initiated by pushbuttons. A Nixie tube readout on the panel displays the contents of the address and command registers.

Two modes of operation are available, digital or tone mode. In the digital mode, 256 possible PCM/Frequency-Shift-Keying (FSK) commands, at 128 bps, are made available for controlling a spacecraft. True FSK is used to generate the PCM subcarrier signals by forcing a voltage-controlled oscillator to operate at one of two distinct frequencies. Bit synchronization is provided by amplitude-modulating the subcarrier with a 128-Hz source synchronized with the digital bit transition. In the tone mode, there are 343 tone commands available for controlling a spacecraft.

An OGO/ATS encoder was designed to provide the commands required to control OGO and ATS spacecraft. The command to be transmitted to a spacecraft may originate at peripheral equipment (such as a control center computer or paper tape reader) or at the command console control panel (manually). The encoder converts a command request to the required sequence of pulse-code-modulated (PCM) frequency shifts or tone bursts. These are then used to modulate the carrier of the command transmitter.

The sequential tone command generator provides two modes of operation, standard and GRARR. The standard mode of operation, when used with the GRARR system, phase-modulates the transmitted signal to be used to command the IMP series spacecraft. The GRARR mode is used for ranging information.

Four types of command transmitters are currently used in the STADAN: The Collins Model 242 with 200-watt output; the ITA VHF-120H with 2.5-kw output; the Hughes HC-300B with 2.5k-w output; and the General Electric 4BT91A1 with 5-kw output. The ITA VHF-120H, Hughes HC-300B, and General Electric 4BT91A1 transmitters are capable f high-level PCM/AM/AM and low-level PCM/AM modulations. The Collins Model 242 transmitter is operated in the PCM/AM/AM mode only.

A command system is normally equipped with an immediately selectable alternate transmitter for back-up support. Each of these standard VHF transmitters operates in the 122- to 123-MHz and 147- to 155-MHz frequency ranges. All installations are equipped with the peripheral equipment necessary for remote control and monitoring of operational parameters.

The Hughes transmitters installed at ATS support stations have the additional capability of being frequency modulated.

The STADAN command antennas are relatively narrow-beam radiating devices that require accurate pointing. A dual-yagi command antenna and the SATAN command arrays are equipped with separate steerable mounts which are controlled from the command operations area. Single disc-on-rod crossed dipole antennas are mounted on the periphery of an 85-foot dish or in the center of a multi-element receive antenna to permit simultaneous positioning of command and receive antennas by a single mount. The SATAN command array consists of nine radiating elements with a nominal gain of 20db. Polarization capabilities of all antenna subsystems are right-hand circular, left-hand circular, linear horizontal, and linear vertical. Polarizations of the SATAN command array, the disc-on-rod (85-foot dish), and the ATS telemetry and command (T&C) subsystems may be switched from the operating position. The polarization of dual-yagi command and single command elements of multi-element receive arrays can be changed only be manually substituting critical-length feed cables. The nominal gain of each of the single-boom command antennas is 13db.

6-0446
9-68

TECHNICAL REQUEST/RELEASE	FROM W.J. Kubicki	Page 16 of 17 DATE 8/26/70

As is the practice in telemetry reception, command transmission is generally not attempted below the local horizon elevation angle of 10°.

At those select stations having the GRARR capability, command transmission is possible at 1800 MHz. The command equipment exists however, its use is not considered standard in the STADAN. It has not been seriously contemplated but the antenna feeds could be modified for operation in the Unified S-Band frequency range.

The STADAN is currently supporting approximately 40 satellites, both United States and international. If current budgetary and technical trends continue, the possibility exists that the numbers of operational ground stations will be reduced and the number of satellite programs supported will be reduced accordingly. Continued advances in tape recording techniques and higher bit rate transmission minimize the real time viewing requirement on ground stations. The operational or hard core stations will probably be confined to those presently having and those predicted to have the S-Band capability. The preceeding statements are speculative and should be confirmed before mission definitions are formulated.

TABLE 1 - STADAN Ground Stations

Station Location & Tracking Number		Code Name	Latitude	Longitude (East)	Minitrack Antenna	Range and Range Rate	40' Antenna	85' Antenna
Fort Myers, Florida	3	FTMYRS	23° 32' 53.516"N	278° 08' 03.887"	1	0	0	0
Quito, Ecuador	5	QUITOE	00° 37' 21.751"S	281° 25' 14.770"	1	*Pred	1	0
Santiago, Chile	8	SNTAGO	33° 08' 58.106"S	289° 19' 51.283"	1	1	1	0
Winkfield, England	15	WNKFLD	51° 26' 44.122"N	359° 18' 14.615"	1	0	0	0
Johannesburg, South Africa	16	JOBURG	25° 52' 58.862"S	027° 42' 27.931"	1	*Pred	1	0
Barstow, California	17	MOJAVE	35° 19' 48.525"N	243° 06' 02.776"	1	0	0	0
Fairbanks, Alaska	19	ALASKA	64° 58' 36.572"N	212° 29' 05.794"	1	0	0	0
Rosman, North Carolina	20	ROSMAN	35° 12' 00.710"N	277° 07' 41.230"	0	1	0	1
Canberra, Australia	21	ORORAL	35° 37' 52.718"S	148° 57' 20.867"	1	0	0	0
Tananarive, Malagasy Republic	22	MADGAR	19° 00' 27.109"S	047° 18' 00.468"	1	1	0	1
Greenbelt, Maryland	24	NTTF	38° 59' 59.638"N	283° 09' 29.965"	1	0	0	0
Kauai, Hawaii	37	KAUAIH	22° 07' 26.546"N	202° 19' 50.970"	0	0	0	0
Fairbanks, Alaska (NESC/CDA)	49	GILMOR	64° 58' 42.667"N	212° 30' 18.045"	0	1	1	1
Carnarvon, Australia	52	CARVON	24° 54' 14.860"S	113° 42' 55.020"	0	1	0	1
Cooby Creek, Australia	66	TOOMBA	27° 32' 50.496"S	151° 56' 16.153"	0	0	0	0
Kashima Machi, Japan		KASATS			0	0	0	0
Singapore		SNPORE			0	0	0	0
South Atlantic Ocean		SOLANT			0	0	0	0

* Predicted