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ELECTRONICALLY AGILE RADAR (EAR) INTERFACE INVESTIGATIONS

BOEING WICHITA COMPANY
A DIVISION OF THE BOEING COMPANY
WICHITA, KANSAS 67210

APRIL 1977

FINAL REPORT JANUARY 1976 - JULY 1976

AIR FORCE AVIONICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

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A flight test interface for the Electronically Agile Radar (EAR) in a model B-52G strategic weapon system test bed is defined. Space provisions were developed considering flight test requirements and, for the nose radome area only, the requirements of B-52 defensive subsystems. The EAR and instrumentation require added B-52 environmental control system capability. An analysis of radome characteristics indicates beam deflection compensation requirements are reasonably continuous and change little with frequency

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excursions. Most of the bombing-navigation system will be removed to make room for the EAR, but a doppler-inertial navigation capability independent of the EAR Test Article will be retained. Flight test planning includes delineation of the radar contractor and test bed contractor responsibilities for materials and services.

FOREWORD

This report describes the results of a flight test interface investigation for the Electronically Agile Radar (EAR). The study was performed by The Boeing Company, Wichita Division, Wichita, Kansas on Contract Number F33615-76-C-1145 during the January to July 1976 time period. The sponsoring agency was the Air Force Avionics Laboratory, Air Force System Command, United States Air Force, Wright-Patterson AFB, Ohio. The project engineer was Mr. John T. Cogan, AFAL/RW-EAR. The study involved a broad scope of airplane technical disciplines. The report includes contributions, not otherwise credited, from representatives of several technical areas.

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SECTION I

INTRODUCTION AND SUMMARY

The Air Force is currently developing an Electronically Agile Radar (EAR) for possible incorporation into a strategic weapon system in the 1980 time frame. The EAR is an X-band, multimode radar designed to perform air-to-ground mapping, velocity measurements, and terrain following/terrain avoidance functions in accordance with strategic weapon system requirements. The candidate strategic weapon systems which EAR addresses are the B-1, FB-111, and B-52. The EAR development plan includes flight testing in a B-52G Test Bed aircraft.

The Boeing Company has conducted a Flight Test Interface Investigation under Air Force Contract F33615-76-1145. The objective of the investigation was to provide data for formulation of a flight test plan by the EAR development contractor (Westinghouse Electric Corporation) and to compile data for follow-on activity concerned with Class II aircraft modification and EAR flight test. The investigation addressed installation provisions and interfaces for the EAR Test Article and instrumentation as well as test planning and test support requirements. The time phasing of the interface investigation paralleled the final months of EAR Service Test Model detail design and preliminary design of EAR flight test instrumentation. Coordination with the Westinghouse Electric Corporation was continuous throughout the investigation.

Space provisions were developed for the EAR test article and instrumentation based on EAR installation requirements. In the nose radome cavity, these provisions were also constrained to critical EAR equipment locations that are feasible for any subsequent B-52 implementation considering other B-52 systems (e.g. ECM). All other locations considered only flight test bed requirements. The selected antenna location is on the Station 96 bulkhead centered at Buttock Line 0 and Waterline 165.8 with related electronic equipment in appropriate proximity. A new rigid frame will mount the antenna and inertial motion sensor. An EAR operator's station is provided at the B-52 radar navigator's station, the EAR test engineer's console is at the B-52 gunner's station.

The EAR Test Article and instrumentation are projected to give a net increase of 13+ kilowatts in the cabin heat load which exceeds B-52 environmental control system reserves. The installation provisions will include a liquid cooling system and an additional air-conditioning pack.

Major elements of the B-52 bombing/navigation system (BNS) must be deleted to make room for the EAR. A B-52 navigation capability independent of the Test Article and a source of airplane state data for Test Article interface is provided based on use of the AGM-69A carrier navigator, the ancillary doppler and compass systems, and selected BNS components.

Flight test planning, including a structured definition of associate contractor responsibilities for materials and services, was worked out in coordination with Westinghouse. The EAR development contractor will determine conditions and objective for each flight and operate all EAR associated systems and instrumentation. Boeing will accomplish airplane operations and detailed scheduling in consonance with Westinghouse requirements. The handling of test data includes quick look analysis at Wichita and in depth analysis at Baltimore.

The interface investigation also provided B-52 technical data affecting EAR service test model design activities. These include aerodynamic, engine performance, and flight control system characteristics as well as determination of radome beam deflection compensation requirements.

SECTION II

STUDY TASKS

The specific tasks to be accomplished were:

1. TASK 1 - EAR INSTALLATION

    1) Provide a general configuration for installation of EAR in the B-52 nose area in terms of form fit, function, weight and environmental control.

    2) From this general configuration determine line lengths to interface EAR components, radome characteristics, and power and cooling provisions.

    3) Provide B-52 flight characteristics data for implementation of EAR TF/TA parameters.

    4) Determine feasibility and requirements for synchronization of EVS to EAR.

2. TASK 2 - MOTION SENSOR/NAVIGATOR INSTALLATION

    1) Provide a preliminary design configuration for installation of motion sensor/navigator which takes into account the EAR motion compensation requirements.

    2) Provide data related to physical installation as it relates to the EAR contractor's test planning.

    3) Determine the interface avionics required for interface of the motion sensor/navigator to resident airplane systems.

3. TASK 3 - INSTRUMENTATION INSTALLATION

    1) Provide a general configuration for installation of EAR instrumentation, ancillary equipment and test engineer's console.

    2) Determine availability of prime console space for test engineer's console from consideration of required equipment retentions at the RN/N stations.

    3) Determine general requirements for test bed airplane and environment instrumentation.

4. TASK 4 - FLIGHT TEST PLANNING

    1) Provide test planning data in the areas of airplane basing, crew requirements, data handling and ground station, airplane maintenance (per Attachment 1 of AFSCR 66-24), EAR maintenance, test instrumentation maintenance, EAR support facilities, flight safety, test range coordination, EMI test and B-52 peculiar overall test planning.

2) Prepare a work statement and schedule defining test bed contractor activities and materials required to prepare the airplane and support the planned test.

3) Assist the Air Force and EAR contractor in defining test range requirements, including remote/portable range instrumentation systems and data recording/processing needs.

4) Support EAR instrumentation planning and design in terms of time coordination, ground station compatibility and environmental considerations.

SECTION III

INSTALLATION REQUIREMENTS

1. TEST ARTICLE

The Test Article is defined as the EAR, the motion sensor/navigator and associated ancillary equipment. The configuration of the test article is controlled by the EAR development contractor. The installation approach developed in this study was determined considering the following installa-ation requirements which evolved through continuing coordination with Westinghouse:

1) The EAR antenna and the IMU should be installed on common, rigid structure in close proximity to one another to minimize their relative motion.

2) The BIT horn antenna will be mounted as a part of the EAR antenna.

3) The distance between the Transmitter and the Transmitter High Voltage Power Supply should be minimized because of the high voltage cabling between the two units. The high voltage cable will be supplied by Westinghouse.

4) The maximum length of 10-30498 cable between the digital assembly and the Low Power RF Assembly (LPRF) shall be 30 feet because of the (25 MHz) data rate carried by these interconnecting cables.

5) The maximum length of 10-30498 cable between the Microwave assembly and the LPRF shall be 12 feet to minimize the impact on receiver noise figure.

6) The maximum length of waveguide run between the Transmitter and the Microwave assembly shall be 10 feet.

7) The maximum length of the three "phase matched" waveguide runs between the Microwave assembly and the antenna shall be 6 feet. These phase matched waveguides will be supplied by Westinghouse.

8) The maximum length of 10-30498 cable between the Transmitter and the LPRF shall be 8 feet.

9) The Scan Converter/Pattern Raster Generator (SC/PRG) will be located in the crew compartment so that it is readily accessible during flight.

10) Except for the IMU, all of the Inertial Navigation Units and the EAR units will be hard mounted directly to structure. The IMU isolators, however, are part of the GFP supplied IMU unit.

11) The following units may be ram air cooled:

    - Transmitter high voltage power supply (HVPS) 2 each

- Interface Electronics Unit (IEU)
- Digital Computer Unit DCU (ROLM 1602 Computer)
- Digital Assembly
- 5 Volt inverter power supplies (HO-3) 4 each
- Low Voltage Bulk Power Supply (HO-2)
- Process and Display Unit (PDC)
- Scan Converter/Pattern Raster Generator (SC/PRG)
- Liquid Coolant Heat Exchanger
- Low Power RF Unit

12) Inertial Nav Unit IMU to IEU shall be 20 ft. maximum length.

13) The maximum length of flexible 10-30498 cable between the Driver power supply and the antenna shall be 8 feet.

14) The following units will be interconnected with flexible 10-30498 RF coaxial line:

- LPRF and Transmitter
- LPRF and M:crowave Assembly

15) The EAR Test Article Antenna and FPA/P&C main and redundant transmitters will be liquid cooled.

2. RESIDENT SYSTEMS

Resident systems are defined as those systems presently installed in the B-52. In order to install the Test Article in the B-52, several of these systems will be relocated or removed.

The following installation requirements pertain to ECM equipment in the nose radome area. Although the ECM systems are not required to be operational for the Flight Test, these requirements were considered to assure that locations of critical EAR equipments were feasible in terms of any subsequent B-52 implementation so that measured EAR capability might be representative of a strategic weapon system implementation.

It is intended that during test installation design, complete installation provisions will not be engineered for the relocated nose radome equipment.

a. AN/ALQ-117

1) The maximum waveguide length between the PD transmit antenna in the eyebrow window and the PD transceiver is 10 feet.

2) The maximum RF coaxial line length between the PD receiver antenna and the PD transceiver is 7 feet.

3) The maximum coaxial line/waveguide length between the PR receiver and the PR receive antenna is 5.5 feet (2 feet flex guide plus 3.5 feet of 10-30498 coax). This assumes an increase in the length of only the coaxial line portion of the transmission line system.

4) The waveguide lengths between the PR transmitter and the antenna selector switch; and between the antenna selector switch and the six antennas cannot be increased beyond their present lengths.

5) The maximum length of the 10-30498 coaxial line between the PR transmitter and the PR receiver is 4 feet.

b. AN/ALT-28

The length of the waveguides between the dual horn antenna and the existing isolators should not be increased.

SECTION IV

TEST SYSTEM INSTALLATION

Space provision layouts for the EAR test article and EAR instrumentation were developed. The layouts evolved in response to installation requirements and test planning jointly developed with Westinghouse as the test system design progressed. The EAR test installation involves deletion of not only the B-52 forward looking radar, but major elements of the bombing/navigation computer group as well, including the remote module racks, the power modules rack, and numerous controls and displays.

1. ANTENNA LOCATION

The placement of the EAR antenna - SPN/GEANS IMU and associated equipment in the nose radome area for the development flight test will require removal of the complete complement of ECM equipment and racks in that area.

A new rigidized frame assembly will be required to mount the EAR antenna and the SPN/GEANS IMU within the allowable relative motion tolerances. The Station 96 bulkhead and supporting structure adjacent to the cabin floor will be modified to provide precision attachment points for the antenna mount and to maintain the required alignment. The IMU mounting plate will provide attachment points for optical alignment fixtures to permit an alignment check between the mount and IMU to track the fatigue rate of the IMU shock mount.

An early determination was made that the EAR and ECM antennas should be located as far aft as possible (near the B.S.96 bulkhead) in order to maximize the frontal area available for assignment.

Initially, four B-52 nose radome configurations were developed taking into consideration spatial dimensions, installation requirements, and B-52 resident system requirements. Frontal views of the four configurations, respectively labeled 1A, 2A, 3A, and 4A are shown in Figures 1 and 2. A preliminary microwave analysis of these four configurations was accomplished. This analysis included Radome Boresight Error, Inter and Intra System Isolation, and Antenna Pattern/Gain. Tables I, II and III, respectively, summarize the results. From these data, Configuration 1A was selected. The selection was especially influenced by the Isolation analysis which shows Configuration 1A to have no degraded systems. Antenna Pattern coverage and gain analyses showed both Configurations 1A and 2A to be acceptable. The radome boresight error data indicated that the apparent complexity of the required beam deflection compensation was similar for locations 1A, 2A, and 3A, and only moderately complicated by location 4A.

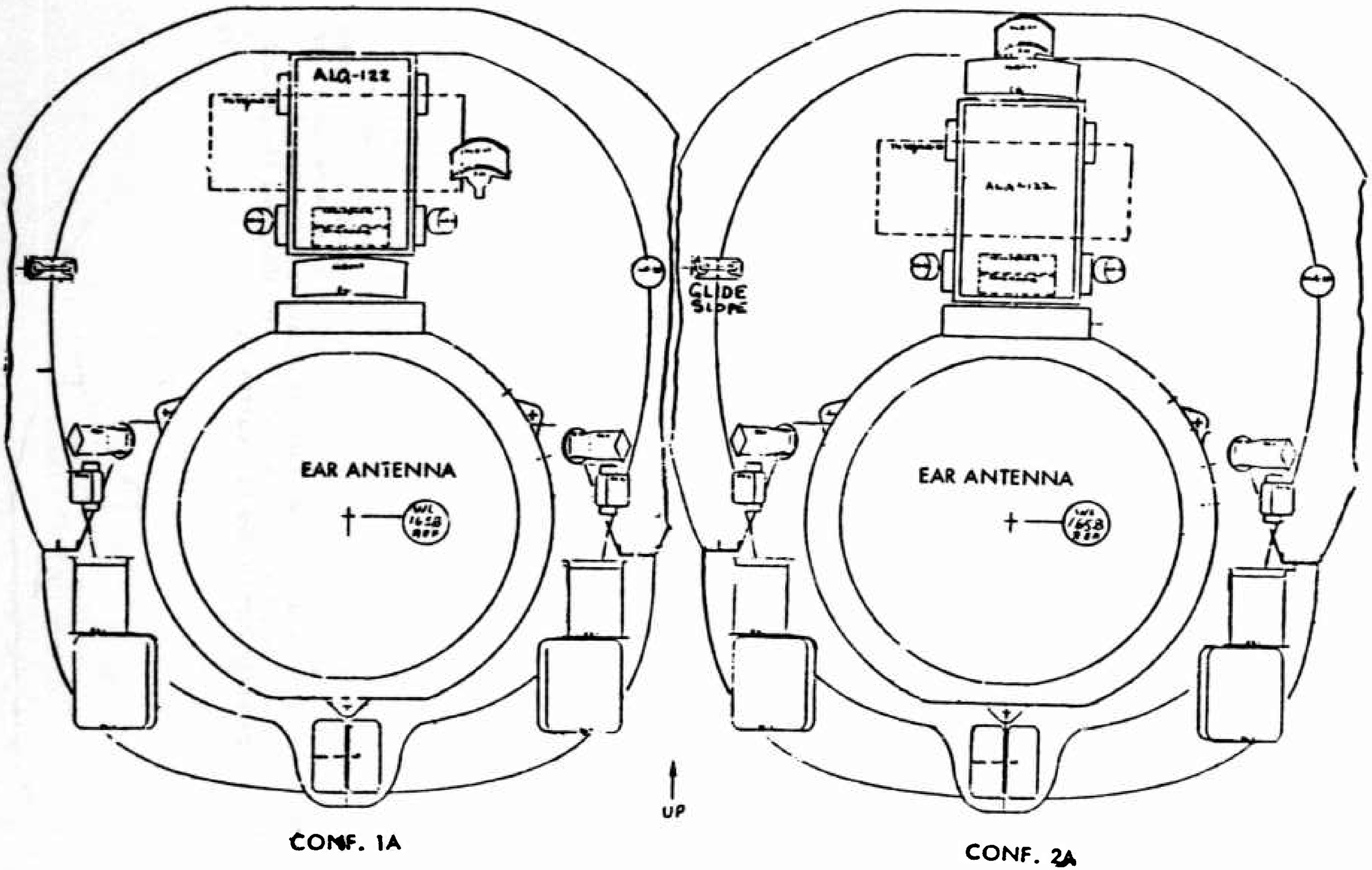


Figure 1. Antenna Arrangements - Front View

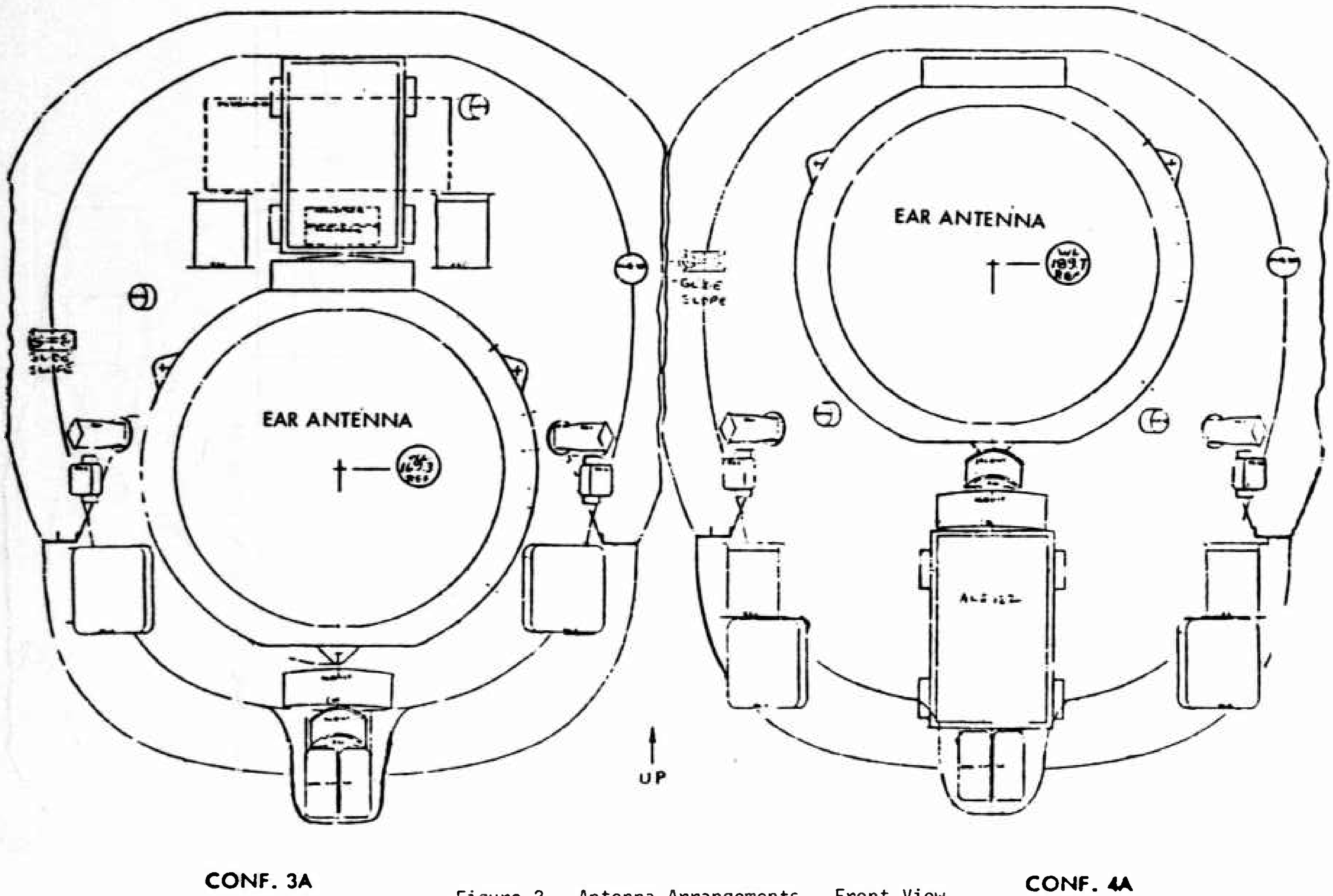


Figure 2. Antenna Arrangements - Front View

TABLE I

EAR RADOME BORESIGHT ERROR ANALYSIS

ELEV. ANGLE	POLARIZATION	ERROR DIRECTION	CONFIGURATION								
			1A & 2A			3A			4A		
			AZ. ANGLE			AZ. ANGLE			AZ. ANGLE		
			0°	20°	45°	0°	20°	45°	0°	20°	45°
+45°	VERT.	VERT.	+0.5	-2.0	—	+0.7	+0.8	—	+0.1	-2.4	—
		HORIZ.	0.0	+2.3	—	0.0	+1.5	—			—
	HORIZ.	VERT.	—	—	—	—	—	—	—	—	—
		HORIZ.	—	—	—	—	—	—	—	—	—
+20°	VERT.	VERT.	-2.3	-1.2	-0.9	-1.1	-1.3	-0.3	+1.7	+0.9	+1.5
		HORIZ.	0.0	+2.7	+0.9	-0.1	+0.4	-0.3	0.0	+3.2	-1.9
	HORIZ.	VERT.	—	—	—	—	—	—	—	—	—
		HORIZ.	—	—	—	—	—	—	—	—	—
0°	VERT.	VERT.	-1.6	+0.3	+0.2	-1.5	+0.3	-0.2	+1.8	-0.3	+1.0
		HORIZ.	0.0	+1.4	+2.7	0.0	+1.3	+0.7	0.0	+0.8	+0.1
	HORIZ.	VERT.	-4.1	-2.9	—	-4.0	-2.2	—	+4.6	+1.8	—
		HORIZ.	0.0	-1.5	—	0.0	-0.8	—	0.0	-0.2	—
-20°	VERT.	VERT.	-0.4	-0.3	0.0	-0.4	+0.4	+0.6	-0.2	+1.5	+1.4
		HORIZ.	0.0	+0.4	+1.0	0.0	+2.1	+0.3	0.0	+2.6	+1.1
	HORIZ.	VERT.	-1.3	-1.4	—	-1.2	-0.85	—	-0.6	+0.9	—
		HORIZ.	0.0	-1.4	—	0.0	-0.6	—	0.0	+1.0	—
-45°	VERT.	VERT.	-0.2	-3.6	—	-0.2	-1.6	—	-0.1	+1.1	—
		HORIZ.	-0.1	+0.7	—	0.0	+0.6	—	0.0	+1.9	—
	HORIZ.	VERT.	—	—	—	—	—	—	—	—	—
		HORIZ.	—	—	—	—	—	—	—	—	—

NOTE: BORESIGHT ERROR GIVEN IN MILLIRADIANS.

TABLE II

ISOLATION PATHS

XMIT	REC	ANTENNA CONFIGURATION			
		1A	2A	3A	4A
EAR ANT & BIT HORN	ALQ-117 (PDR)	B	B	B	B
" " " "	" (PRR)	B	B	B	B
" " " "	" (PRT)	B	B	B	B
" " " "	ALR-20 (10-30257)	B	B	B	B
" " " "	ALR-46	B	B	B	B
ALQ-117 PDT	EAR ANT	B	B	B	B
" " "	ALQ-117 (PDR)	M	D		
" " "	" " (PRR)		D		
" " "	ALR-20 (10-30257)				
" " "	ALR-46				
ALQ-117 PRT	EAR ANT	B	B	B	B
" " "	ALQ-117 (PDR)				
" " "	" " (PRR)			M	M
" " "	ALR-20 (10-30257)				M
" " "	ALR-46				M
ALQ-122	ALR-20 (RL-1)				D
" "	" " (RS-1)			D	D
ALT-28	ALQ-117 (PDR)			M	M
" "	" " (PRR)			M	M
" "	ALR-20 (10-30257)	M	M	M	M
" "	" " (RC-1)	M	M	M	D
" "	" " (RS-2)				M
" "	ALR-46				D

\- NO DEGRADATION   M - MARGINAL   D - DEGRADED   B - BLANKING REQUIRED

TABLE III

PATTERN/GAIN

ANTENNA	ANTENNA CONFIGURATION							
	1A		2A		3A		4A	
	PAT	LL	PAT	LL	PAT	LL	PAT	LL
EAR								
ALQ-117 - PDT								
ALQ-117 - PRT (6 EA.)								
" " - PRR	M	M	M	M	D	M	D	
" " - PDR		M		M	D	M		
ALR-20 - RL1 (2 EA.)								
" - RS1 (2 EA.)					M			
" - RS2 (2 EA.)								
" - RC1 (2 EA.)								
" - 10-30251								
ALR-46 - (2 EA.)	M		M		M	M	M	M
ALQ-122							D	
ALT-28 - (2 EA.)								
GLIDE SLOPE								

NOTE: LINE LENGTH ANALYSIS WAS PERFORMED USING LRU CONFIGURATION D.

\- NO SIGNIFICANT DEGRADATION
M - MARGINAL
D - DEGRADED

2. EQUIPMENT INSTALLATION

Drawing SK W-AI-452, sheets 1 thru 7, shows the planned location of the EAR Test Article equipment. See Appendix. The mock ACUC and its peripherals are installed in an instrumentation rack as shown in section IV.3.

A new equipment rack will be provided in the radome cavity between the Station 96 and Station 150 bulkheads below the cabin floor. This rack will provide mounting and air cooling provisions for the Microwave Assembly, Antenna Driver Power Supplies 1 and 2, Transmitter Low Voltage Power Supply and both main and redundant High Voltage Assemblies (HVA) and Protection & Control/FPA assemblies. The radar pressurization pump/system will be located on this rack also.

Equipment to be located in the area immediately aft of the Station 175 bulkhead in the lower crew compartment are:

1) Low Power RF
2) Low Voltage Bulk Power Supply
3) GEANS Interface Electronics Unit (IEU)
4) GEANS IEU Shock Mount

The forward BNS Console will be modified as required to install the following equipment.

1) Processor Display & Control
2) Hand Control and Electronics
3) GEANS Control Display Unit (CDU)
4) GEANS Power Distribution Panel (or equivalent)

The existing BNS Power Supply rack will be removed and a new rack fabricated and installed. The doppler amplifier will be relocated from the lower forward end of the crew compartment and installed in the aft end of the new rack. The GEANS Avionics Interface Unit and Digital Computer (DCU) will be installed in the rack above the doppler amplifier and the DCU will be shrouded and cooled with forced air. A pivoted assembly will be incorporated into the forward end of the rack area and will provide equipment mounting and cooling provisions for the Digital Assembly, SC/PRG, IAU and four power supplies. The assembly will pivot to permit maintenance/removal of the IAU and four power supplies. The Transient Battery Box will be mounted on the fix rack structure just aft of the Radar Operator's seat. Estimated interconnecting line lengths for this configuration are shown in Table IV.

TABLE IV

B-52/EAR TEST ARTICLE WIRE BUNDLES

FROM	CABLE REF.	TO	APPROX. LENGTH IN FEET
B-52 AC Power		EAR #1 Driver Power Supply J1 HO 1 (Ref. 21)	24
B-52 AC Power		EAR #2 Driver Power Supply J1 HO 1 (Ref. 22)	24
B-52 AC Power		EAR LV Bulk Power Supply J1 HO 2 (Ref. 11)	17
B-52 AC Power		#1 5V Inverter HO 3 (Ref. TBD)	20
B-52 AC Power		#2 5V Inverter HO 3 (Ref. TBD)	20
B-52 AC Power		#3 5V Inverter HO 3 (Ref. TBD)	20
B-52 AC Power		#4 5V Inverter HO 3 (Ref. TBD)	20
B-52 3Ø AC Power		EAR Low Voltage Power Supply J1 HO 4 (Ref. 32)	24
B-52 3Ø AC Power		EAR Main Transmitter HVPS J1 (Ref. 33)	20
B-52 3Ø AC Power		EAR Redundant Transmitter HVPS J1 (Ref. 34)	25
B-52 AC Power		EAR Main Transmitter J3 (Ref. 37)	22
B-52 AC Power		EAR Redundant Transmitter J3 (Ref. 38)	26
B-52 Squat Switch		EAR Main Transmitter J2 (Ref. 44)	35
B-52 Squat Switch		EAR Redundant Transmitter J2 J2 (Ref. 45)	40
B-52 AC Power		EAR TF/TA Control (Ref. 52)	15
B-52 AC Power		EAR Microwave J1 (Ref. 61)	20
B-52 AC Power		EAR Multimode Display (Ref. 64)	10
B-52 AC Power		EAR Test Control & Monitor Panel (Ref. 83)	20
B-52 AC Power		EAR Oscillograph (Ref. 85)	18
B-52 AC Power		SPN/GEANS Power Distribution Panel 4J1 (Ref. TBD)	10
EAR LPRF J1	1	EAR Digital Assembly Synchronizer	24
EAR LPRF J1	2	EAR Digital Assembly RSP	24

TABLE IV (Contd.)

FROM	CABLE REF.	TO	APPROX. LENGTH IN FEET
EAR LPRF J2	3	EAR Digital Assembly Synchronizer	24
EAR LPRF J2	4	EAR Digital Assembly RSP	24
EAR LPRF J2	5	EAR Digital Assembly RDP	24
EAR Digital Assembly Synchronizer	6	EAR Scan Converter and Pattern Raster Generator	5
EAR Microwave Unit J2	7	EAR Digital Assembly Synchronizer	23
EAR Microwave Unit J2	8	EAR Digital Assembly Synchronizer	23
Low Voltage Bulk P.S. HO 2 J3	9	LPRF J3	3
Low Voltage Bulk P.S. J2	10	Microwave Unit (SA1) J1	14
Digital Assembly, J TBD		5V Inverter #1 HO 3	4
Digital Assembly, J TBD		5V Inverter #2 HO 3	4
Digital Assembly, J TBD		5V Inverter #3 HO 3	4
Digital Assembly, J TBD		5V Inverter #4 HO 3	4
Digital Assembly, BSC J1	12	Antenna, J TBD	23
Digital Assembly, BSC J2	13	Antenna, J TBD	23
Digital Assembly, BSC J3	14	Antenna, J TBD	23
Digital Assembly, BSC J4	15	Antenna, J TBD	23
LPRF J2	16	Microwave Unit J2	15
Digital Assembly BSC J6	17	#2 Driver PS J2	24
Digital Assembly BSC J6	18	#1 Driver PS J2	26
#1 Driver PS J4	19	#2 Driver PS J3	6
#1 Driver PS J3	20	Antenna J TBD	8
Digital Assembly RDP J TBD	23	#1 Driver Power Supply J2	26
Digital Assembly RDP J TBD	24	#2 Driver Power Supply J2	24
Digital Assembly RDP J TBD	25	Microwave Unit J2	25
Digital Assembly RDP J TBD	26	Main Transmitter J1	21
Digital Assembly RDP J TBD	27	Redundant Transmitter J1	24
Digital Assembly RDP J TBD	28	SC/PRG J TBD	6
Digital Assembly RDP J TBD	29	Radar Multimode Display J TBD	19
Transmitter LVPS J2	30	Main Transmitter HVPS J1	10
Transmitter LVPS J3	31	Redundant Transmitter HVPS J1	10
Main Transmitter HVPS J1	35	Digital Assembly Synchronizer J TBD	21
Redundant Transmitter HVPS J1	36	Digital Assembly Synchronizer J TBD	24

TABLE IV (contd.)

FROM	CABLE REF.	TO	APPROX. LENGTH IN FEET
Microwave Unit J6	79	Redundant Transmitter J TBD	12
Microwave Unit J7	80	Antenna J TBD	6
Microwave Unit J8	81	Antenna J TBD	6
Digital Assembly RDP J TBD	82	Test Control and Monitor Panel J2	20
Test Control & Monitor Panel J3	84	Oscillograph J TBD	8
Digital Assembly RDP J TBD	86	IAU J TBD	10
IAU J TBD	87	Mock ACUC J TBD	20
Mock ACUC J TBD	88	SPN/GEANS IEU 2J2	21
IMU 1J3 W0003	63	IEU 2 J5	16
AIU 6J2 W0007	68	CDU 3 J1	16
IMU 1J2 W0023	61	IEU 2J7	16
IMU 1J1 W0024	62	IEU 2J6	16
IEU 2J1 W0025	65	AIU 6J3	22
AIU 6J1 W0026	67	DCU J5	23
AIU 6J5 W0027	66	DCU J1	24
IEU AUX I/O 2J2 W TBD	88	Mock ACUC J TBD	18
AIU 6J4 W TBD	69	PCP SR1	22
IEU 2J4 W TBD	70	PCP TBD	10
Battery J1 W TBD	71	PCP TBD	22
Battery J2 W TBD	72	PCP TBD	22
IMU Interface EAR 1 JG W TBD		EADR DIGITAL ASSY	20

TABLE IV (Contd.)

FROM	CABLE REF.	TO	APPROX. LENGTH IN FEET
Main Transmitter HVPS J1	39	Transmitter LVPS J3	10
Redundant Transmitter HVPS J1	40	Transmitter LVPS J2	10
Main Transmitter J2	41	Microwave Unit J2	6
Redundant Transmitter J2	42	Microwave Unit J2	12
Redundant Transmitter	43	Main Transmitter J2	10
LPRF Unit J4	46	Microwave Unit J TBD	16
LPRF Unit J5	47	Microwave Unit J TBD	16
LPRF Unit J6	48	Microwave Unit J TBD	16
Digital Assembly RSP J TBD	49	SC/PRG J TBD	6
SC/PRG J TBD	50	Radar Multimode Display J TBD	23
SC/PRG J TBD	51	Radar Multimode Display J TBD	23
Radar Multimode Display J TBD	53	TF/TA Control J TBD	12
Hand Control J TBD	55	Radar Multimode Display J TBD	6
Display J TBD	58	Display Repeater J TBD	TBD
Radar Multimode Display J TBD	59	TF/TA Recorder J TBD	TBD
Radar Multimode Display J TBD	60	Video Distribution J TBD	19
Microwave Unit J2	62	Antenna J3	6
Microwave Unit J2	63	Antenna J2	6
Microwave Unit WG TBD	65	Antenna WG TBD	6
Microwave Unit WG TBD	66	Antenna WG TBD	6
Microwave Unit WG TBD	67	Antenna WG TBD	6
Microwave Unit WG TBD	68	Redundant Transmitter WG TBD	10
Microwave Unit WG TBD	69	Main Transmitter WG TBD	6
LPRF Unit J11	70	Test Point Panel J TBD	18
LPRF Unit J12	71	Test Point Panel J TBD	18
LPRF Unit J13	72	Test Point Panel J TBD	18
LPRF Unit J14	73	Test Point Panel J TBD	18
LPRF Unit J15	74	Test Point Panel J TBD	18
Digital Assembly RSP J TBD	75	SC/PRG J TBD	6
LPRF Unit J7	76	Main Transmitter J TBD (OSM)	10
LPRF Unit J8 (TNC)	77	Redundant Transmitter J TBD (OSM)	10
Microwave Unit J5	78	Main Transmitter J TBD	4

3. INSTRUMENTATION

The instrumentation required for the Flight Test program is being determined through a coordinated effort. Westinghouse has the responsibility to identify required instrumentation for the Test Article. Boeing's responsibility is to identify B-52 Test Bed instrumentation.

A list of the equipment to be used is shown in Table V. The AR-200 analog tape recorder, signal conditioning, instrumentation, control panel, and SRAM CAE digital instrumentation are provided for test bed instrumentation. Table VI lists the airplane test bed and environmental signals to be instrumented.

The current instrumentation is shown in Figures 3, 4, 5, and 6. The basic criteria used to develop this configuration were spatial dimensions, and in-flight accessibility to particular units. Figure 7 shows a general overview of the airplane with sectional instrumentation locations broken out. Most of the equipment will be installed in the 41 section except for a camera mounted on the vertical fin, inverter installation on the alternator deck, TR unit mounted in the forward wheel well, and CIRIS equipment which will be a bomb bay platform installation and in the Navigator's Console.

The B-52 airplane does not provide a source of 115 VAC 60 Hz power. An inverter with an output of at least 3000 watts will be required to accommodate the requirements of the test article. Unitron Model PS62-600 can meet this requirement.

Instrumentation in the 41 section will be in four areas and will require three installation racks. Figure 3 shows the installation in the upper 41 section right hand side directly forward of the EWO seat. Equipment mounted in these racks will be:

- Video Mixer
- Floating Point Processor
- Video Recorder
- Mock ACUC Computer
- CA Power Supply

TABLE V

EAR INSTRUMENTATION EQUIPMENT LIST

	* DIMENSIONS (inches)
RDP CONSOLE	
Intell. Unit	20 W x 13.5 H x 26 D
Keyboard	18.5 W x 3.5 H x 6.5 D
CRT Display	18.5 W x 12.5 H x 16 D
Floppy Disc	17 W x 9.7 H x 19.25 D
MOCK ACUC	
Computer	19 W x 8.7 H x 17.4 D
PWR Supply	19 W x 5.25 H x 18 D
Punch Tape Reader	19 W x 5.3 H x 11 D
Floating Point Processor	19 W x 7 H x 17.4 D
OSCILLOSCOPE	19 W x 5.3 H x 22.8 D
SPECTRUM ANAL.	19 W x 16 H x 18.4 D
FREQ. COUNTER	16.8 W x 3.5 H x 13.2 D
DIGITAL MULTIMETER	8.7 W x 6 H x 15.3 D
TEST POINT PANEL	19 W x 3 H x 6 D
TARGET SIMULATOR	19 W x 24H x 24 D
TEST MONITOR AND CONTROL PANEL	16.5 W x 12 H x 13.1 D
INST. CONTROL PANEL	5.75 W x 14 H x 8 D
DIGITAL BUFFER AND ANALOG SIG. COND.	19 W x 24 H x 24 D
RADAR STATUS PANEL	5.75 W x 6 H x 6 D
PATCH PANEL	19 W x 5.2 H x 12 D
TCG IRIG. "A"	19 W x 3.5 H x 22 D
ANALOG TAPE RECORDER	19 W x 21 H x 12.5 D
X-Y MONITOR	16.8 W x 12.2 H x 24 D
VIDEO RECORDER	19 W x 15.6 H x 8.6 D
VIDEO MIXER	10 W x 3.5 H x 10.8 D
TCG IRIG. "B"	5 W x 8 H x 15 D
⊛ AR-200 ANALOG TAPE RECORDER	10 W x 5H x 24 D
VOICE TAPE RECORDER	5 W x 8 H x 15 D
⊛ SIGNAL CONDITIONING	19 W x 5 H x 22 D
35MM CAMERA	3 W x 12 H x 12 D (EST.)
T.V. CAMERA	3 W x 5 H x 10 D (EST.)
T.R. UNITS (2 EACH)	6 W x 6 H x 15 D
INVERTOR	9 W x 10 H x 22 D
⊛ INSTR. CONTROL PANEL	5.75 W x 12 H x 5 D (EST.)
CIRIS	? x ? x ?
⊛ SRAM CAE DIGITAL INST.	5.75 W x 2.25 H x 4 D

⊛ Provide for Test Bed Instrumentation Signals

* Dimensions are only estimates of equipment Westinghouse would like. Some equipment not designed or built yet.

TABLE VI

PRELIMINARY MEASUREMENT LIST
FOR A/C TEST BED & ENV. SIGNALS

1. Pressure Altitude
2. Airspeed
3. Mach Number
4. Radar Altitude (APN-194)
5. Radar Altitude Validity
6. NACG
7. Pitch Angle
8. Roll Angle
9. Angle of Attack
10. Gross Weight and C.G. (Manual)
11. Heading
12. SRAM Channel 7 Data
13. Control Column Position
14. Control Wheel Position
15. Rudder Pedal Position
16. Environmental Temperatures (26)
17. Environmental Pressures (16)
18. Environmental Air Flows (13)
19. OAT
20. 400 Hz Airplane System Voltage
21. 28 VDC Airplane System Voltage
22. True Airspeed
23. Forward Viewing Camera
24. Downward Viewing Camera
25. Oscilloscope Camera

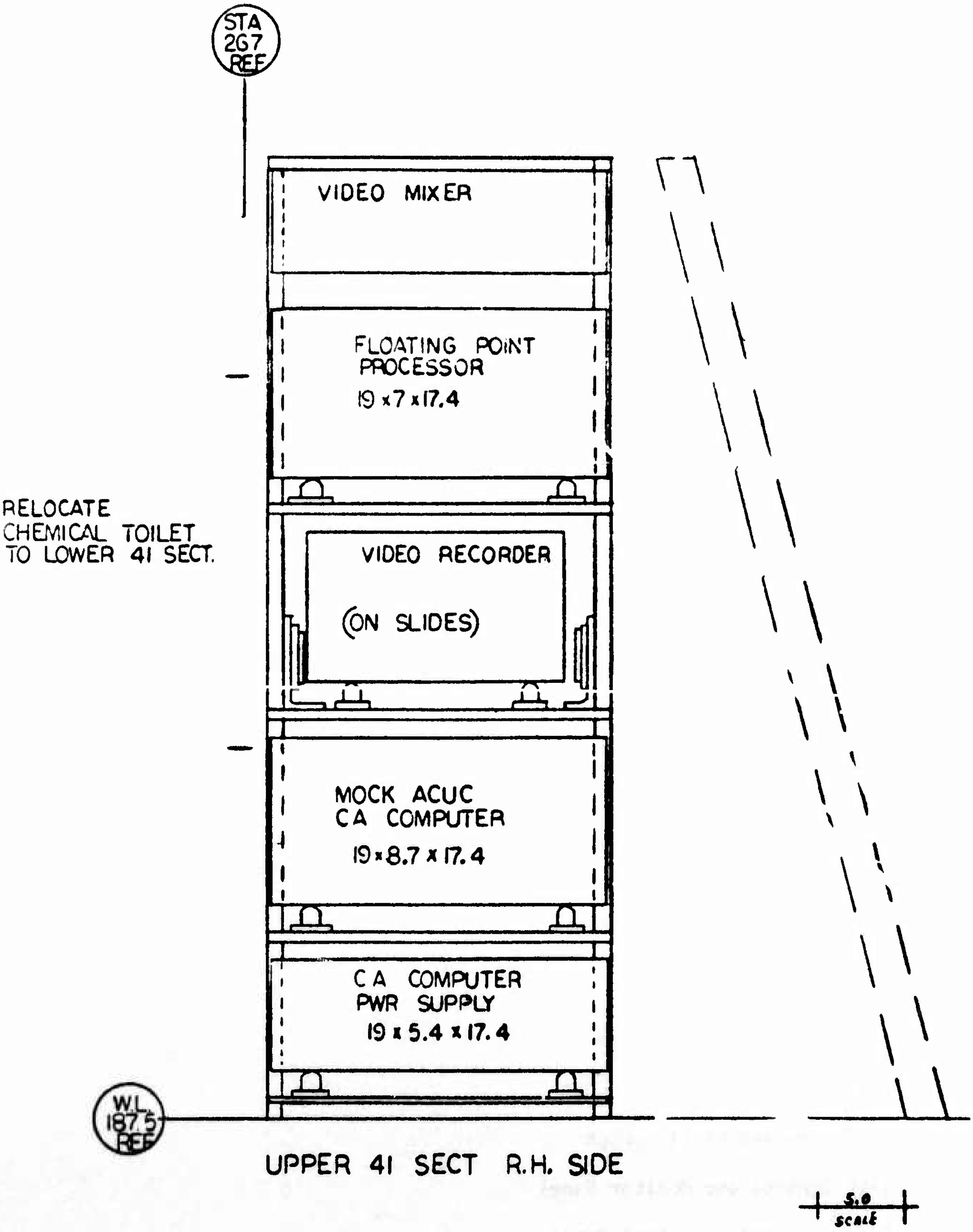


Figure 3. EAR Instrumentation

Figure 4 shows the installation in the upper left-hand side of the 41 section. Available space and mounting provisions (with minor modifications) are sufficient at this location to install equipment without adding a rack. Equipment mounted in this area will be:

Voice Recorder

RDP Console Intell. Unit

Floppy Disc Recorder

IRIG "B" TCG

IRIG "A" TCG

Patch Panel

AR 200 Electronics

An AR 200 Tape Recorder is to be floor mounted in the aisle in front of the other equipment. There are also space provisions in this area for tape stowage.

Figure 5 shows the installation in the lower 41 section. Two new racks will be installed in the area now occupied by the BNS Remote Modules Rack. Equipment mounted in these racks will be:

Digital Buffer & Analog Sig Conditioner

Target Simulator

PC 500 Tape Recorder

2 Test Equipment

Space for Analyzer Plug-In Stowage

Figure 6 shows the configuration of the Test Engineer's Console at the Gunner's Station. New panels and equipment to be installed at this station will be:

RDP Console CRT Display

EVS Monitor

Time Display

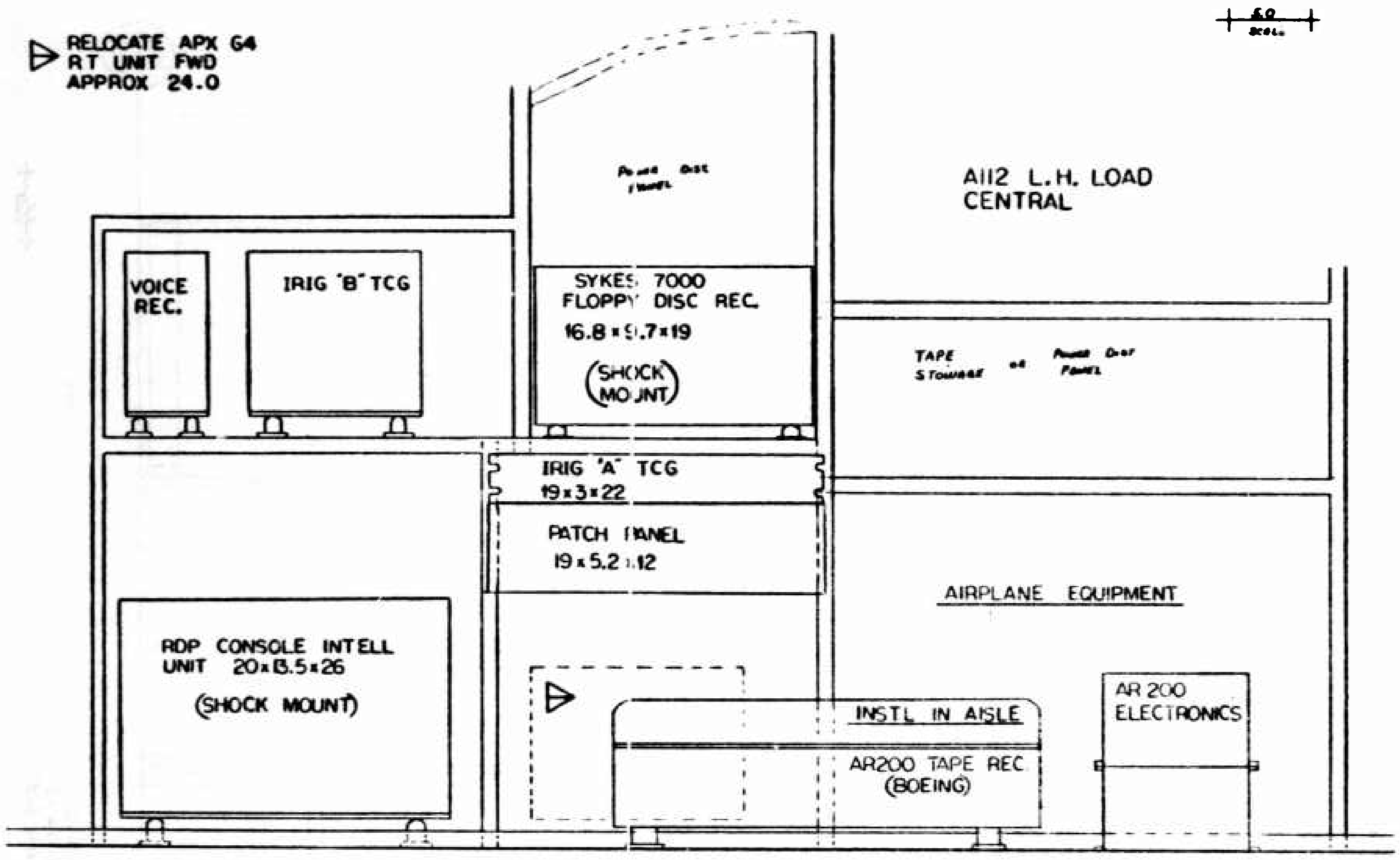
X-Y TV Monitor

Direct Record Oscillograph

Test Control and Monitor Panel

Instrumentation Control Panel

This installation will require a complete redesign and rework of the Gunner's Station Console.



UPPER 41 SECT L.H. SIDE

Figure 4. EAR Instrumentation

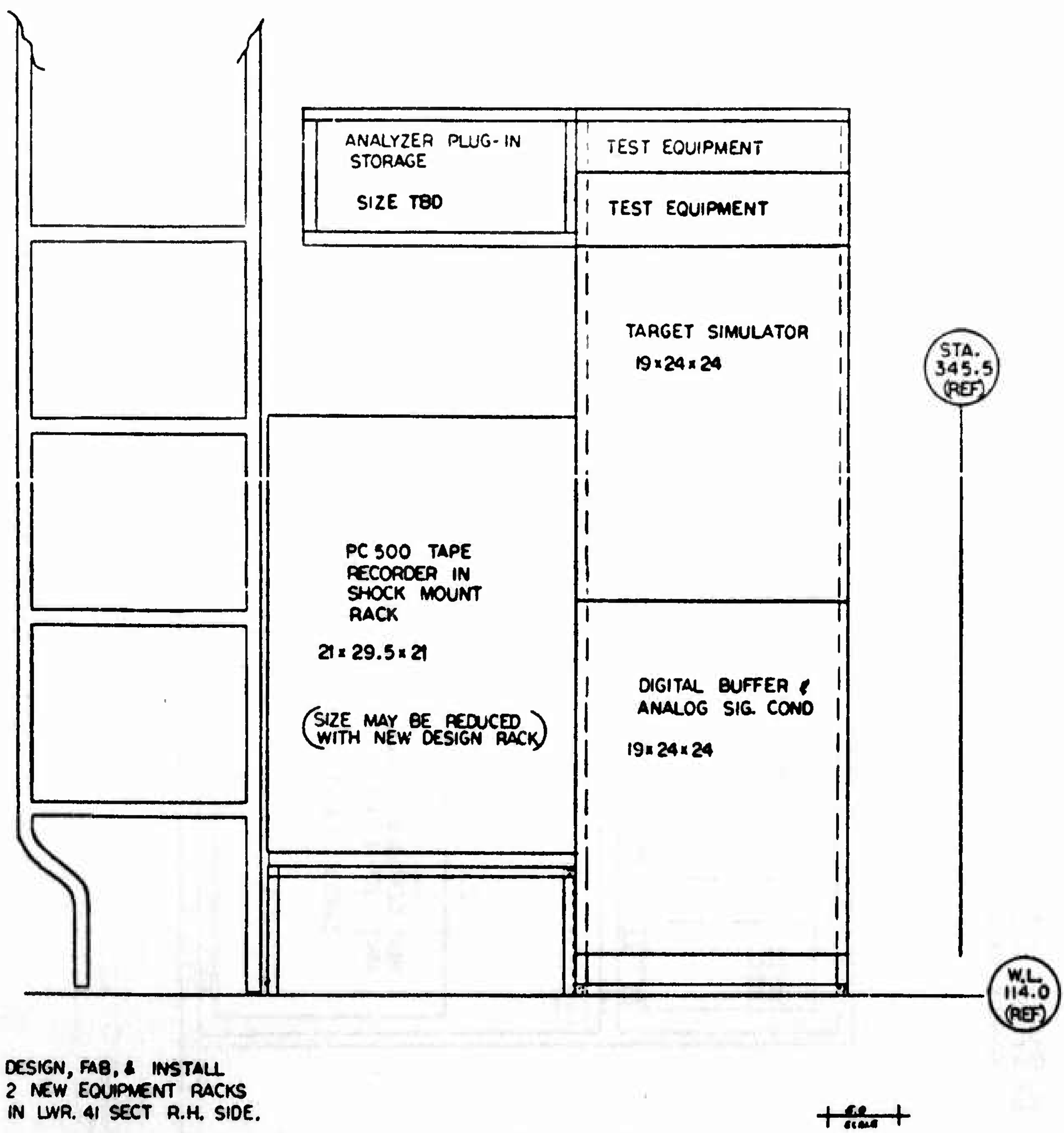


Figure 5. EAR Instrumentation

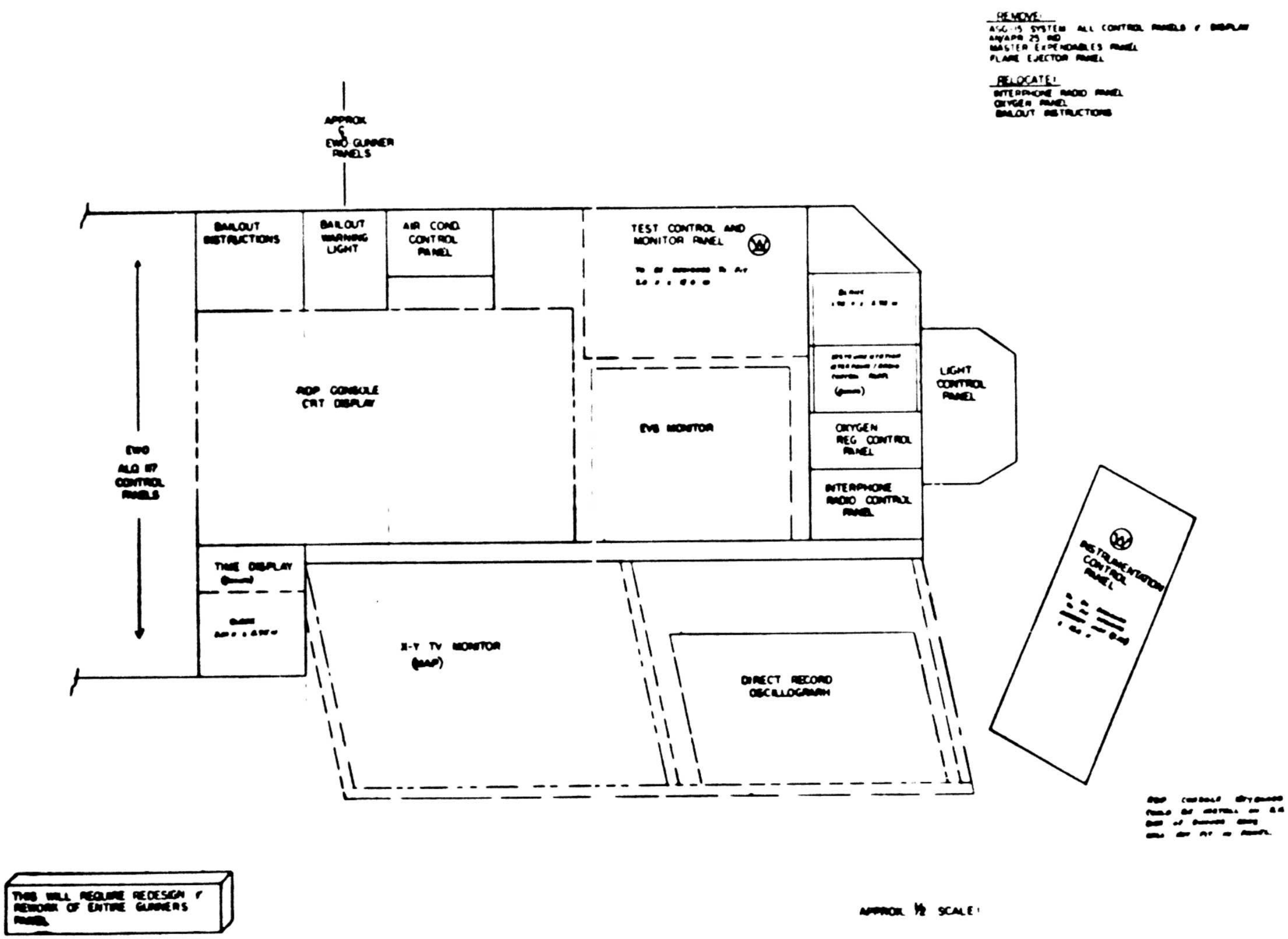


Figure 6. EAR Instrumentation - Air Instrumentation - Test Engineers' Console

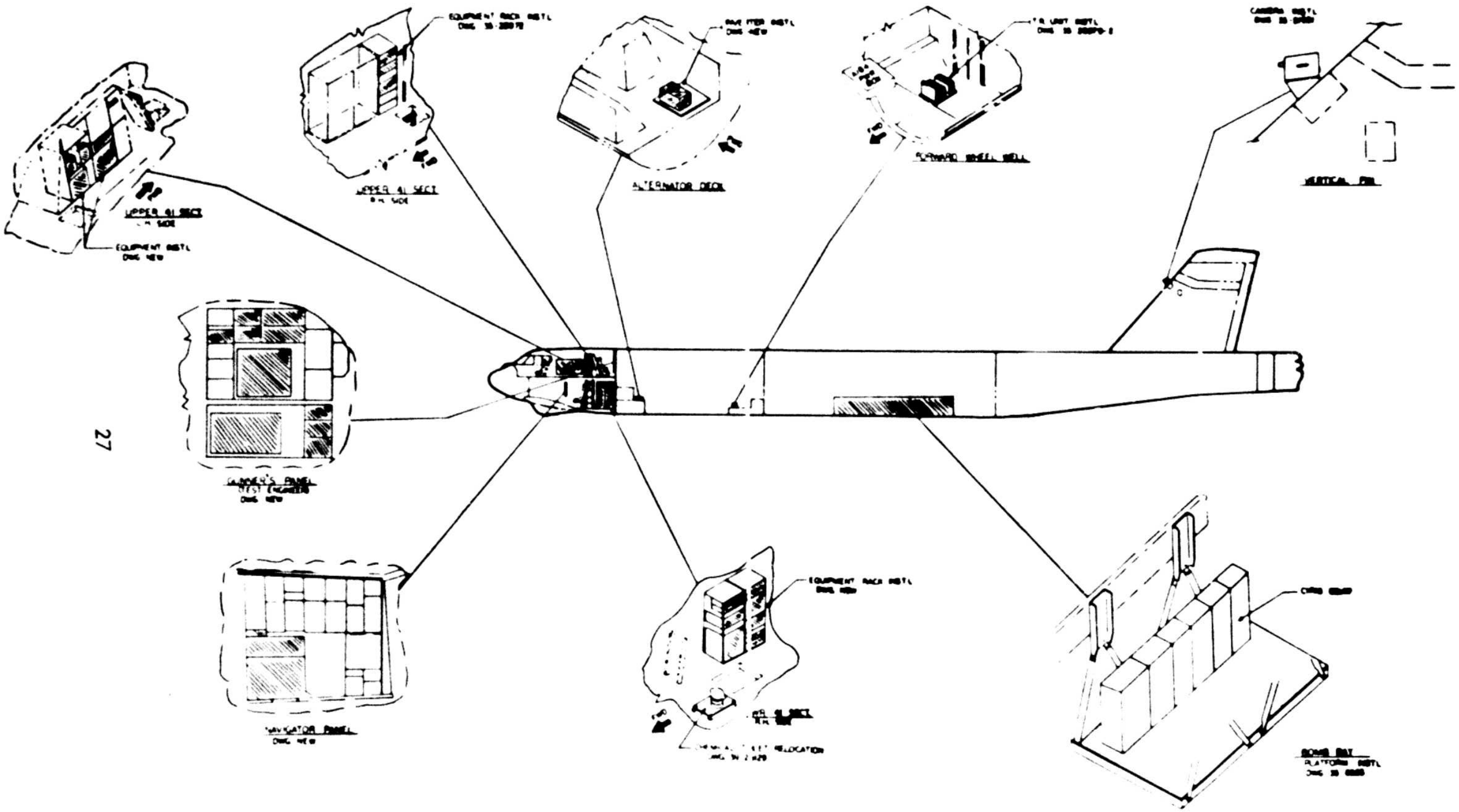


Figure 7. Instrumentation Overview

SECTION V

ENVIRONMENTAL CONTROL PROVISIONS

Nearly all of the equipment and instrumentation are located in the cabin and nose radome compartments. A list of EAR and ancillary equipment is shown in Table VII.

1. LIQUID COOLING REQUIREMENTS

Three items of EAR equipment, the antenna, transmitter FPA/P&C main, and transmitter FPA/P&C redundant, must be liquid cooled. Boeing will be responsible for providing the entire liquid circulation and cooling system. The liquid cooled components will be located in the nose radome but, since liquid coolant temperature at equipment inlets cannot exceed 100°F, conditioned air is required for a heat sink. Therefore, the heat exchanger must either be located in the cabin or conditioned air must be piped to the nose radome.

There are three modes of operation, two of which involve a 500-watt dummy load. The modes and heat dissipation as defined by Westinghouse are as follows:

Mode 1 - Normal Operation

Antenna	790 Watts
Transmitter FPA/P&C, Main	1,420 Watts
Transmitter FPA/P&C, Redundant	300 Watts
Dummy Load	0
TOTAL	2,510 Watts

Mode 2 - Dummy Load or no RF Drive Operation

Antenna	635 Watts
Dummy Load (dissipated in antenna)	500 Watts
Transmitter FPA/P&C, Main	1,420 Watts
Transmitter FPA/P&C, Redundant	300 Watts
TOTAL	2,855 Watts

Mode 3 -

Antenna	635 Watts
Dummy Load (dissipated in transmitter, main)	500 Watts
Transmitter FPA/P&C, Main	1,420 Watts
Transmitter FPA/P&C, Redundant	300 Watts
TOTAL	2,855 Watts

The system is shown schematically on Figure 8 with resulting system temperatures for Mode 2.

TABLE VII

EAR AND SPN/GEANS EQUIPMENT LIST [1]

COMPONENT	NO. REQ'D	VENDOR DWG. NO.	TYPE OF COOLING [2]	HEAT DISSIP. WATTS	APL LOCATION [3]
EAR COMPONENTS					
1 Antenna	1	641R421	LIQ	1135*	NR
2 Driver P.S. No. 1	1	641R422	FA	300	NR
3 Driver P.S. No. 2	1	641R422	FA	300	NR
4 Transmitter, HVPS Main	1	641R475	FA	650	NR
5 Transmitter, HVPS Redundant	1	641R475	FA	650	NR
6 Transmitter, LVPS	1	641R422	FA	300	NR
7 Microwave	1	641R474	FA	800	NR
8 Low Power RF	1	641R477	FA	845	C
9 Low Voltage Bulk P.S.	1	641R422	FA	600	C
10 Transmitter, FPA/P&C Main	1	641R476	LIQ	1420	C
11 Transmitter, FPA/P&C Redundant	1	641R476	LIQ	300	C
12 Digital Assembly	1	641R427	FA	2410	C
13 Inverter, 5V No. 1	1	641R422	FA	350	C
14 Inverter, 5V No. 2	1	641R422	FA	350	C
15 Inverter, 5V No. 3	1	641R422	FA	350	C
16 Inverter, 5V No. 4	1	641R422	FA	350	C
17 Pattern Raster Gen.	1		FA	425	C
18 Processor Assembly	1		FA	700	C
19 TF/TA	1		FA	10	C
SPN/GEANS					
1 DCU	1		AMB	210	C
2 AIU	1		FA	350	C
3 IEU	1		FA	170	C
4 IMU	1		FA	212	NR
5 CDU	1		AMB	50	C
6 Battery	1		AMB	-	

[1] This chart based on Westinghouse chart dated 12 January 1976

[2] LIQ - Liquid Cooled; FA - Forced Air; AMB - Ambient Air Cooled by free convection.

[3] NR - Nose Radome; C - Cabin

* Includes 500W for dummy load.

LIQUID COOLING SYSTEM

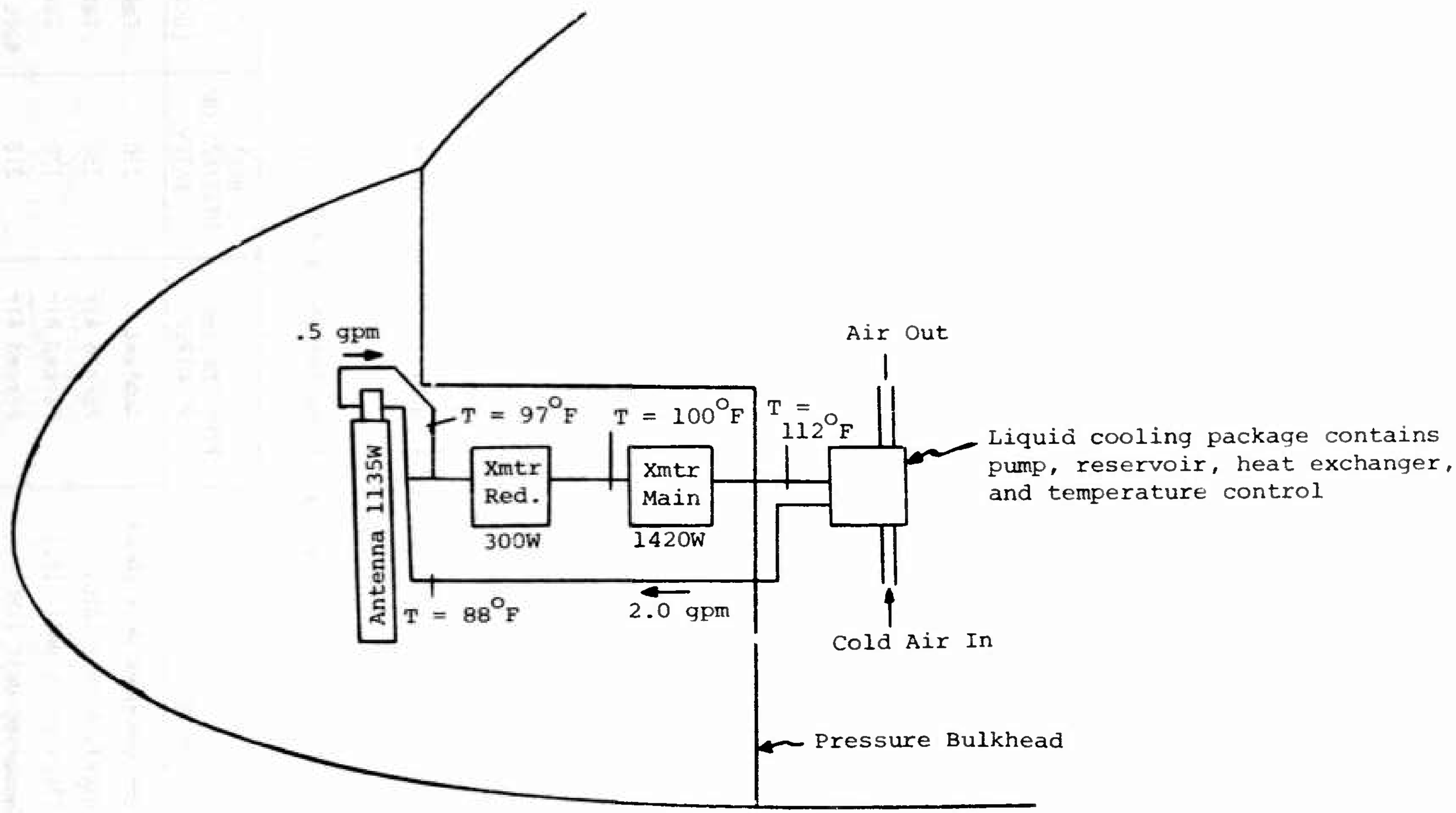


Figure 8. EAR Liquid Cooling System Schematic

Additional requirements for the liquid system are as follows:

	ANTENNA	TRANSMITTER FPA/P&C MAIN	TRANSMITTER FPA/P&C REDUNDANT
Type of Liquid Coolant	Coolanol 25	Coolanol 25	Coolanol 25
Liquid flow required, minimum-gpm	.5	2.0	2.0
Maximum heat dissipation - watts	1135	1420	300
Maximum allowable coolant inlet temp - °F	100	100	100
Liquid pressure drop at minimum required flow - psi	.5	15	15
Volume of liquid in equipment - $in.^3$ (approx.)	50	100	100

Contamination of the liquid coolant must be controlled such that the dielectric strength does not become less than 20 kilowatts per .10 inch gap at 25°C.

Liquid coolant system is not required to operate during airplane taxi or takeoff.

2. SPN/GEANS EQUIPMENT REQUIREMENTS

The SPN/GEANS equipment and cooling information are as follows:

COMPONENT	TYPE COOLING REQUIRED	HEAT DISSIPATION WATTS	LOCATION
ROLM Digital Computer Unit (DCU)	Ambient	210	Cabin
Avionics Interface Unit (AIU)	Forced Air	350	Cabin
Interface Electronics Unit (IEU)	Forced Air	170	Cabin
Inertial Measuring Unit (IMU)	Forced Air	212	Nose Radome
Control Display Unit (DCU)	Ambient	50	Cabin

Required airflow as function of ambient and inlet air temperatures are shown on Figures 9, 10, and 11. Pressure drop characteristics are as follows:

IMU $\sigma\Delta P$ = 1.8 in H_2O @ 3.0 lb/min.

AIU $\sigma\Delta P$ - 1.75 in H_2O @ 2.0 lb/min.

IEU $\sigma\Delta P$ = 1.75 in H_2O @ 2.0 lb/min.

The IMU must be located in the nose radome compartment and also must be cooled with refrigerated air. Therefore, since the required airflow is relatively low, the original concept was to bleed cabin air through the IMU as shown on Figure 12. Since SPN/GEANS must operate during takeoff, taxi, and low level flight where the cabin air is unpressurized, a fan is required. Also, in order to limit the flow when the cabin is pressurized, a shutoff valve is required. A check valve is required to prevent blower recirculation. This appears to be a feasible system, however, now that an additional air conditioning pack will be used, this concept may be changed. (Note: the narrow band of temperatures and flow rates shown on Figures 10 and 11.)

3. NOSE RADOME EAR AIR COOLED EQUIPMENT

EAR air cooled equipment located in the nose radome consists of six units. These units were originally intended to be cooled with ram air as shown on Figure 13. There has been continued interest in providing the cooling margin afforded by air conditioned air if the impact on B-52 modification complexity was not severe. Since an additional air conditioning pack was required for cabin equipment, even without the nose radome equipment, the standing decision now is to procure an air conditioning package capable of cooling cabin and nose radome equipment.

4. CABIN SYSTEM REQUIREMENTS

A list of EAR equipment and cooling information, as received from Westinghouse, is shown on Table VIII. Also, the following verbal information was obtained:

1) Cooling airflow required, in a general sense, is 3.4 lb/min per KW heat dissipation with inlet air temperature of 85°F. However, for the study, airflow required as a function of inlet air temperature may be calculated assuming 160°F outlet air temperature. This applies to all basic EAR air-cooled equipment except the digital assembly. Maximum allowable outlet temperature for this digital assembly should be that determined from the chart using 85°F inlet temperature.

2) Pressure drop of the units is $\sigma\Delta P$ = 2.0 inches of water at the flows stated in the chart.

3) EAR equipment is not required to operate during taxi.

Data on Tables IX, X, and XI summarize total cabin heat load changes as used in the study.

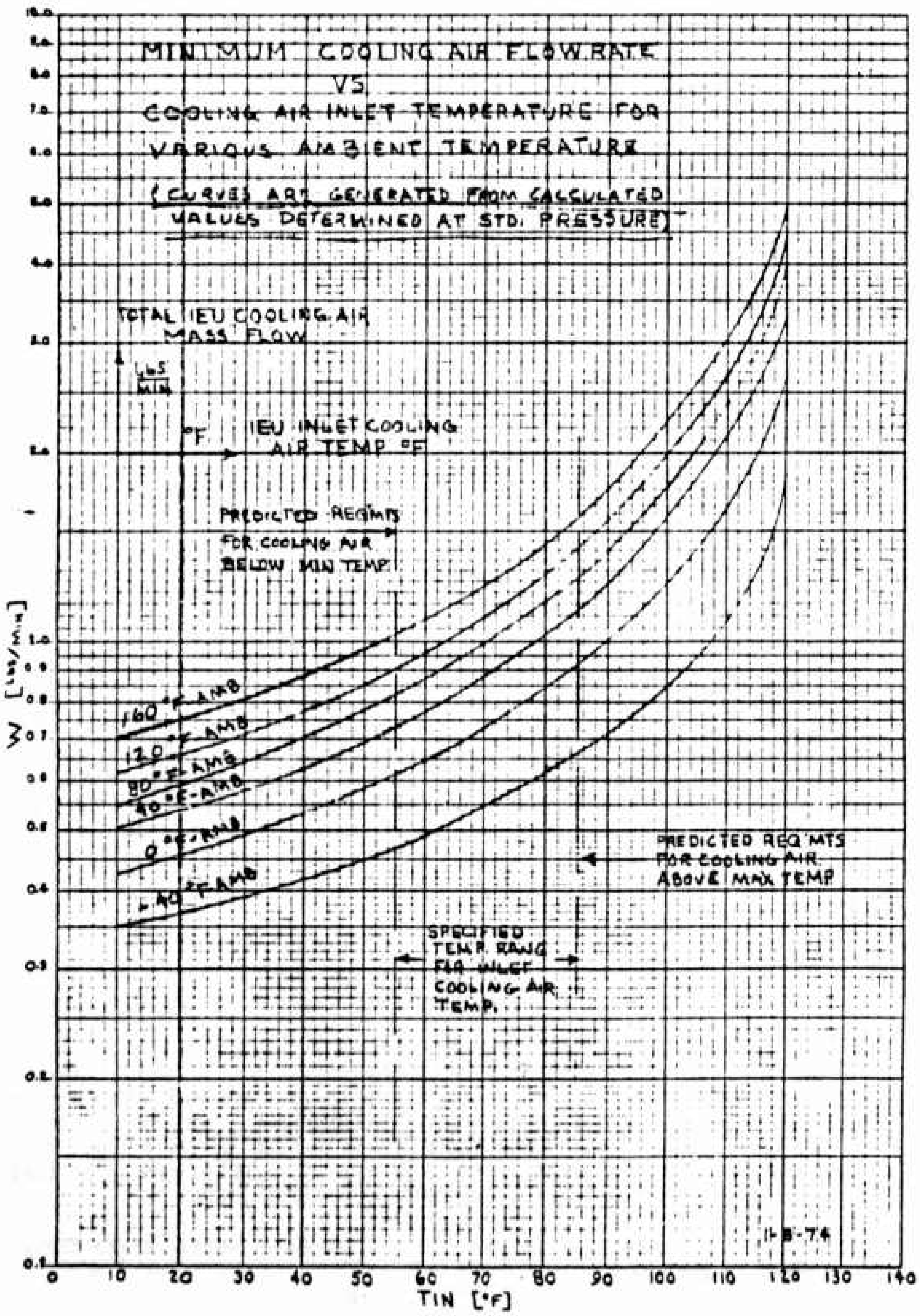


Figure 9. IEU and AIU Flow Rates

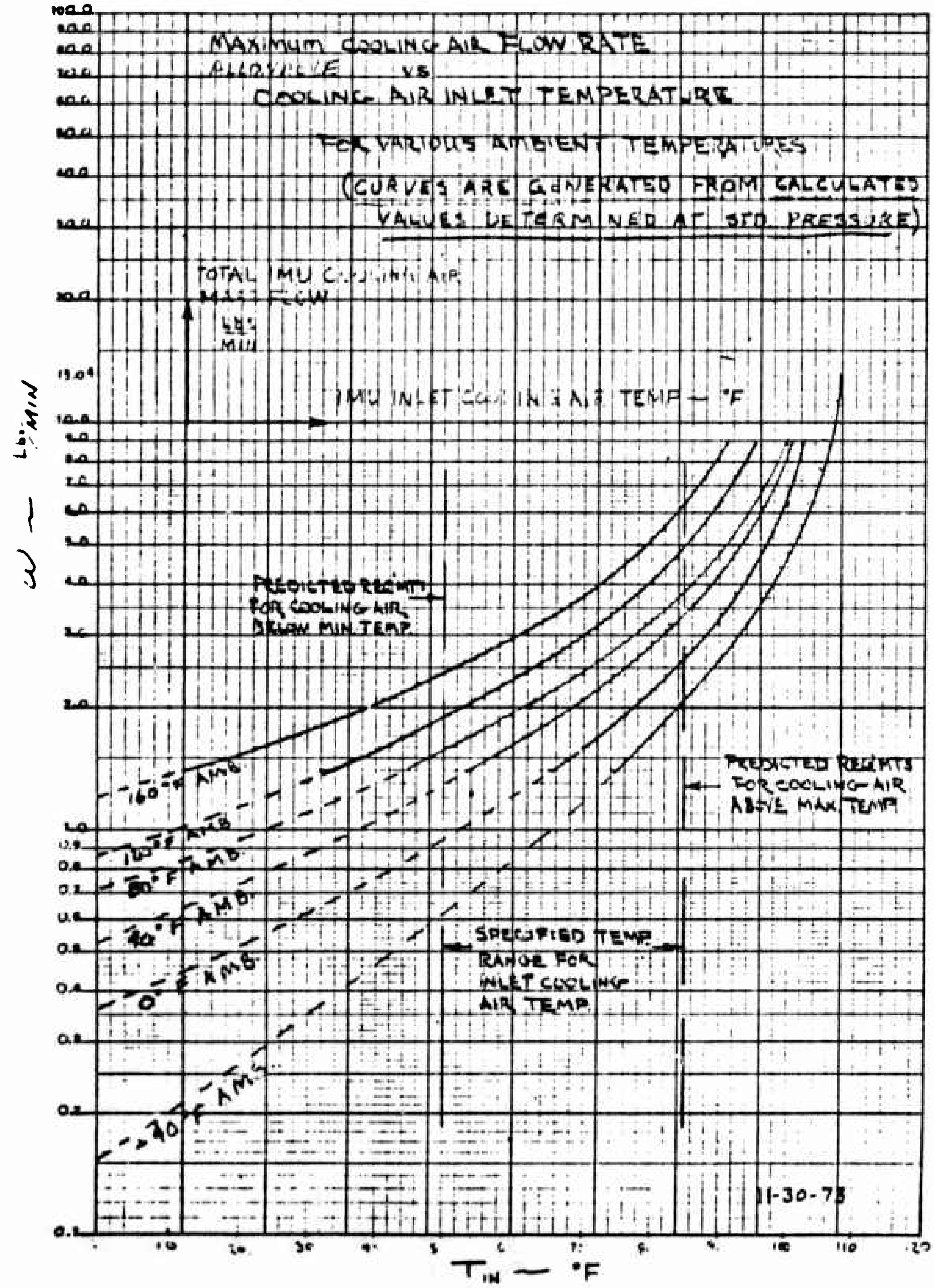


Figure 10. IMU Airflow Rates (Sheet 1 of 2)

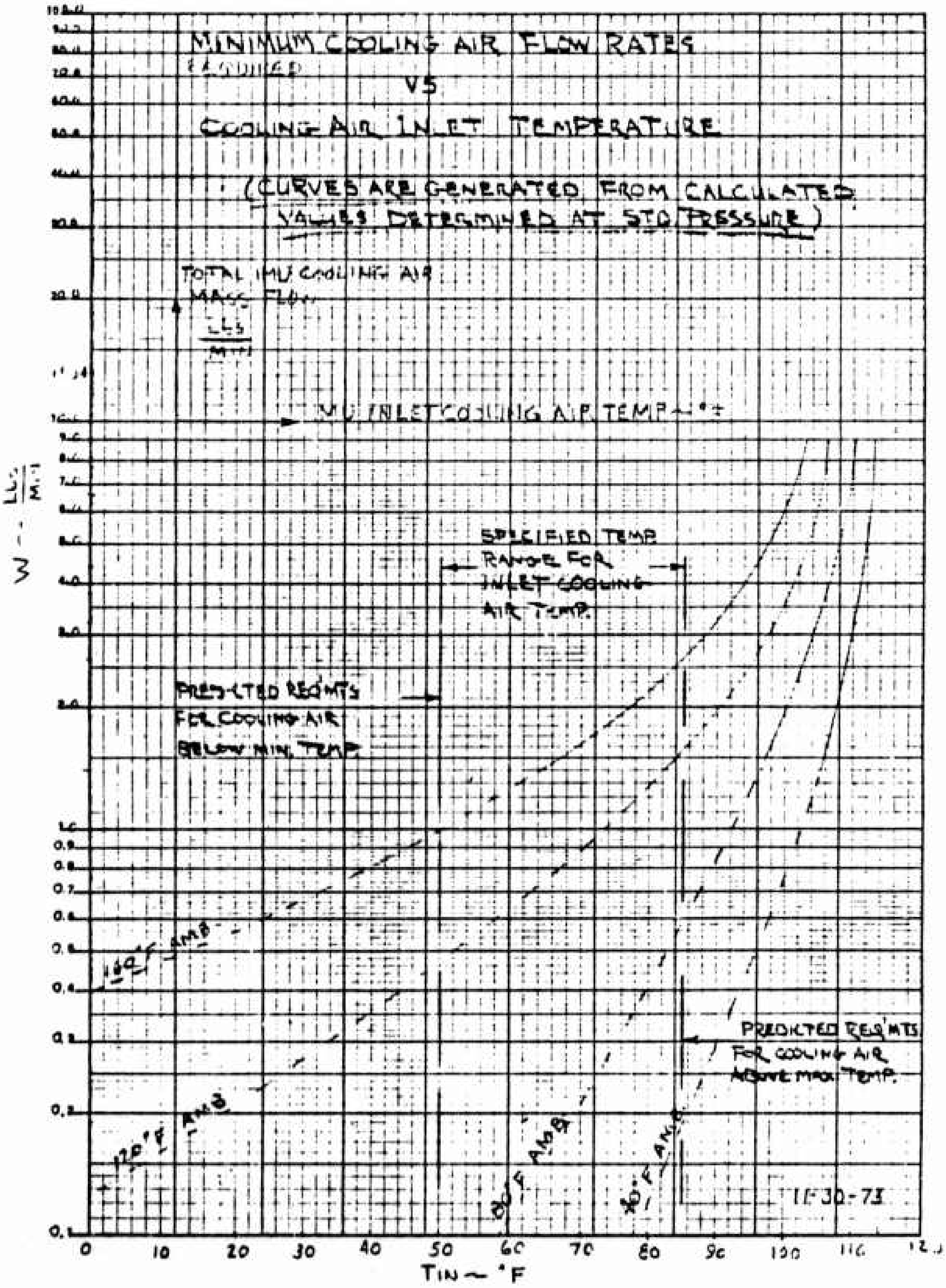
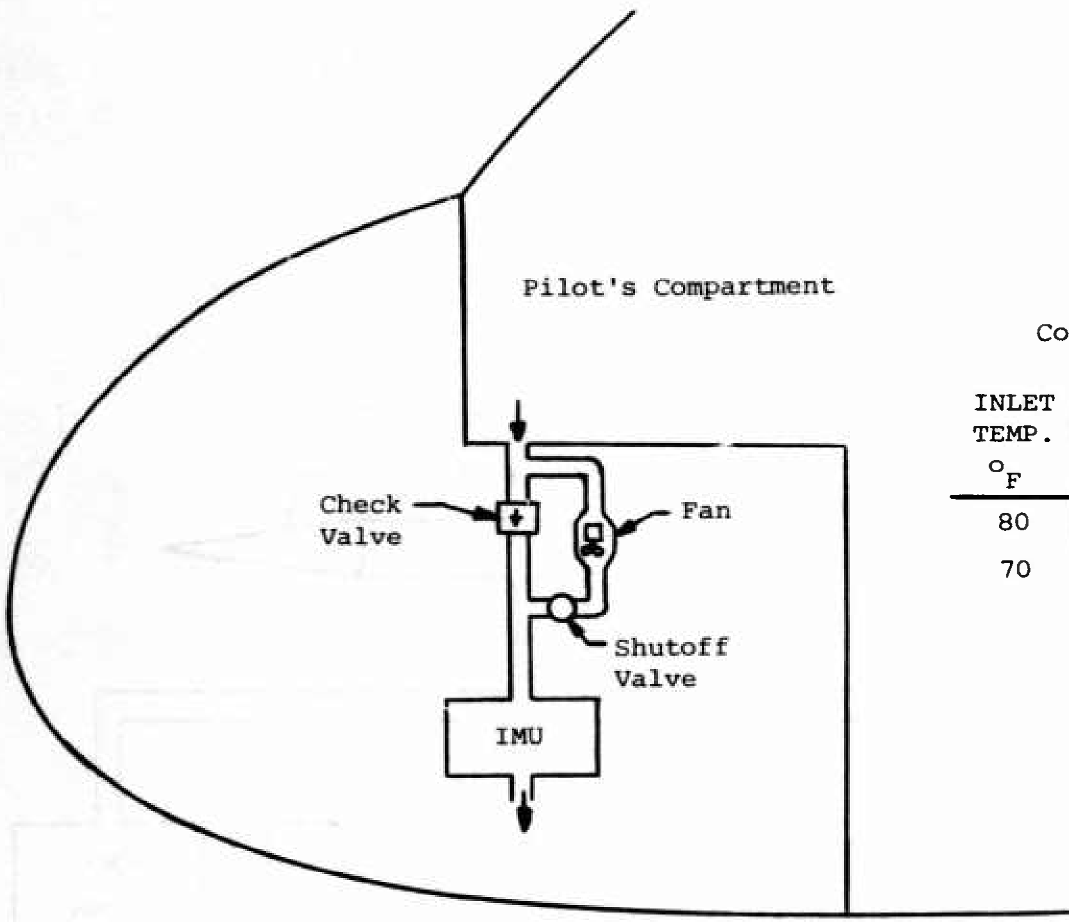


Figure 11. IMU Airflow Rates (Sheet 2 of 2)



Cooling Air Requirements

INLET TEMP. °F	AMBIENT TEMP. °F	MIN. REQ'D FLOW LB/MIN	MAX. ALLOWABLE FLOW LB/MIN
80	120	1.20	4.05
70	40	.03	2.10

Figure 12. SPN/GEANS IMU Cooling System Schematic

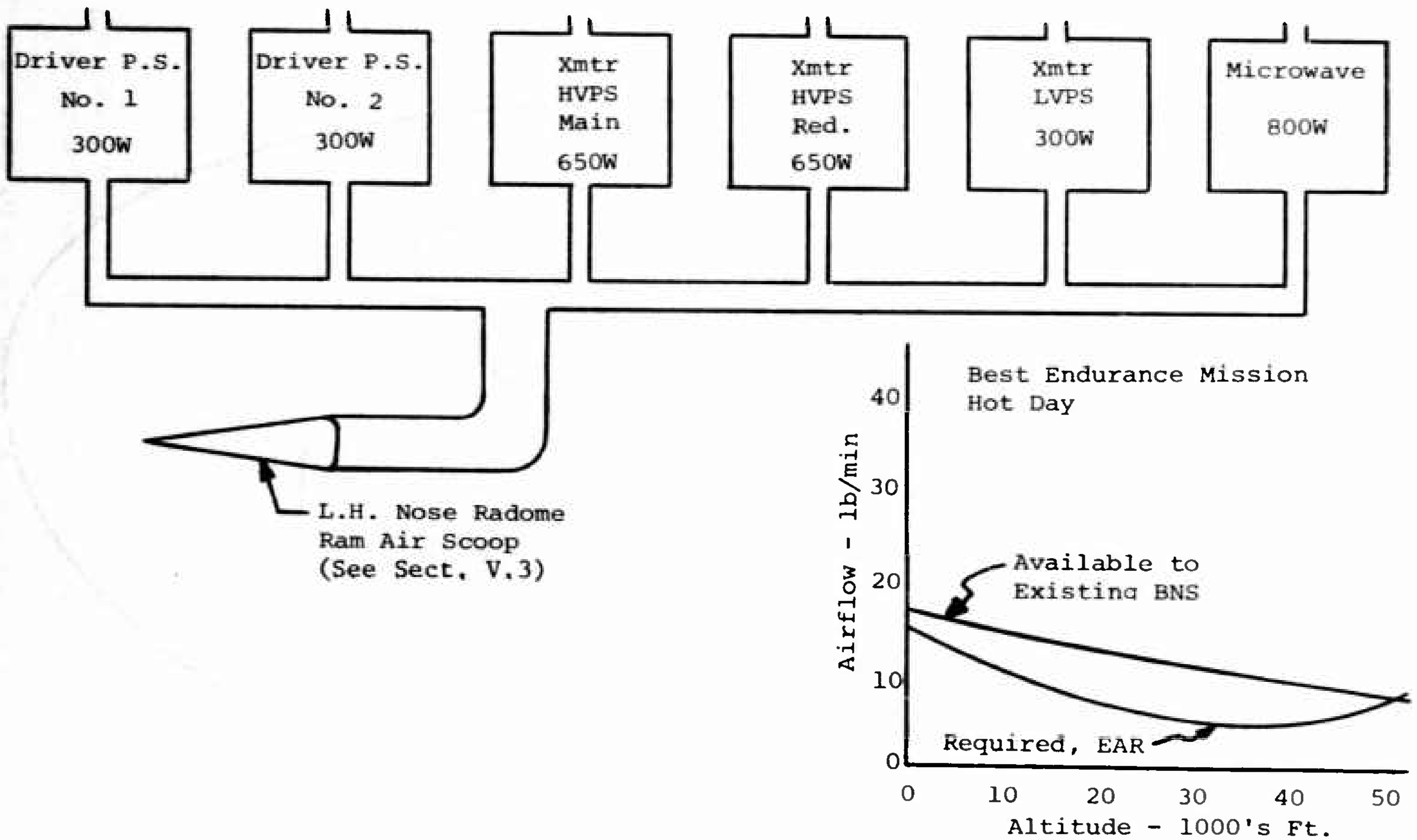


Figure 13. Nose Radome Air Cooled Equipment Schematic

TABLE VIII

EAR CONFIGURATION
REDUNDANT SYSTEM

		WEIGHT LBS.	VOL. IN^3	DISS. WATTS	TYPE OF COOLING	AIRFLOW LB/MIN	COOLANOL gpm	I/O DRAWING
Antenna		317	(4)	790	Liq.(1)		0.5	641R421
Driver P.S. No. 1		25	695	375	Air	1.2		641R422
Driver P.S. No. 2		25	695	375		1.2		
Low Power R.F.		78	4330	1180	Air	4.0		641R477
Low Voltage Bulk P.S.		22	655	440	Air	1.5		641R422
Transmitter FPA/P&C	- Main	64	1940	1420	Liq. (1)		2.0	641R476
	- Red.	64	1940	300			2.0	
Transmitter HVPS	- Main	69	1960	650	Air	2.2		641R475
	- Red.	69	1960			2.2		
Transmitter LVPS		19	645	320	Air	1.0		641R422
Microwave		55	2995	685	Air	2.3		641R474
Digital Assembly		80	2500	2070	Air	9.8		641R427
Digital Assy Inverter	- No. 1	24	645	630	Air	2.1		641R422
	- No. 2	24	645	630	Air	2.1		
	- No. 3	24	645	630		2.1		
	- No. 4	24	645	630		2.1		
SC/PRG		25	875	550	Air	2.6		
TOTALS		1008	23770 ($13.75\ ft^3$)	11675		36.4(3)	2.0(3)	
Without SC/PRG and Inverter No. 4		959	22250	10495		31.7(3)		

(1) Increase by 0.5 KW for testing into dummy load or no RF drive. The dummy load is on the antenna. With no RF drive, the additional dissipation is in the TWT collector.

(2) Total flow of 2 gpm may be by series flow through the three LRU's if sufficient pressure drop is available.

(3) Cooling air at 85°F.

(4) Antenna volume is 13,500 $in.^3$.

TABLE IX

SUMMARY OF CABIN HEAT LOADS

- COMPARTMENT U - (Modules Rack Area)

<u>Deleted</u>:

Aft Modules Rack	918 watts	
Forward Modules Rack	1,348 watts	
Total	2,266 watts	
Piped	1,521 watts	
Ambient	745 watts	

<u>Added</u>:

Digital Assembly	2,410 watts	(T_{out} max. = 140°F)
Inverter 5V, No. 1	350 watts	T_{out} max. = 160°F
Inverter 5V, No. 2	350 watts	
Inverter 5V, No. 3	350 watts	
Inverter 5V, No. 4	350 watts	
GEANS DCU	210 watts	
GEANS AIU	350 watts	
TOTAL	5,370 watts	
Instrumentation	3,800 watts *	
EAR + Instr.	9,170 watts	

<u>Originally Piped Equipment which Remains</u>:

6209947 Indicator Elec. Unit Frame . . .	92.5 watts
6605600 Heading Vel. Elec. Unit Frame. .	97.8 watts
6609600 Regulator Elec. Unit Frame . . .	36.0 watts
TOTAL	226.3 watts

*2100 watts may be in back bay ambient cooled, 120°F maximum ambient.

ΔPiped Total = 5750 - 1521 = 3849 $\begin{cases} \text{2410 watts is } T_{out} = 140°F \\ \text{1439 watts is } T_{out} = 160°F \end{cases}$

ΔAmbient = 1700 - 745 = 955 watts (or 3800 - 745 = 3055 watts if all in cabin)
Total Original Piped = 1521 + 226.3 = 1747.3 watts
Total Final Piped (160°F) = Orig. + D = 1747.3 + 1439
= 3186 watts (10,871.6 btu/hr.)
Total Final Piped (140°F) = 2410 watts (8222.9 btu/hr.)
Original Total Heat Load = 3250 watts (11,090 btu/Hr.)
Original Total Ambient Load = Total-Piped = 3250-1747.3 = 1502.7 watts
Final Total Ambient = Orig. + Instr. - Deleted = 1502.7 + 1700-745 = 2457.7 watts
Final Total Heat Load = 3186 + 2410 + 2457.7 = 8053.7 (27,479.22 btu/hr.)

TABLE X

SUMMARY OF CABIN HEAT LOADS

- COMPARTMENT W - (Wine Cellar)

Deleted

Nav. Sta.	1,100 watts	
Radar Nav. Sta.	841 watts	
Total	1,941 watts	
Piped	1,115 watts	
Ambient	826 watts	

Added

Pattern Raster Generator	425 watts	1305 watts T_{out} max. = 160°F
Processor Display (PDC)	700 watts	
TF/TA	10 watts	
GEANS IEU	170 watts	
Liq. System Hx	2,855 watts	
Instrumentation	2,067 watts	(Amb. $T_{max.}$ Amb. = 120°F)
Total	6,227 watts	

Original Piped Equipment Which Remains:

AM946 Doppler	140 watts

ΔPiped = 1305 - 1115 = 190 watts (assume all is T_{out} = 160°F)
Total Original Piped (160°F) = 140 + 565 + 550 = 1255 watts
Total Final Piped (160°F) = Orig. + Δ = 1255 + 190 = 1445 watts (4930.3 btu/hr.)
Total Liq. System = 2855 watts (9741.2 btu/hr.)
Original Total Heat Load = 6171 watts (21,057 btu/hr.)
Original Total Ambient Heat Load = Total-piped = 6171-1255
= 4916 watts (16,773 btu/hr.)
Final Total Ambient = Original + Instr.-Deleted = 4916 + 2067-826 = 6157 watts
(21,007 btu/hr.)
Final Total Heat Load = 6157 + 2855 + 1445 = 10,457 watts (35,679.3 btu/hr.)
ΔAmb. = 2067 - 826 = 1241 watts (4234.3 btu/hr.)

Compartment V - (BNS Sta.)
Deleted = 0
Added = 450 watts (Instrumentation)

Gunner's Station -
Deleted = 0
Added = 1508 watts (Instrumentation).

TABLE XI

SUMMARY OF CABIN HEAT LOADS

LOCATION WITHIN CABIN	BEFORE (btu/hr.)	EAR (watts)	AFTER BASIC EAR SYSTEM (watts)	AFTER EAR PLUS INSTR. (watts)
Pilot Station	9,979.3	2,924.76	2,924.76	2,924.76
Upper Aisle	341.0	99.94	99.94	99.94
Gunner Station	4,250.0	1,245.60	1,245.60	2,753.0
Wine Cellar (Compartment W)	21,057.0	6,171.45	8,390.0	10,457.0
Nav. Station (Compartment V)	1,415.0	414.71	414.71	864.0
Compartment U	11,090.0	3,250.29	6,354.0	10,154.0
Aft Elec. Compartment	4,266.4	1,250.41	1,250.41	1,250.41
Compartment M	952.2	279.07	279.07	279.07
Compartment X	4,263.0	1,249.41	1,249.41	1,249.41
RH Elec. Compartment	10,668.0	3,126.61	3,126.61	3,126.61
Compartment Z	2,953.0	865.47	865.47	865.47
Compartment Y	1,536.0	450.17	450.17	450.17
TOTAL	72,770.9	21,327.89	26,650.14	34,473.84
Upper Deck Total		11,491.44	11,491.44	12,998.84
Lower Deck Total		9,836.45	15,158.70	21,475.00

ΔEAR Basic = + 5322.25 watts (all in lower deck)

ΔInstr. = + 7823.7 watts; 1507.4 watts upper deck, 6316.3 watts lower deck

ΔEAR + Instr. = + 5322.25 + 7823.7 = + 13,145.95 watts.

A computer program, capable of calculating air conditioning system performance and the resulting cabin compartment temperatures, was used to analyze the EAR configuration. Initially, the system was analyzed to determine whether adequate cooling could be provided by the existing air conditioning system. For this configuration, all possible equipment was assumed cooled with ram air. Changes in the configuration and operating conditions such as modified precooler, modified 38°F control, relaxing design goals for crew compartment ambient temperatures, narrowing the flight speed/altitude envelope, and ambient temperature limitations were required. The conclusion was that the necessary limitations would not be acceptable. Figure 14 shows an allowable envelope of speed versus free stream ambient temperature for sea level flight. This chart shows that, for example, in order to fly at Mach No. = .4 at sea level, ambient temperature cannot be less than -12°F or greater than 69°F. Figure 14 is for cabin airflow distribution optimized for low level flight and may not be the desirable distribution for altitude operation. Many conditions at low level and high altitudes were analyzed and serious limitations at high altitudes were encountered.

It was concluded that an additional air conditioning pack was needed. System flow requirements, as a function of equipment inlet temperature, were developed. Results are shown in Figure 15. The three major manufacturers of air cycle refrigeration packs, AiResearch, Hamilton Standard, and Stratos have been given preliminary requirements and asked for information as to availability of equipment. Responses are incomplete at this time and a pack has not been selected. A preliminary investigation of availability, capability, cost, and procurement time has been accomplished. It appears that a modified pack from a B-47 aircraft offers a feasible solution. The apparent required modifications include modified flow control, 38°F air line relocation, and added shut-off valve, and perhaps a safety shroud.

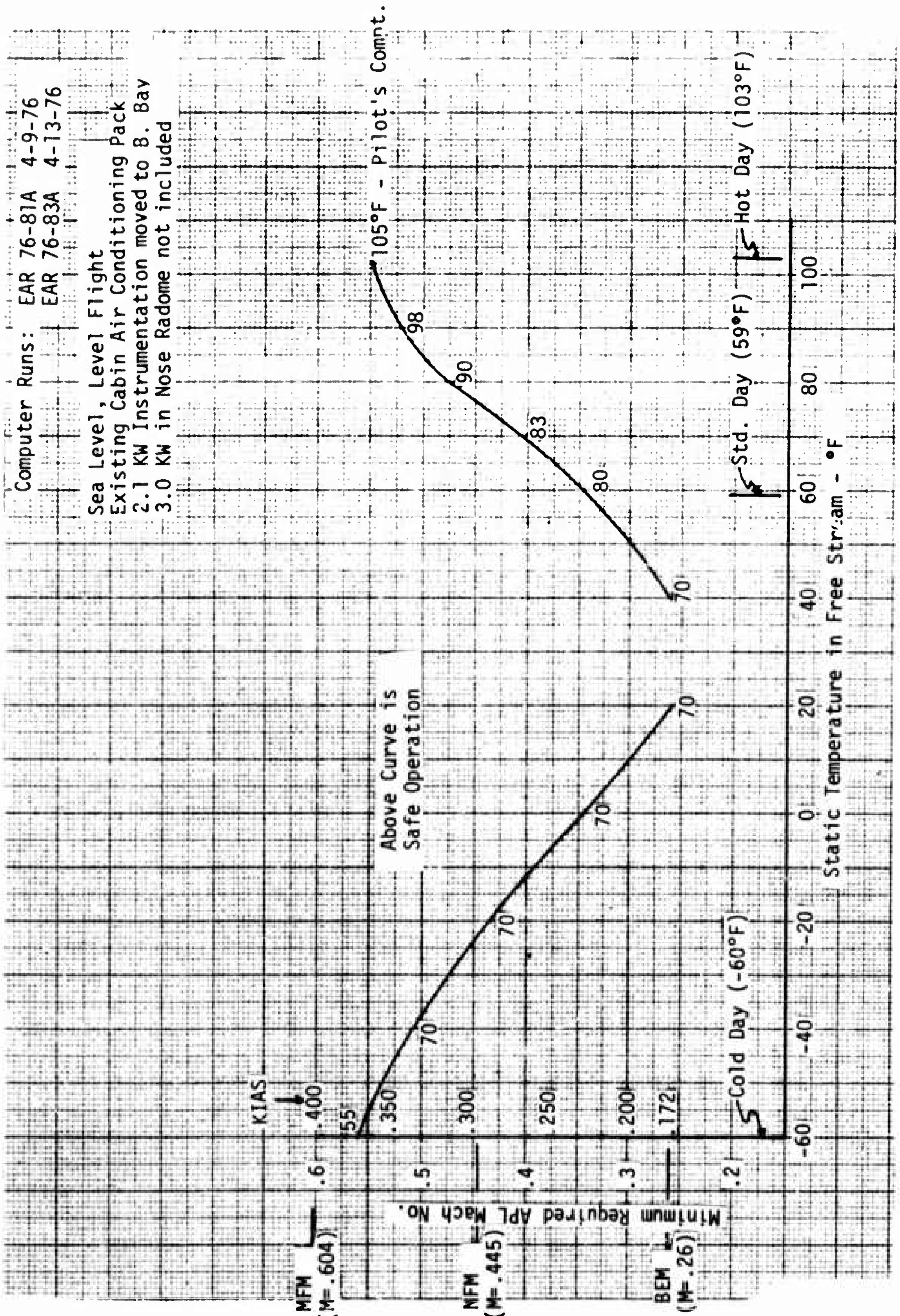


Figure 14. Flight Speed/Ambient Temperature Envelope With Existing A/C System For Sea Level Flight and Minimum Heat Load

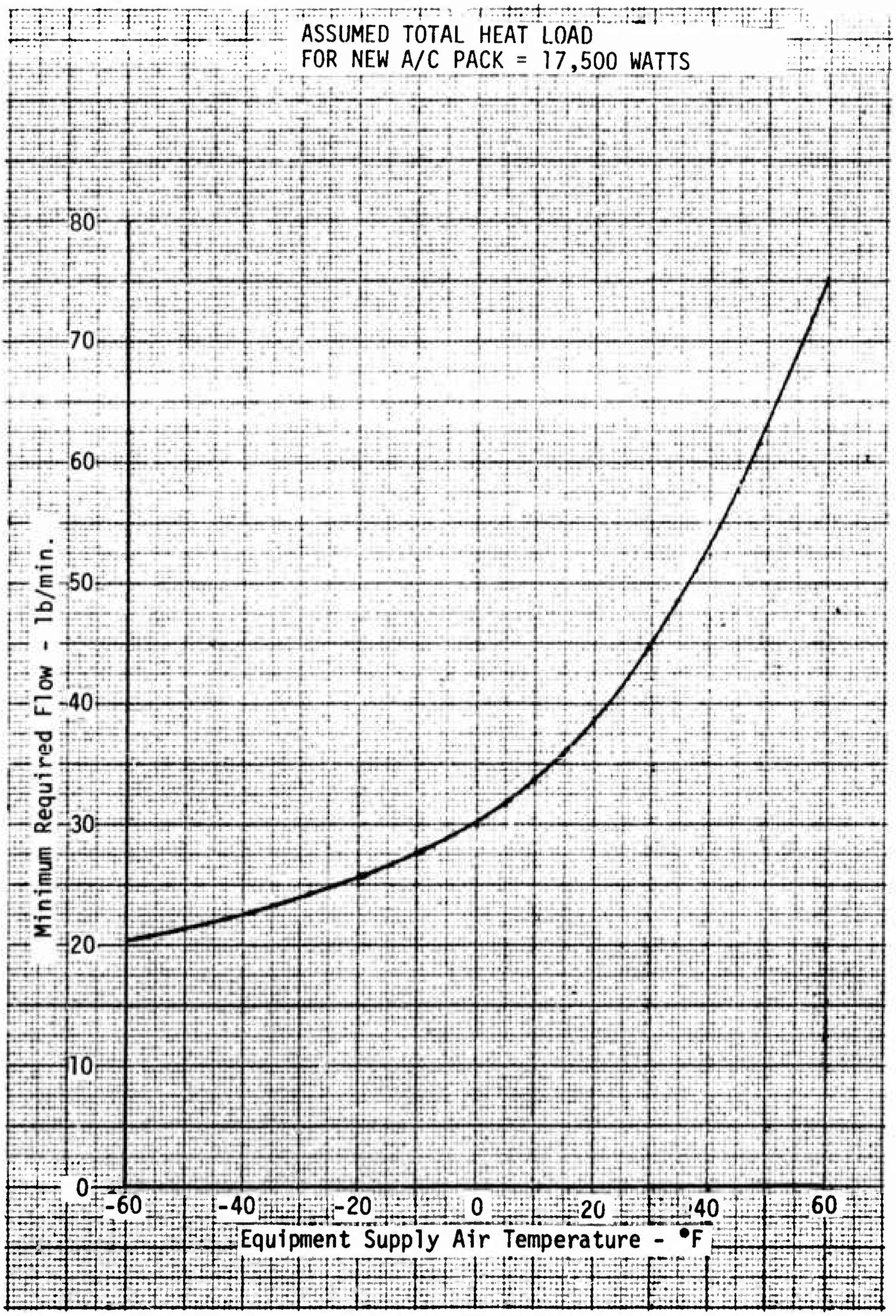


Figure 15. Preliminary Total Cooling Airflow Requirement Versus Inlet Temperature For EAR System Including Installation

SECTION VI

AVIONICS INTERFACES

1. TEST ARTICLE INTERFACES

The EAR and SPN/GEANS systems will require input data from and supply data to B-52 systems during the flight test program. These data signals are listed in Table XII and further discussed in the paragraphs of Section VI.2 thru VI.17 below. Table XIII lists the operational systems impacted by this interface.

2. RADAR ALTITUDE (H_R)

Radar altitude will be provided by an APN-194 radar altimeter. This altimeter will replace the APN-150 and supply altitude data through the interface unit to the EAR. See Figure 16. The altitude signal format is digital serial binary and is readout upon receipt of a clock signal from the EAR.

3. RADAR ALTITUDE VALIDITY

Radar altitude validity is required by EAR. The APN-194 will supply a 5 volt positive DC signal when valid radar altitude data is available. See Figure 16.

4. TRUE AIRSPEED

True airspeed will be computed for the EAR with selected BNS components (See Section VI.18). The application of existing components for this function eliminates the need for new functional circuit design and does not disturb the airspeed interface with other functions. (BNS computed airspeed is used in the establishment of a vertical reference platform for pitch and roll data generation). See Figure 22. Figure 17 depicts the data flow between BNS components as required for airspeed computation. Two servo loops, $f_1(M)$ and V_A, are used to compute true airspeed. The temperature transducer, static pressure transducer, static pressure calibrator, and dynamic pressure transducer provide inputs to the airspeed computer. This electro-mechanical computer is driven by amplifiers relocated to the Heading Electronics Frame. Required voltage amplitudes and phases are developed by the AC signal power supply, AC excitation power supply, and +300V DC power supply.

An alternate source of true airspeed for the Test Article is the C-2A True Airspeed Computer. It would provide true airspeed as synchro data. The quality and format of the true airspeed signal desired for the Test Article will determine which source will be utilized.

5. MACH NUMBER

Mach number (M) is computed by the C2A Computer. Its synchro output is supplied to the EAR radar.

TABLE XII

PARAMETER REQUIREMENTS

EAR

Signal	Symbol	Discussed in Paragraph
Radar Altitude	H_R	VI.2
Radar Altitude Validity		VI.3
True Airspeed	V_A	VI.4
Mach Number	M	VI.5
Normal Acceleration	NACG	VI.6
Center of Gravity Location	CG_L	VI.7
Pilot Control Stick Position		VI.8
Angle of Attack	AOA	VI.9
Audio Warning		VI.10
Terrain Following Failure Indication		VI.11
Display Synchronization		VI.12
Video		VI.13
Power		VI.14

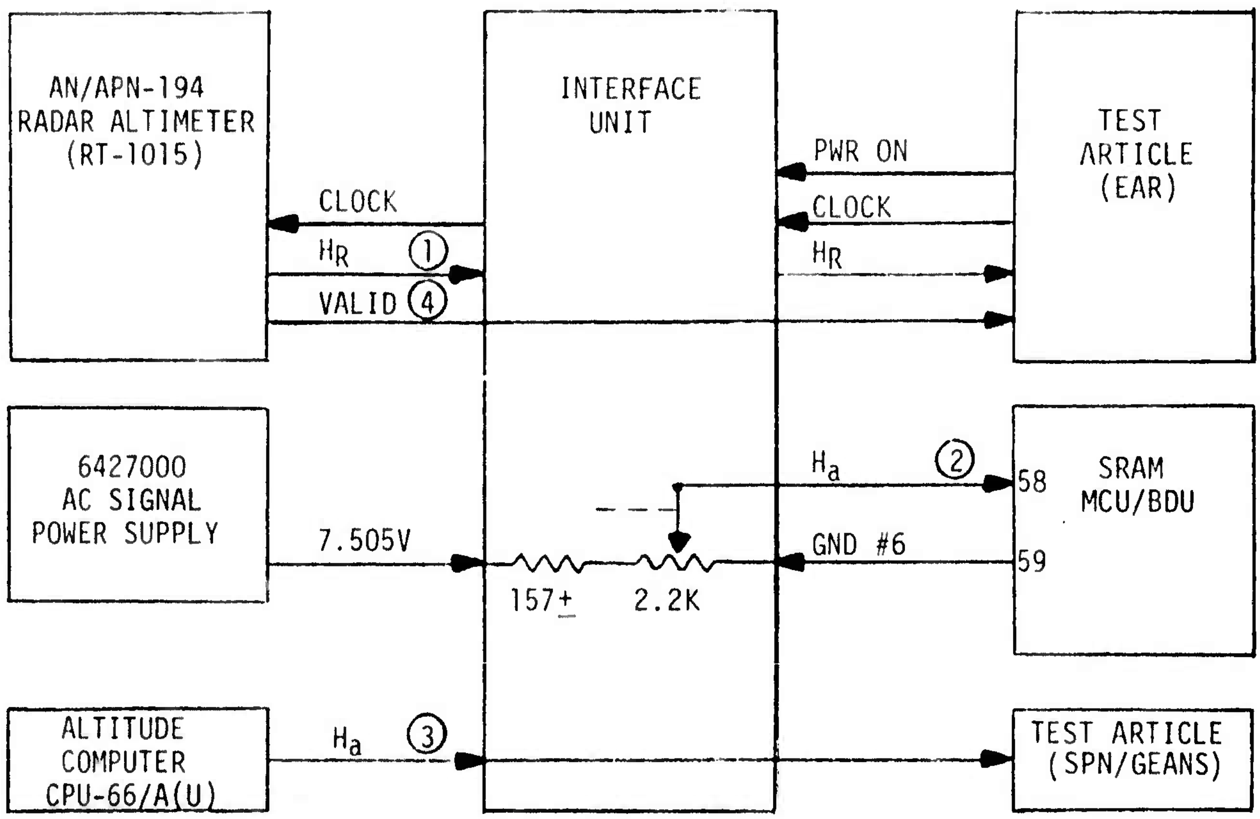
SPN/GEANS

Barometric Altitude	Ha	VI.15
Magnetic Heading	h_m	VI.16
Airplane Pitch	p	VI.17
Airplane Roll	r	VI.17
Power		VI.14

TABLE XIII

INTERFACE IMPACTED OPERATIONAL SYSTEMS

AN/APN194	Radar Altimeter	Replaces AN/APN-150
AN/ASB-16 (Part)	Bombing Navigational Computer	Selected Components Installed in a non-standard configuration
C-2A	True Airspeed Computer	
	Angle-of-Attack	
AN/ASQ-151	Electro Optical Viewing System	Some symbology not available. Manual steering only
CPU-66/A( )	Altitude Encoding Altitude Computer	
AGM-69A	SRAM	No change in required data format
N-1	Compass System	
AN/AJA-1	True Heading Computer	
MD-1	Astro Compass	
AN/APN-89A	Doppler Radar	No BNS Memory Mode Input



① Digital, 0 to 5000 feet. Serial binary.

② Linear Analog, 0 to 50000 feet; 10000 feet/volt.

③ Synchro output, 26V excitation.

④ +5V DC.

Figure 16. Altitude Data Interface

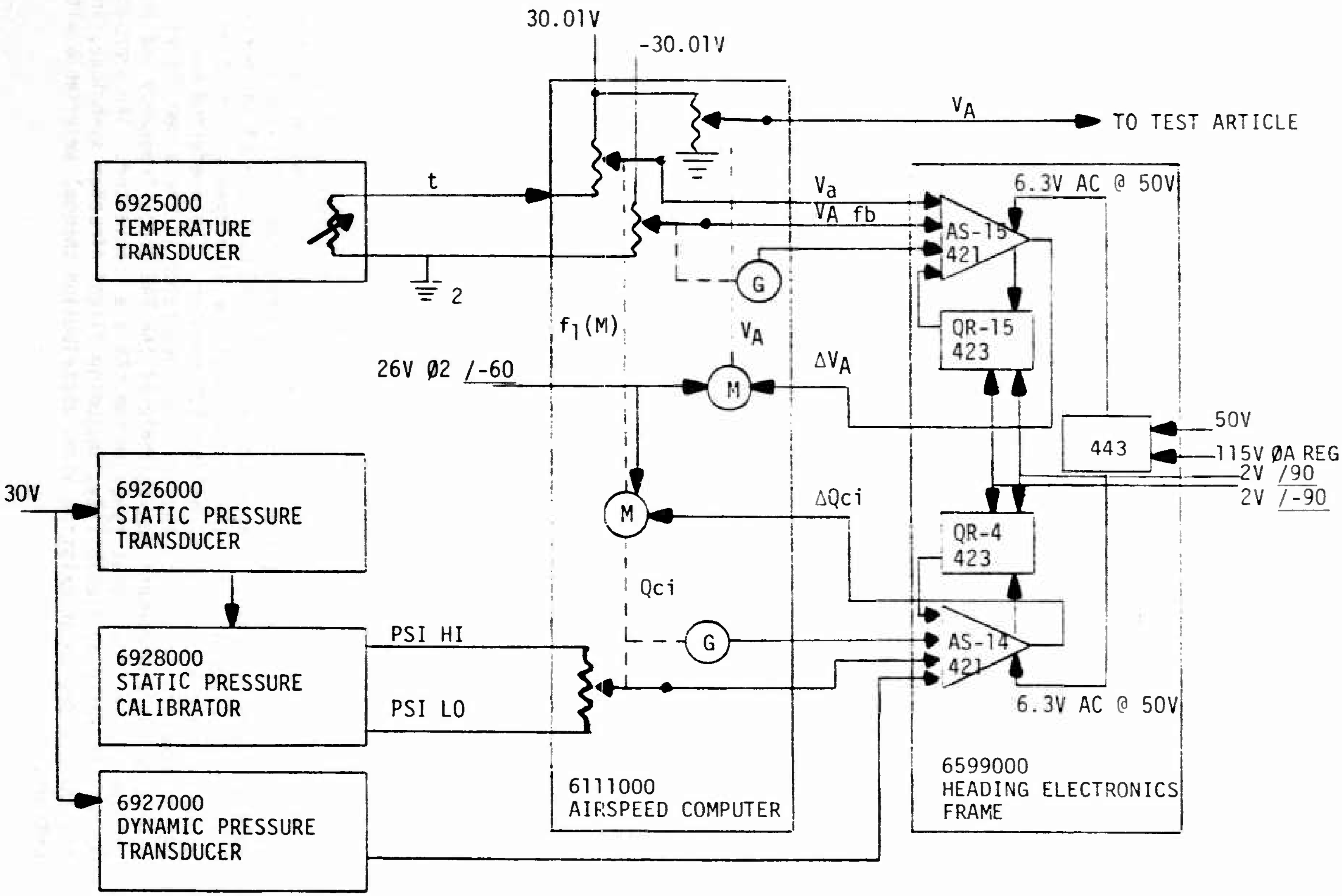


Figure 17. True Airspeed Computation and Interface

6. NORMAL ACCELERATION (NACG)

Normal acceleration (NACG) will be supplied to the EAR from flight test instrumentation.

7. CENTER OF GRAVITY LOCATION (CG_L)

Center of gravity location (CG_L) can be computed during flight by translating recorded total fuel readings to CG location. Normal fuel distribution and usage will be required.

8. PILOT CONTROL STICK POSITION

Pilot control stick position will be monitored with a linear potentiometer added to the control stick linkage. This technique has been successfully utilized on previous test programs.

9. ANGLE OF ATTACK (AOA)

Angle of Attack (AOA) will be computed by the Angle of Attack Computer and transmitted as synchro data to the EAR. The same signal will be used which normally is supplied to the BNS radar antenna and TA servos. The power source for this signal is from the BNS regulated power units.

10. AUDIO WARNING

In the TF mode, a warning signal from the EAR to the interphone system will be supplied which will inform the pilot to initiate either climb or descent to maintain a desired profile.

11. TERRAIN FOLLOWING FAILURE INDICATION

In the TF mode, a signal will be provided from the EAR to turn on a warning light when reliable data is not available.

12. DISPLAY SYNCHRONIZATION

The EAR TA/TF video can be displayed on the present B-52 EVS. Both the EVS and the EAR scan converted video are in 875 line RS-170 standard TV format, however the clock frequencies differ. The EVS clock is at 13.44 MHz, or the 512 multiple of the horizontal frequency (512 H), while the EAR clock is available at 33.75 MHz which is approximately the 1286 multiple of the horizontal frequency (1286 H). The TA/TF video could be displayed by providing EAR composite synch direct to the pilot's EVS monitor. The preferred configuration, however, is to lock the EVS synch generator in the Video Distribution unit to the appropriate sub-multiple of the EAR clock frequency and in turn provide EVS vertical reset synch to the EAR scan converter. The approach assures full EVS display flexibility, including flight director symbology, and puts the TA/TF video on an existing video distribution channel selectable with EVS controls.

The feasibility of phase-locking the EVS clock has been established. Circuitry developed for phase locking the clock in the EVS Automated Test Equipment is applicable. Since treating the EAR clock frequency as 1286H provides a prime divisor of 643, some consideration will be given to treating 33.75 MHz as 1280H to simplify the interface design. The resulting equivalent EVS clock frequency of 13.50 MHz represents a 0.45 percent deviation from nominal which is probably excessive.

Current EAR scan converter design does not consider an external vertical reset derived from the EAR clock. If the scan converter cannot be so adapted, it will be necessary to modify the EVS video distribution unit to receive synch pulses from the EAR in addition to the clock signal. The resulting configuration would still provide the desired EVS display flexibility including video selection.

13. VIDEO

Video data from the EAR will be supplied to EVS. The RS-170 format is compatible with both systems.

14. ELECTRICAL POWER INTERFACE

The following electrical power types and quantity shall be provided to EAR Test Article. The B-52 Test Bed aircraft shall provide 118 volt nominal L-N, 3Ø 400 Hz power and 28 volt nominal TR DC power, in accordance with Boeing Document D3-3908 Section V, to the equipment.

The aircraft shall supply 5 volt nominal AC 400 Hz dimmable lighting power from a light dimming control. This power shall be used for panel edge lighting only.

15. BAROMETRIC ALTITUDE (Ha)

Barometric altitude will be provided by the CPU-66. See Figure 16. Its synchro output will be converted to an AC analog of 10,000 feet/volt in the Interface Unit for direct interface with the existing SRAM MCU/BDU. An identical voltage divider network and voltage source will be used to output this signal that is used in the standard configuration. Ha can also be inserted manually into the SRAM Carrier Navigator with a Special Instruction Code (SIC). This data will be supplied to the Test Article as either a synchro signal or as AC analog. The CPU-66 also has a digital output that could be converted to analog and used as a source for altitude data in lieu of the above synchro signal.

16. HEADING (h, h_m)

Airplane true heading (h) referenced to true north is determined by the standard configuration N-1, AJA-1, MD-1 heading systems. The most accurate heading available is normally used and is supplied to the Doppler Radar, the BNC Heading Data Computer 6106400, and the AGM-69A SRAM Carrier Navigator. This heading can also be provided to the Test Article. The Heading Select Panel is used to select the heading system which is to supply heading data.

Magnetic heading (h_m) will also be supplied as a synchro signal from the N-1 Master Indicator through the Interface Unit to the Test Article. See Figure 21.

17. PITCH AND ROLL (p and r)

These stabilization signals will be developed by selected components of the BNS. See Figure 22. The Stabilization Data Generator 6129000 contains a platform which is gyro stabilized and torqued to a level position by sensing earth's gravity vector and correcting for aircraft latitude, speed and heading. The Pitch and Roll Computer 6140000 outputs pitch and roll data for use by the doppler, astro tracker, and Test Article. This configuration was selected to have essentially no impact on existing interfaces.

18. B-52 TEST BED NAVIGATION

The installation of the EAR, SPN/GEANS, and associated instrumentation in the aircraft for this test program could not be reasonably accomplished without the removal of certain equipment. Considerable space was gained in the lower compartment by removal of the AN/ASB-16 Bombing Navigation System (BNS). A navigational capability was retained by keeping the APN-89A Doppler Radar, N-1/AJA-1 True Heading System, MD-1 Astrocompass and the AGM-69A Carrier Navigator inertial system operable. The Electro-Optical Viewing System (EVS) will also be operable. These systems have a normal interface with the BNS as shown in Table XIV. The interface developed for these parameters for the flight test program is discussed in the paragraphs as shown in Table XIV.

The resulting configuration will provide a test bed navigational capability independent of the Test Article as well as the necessary airplane and system data for interface with the Test Article.

BNC built-in bombing and navigational test problems, unit heaters for units located in the pressurized sections, and certain interlock circuits will not be functional.

19. REQUIRED BOMBING NAVIGATIONAL COMPUTER (BNC) COMPONENTS

Several BNC data parameters and control discretes normally supplied to other systems are required by this installation. This requirement will be fulfilled with the reinstallation of certain BNS components interconnected with modified aircraft wiring and with the addition of other equipments. See Table XV.

Many BNS electrical and electronic functions are packaged in small cylindrical cans which are mounted in Electronic Unit Frames where they are cooled by forced air. The 18 cans required by this installation are taken from seven different frames and relocated into three frames to save space.

TABLE XIV

BNS DELETION AFFECTED DATA INTERFACE

Signal	Symbol	Discussed in Paragraph
Doppler		
Heading	h	VI.16
Ground Speed	V_{RG}	VI.21
Ground Track Direction	h_{gta}	VI.22
Pitch	p	VI.17
Roll	r	VI.17
N-1/AJM-1		
Airplane Loading	h	VI.16
Latitude	λ	VI.23
Longitude	$\emptyset$	VI.23
MD-1 Astro Compass		
Latitude	λ	VI.23
Longitude	$\emptyset$	VI.23
Heading	h	VI.16
SRAM Carrier Navigation		
Heading	h	VI.16
Easterly Velocity	V_E	VI.24
Northerly Velocity	V_N	VI.24
Easterly Range to Radar Designated Crosshair Location	R_E	VI.25
Northerly Range to Radar Designated Crosshair Location	R_N	VI.25
Discretes		VI.26
Altitude	H_a	VI.15
EVS		
Hand Control Signals		VI.9
Time-To-Go	TTG	N/A
Heading Error	h_e	N/A
Roll	r	VI.17
Azimuth and Elevation Limits		N/A
Azimuth of Line of Sight		N/A
Elevation of Line of Sight		N/A

TABLE XV

REQUIRED BNC COMPONENTS

Hand Control	6105000
Heading Data Computer	6106400
Airspeed Computer	6111000
Stabilization Data Generator	6129000
Pitch and Roll Data Computer	6140000
BNS Power Control	6145000
Data Set Control	6147000
Voltage Regulators (3)	6152000
Heading Error Computer	6208200
Computer 300V Power Supply	6402700
Computer 150V Power Supply	6402800
AC Excitation Power Supply	6426000
AC Signal Power Supply	6427000
Memory Point Electronics Frame	6599200 *
Heading Electronics Frame	6593000 *
Regulator Electronics Frame	6609000
Temperature Transducer	6925000
Static Pressure Transducer	6926000
Differential Pressure Transducer	6927000
Static Pressure Calibrator	6928000
Heading Select Panel	
BNS Lighting Control	A328

* These units contain a non-standard configuration of cans.

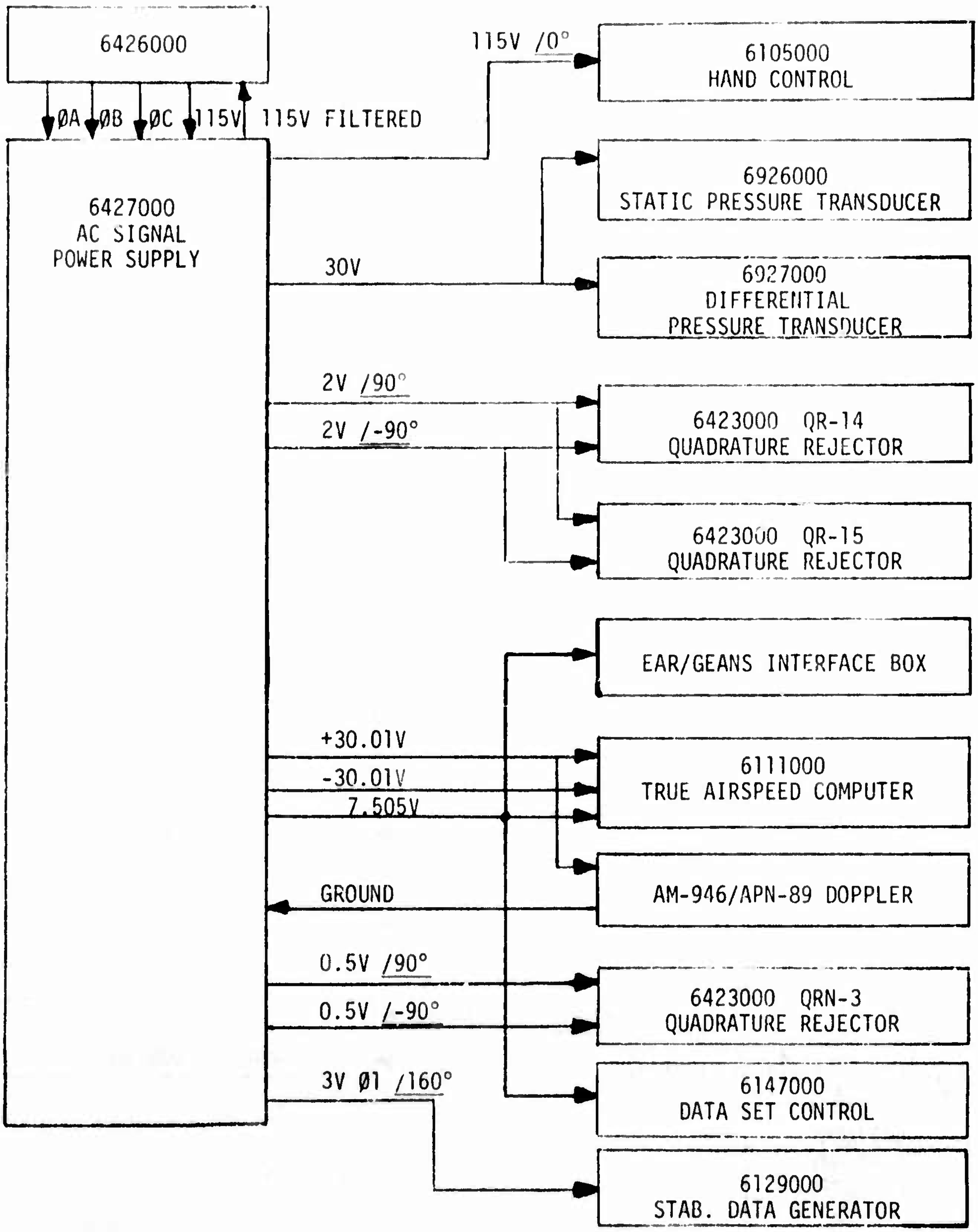


Figure 18. AC Signal Power Supply Interface

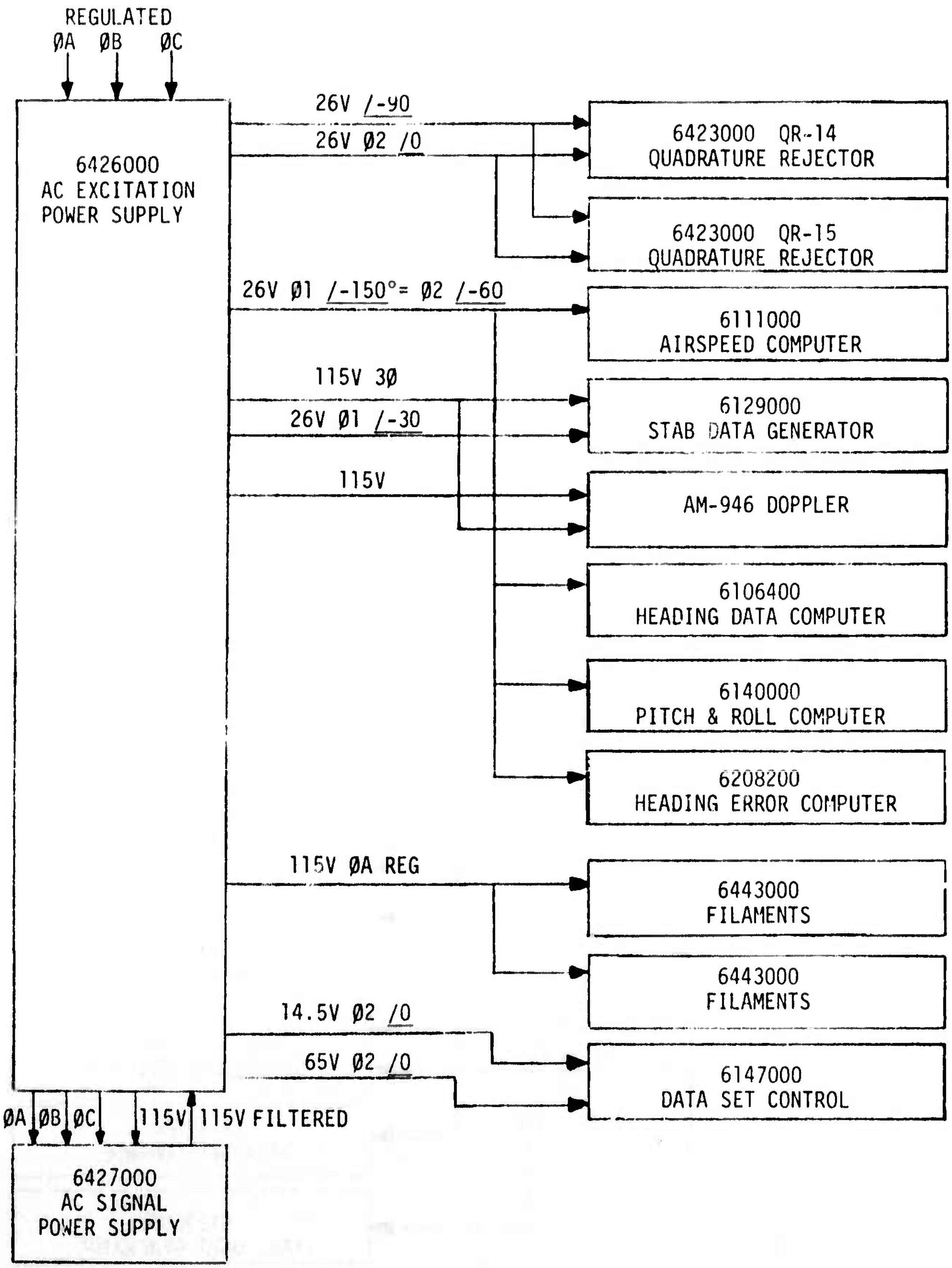


Figure 19. AC Excitation Power Supply Interface

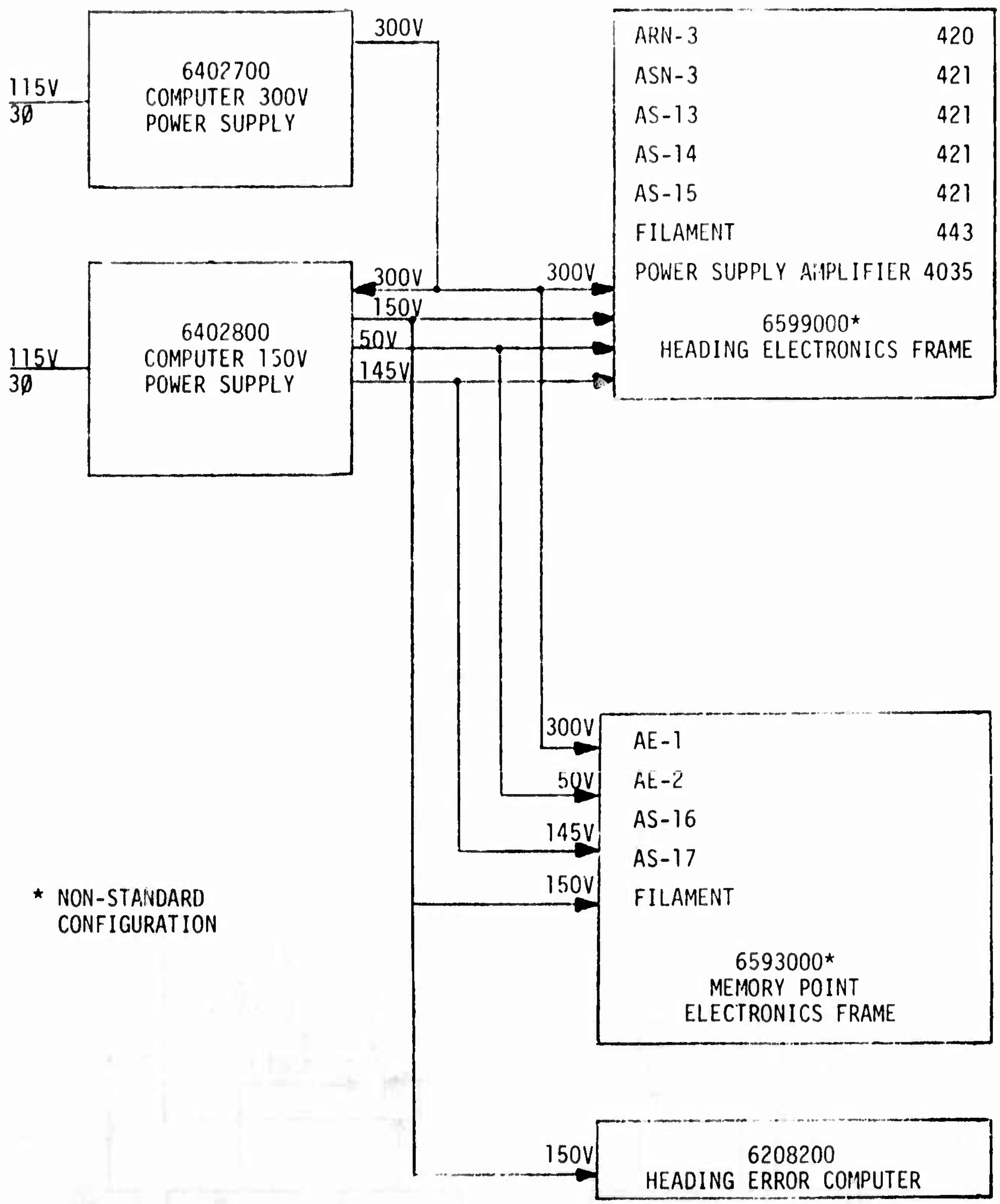


Figure 20. DC Power Distribution

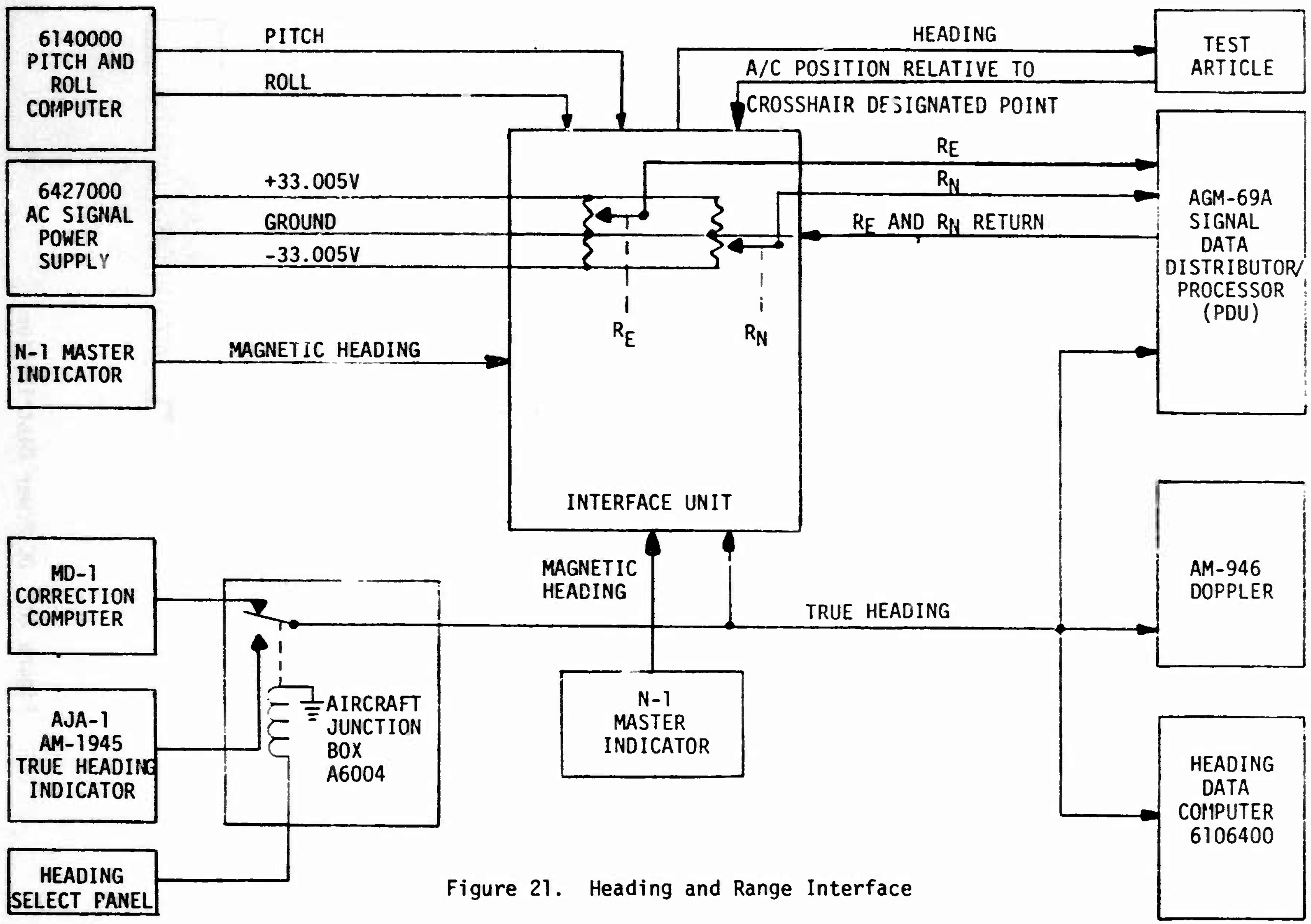


Figure 21. Heading and Range Interface

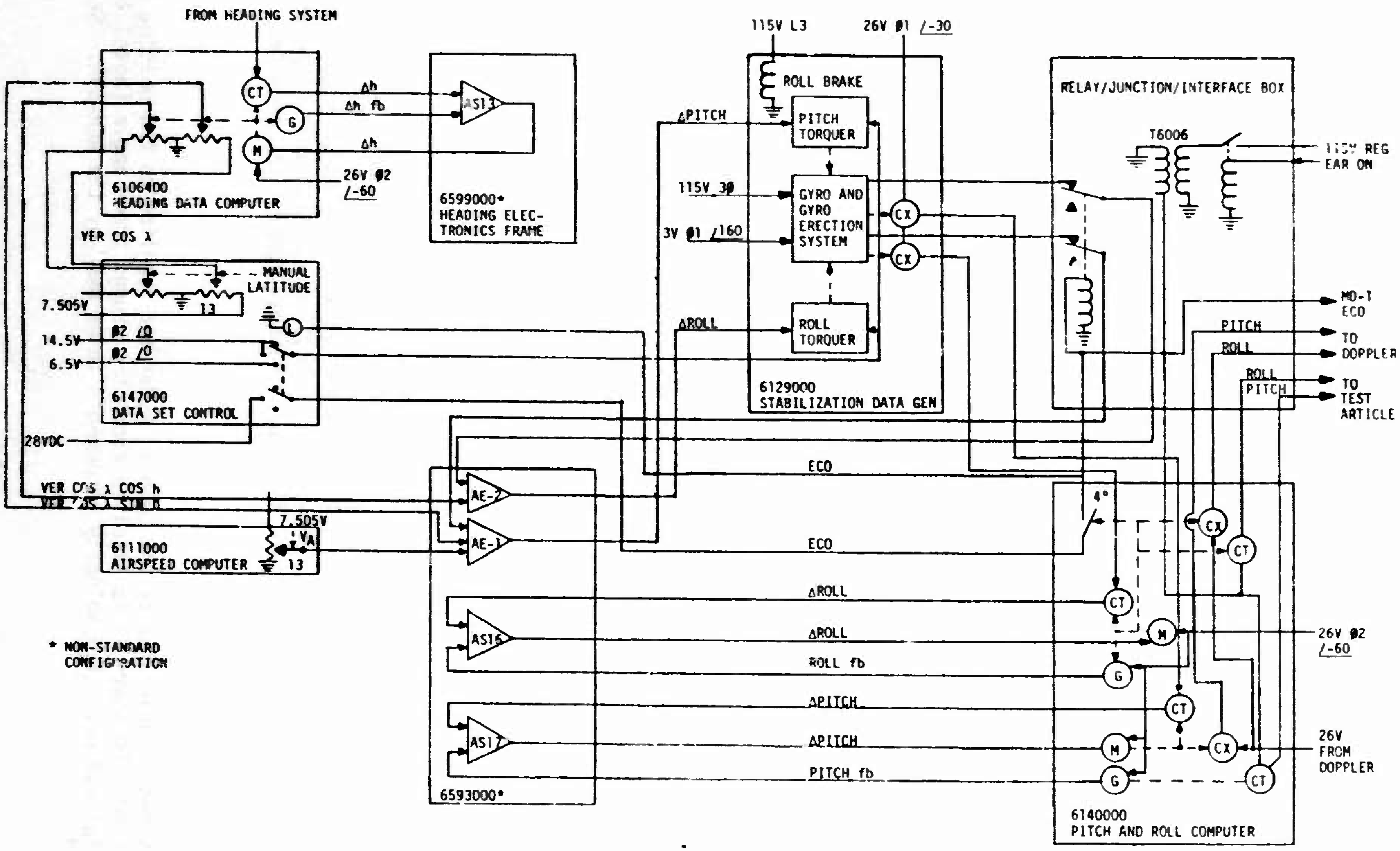


Figure 22. Stabilization Interface

The three selected frames are the Heading Electronics Unit Frame, 6593000, the Memory Point Electronics Unit Frame, 6599200, and the 6609000 Regulator Electronics Frame. These were chosen because the openings in the frames have the proper sizes to receive the required cans.

Table XVI lists the cans required, their functional application, and their locations in the normal and test installations. Two can locations are to receive a different type can than in the normal installation. ARN-3 type 420 replaces AT-3 type 422 and 4035 replaces a dummy plate. The diameters of the replacements are the same as the replaced parts so no physical interface problem exists.

The cans are normally connected to cables which are an integral part of the Relay Frames. Since these relay frames are not used in the test installation, the cans will connect directly with airplane wiring. A special wiring harness will be manufactured for this installation. A few relays, normally a part of the relay frames, will be required and will be physically located in the Interface Unit.

All other BNC components required for this installation are panel mounted and do not require cooling. These are identified with the paragraphs that discuss parameter generation interface.

20. POWER REQUIREMENTS FOR BNS COMPUTER

Power requirements for the Bombing Navigational Computer are 115V 3Ø, 115V 1Ø, and 28 VDC. The BNS components require critical AC voltage amplitude ratios as developed by the AC Signal Power Supply 6427000. See Figure 18. Voltages whose phase and amplitude are critical are provided by the AC Excitation Power Supply 6426000. See Figure 19. A regulated power supply controls the amplitude of each phase of a 115V 3Ø source from which other voltages are developed.

The regulated power supply is composed of three voltage regulators 6152000 and three amplifiers 6153000 (one for each of the 3Ø inputs). The regulators are panel mounted and the amplifiers are installed in the Regulator Electronics Unit Frame 6609000.

DC voltages are generated by B-52H BNS power supplies. These DC supplies were chosen over the standard B-52G BNS power supply because they are air cooled rather than oil cooled, and because they are readily available. This negates a requirement for an oil pump, radiator, and plumbing. The B-52H power supplies are +300 volt Computer Power Supply 6402700 and +150 volt Computer Power Supply 6402800. See Figure 20. The +150 volt supply also provides +145 and +50 volts DC.

21. GROUND SPEED (V_{RG})

Ground speed is developed by the Doppler Radar and supplied to the Heading Error Computer 6208200 where it is resolved above the ground track angle (hgta) into north (V_N) and east (V_E) velocity components to be used by the AGM-69A. See Figure 23.

TABLE XVI

REQUIRED COMPONENTS OF ELECTRONIC UNIT FRAMES

Function	Required Component Application & Part No.		Standard Location	Test Installation BNS Frame and Can Location	
Airspeed	AS-14	421	6597000	6599200	AS-4
Airspeed	AS-15	421	6597000	6599200	AS-6
Airspeed	QR-14	423	6597000	6599200	QR-1
Airspeed	QR-15	423	6597000	6599200	QR-4
Heading	AS-13	421	6593000	6599200	AS-3
Velocity Components	ASN-3	421	6605000	6599200	AS-1
Velocity Components	QRN-3	423	6605000	6599200	QR-3
Velocity Components	ARN-3	420*	6605000	6599200	AT-3
Power Supply Filter Amplifier	Amp	4035*	6403300	6599200	Dummy
Filament Transformer	Fil	443	6599200	6599200	Fil
Stabilization	AE-1	430	6593000	6593000	AE-1
Stabilization	AE-2	430	6593000	6593000	AE-2
Stabilization	AS-16	421	6595000	6593000	AS-11
Stabilization	AS-17	421	6595000	6593000	AS-13
Filament Transformer	Fil	443	6593000	6593000	Fil
Voltage Regulator	3-Amplifier	153	6609000	6609000	3-Amplifier

* These components are different from those normally installed at this frame location.

22. GROUND TRACK DIRECTION (hgta)

The azimuth of the ground track from true north is the sum of true heading and doppler drift. Heading is taken from the Heading Data Computer and added to the drift angle in the dopper AM-946. The output ground track angle is sent to the Heading Error Computer 6208200 where it is used in the generation of V_N and V_E. See Figure 23.

23. LATITUDE AND LONGITUDE (λ and $\emptyset$)

Latitude and longitude are normal inputs to the N-1/AJA-1 and MD-1. The systems have provisions to manually update these values when the BNS signals are not available. During the test program these values will be manually set. The vertical reference gyro also requires latitude data. This will be manually set on the Data Set Control 6147000.

24. VELOCITY EAST AND VELOCITY NORTH (V_E and V_N)

The velocity components for SRAM will be generated by the Heading Error Computer 6208200 in the same manner as in the normal installation. A BNS memory mode will not be available but SRAM is not programmed to use V_E and V_N when this secondary mode is employed. See Figure 23.

25. RANGE EAST AND RANGE NORTH (R_E and R_N)

These signals will be supplied from the Interface Unit to the AGM-69A if relative position data is available from the EAR which can be used by the Interface Unit to compute these range values. The visual check point mode is, of course, also available in the AGM-69A Carrier Navigator for initialization and update.

The EAR data would be converted as required to represent R_E and R_N values as determined in a horizontal plane at the aimpoint elevation. The BNC values of true heading, airplane roll, airplane pitch and altitude would be used as required to compensate the EAR data for aircraft state. The Interface Unit would then supply analog values of R_E and R_N using the same reference voltage, impedance, and scale factor as on a standard B-52G aircraft.

26. DISCRETES

The AGM-69A and the EVS normally receive certain discretes and data in addition to that already discussed. The AGM-69A discretes of Low Altitude Calibrate, Altitude Calibrate, Bomb Door Open, and Bomb Mode will not be supplied by this configuration as they are not needed. Some symbology on the EVS displays will not be generated as sources for this data are not available, nor is this data required. Missing symbology includes Time-To-Go and Heading Error. BNS steering limit discretes also are not available or needed.

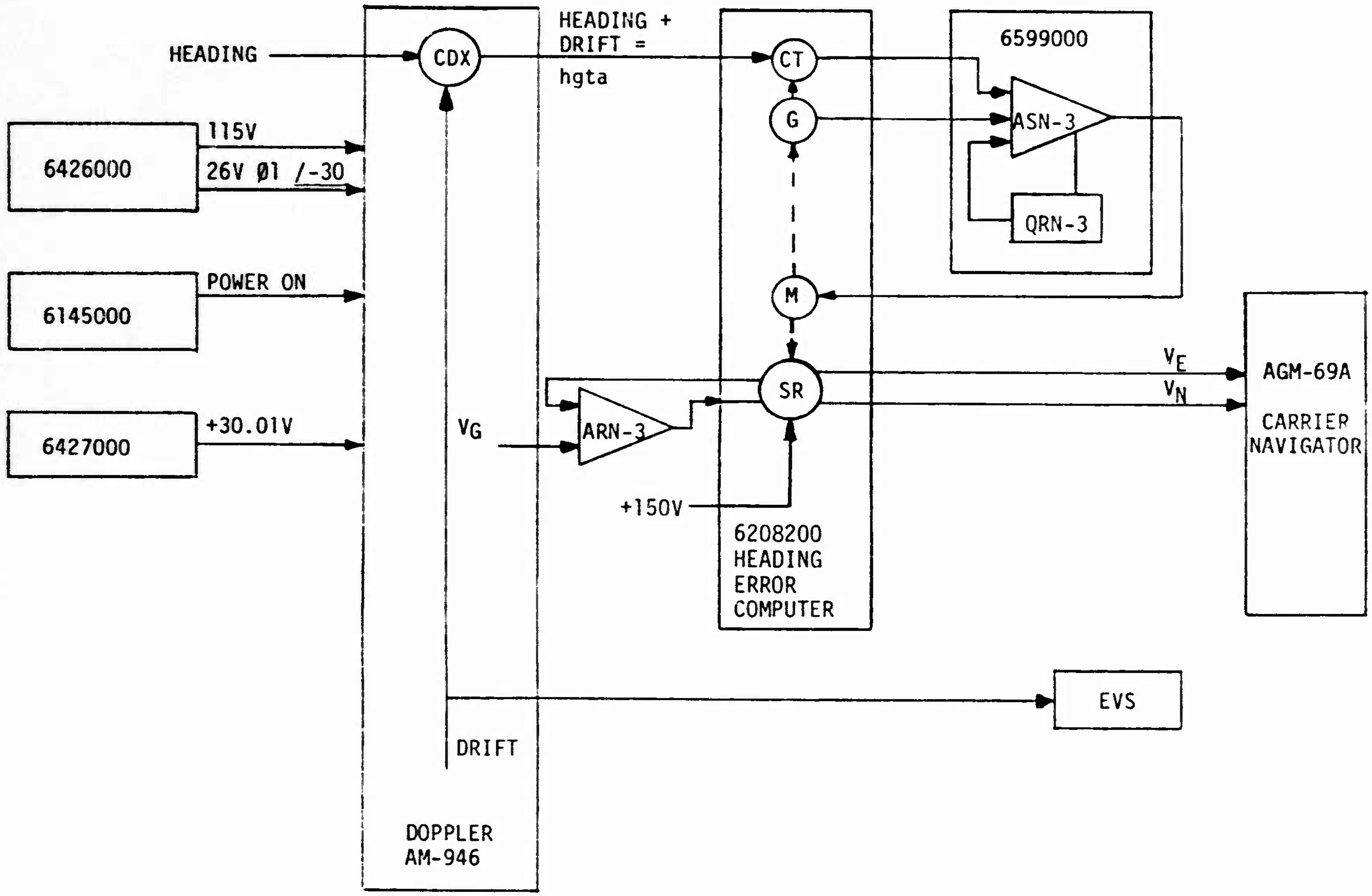


Figure 23. Doppler Radar Interface

27. HAND CONTROL SIGNALS

The BNS Hand Control 6105000 will be used to supply steering control for the EVS. The BNS AC Signal Power Supply will provide the excitation voltage as in a standard installation.

28. INTERFACE UNIT

The composite functions of the Interface Unit are shown in Figure 24.

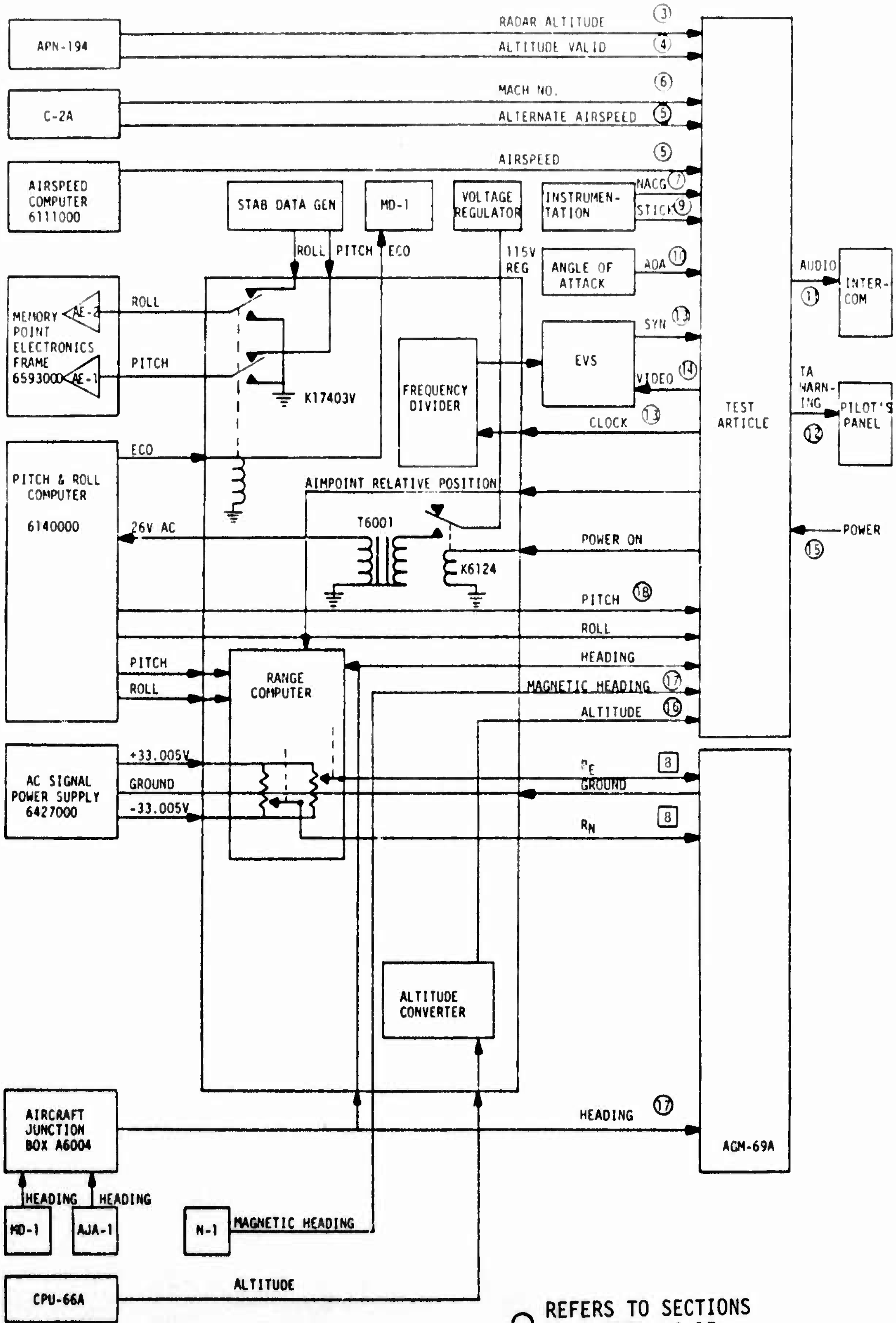


Figure 24. Interface Unit

○ REFERS TO SECTIONS VI.1 THRU VI.17

□ REFERS TO SECTIONS VI.18 THRU VI.28

SECTION VII

FLIGHT TEST PLANNING

The following information is provided to aid the EAR development contractor with the Flight Test Planning. The information was generated using Boeing's past experience from other B-52 Flight Test Programs. It specifically describes requirements and available facilities relative to the airplane being based at Wichita and covers Crew Requirements, Data Handling, Data Processing Station, Airplane Maintenance, EAR Maintenance/Support Facilities, Flight/Ground Safety, Test Range Coordination, EMI/EMC Testing, and B-52 Flight Planning.

1. AIRPLANE BASING

The EAR B-52 test airplane is scheduled to be based at the Boeing-Wichita facility, Wichita, Kansas. This location is centrally located to an array of government ranges for potential usage. The location is adjacent to several military low altitude, high speed training routes (Olive Branch) and VFR low altitude training routes (TR) including six originated by Boeing-Wichita for test and evaluation of the B-52 airplane and its systems in the low level environment. Figures 25 and 26 show the relative locations of these ranges and routes. Although operations can be anticipated over the entire continental United States and adjacent waters, the concentration of operation can be expected to be within a 500-nautical mile range of Wichita.

The Boeing-Wichita facility (AF Plant No. 13) is located adjacent to McConnell AFB, Kansas. Facilities and services to be made available to the EAR program include the following: Figure No. 27 shows the Boeing complex and indicates the current planned locations of many of the facilities.

1) The test aircraft will be located on the north end of Ramp No. 300, Positions 318, 319, or 320. The position is backed by a blast fence and includes air and electric utilities, tie down, and ramp flood lights. A line shack will be located adjacent to the aircraft.

2) Avionic Laboratory facilities are located approximately 200 yards north of the airplane position. B-52 avionics and the inertial system bench maintenance can be accomplished in this building, primarily on the second floor. All standard Avionic Laboratory equipment is available as well as many unique facilities.

3) The Flight Operations building contains facilities unique to flight crew requirements such as map room, locker and shower rooms, personal equipment storage and repair, ready room, weather briefing equipment, and radio communications equipment.

4) Engineering offices are located on the third floor of the Administration Building, Cafeteria Building, and Experimental Flight Hangar.

5) The Data Processing Ground Station is located in the Experimental Flight Hangar.

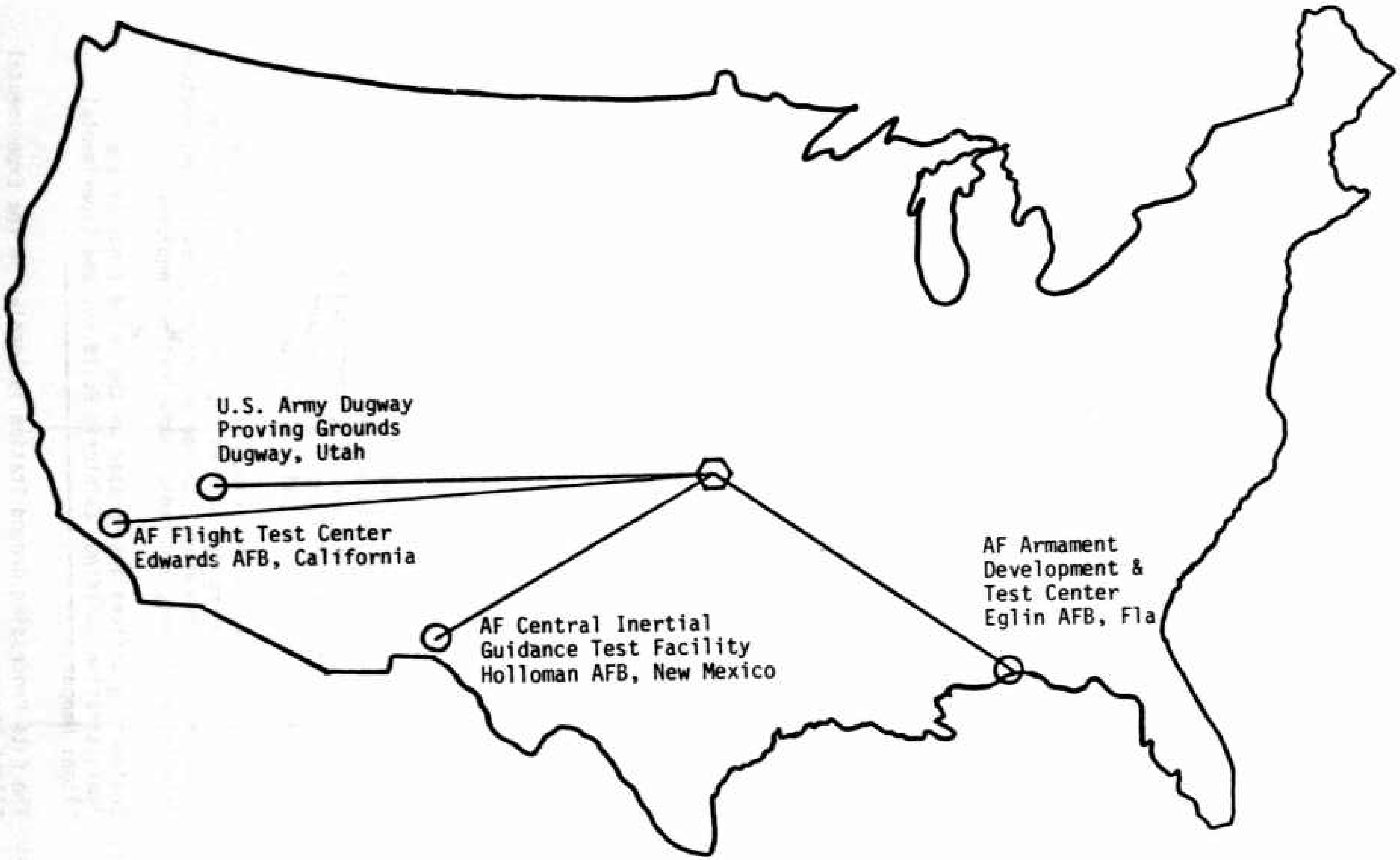


Figure 25. Candidate Government Ranges

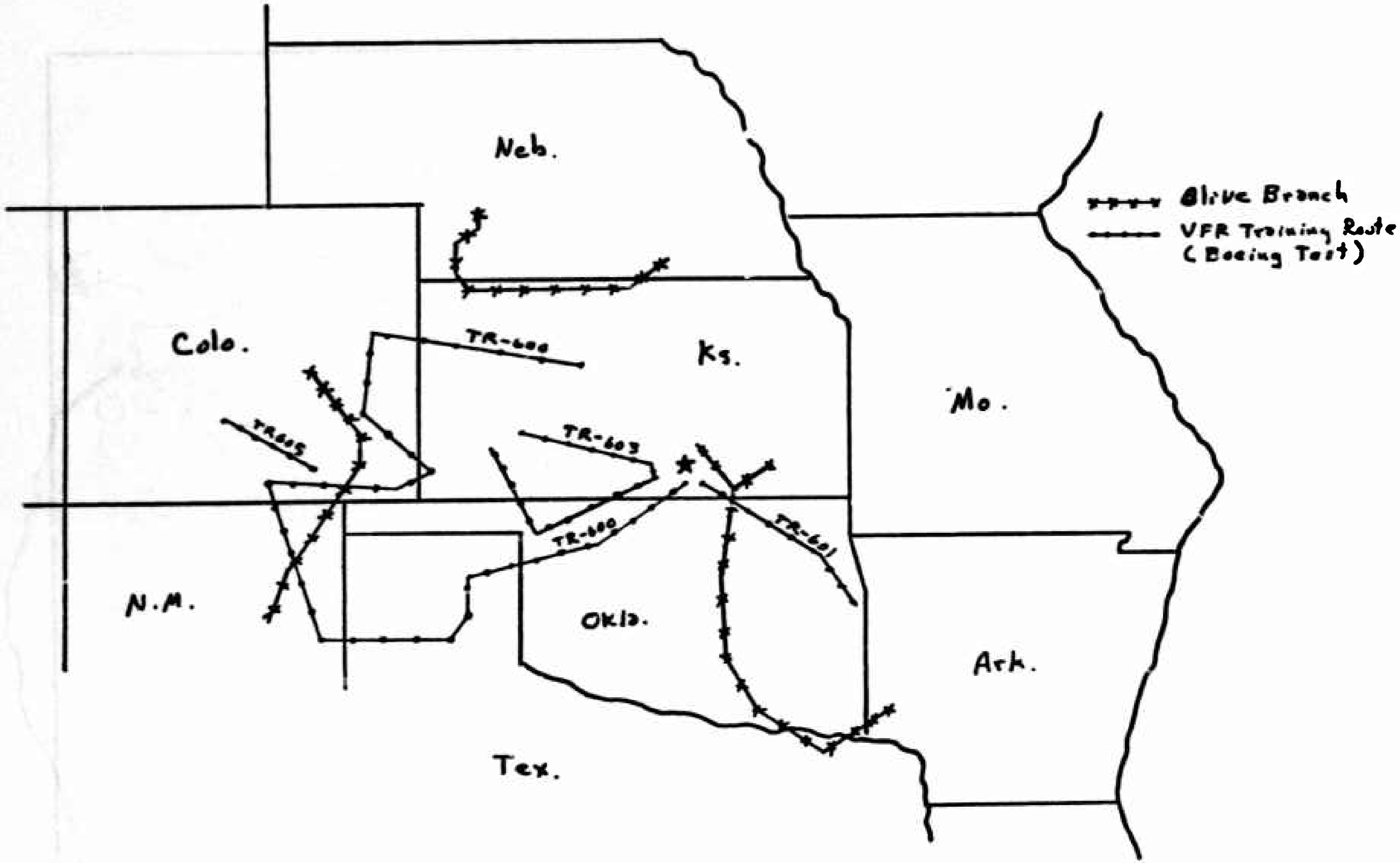


Figure 26. Low-Level Routes - Wichita Area

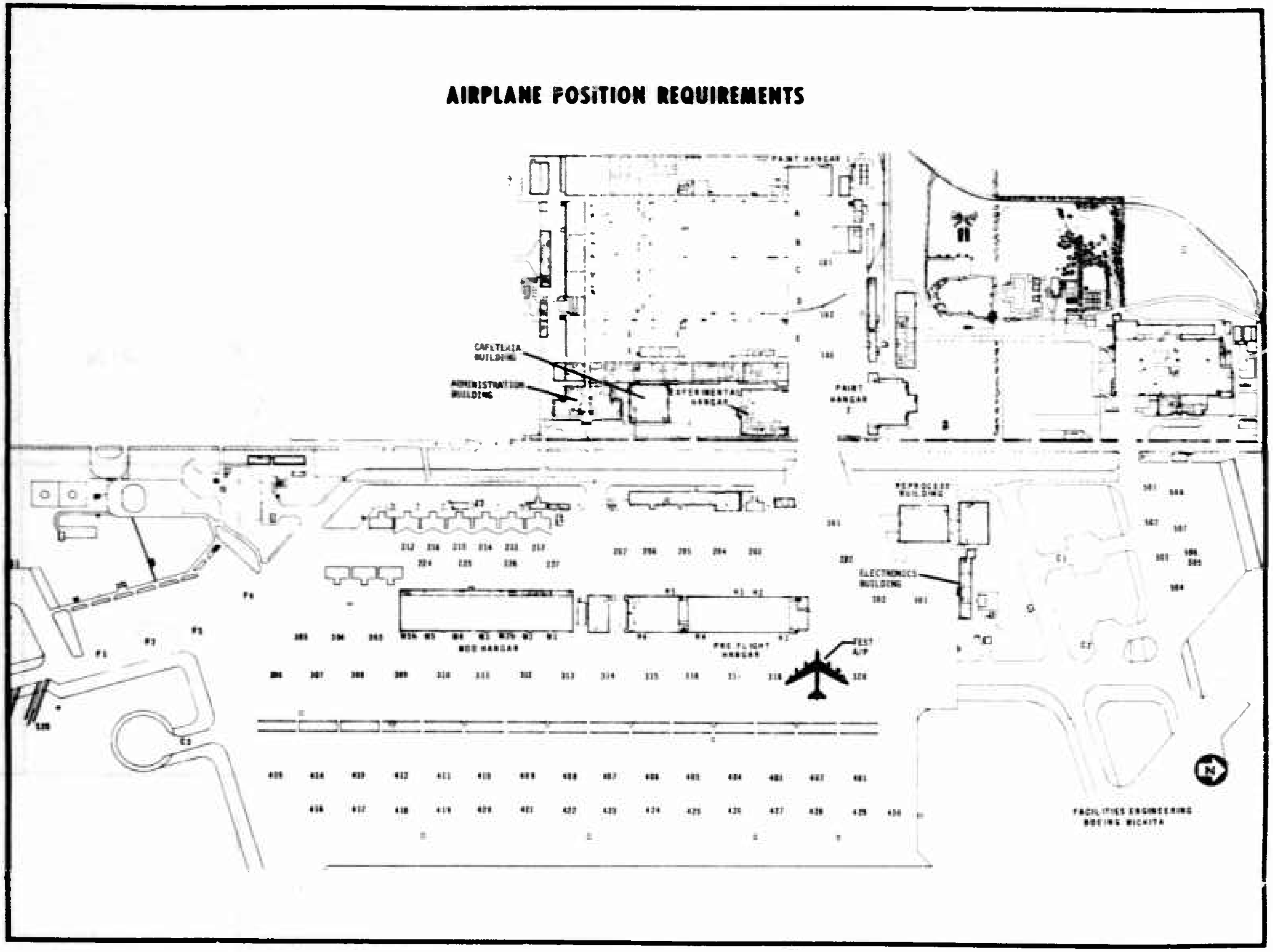


Figure 27. Boeing Plant Layout

2. CREW REQUIREMENTS

The flight crew consists of the command pilot (Boeing), second pilot (Boeing), navigator (Boeing), test system operator (Westinghouse), and test engineer (Westinghouse). The test system operator occupies the station normally occupied by the radar-navigator on the B-52 aircraft, and the test engineer occupies the station normally occupied by the gunner in the B-52G/H aircraft.

A sixth position is available for an observer at the station normally occupied by the EWO in B-52G/H aircraft.

The Boeing flight personnel will be current on all applicable civil and military requirements for operation of the test aircraft.

All Westinghouse and other non-military personnel will be classified as observers and will be required to have current the following to participate in the test flights:

1) A current Class II physical issued by an FAA certified flight physician.

2) A current military altitude chamber indoctrination.

3) Training in operation of B-52G ejection systems and emergency egress procedures.

Material relative to the latter requirement and training will be provided by Boeing Flight Operations personnel. Training in use and operation of personal flight equipment and flight station utilities will also be provided by Boeing.

Personal flight equipment consisting of parachute, helment, oxygen equipment, and flight suit will be provided by Boeing.

3. DATA HANDLING

Facilities and procedures are available at Boeing-Wichita for handling the various types of data obtained during a flight test program. A typical flight test data flow chart is shown in Figure 28.

Voice tape recordings will be made of all flight crew comments made during the course of a flight. Recordings will also be made of all communication between the test airplane and the ground. A file of these recordings is maintained for each test airplane and is available for review of any significant event occurring during a flight.

Manual notes recorded by members of the flight crew and radio room notes will become part of the Plans, Conferences and Data Document (PC&D). This document provides an organized source of crew comments and data to supplement tape recordings.

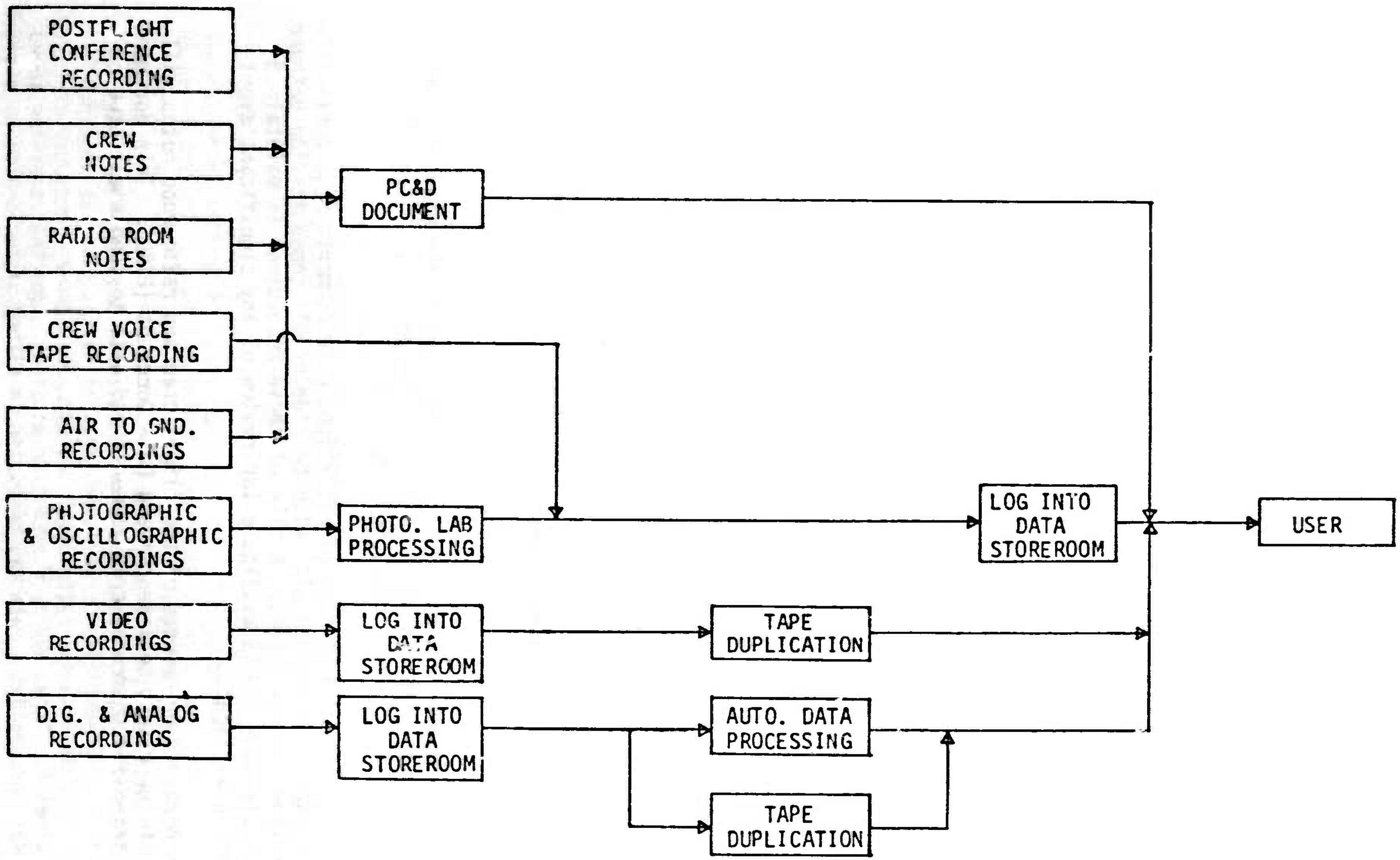


Figure 28. Typical Flight Test Data Flow

Photographic and oscillographic records obtained during flight will be filed and maintained in the Data Reduction Storeroom. Data recorded on magnetic tape (digital, analog and video) will be logged into the data storeroom. Tapes in the data storeroom will be available to the ground station for data processing or for tape reproduction. Data in all forms will be available to each organization requiring its use.

Boeing will provide quick-look data processing of the aircraft recorded data. The processing will be accomplished on the two Westinghouse PCM systems (one standard, one non-standard), the vibration signals, and the B-52 PCM and FM systems. Quick-look results will be available for on-site evaluation and for editing. The format of the quick-look data will be general lists and/or plots, event lists, and analog stripouts.

Duplicate copies of aircraft recorded tapes and associated data and editing and processing instructions will be provided to Westinghouse for further processing. Processed data returned to Wichita for review will be logged through the data center.

Boeing will process environmental and B-52 peculiar data.

4. TELEMETRY AND DATA PROCESSING STATION

Ground Station features and processes were reviewed and demonstrated for Westinghouse so that the EAR test plan could make appropriate use of the facilities. The Telemetry and Data Processing Station is equipped to process PCM, FM, and unique Serial/Digital data from both magnetic tapes, which have been recorded on the test vehicle, and from telemetry received in real time from the test vehicle.

1435-1535 MHz(L) band or 2200-2300 MHz(S) band telemetry signals are received by a 10-foot parabolic reflector which is part of an auto-track telemetry system capable of receiving data from an air vehicle at distances up to 200 miles from the Wichita facility. The signals are demodulated by a Microdyne receiver and routed through an input selector panel to appropriate processing equipment.

Magnetic tapes which have been recorded on a test vehicle may be reproduced on a 28-track wide band recorder, a 14-track wide band recorder, or a 14-track medium band recorder. Recorder speeds from 1-7/8 to 120 IPS are available with limited capability at 15/16 and 240 IPS.

Data are normally recorded as Narrow Band FM (IRIG Bands 2 through 18 - Proportional $\pm 7.5\%$; Bands 3B through 21B - Constant Bandwidth ± 4 KHz/ Bands 7C through 35C - Constant Bandwidth ± 8 KHz), Wide Band FM (450 KHz $\pm 30\%$), Single Carrier FM (3.375 to 108 KHz $\pm 40\%$), and Serial PCM. An EMR Discriminator System is used to process the FM data and PCM data are routed to an EMR or Stellarmetrics System for decommutation. NRZ, Bi-Phase, and Delay Modulation codes are available for use. An additional capability for reducing data at faster speed is provided by an Astrodata FM discriminator system equipped with IRIG center frequency channels multiplied times 8. Tapes may be played back at eight times the recording speed for faster data processing. Data may be routed to either of two analog to digital converters for input to the computers. Twenty-four digital to analog converters (DAC's) are available in the EMR PCM system for output data and twenty DAC's are similarly available in the Stellarmetrics PCM System. A Boeing-designed Serial/Digital Processor built for the Air Launched Cruise Missile (ALCM) Program is available.

Both IRIG-B and Eglin 17-bit time codes may be used through two Astrodata and one Datum Time Code Translators. IRIG-A time codes may be decoded by utilizing additional plug-in modules.

Data Formatting and display can be accomplished through two paths, allowing simultaneous processing of two data streams. Analog outputs may be displayed on two 8-Channel Brush Strip Chart Recorders, an 8-Channel Beckman Chart Recorder, and/or a Honeywell 18-Channel oscillograph.

Data that are recorded on airborne analog tape recorders must be pre-processed to convert it to computer compatible format. This "formatting" process takes the output data from the PCM frame synchronizer, analog-to-digital converter, or special detection equipment and writes the data onto computer compatible digital tapes. This task is accomplished on a PDP-11/20 or PDP-11/45 minicomputer. The data on the digital tapes are in binary format in units often referred to as "counts".

The computer compatible tapes may then be processed through the minicomputer or an IBM 370 computer to perform various computational functions. The first task is to convert the data to engineering units in scale/calibrate routines. The task is to apply instrument calibration data or scale factors to the formatted data in counts with the result being data in engineering units (i.e., knots, psi, feet, degrees, etc.). The engineering unit tape may then be used as input to general purpose list and/or plot programs and data analysis programs.

Data analysis programs analyze the data from a single input source or from multiple input sources such as between navigation system data and ground radar tracking data. They may perform such functions as digitial filtering, aircraft performance, power spectral density, and navigation systems analysis, to name but a few examples. The output of the data analysis programs may be on digital tapes or disks, on-line composite lists, general time history lists of data grouped for easy analysis, and/or time history and/or variable vs. variable plots. The engineering unit tapes and the analysis program tapes may be retained for long-term storage or re-used as required.

Photographic data may be used as input to analysis programs to allow for input of special gage or dial readings or strike camera readings. Gage or dial readings are manually read from photopanel film and the results input to such programs as engine performance, airspeed/altitude calibration programs, and others of similar nature.

Strike camera film frames normally contain images of known and surveyed points. By digitizing the X-Y film coordinates of each surveyed point and correlating these readings with the geometric coordinates and altitude of the point, the aircraft position data may be determined. These position data are then input to the navigation analysis program, corrected for aircraft roll and pitch angles, and are then used in the program to determine position errors of the aircraft. The results of these computations are then handled the same as other analysis program output.

Data turn-around times are dependent upon the length of the flight or mission, the amount of data to be processed, the number of interfaces to be processed, and other factors. In general, quick-look data comprised of stripouts and limited lists, may be available from within a few hours to overnight. Computer compatible tapes and scale/calibrate data may be available twenty-four to seventy-two hours after receipt of detailed data requests. Analysis program outputs are dependent on the number of input data sources and the availability of such items as ground radar tracking data. These schedules should only be used as indicators of what to expect because they may be shortened or lengthened depending on specific contract requirements for data.

5. AIRPLANE MAINTENANCE

Maintenance functions performed on the test airplane will include:

1) Receive airplane and accomplish a receiving inspection.

2) Maintain aircraft and historical records and forms in accordance with applicable maintenance Technical Orders.

3) Accomplish normal maintenance during modification, checkout, test and redelivery periods. This will consist of preventative maintenance and compliance with all applicable technical orders as well as correction of all flight squawks.

4) Accomplish airplane preflights, postflights, fueling and support EAR maintenance, ground testing, flight testing and airplane re-delivery.

5) Process, order T.O. kits and parts and accomplish organizational and intermediate level TCTO's. (Depot level TCTO's will not be accomplished unless authorized by the Air Force.)

The Ground Operations Engineer will schedule and control all test oriented work accomplished on the test airplane. This provides a focal point to ensure a smooth work flow and expeditious accomplishment of the test program.

6. EAR MAINTENANCE/SUPPORT FACILITIES

The EAR system will be maintained by Westinghouse and the SPN/GEANS system by Honeywell. It is anticipated that Boeing-Wichita will maintain the GFP APN-194 radar altimeter equipment. Laboratory maintenance facilities will be provided for Westinghouse and Honeywell in the Electronics Building which is approximately 200 yards from the B-52 parking area. Transportation of equipment between the test airplane and the Electronics Building can be provided by Boeing. Work benches, desks and chairs will be provided, as well as standard utilities including: telephone, lights, heating, cooling, electrical power (60 and 400 Hz, AC and 28 VDC) and compressed air. Laboratory technician support will be provided by Boeing, as required by contract. Certification and calibration of Westinghouse and Honeywell test equipment can be provided by the Boeing Certification and Calibration Laboratories.

Support of Westinghouse and Honeywell personnel performing maintenance on board the test airplane will be provided by Boeing. This support will include: operation of ground support equipment (such as ground cooling and air conditioning, electrical power and lighting), operators for airplane systems, other than the test system, and mechanical and electrical shop support as required.

In addition to the facilities mentioned above, Boeing-Wichita will provide the following facilities/space, services, supplies and equipment for the EAR Flight Test Program:

Facilities/Space

- Office Area
- Conference Rooms
- Test Data Center
- Document/Drawing/Secret Control Area
- Reproduction Area
- Parking

Services

- Industrial Safety
- Security
- Fire Protection
- Emergency Medical
- Mail
- Reproduction
- Audio Visual
- Telephones/Communications
- Plant Services
- Photographic
- Repair of general purpose and other test equipment
- An Instrumentation, Maintenance and Calibration Lab for the instrumentation on the B-52 shall be provided and maintained

Supplies

- Office
- Petroleum, Oil and Lubricants (POL) (GFP)

7. FLIGHT AND GROUND SAFETY

Boeing's Flight Operations procedures are documented in D3-2918, "Boeing-Wichita Flight Test Procedures - (Military Contracts)". These procedures show compliance with applicable Air Force regulations and safety directives. These procedures are reviewed and approved frequently by the resident Government Flight Representative of Detachment 21, AFLC, at Wichita. This document covers crew training qualifications and flight management.

Any in-flight emergencies for the B-52 will be supported by a safety team composed of a current pilot and technical representatives of the required design or staff areas.

The Kansas City Air Traffic Control Center is connected directly to the Boeing Control Tower and radio rooms by telephone lines. By special arrangement, an Air Traffic Controller communicates with the Boeing flight crew through the Boeing transmitters on the same frequency used by test operations engineers in the radio room. This very close coordination provides for expedited traffic clearances, constant traffic surveillance, minimum interruptions of test conditions and increased safety while in the Boeing flight test areas.

The test airplane will be operated in accordance with the safety procedures described in the B-52G Flight Manual, T.O. 1B-52G-1.

Certification of personnel for critical handling and ground operations is documented in D3-2741, Certification Document. These operations include: engine starting and operation, braking procedures, egress system maintenance, towing operations and other requirements of AFR 127-101, Ground Accident Prevention Handbook. All industrial safety considerations are directed by the Manager of Industrial Safety and Plant Hygiene.

8. TEST RANGE COORDINATION

Use of national ranges requires that the universal documentation system be followed. The intended range user (EAR) starts the documentation system by preparing a Program Introduction (PI) Document. The range responds with a Statement of Capability (SC) Document. The user responds with a Program Requirements Document (PRD) which provides a complete and detailed statement of the range users requirements. The PRD then provides the basis for the Program Support Plan (PSP) prepared by the range. The Operations Requirements (OR) Document is prepared to place requirements of the range for a specific test. The range then prepares the Operations Directive to provide scheduling of range time and support for the actual test.

It is anticipated that coordination will be required with CIGTIF for use of the CIRIS test range. Coordination with Edwards AFB for use of the RADFAG range as well as other ranges may also be accomplished. It will probably be most convenient and more efficient to establish a corner reflector range in the near vicinity of Wichita. This range could be used any time without prior coordination and scheduling. Reflector arrangement would be flexible and probably could be changed during a flight. The convenience of this range would be especially desirable during early testing when flight scheduling is more uncertain. A candidate corner reflector range layout is shown in Figure 29.

9. EMI TESTS

A minimum safety of flight EMI/EMC test will be accomplished on the B-52G/EAR system prior to first flight. Compatibility of the essential B-52 electronic systems to be used inflight with the EAR system will be accomplished. Testing will be accomplished in accordance with MIL-E-6051A and will be performed as outlined in the EMC Test Plan.

These tests will be limited to a determination of whether or not possible electromagnetic interference caused by the new installations will be great enough to cause unsafe flight conditions and inhibit EAR development testing. B-52 systems will be operated in their most sensitive modes and monitored for interference while the EAR system is exercised in its normal modes.

The airplane's 115 volt, three phase, 400 cycle AC and 28 volt DC electrical power will be monitored to assure that proper voltages are maintained and to observe any excessive transients. Interferences that are found will be examined to determine coupling paths and levels of interference so that appropriate remedial actions can be recommended.

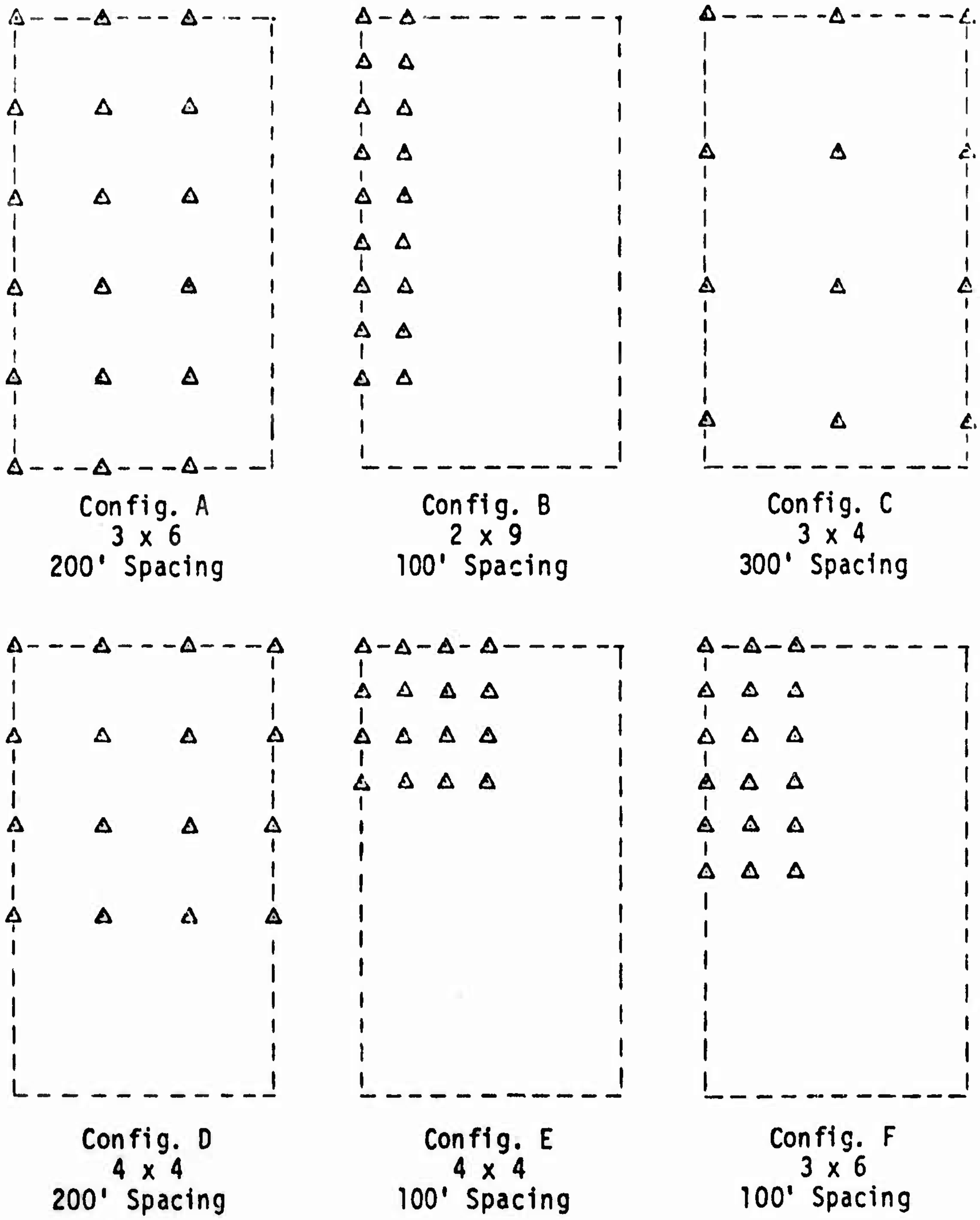
Boeing-Wichita can provide the necessary frequency coordination required with the appropriate Government agencies for the EMI as well as other EAR testing.

10. B-52 FLIGHT PLANNING

EAR flight planning will be accomplished in such a manner that the maximum results can be obtained from a given flight.

The numerous low level routes convenient to Wichita will be utilized as far as possible to achieve the objectives of this program. These routes provide nearly all types of terrain for radar evaluation. There are high mountainous, flat plains, rolling plains, wooded and nonwooded terrains all within easy flying distance of Wichita. All of these routes are sparsely populated and suitable for overflight. A wide variety of manmade features are also available, such as isolated radio and television towers, grain elevators (both steel and concrete), etc.

Mission profile plans will be prepared prior to each flight. This will provide for the optimum use of time and fuel. Airplane gross weight and center of gravity will be monitored and plotted as a time history in the ground radio control room during flight.



This range would be 600 by 1,000 feet. Six different reflector arrangement/spacing configurations could be provided using a maximum of 18 reflectors located on 47 surveyed points.

Figure 29. Candidate Corner Reflector Range

A chart showing the sequence of events required to accomplish a typical flight is shown in Figure 30.

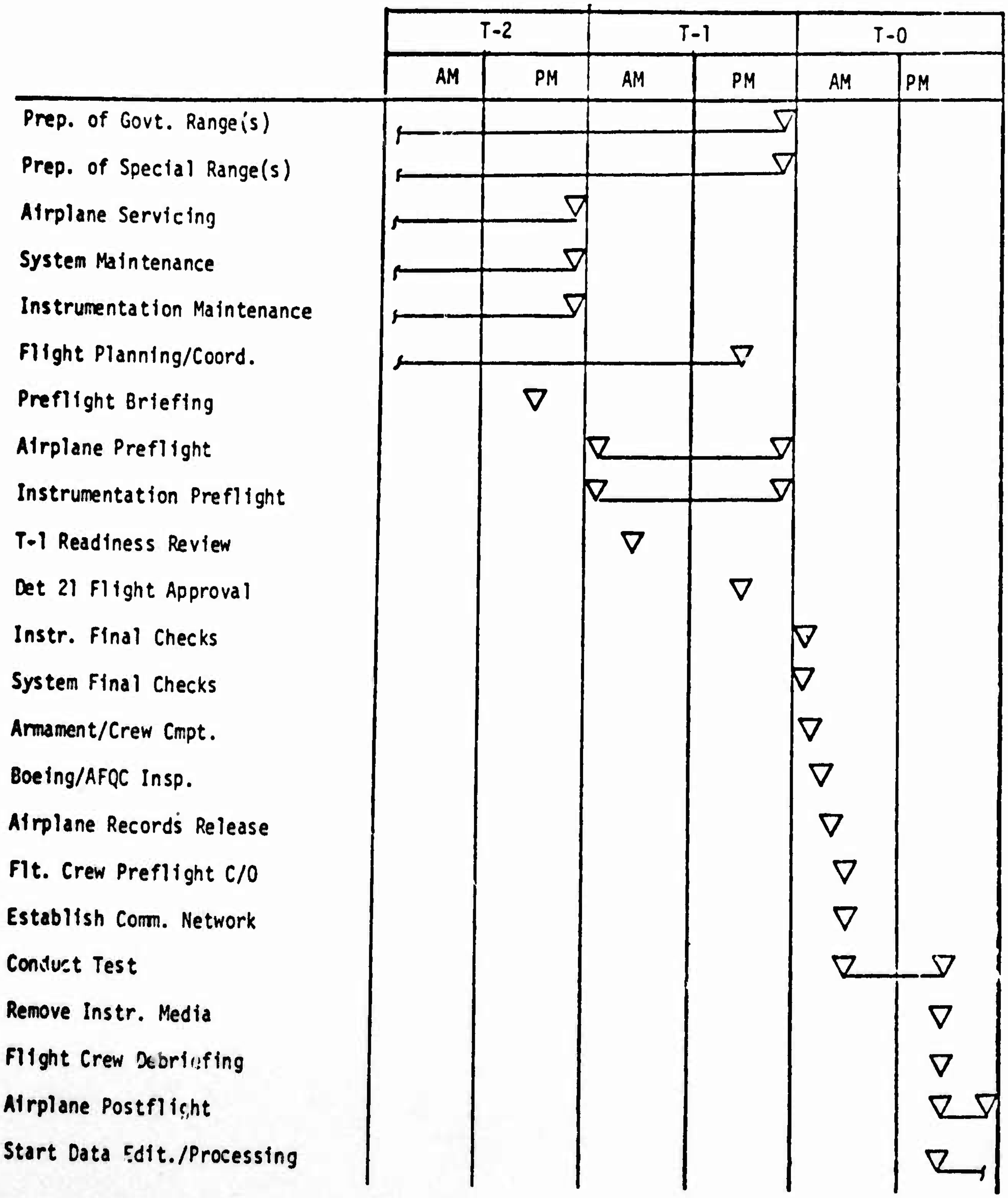


Figure 30. Sequence of Events - Typical Flight Test

SECTION VIII

FLIGHT TEST PROGRAM RESPONSIBILITY

During the study phase, numerous contractor responsibilities were established for modifying the airplane, flight test planning, instrumentation, ground tests and flight tests. The following sections describe the contractor tasks during the Flight Test program.

1. TEST AIRPLANE MODIFICATION

a. Group A Provisions

Boeing - Design, fabricate and install the Group "A" provisions for the test bed airplane. Design, fabricate and perform the installation alignment of the antenna/IMU mount to the airplane centerline. Provide the wiring, coax and waveguide for the airplane modification except for the waveguide between the antenna and microwave unit.

Westinghouse - Support modification design to assure proper physical interface. Provide cabling information and special connectors required for airplane installation. Provide waveguide section between antenna and microwave unit. Support antenna/IMU installation alignment.

b. Group B Installation

Boeing - Install Test Article Group B equipment on the airplane.

Westinghouse - Provide Test Article Group B equipment and support installation on the airplane.

c. Mechanical/Electrical Interface

Boeing - Maintain the Interface Control Drawing (ICD) controlling the Group A provisions installed on the airplane, the interface between the B-52 and Test Article and the B-52 peculiar instrumentation.

Westinghouse - Provide installation requirements and approve the ICD developed by Wichita. Maintain the ICD for the interface within the Test Article; e.g., between EAR, SPN/GEANS, Mock ACUC, IAU, Controls and Displays and associated instrumentation.

Provide wire lists for interface of test article components. Update test article/airplane interface requirements.

d. Instrumentation

Boeing - Provide airplane peculiar and environmental instrumentation. Document instrumentation interface tie-in. Provide movie cameras, Ampex VR 3000 video recorder, 115 VAC 400 Hz, 115 VAC 60 cycle and 28V DC power and IRIG "B" time code generator. Install, checkout and provide initial certification of instrumentation readiness.

Westinghouse - Provide instrumentation peculiar to the test article. Provide connectors for the EAR instrumentation wiring. Support installation and checkout of the instrumentation on the airplane.

e. Group B Equipment

Boeing - Provide airplane parameter sensors and interface electronics for supplying the interface to the Test Article and Test Article data to the airplane navigator. Provide modifications required for displaying EAR TF/TA video on the EVS monitors.

Westinghouse - Provide the EAR, motion sensor, surrogate computer complex and ancillary equipment comprising the Test Article.

f. Reference System Installation

Boeing - Provide installation provisions and install in the airplane.

Westinghouse - Provide requirements for reference system data.

Air Force - Provide installation requirements and wiring information.

g. Airplane Demodification

Boeing - Restore airplane upon AFAL direction.

2. SYSTEM TESTING

a. Radome Tests

Boeing - Modify Antenna Test Range. Conduct antenna/radome tests and document results.

Westinghouse - Provide the antenna and special test equipment required for handling and steering the antenna beam. Support the lab testing.

b. EAR Lab Tests

Boeing - Provide the antenna/IMU mount. The mount will be returned for installation on the airplane for the Flight Test Program.

Provide a set of flexible RF coax cables. A second set of cables will be provided for airplane installation.

c. Operations and Maintenance

Boeing - Ferry airplane to Wichita and conduct a receiving inspection inventory. Provide the pilot, copilot and navigator during the Flight Test Program. Provide airplane maintenance. Maintain and certify instrumentation during the Flight Test. Support special testing of the Test Article.

Westinghouse - Provide radar operator and test engineer to operate the EAR peculiar instrumentation. Maintain the Test Article and support instrumentation maintenance and calibration. Conduct special tests on the Test Article.

d. Test Direction

Boeing - Provide flight and airplane operations direction including airplane scheduling, release and test range coordination. Prepare "Plan of Test" documentation for each flight detailing test conditions to be flown.

Westinghouse - Provide overall test planning and objective. Determine test conditions and objectives for each flight.

The Westinghouse test plan will include an experiment design indicating the various conditions and probable sample size, etc. The flight by flight "Plan of Test" will incorporate those appropriate tests for each flight based on Westinghouse objectives for that flight.

e. Ground Tests

Boeing - Prepare test procedures for electrical power quality, ground vibration, environmental system performance and EMC tests. Conduct ground tests and support Test Article checkout.

Westinghouse - Prepare test procedures and conduct checkout of the Test Article. Support EMC tests and assist in evaluating conditions affecting EAR tests.

f. Flight Tests

Boeing - Conduct checkout flight(s) after airplane modificat Evaluate environmental and EMI data acquired during checkout flight.

Provide airplane operations and operate airplane, environmental and reference system instrumentation during data flights.

Westinghouse - Conduct Test Article checkout on modification acceptance flight(s). Operate Test Article and EAR peculiar instrumentation during data flights.

g. Data Reduction and Analysis

Boeing - Perform data reduction of Flight Test data. Perform data handling and management functions to provide data in formats required. Provide airplane performance evaluations and support "quick look" analysis activities.

Westinghouse - Provide data reduction requirements for Test Article instrumentation. Conduct "quick look" and in-depth system analysis.

SECTION IX

ANALYSES AND TECHNICAL DATA

1. ANTENNA/RADOME INTERACTION

An antenna/radome analysis was performed on the interactions between the EAR antenna and the B-52 Nose Radome for the selected antenna location.

The analysis performed provides the computed boresight errors created by the B-52 Phase VI nose radome for both vertical and horizontal polarization for a multiplicity of look angles within the required window area. This window area is defined as that area required for the EAR antenna to operate within a 60-degree half angle cone about the antenna axis. For the purposes of this analysis, the forward face of the EAR antenna was located at Buttock line 0.0 and Water line 165.8. In addition, the analysis provides computed boresight errors as a function of frequency over the EAR system frequency band for four representative look angles within the RF window, off axis angle slope curves for some 19 look angles within the RF window, and nose radome transmission efficiencies for 19 look angles. Radome boresight errors were calculated for the low, center, and high frequencies of the EAR system, while the off-axis angle slope curves and radome transmission efficiencies were calculated at the center frequency only. The maximum boresight errors calculated were -6.9 milliradians for horizontal polarization and +4.8 milliradians for vertical polarization, both at the center frequency. The minimum transmission efficiency was 71.3 percent at the center frequency. The maximum boresight error delta versus frequency for any one look angle was 0.3 milliradians.

2. ANTENNA/RADOME COMPUTER PROGRAMS

The radome boresight error off-axis angle error slope curves, and radome transmission efficiencies computations were performed utilizing three Boeing developed computer programs. Briefly, two of these programs compute the far field pattern of an antenna, defined by a line of point sources of any amplitude and phase distribution, with the radome characteristics included in one program and deleted in the other. This task is accomplished by generating a near field point source distribution just outside the radome surface where each point is the vector sum of the contributions from each of the antenna aperture distribution points with the insertion loss and phase delay created by the radome included or deleted. It is then a simple matter to generate the far field pattern from this near field distribution. The amplitude and phase distributions of the line of point sources representing the antenna were provided by Westinghouse for both the 0-degree rotation vertical distribution and 0-degree tilt horizontal distribution. Phase adjustments were performed by Boeing to achieve non-zero rotation and tilt angles. The insertion loss and phase delay are determined within the program by applying the Fresnel Equations to a multiple layered media where the complex parameters (permittivity, permeability and dielectric, and magnetic loss tangents) and the thickness of each layer are known. The shape of the B-52 nose radome is defined by 2768 points in three dimensional space which are used along with the location of each antenna aperture point to determine the incidence angle and polarization angle for each ray intersecting the radome. A third computer program was developed to utilize the output of the first two programs to calculate and plot the off-axis angle error slope curves.

3. RADOME BORESIGHT ERROR COMPUTATIONS

The boresight error created by the B-52 nose radome has been computed for 46 look angles in vertical polarization and 16 look angles for horizontal polarization at the center frequency of the EAR system. These errors were calculated using the difference pattern amplitude to obtain the greatest accuracy. The data is tabulated in Tables 17 and 18 and is given in terms of horizontal error and vertical error for each look angle. The errors are given in milliradians with positive numbers indicating the upward direction for vertical errors and the outboard direction for the horizontal errors. The coordinate system utilized to define the look angle is shown in Figure 31.

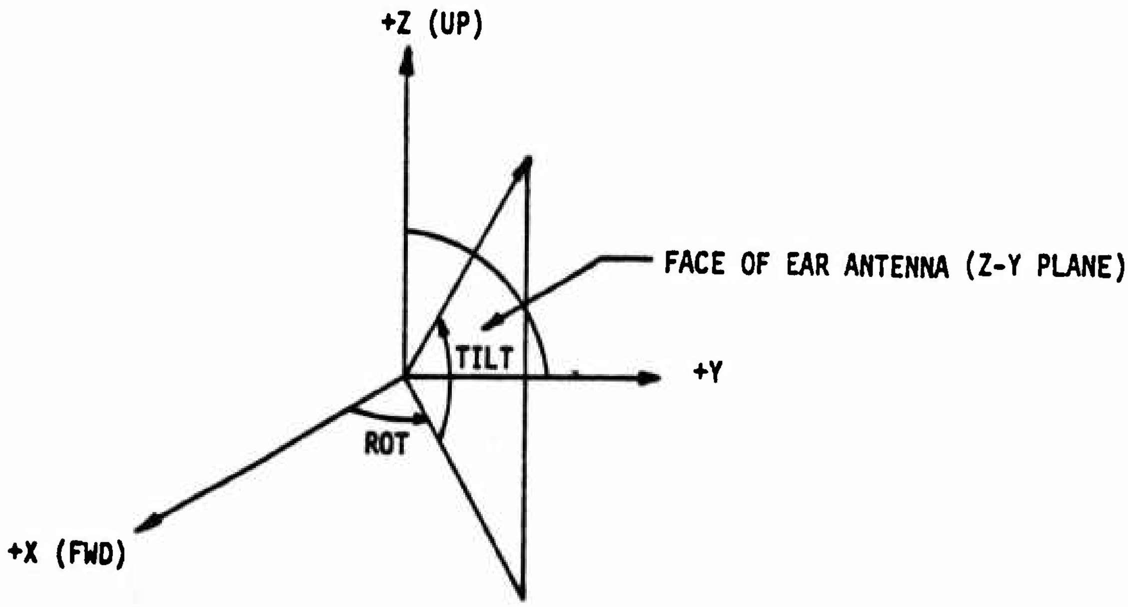


Figure 31. Radome Boresight Coordinate System

4. BORESIGHT ERROR FREQUENCY DEPENDENCE

In addition to the boresight error computations made on the B-52 nose radome at the center frequency of the EAR system, computations of the boresight error were performed at the maximum and minimum frequencies of the EAR system frequency band. The boresight error was computed at these frequencies for four points for both vertical and horizontal polarizations. The look angles chosen exhibited both large and small boresight error and should be representative of the entire window area. These errors are shown in Table 19. These errors again were calculated using the difference pattern antenna distribution for greatest accuracy. The maximum variation indicated across the frequency band of interest is 0.3 milliradians.

TABLE XVII

RADOME BORESIGHT ERRORS (MILLIRADIANS) -
CENTER FREQUENCY

TILT ANGLE	PLANE	ROTATION ANGLE					
		0^0	10^0	20^0	30^0	45^0	60^0
60^0	V	-0.0	--	--	--	--	--
	H	+0.1	---	--	--	--	--
45^0	V	+0.4	-1.9	-2.1	+2.1	--	--
	H	+0.0	+1.6	+2.2	+2.2	--	--
30^0	V	-0.2	-2.5	-1.9	-2.1	-1.0	--
	H	0.0	+4.5	+2.8	+0.7	+0.5	--
20^0	V	-2.1	-1.1	-1.0	-0.8	-0.7	--
	H	0.0	+4.8	+2.6	+0.9	+1.4	
10^0	V	-2.7	-0.6	-0.2	-0.7	-0.8	--
	H	0.0	+2.7	+2.0	+0.6	+0.2	--
0^0	V	-1.5	-0.3	+0.3	-0.5	+0.1	+0.3
	H	0.0	+1.4	+1.3	+1.0	+3.2	+1.4
-10^0	V	-0.8	-0.2	+0.2	-0.7	0.0	--
	H	0.0	+0.1	+1.4	+0.5	-1.2	--
-20^0	V	-0.5	0.0	-0.4	-1.9	-0.1	--
	H	0.0	+0.2	+0.3	+0.5	+1.5	--
-30^0	V	-0.3	-0.1	-2.1	-2.4	-0.3	--
	H	0.0	+0.5	+0.4	-1.1	+0.1	--
-45^0	V	-0.1	-0.8	-3.5	+1.6	--	--
	H	-0.1	+0.4	+0.6	-0.2	--	--
-60^0	V	0.0	--	--	--	--	--
	H	0.0	--	--	--	--	--

VERTICAL POLARIZATION

TABLE XVIII

RADOME BORESIGHT ERRORS (MILLIRADIANS) -
CENTER FREQUENCY

TILT ANGLE	PLANE	ROTATION ANGLE			
		0^0	10^0	20^0	30^0
10^0	V	-6.9	-5.0	-3.6	-2.3
	H	0.0	-0.8	-0.7	-0.4
0^0	V	-4.1	-3.2	-2.9	-2.1
	H	0.0	-1.4	-1.5	-0.8
-10^0	V	-2.4	-2.1	-2.1	-2.0
	H	0.0	-1.3	-1.3	-2.5
-20^0	V	-1.3	-1.3	-1.4	-2.7
	H	0.0	-1.0	-1.4	-0.9

HORIZONTAL POLARIZATION

TABLE XIX

RADOME BORESIGHT ERRORS (MILLIRADIANS) -

FREQUENCY DEPENDENCE

TILT ANGLE	PLANE	FREQ.	ROTATION ANGLE			
			10^{0}		20^{0}	
			VERT. POL.	HORIZ. POL.	VERT. POL.	HORIZ. POL.
0^{0}	VERT.	LO	-0.3	-3.2	+0.3	-3.0
		MID	-0.3	-3.2	+0.3	-2.9
		HI	-0.4	-3.3	+0.1	-2.9
	HORIZ.	LO	+1.4	-1.4	+1.4	-1.6
		MID	+1.4	-1.4	+1.3	-1.5
		HI	+1.4	-1.4	+1.2	-1.4
-10^{0}	VERT.	LO	-0.0	-2.1	+0.4	-2.1
		MID	-0.2	-2.1	+0.2	-2.1
		HI	-0.2	-2.1	+0.1	-2.1
	HORIZ.	LO	+0.0	-1.2	+1.6	-1.4
		MID	+0.1	-1.3	+1.4	-1.3
		HI	+0.2	-1.3	+1.3	-1.3

5. OFF-AXIS ERROR SLOPE COMPUTATION

The off-axis angle function (K) was computed for 19 different look angles within the 60° conical window of the EAR antenna. This angle function is defined by the following equation:

$$K = \frac{|\Sigma| * |\Delta| * \cos(\gamma_\Delta - \gamma_\Sigma)}{\Sigma^2}$$

where λ_{Δ} = Pattern phase angle of the difference pattern

λ_{Σ} = Phase angle of the sum pattern

Σ = Magnitude of the sum pattern

Δ = Magnitude of the difference pattern.

The off-axis angle function was computed for both with radome and without radome (free space). The function was evaluated for an angular variation of plus and minus 2.5° about the designated look angle in both the vertical and horizontal planes.

For the vertical plane curves, the "At ____ Degrees" angle is a theta angle, and the rotation angle is a phi angle. The vertical plane plots compare K vs theta, with the center angle being the "At ____ Degrees" angle. For the horizontal plane plots, the "At ____ Degrees" angle is a phi angle and the tilt angle is a theta angle. The horizontal plane plots compare K vs phi, with the center angle being again the "At ____ Degrees" angle.

The off-axis error slope curves are provided in Figures 32 thru 69.

Transmission efficiency for 19 look angles has been calculated for the B-52 nose radome. The values are provided in terms of percent of power transmitted through the radome and represents contributions from both the vertical and horizontal plane distributions. These transmission efficiencies are shown in Table XX.

TABLE XX

TRANSMISSION EFFICIENCY - CENTER FREQUENCY

TILT ANGLE	ROTATION ANGLE				
	0°	10°	20°	30°	45°
45°	71.3	--	79.0	--	--
30°	--	79.2	--	--	--
20°	82.7	--	82.6	--	85.0
10°	--	84.3	--	84.4	--
0°	89.7	--	86.2	--	85.4
-10°	--	86.5	--	84.9	--
-20°	89.0	--	86.3	--	86.5
-30°	--	86.9	--	--	--
-45°	84.4	--	83.5	--	--

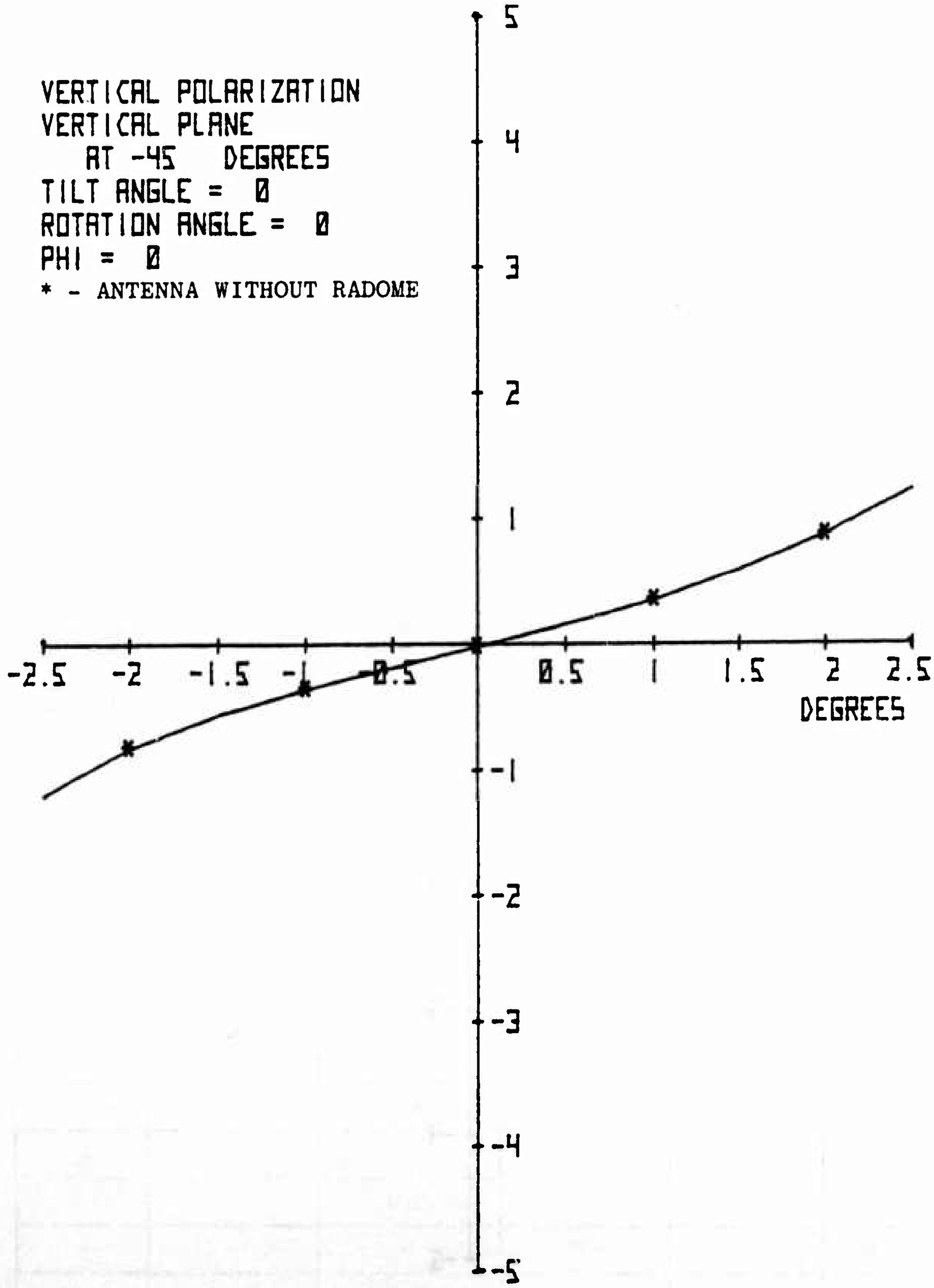


Figure 32. Off-Axis Angle Error Slope

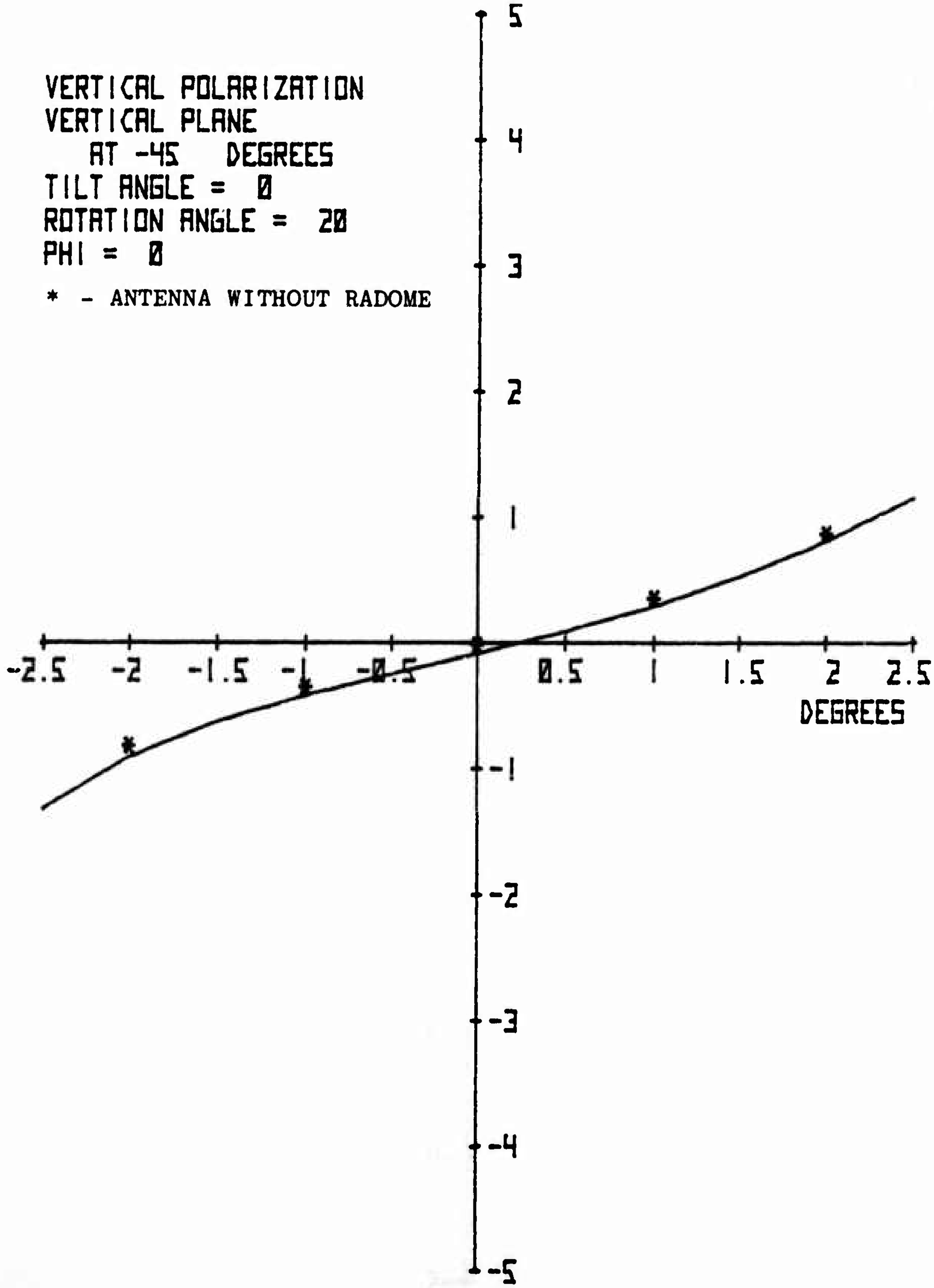


Figure 33. Off-Axis Angle Error Slope

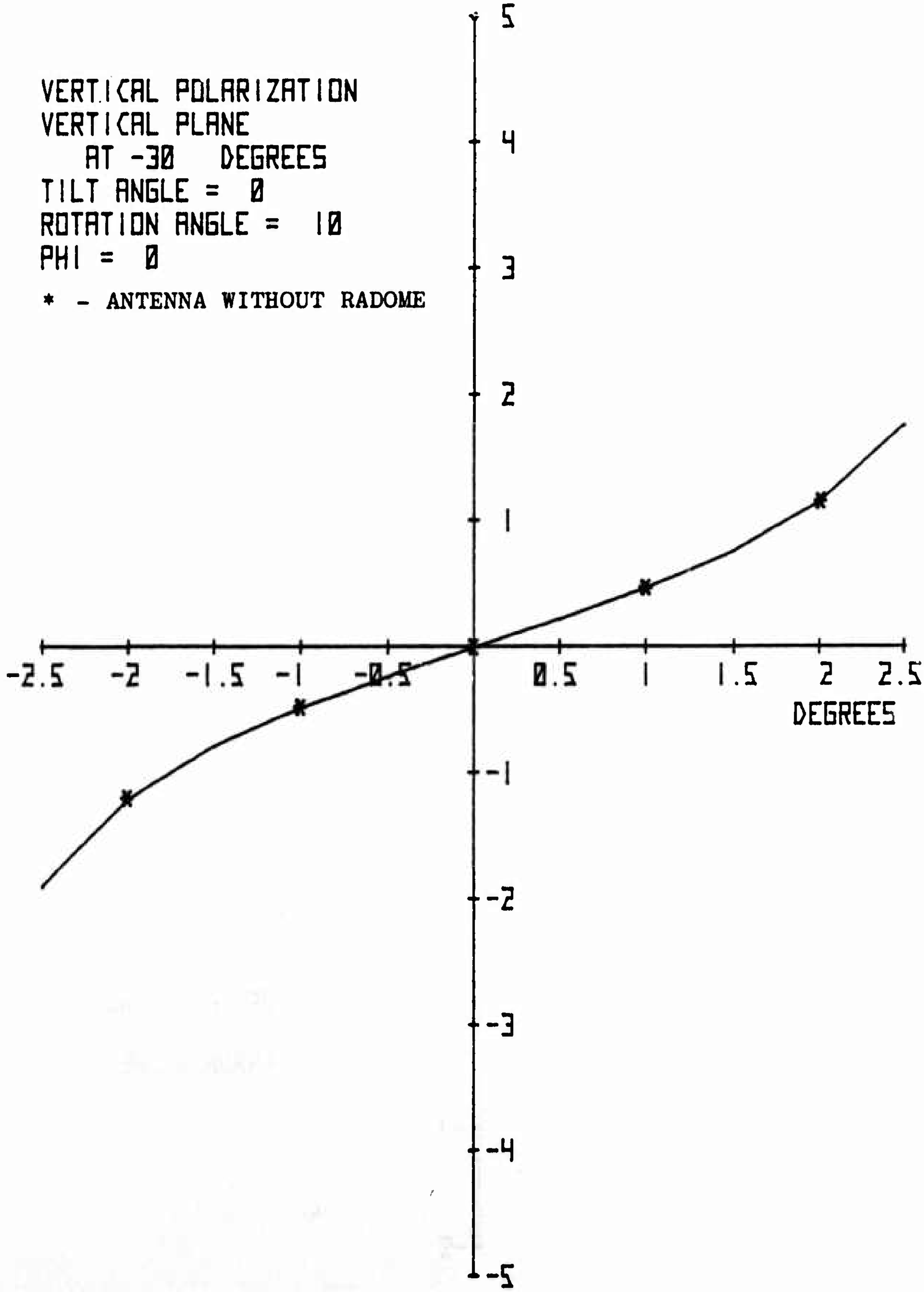


Figure 34. Off-Axis Angle Error Slope

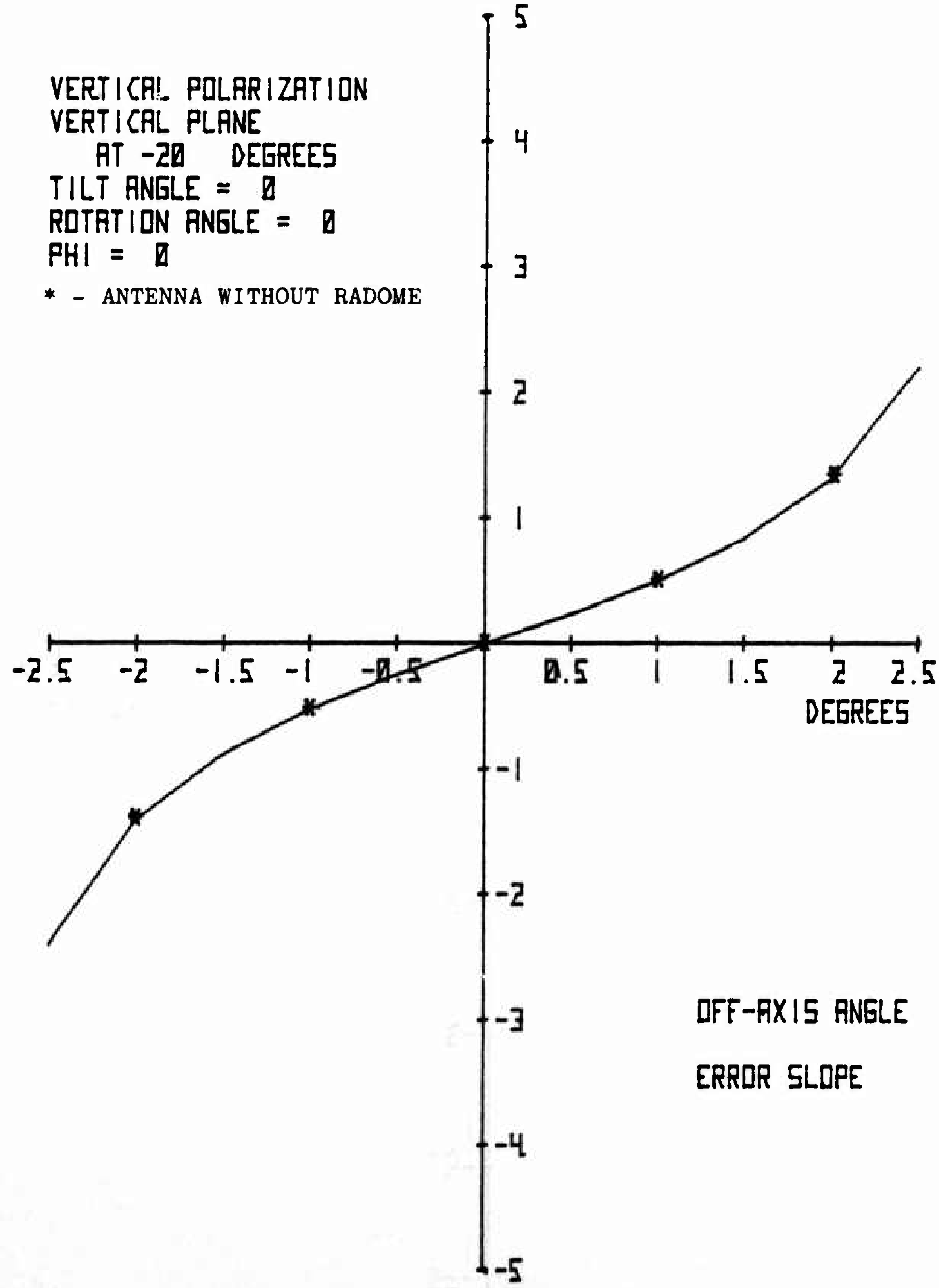


Figure 35. Off-Axis Angle Error Slope

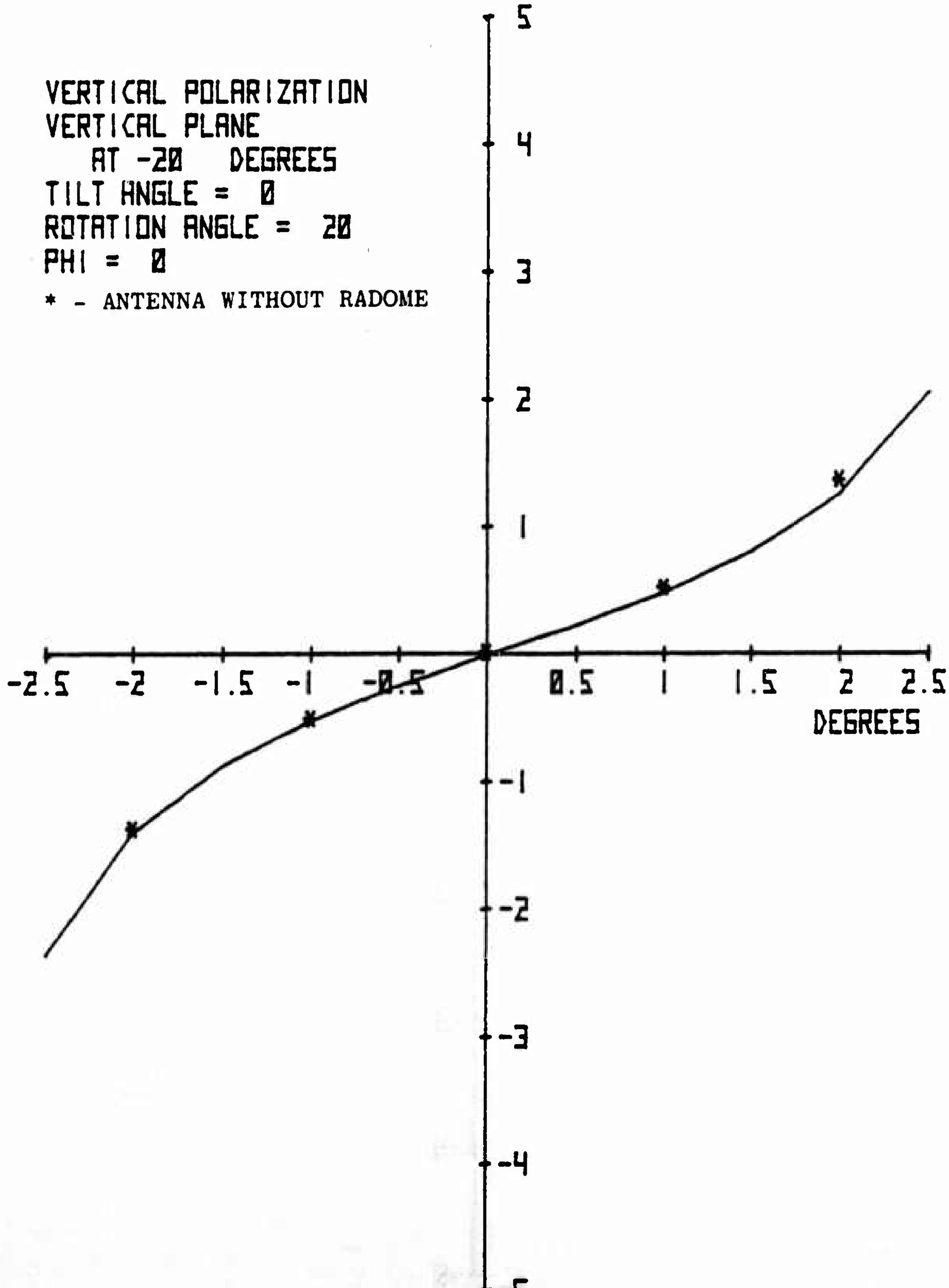


Figure 36. Off-Axis Angle Error Slope

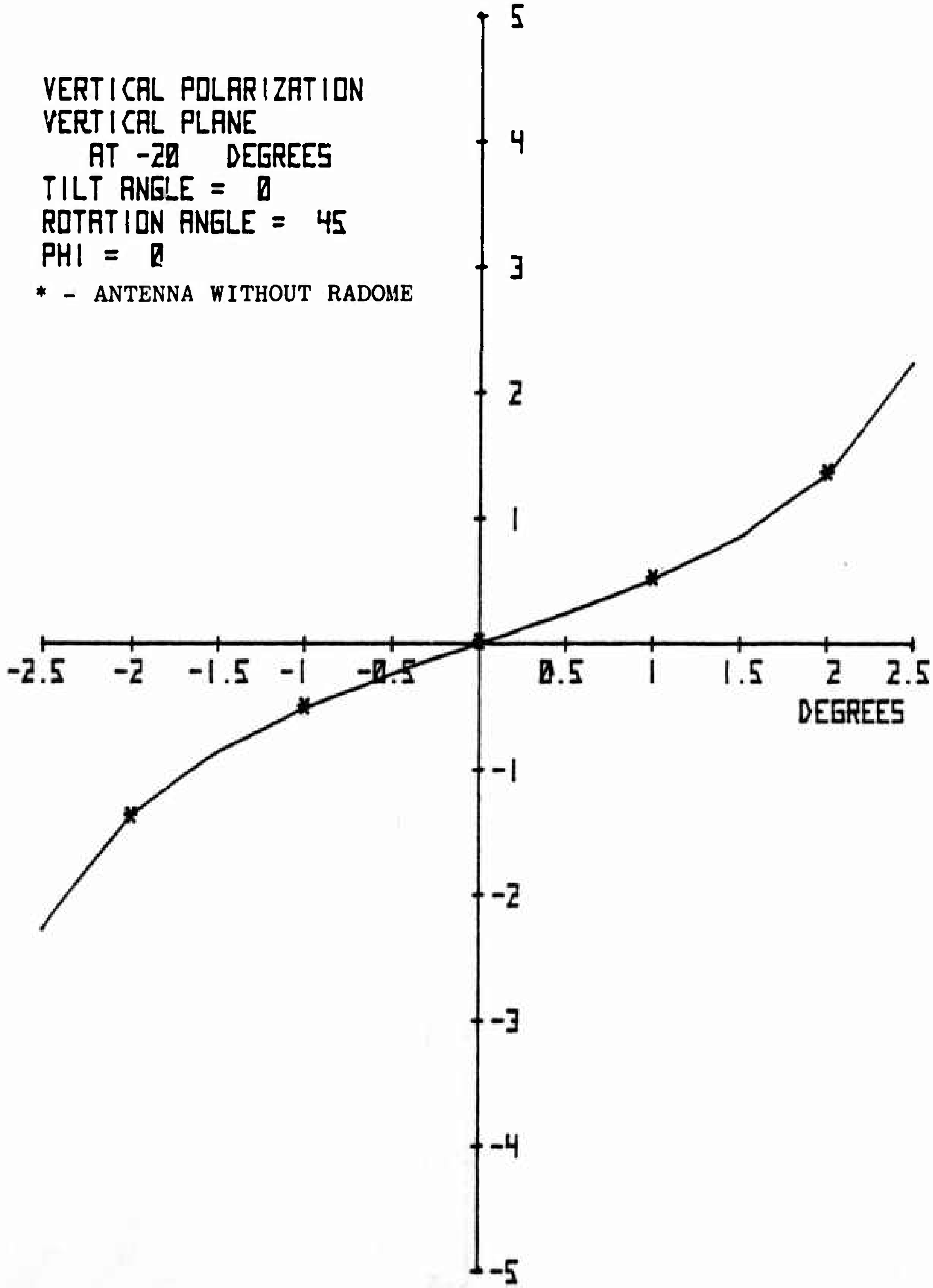


Figure 37. Off-Axis Angle Error Slope

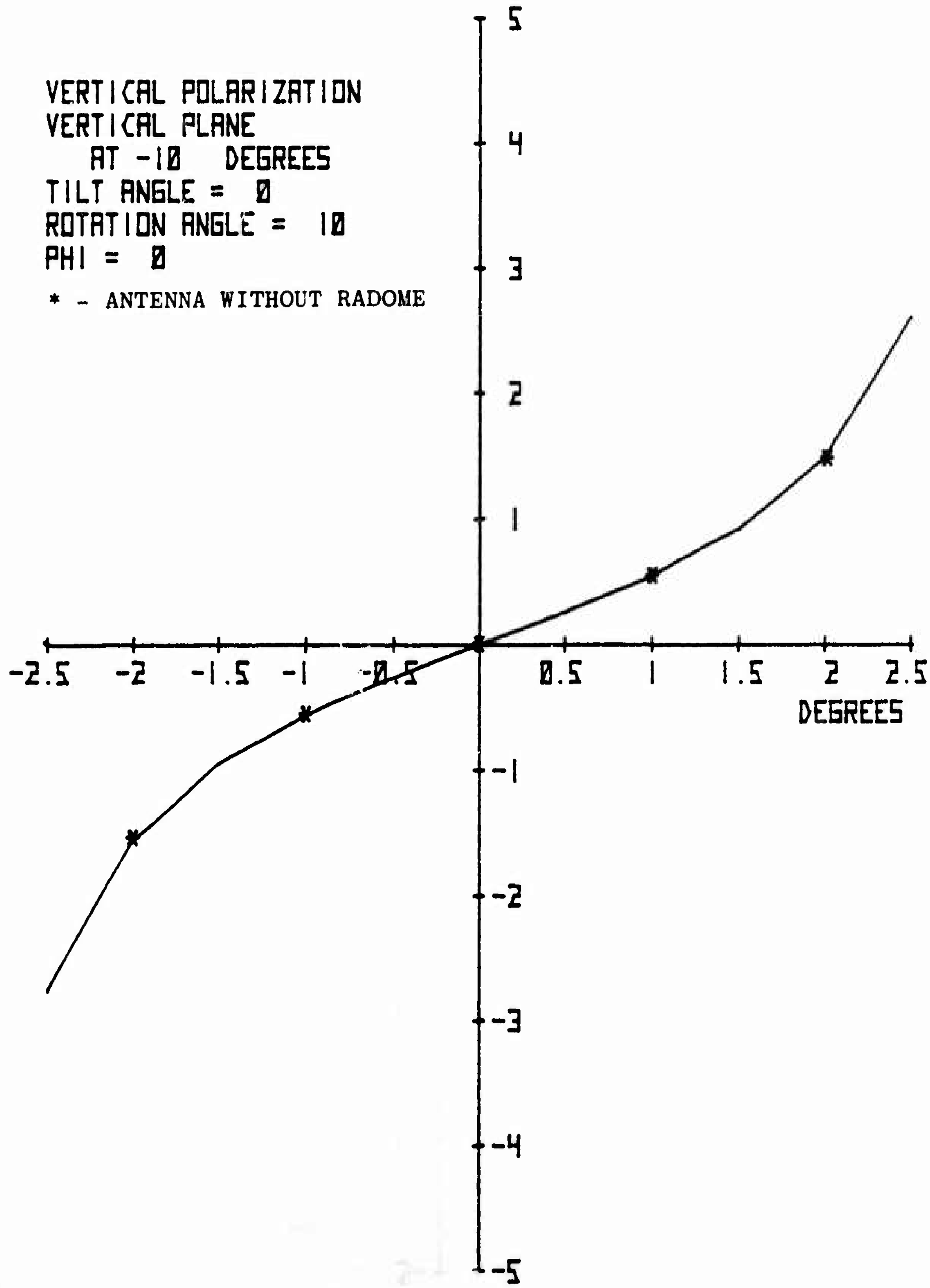


Figure 38. Off-Axis Angle Error Slope

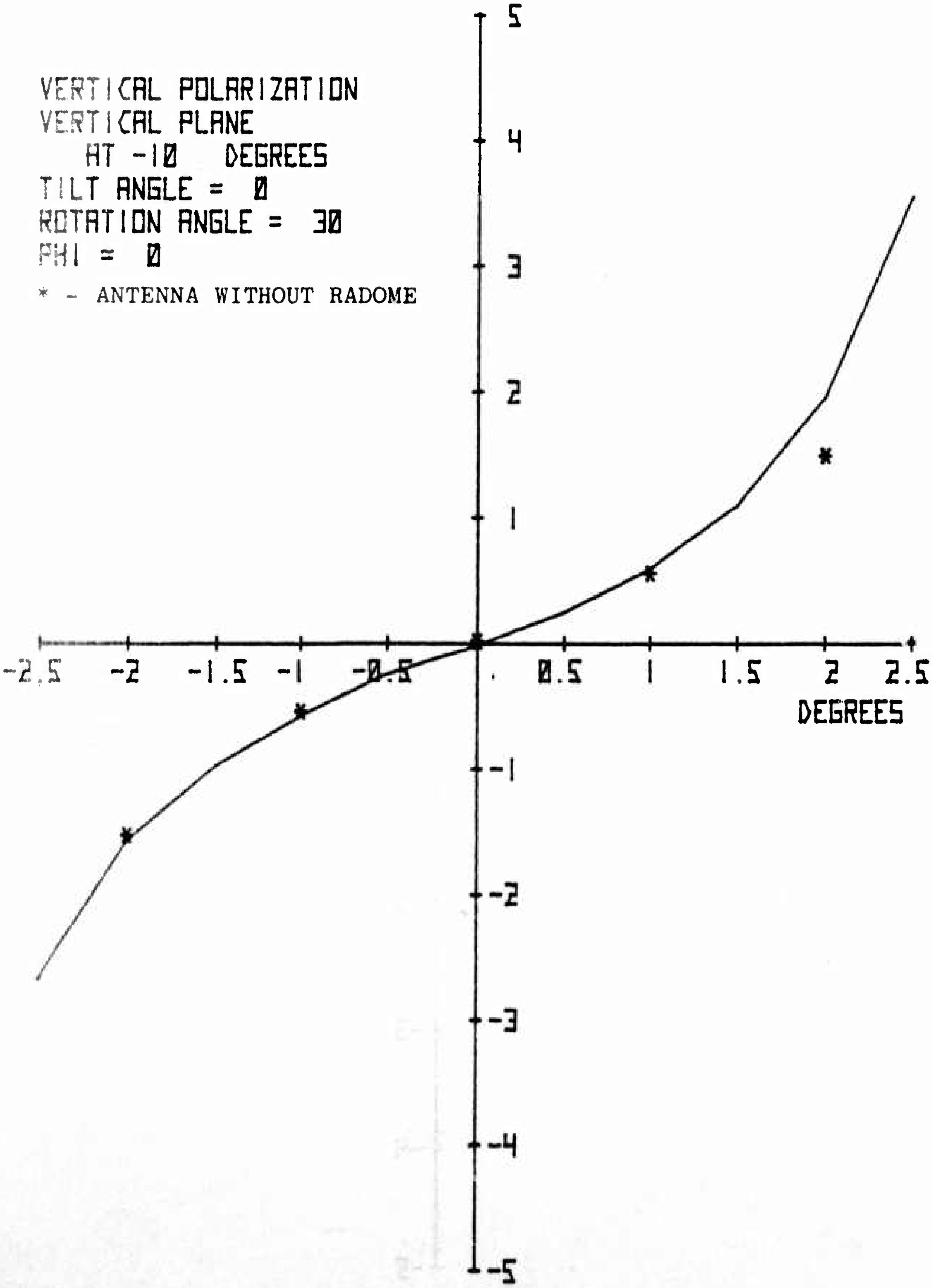


Figure 39. Off-Axis Angle Error Slope

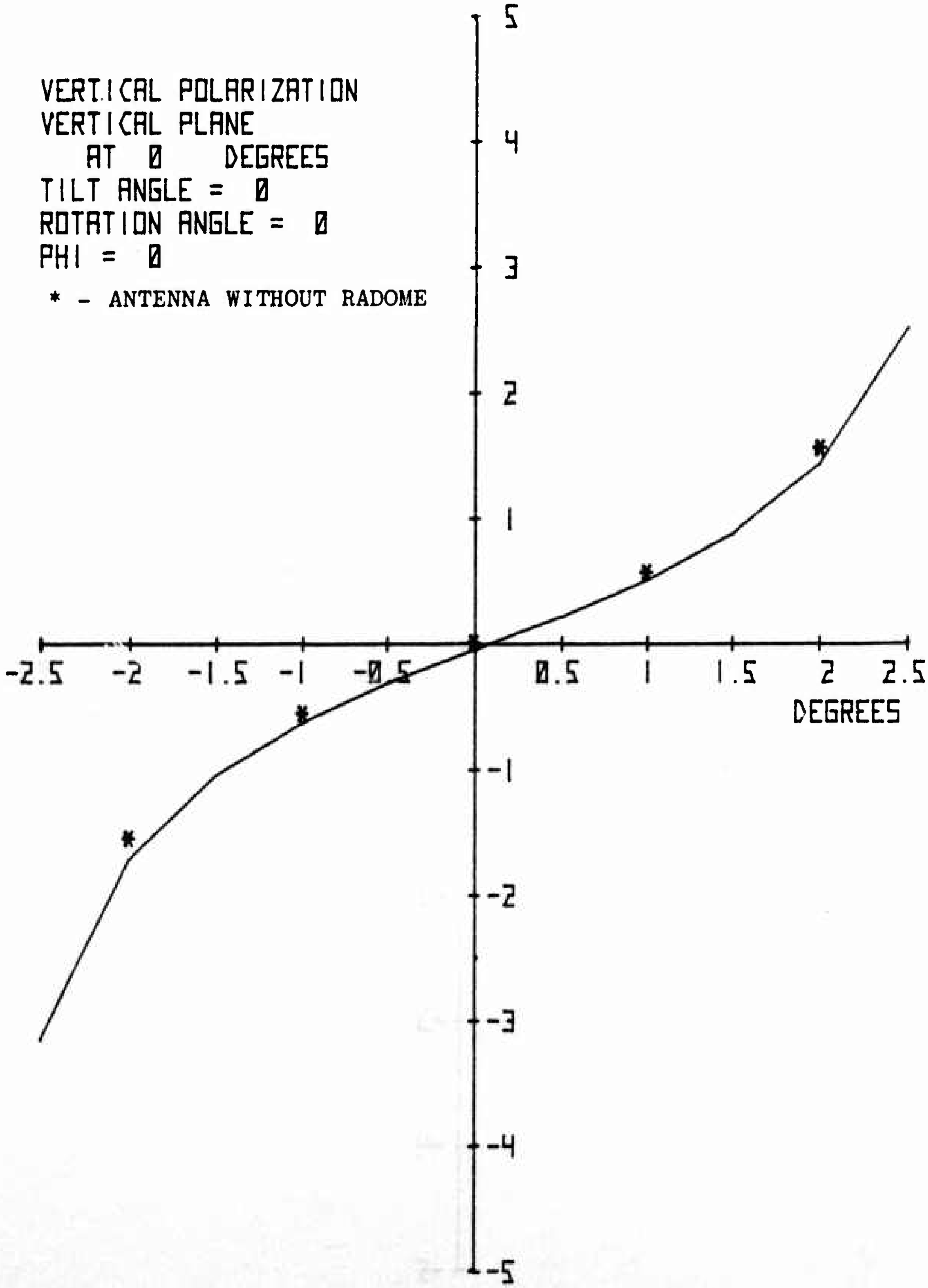


Figure 40. Off-Axis Angle Error Slope

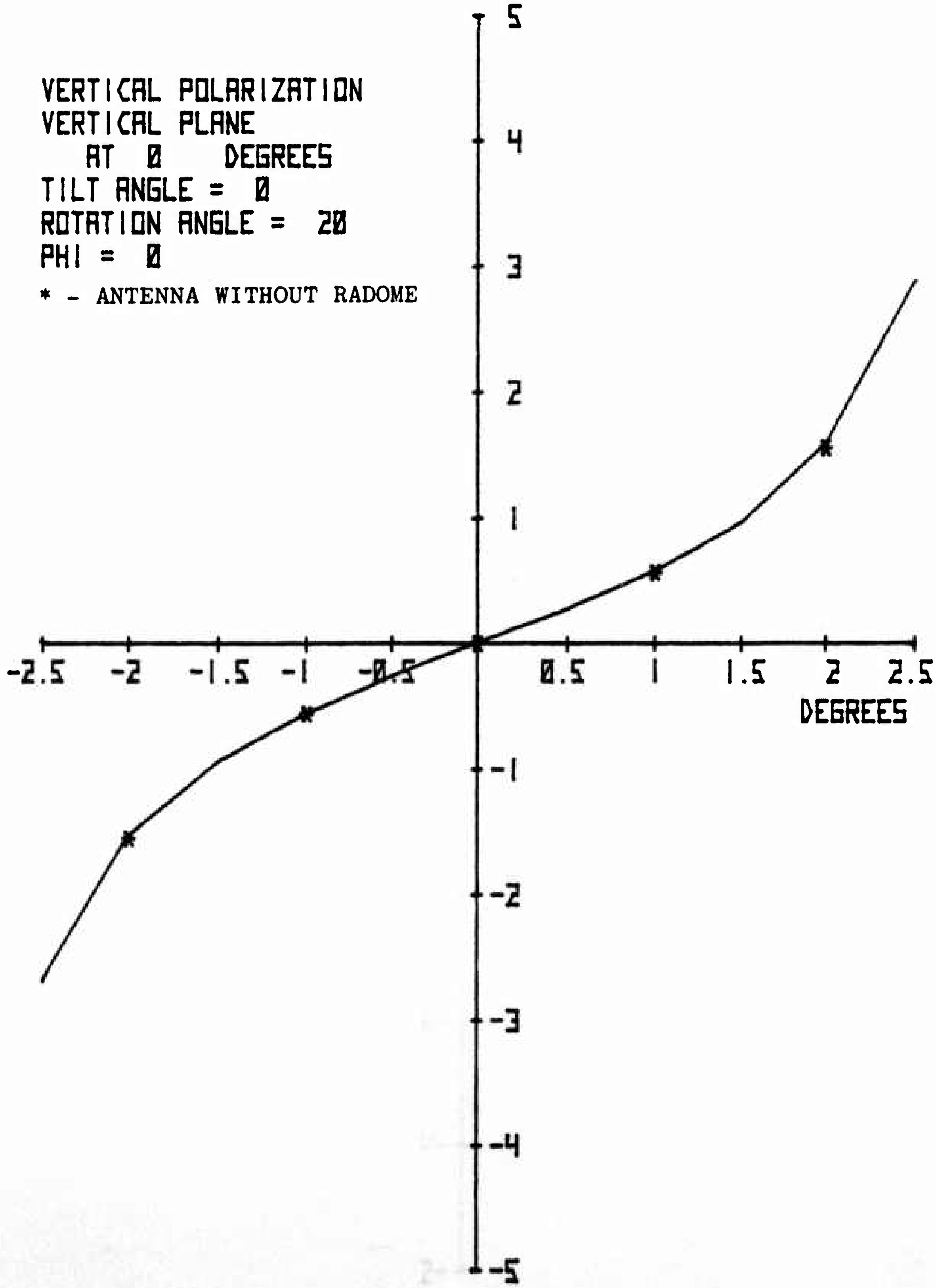


Figure 41. Off-Axis Angle Error Slope

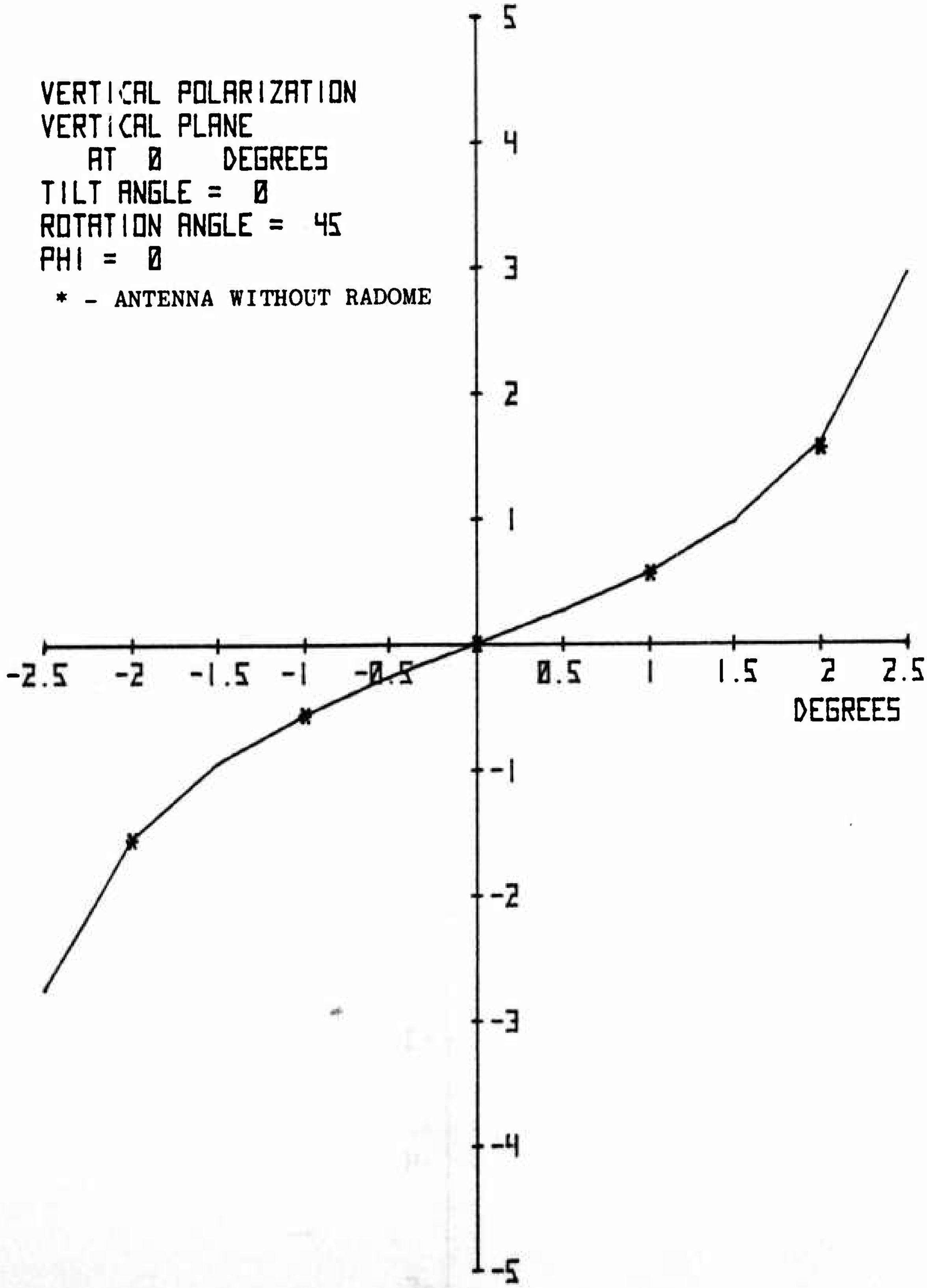


Figure 42. Off-Axis Angle Error Slope

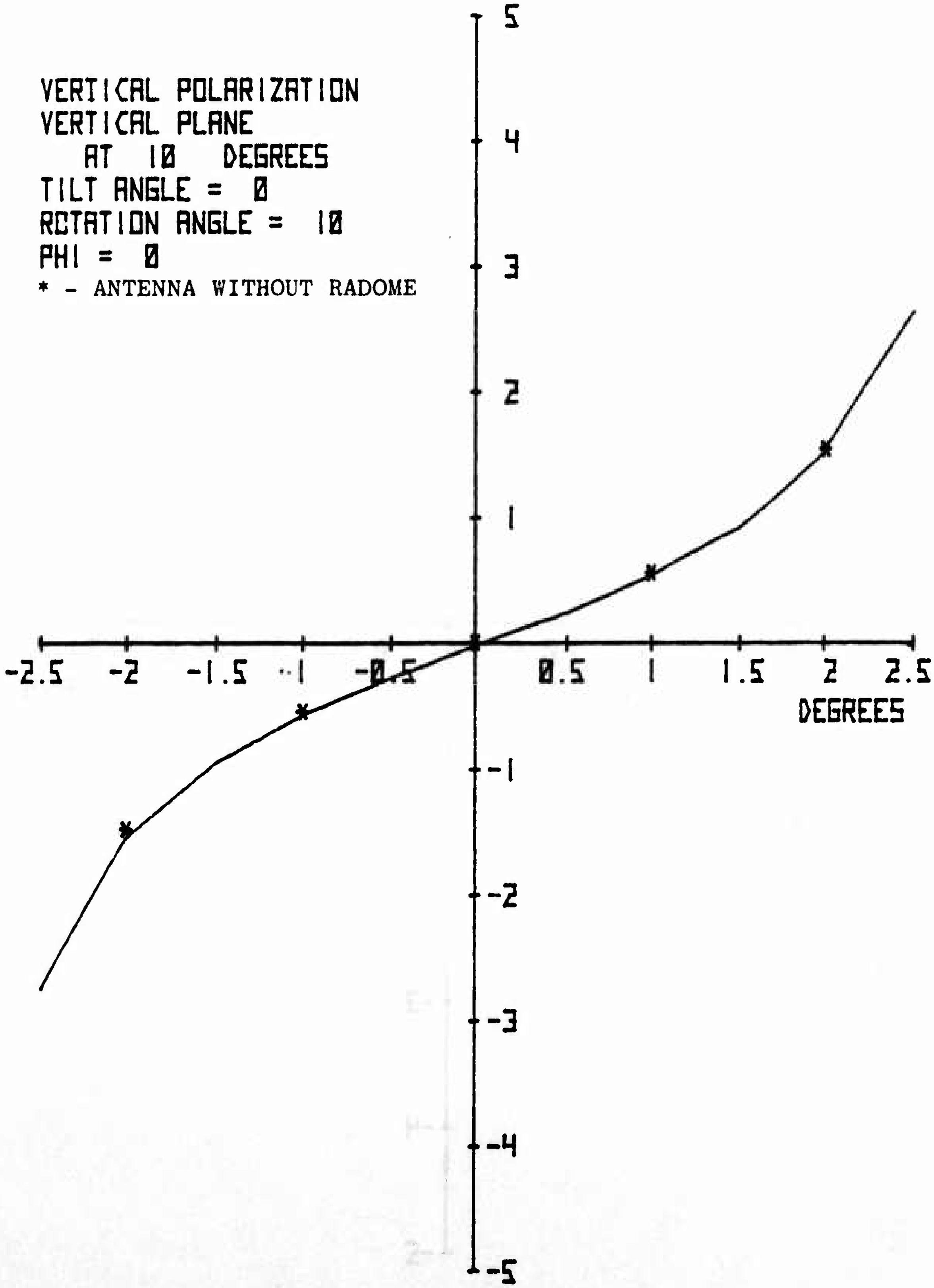


Figure 43. Off-Axis Angle Error Slope

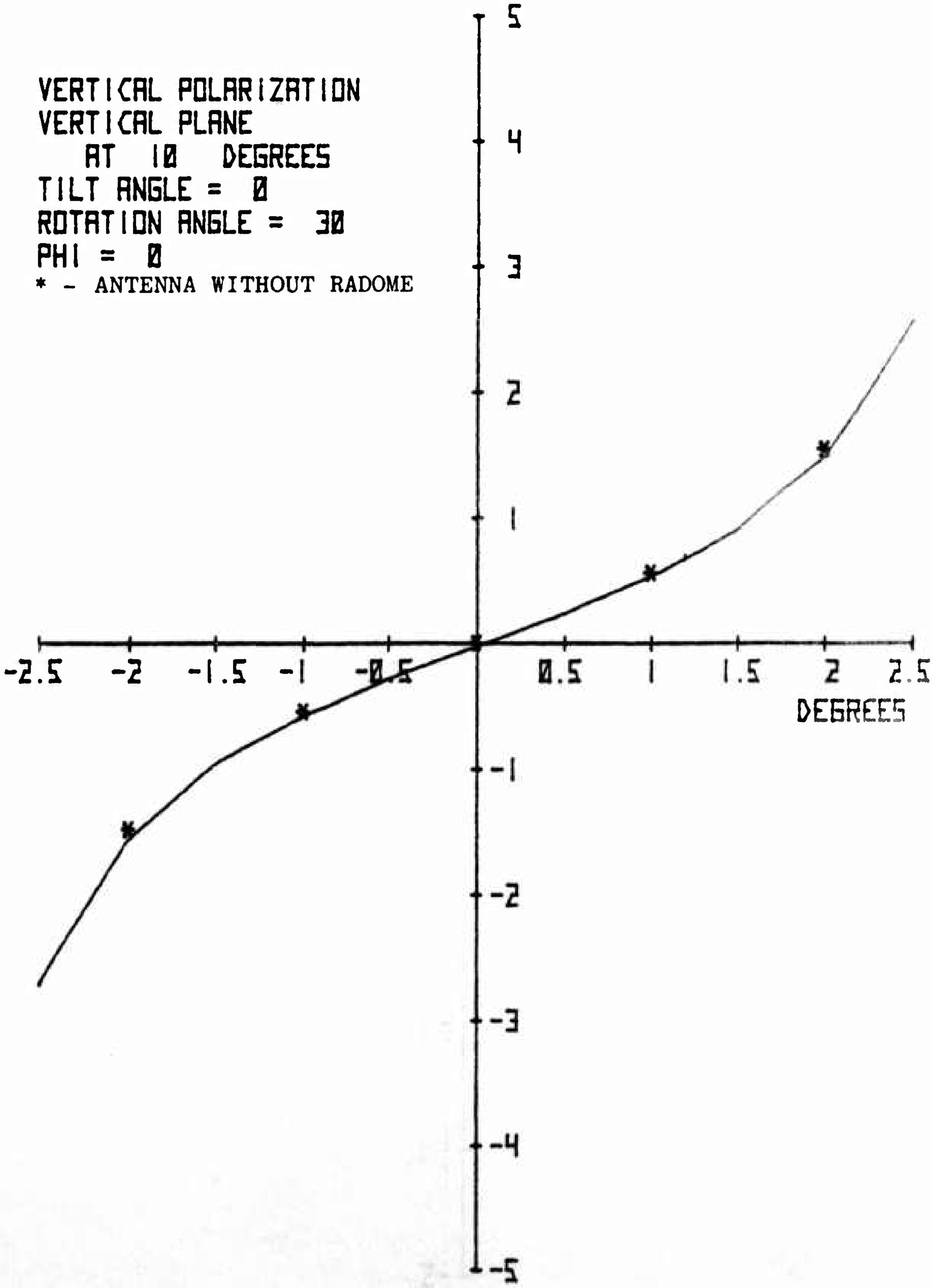


Figure 44. Off-Axis Angle Error Slope

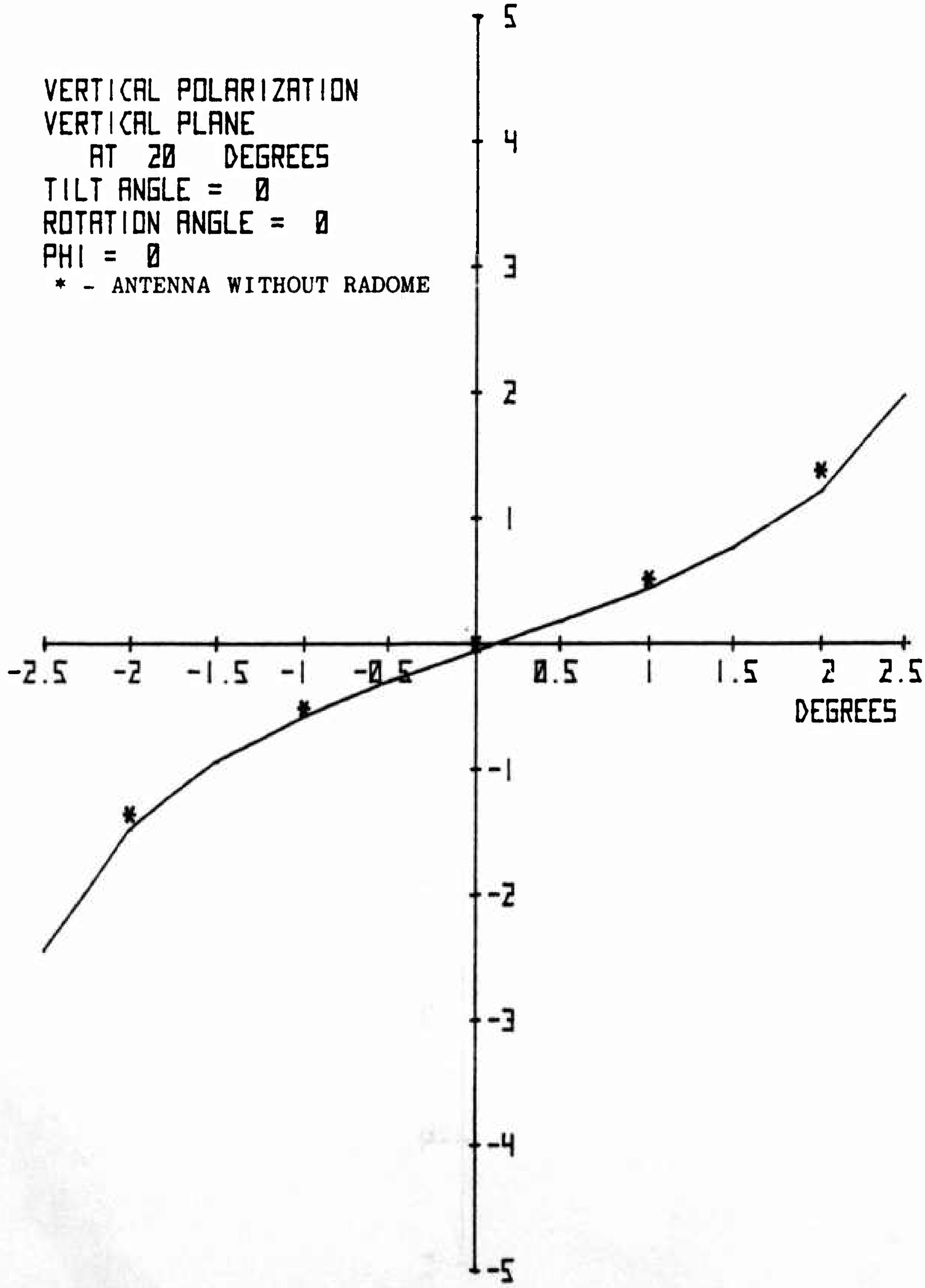


Figure 45. Off-Axis Angle Error Slope

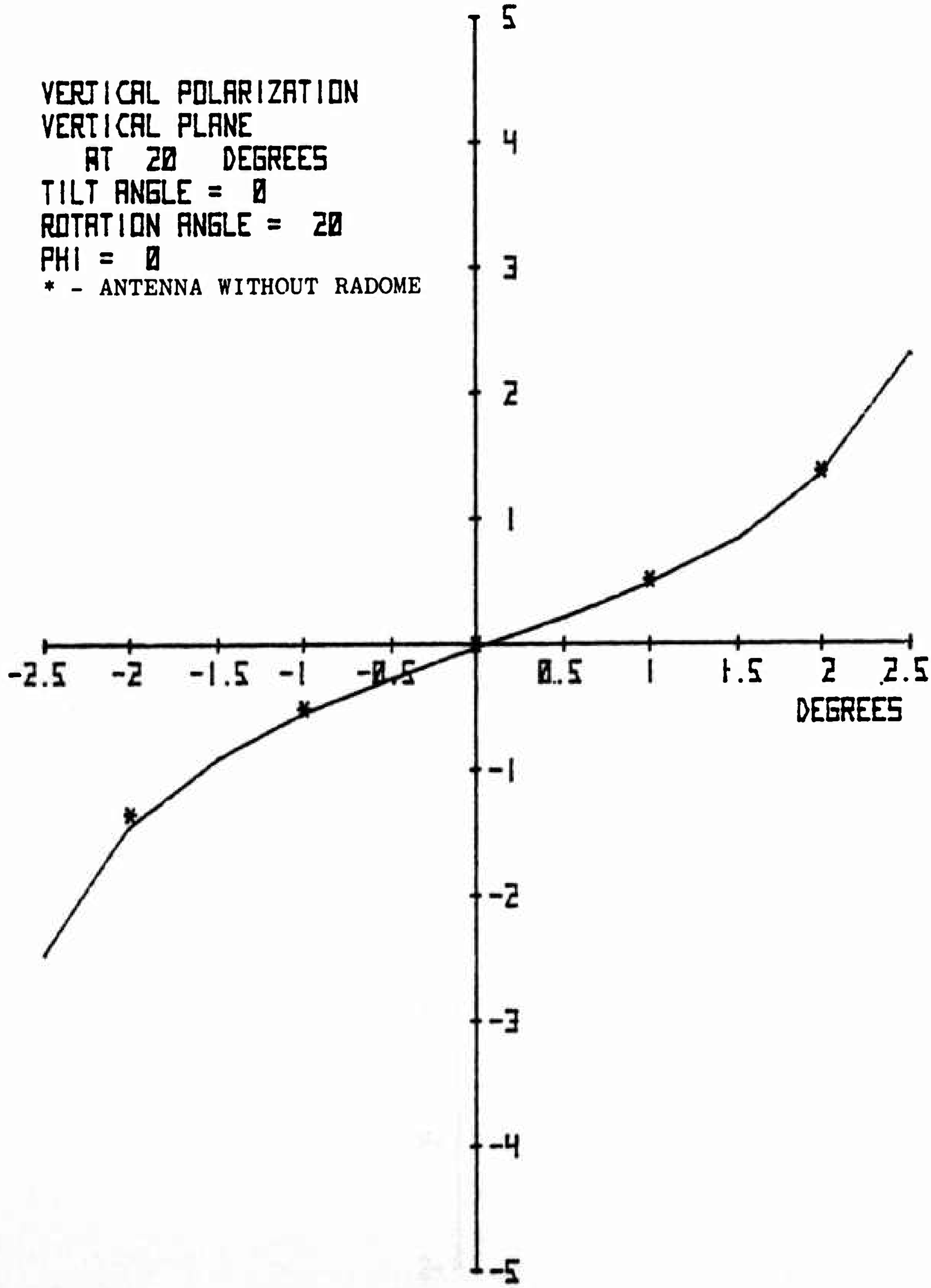


Figure 46. Off-Axis Angle Error Slope

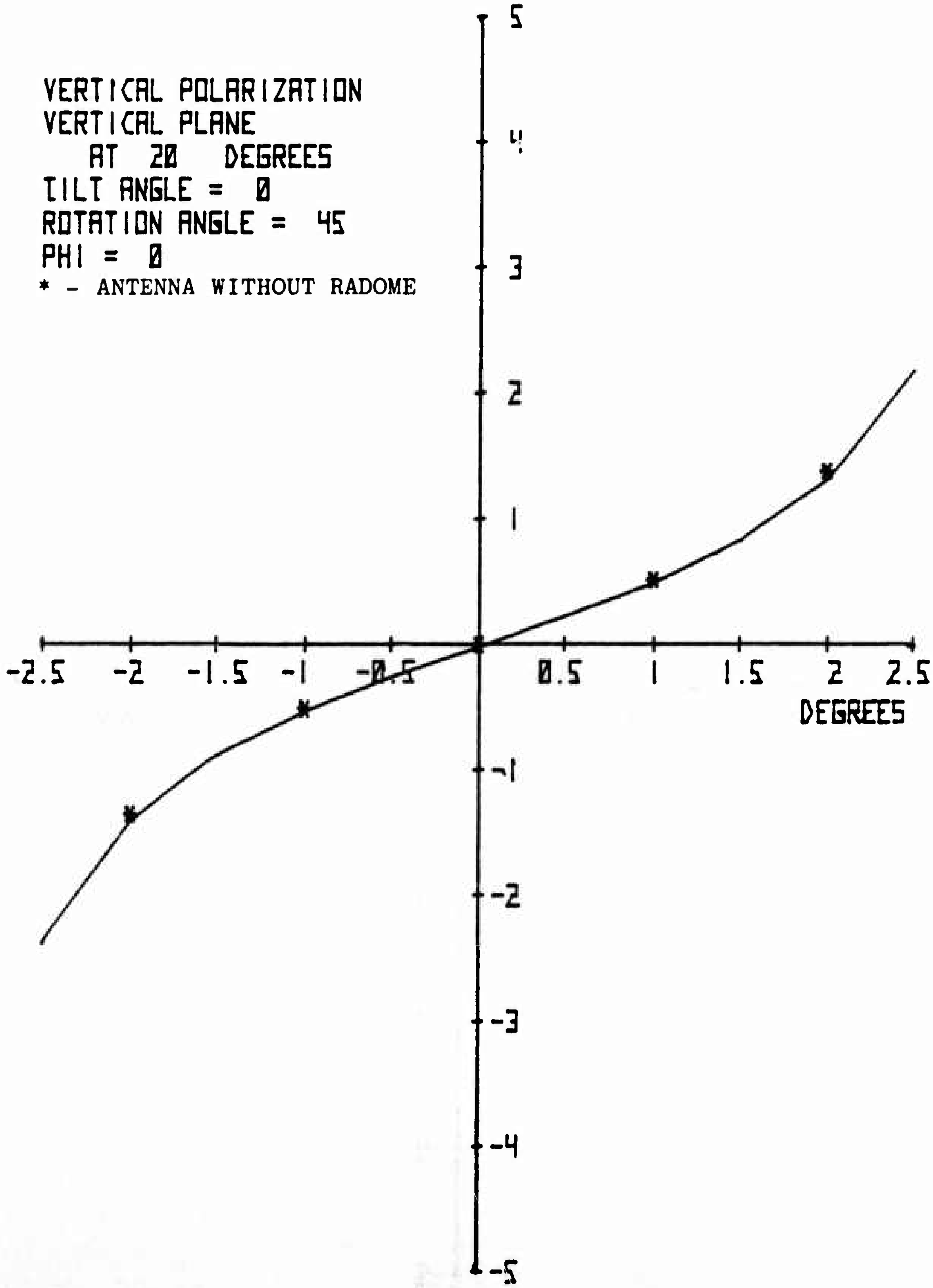


Figure 47. Off-Axis Angle Error Slope

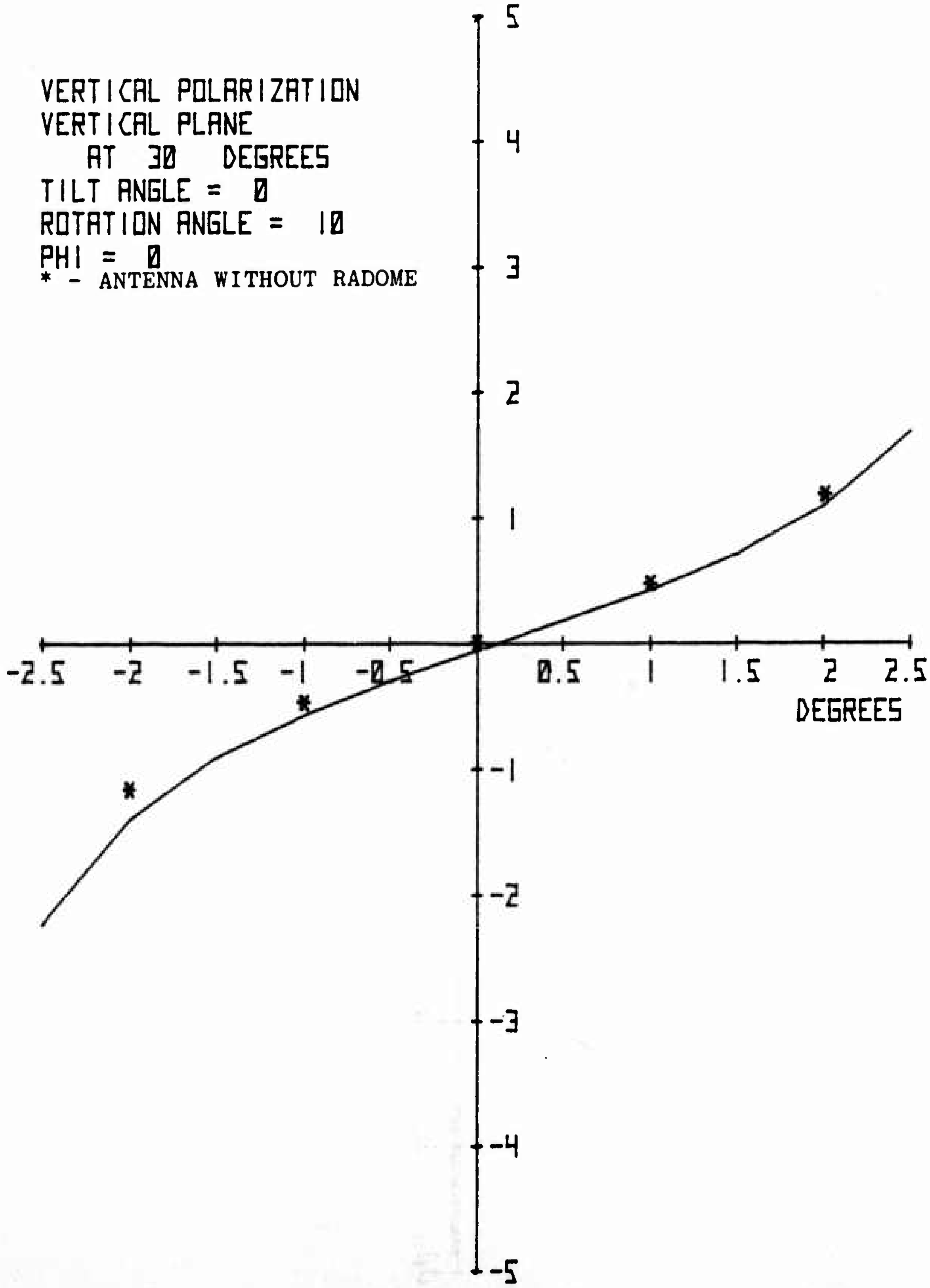


Figure 48. Off-Axis Angle Error Slope

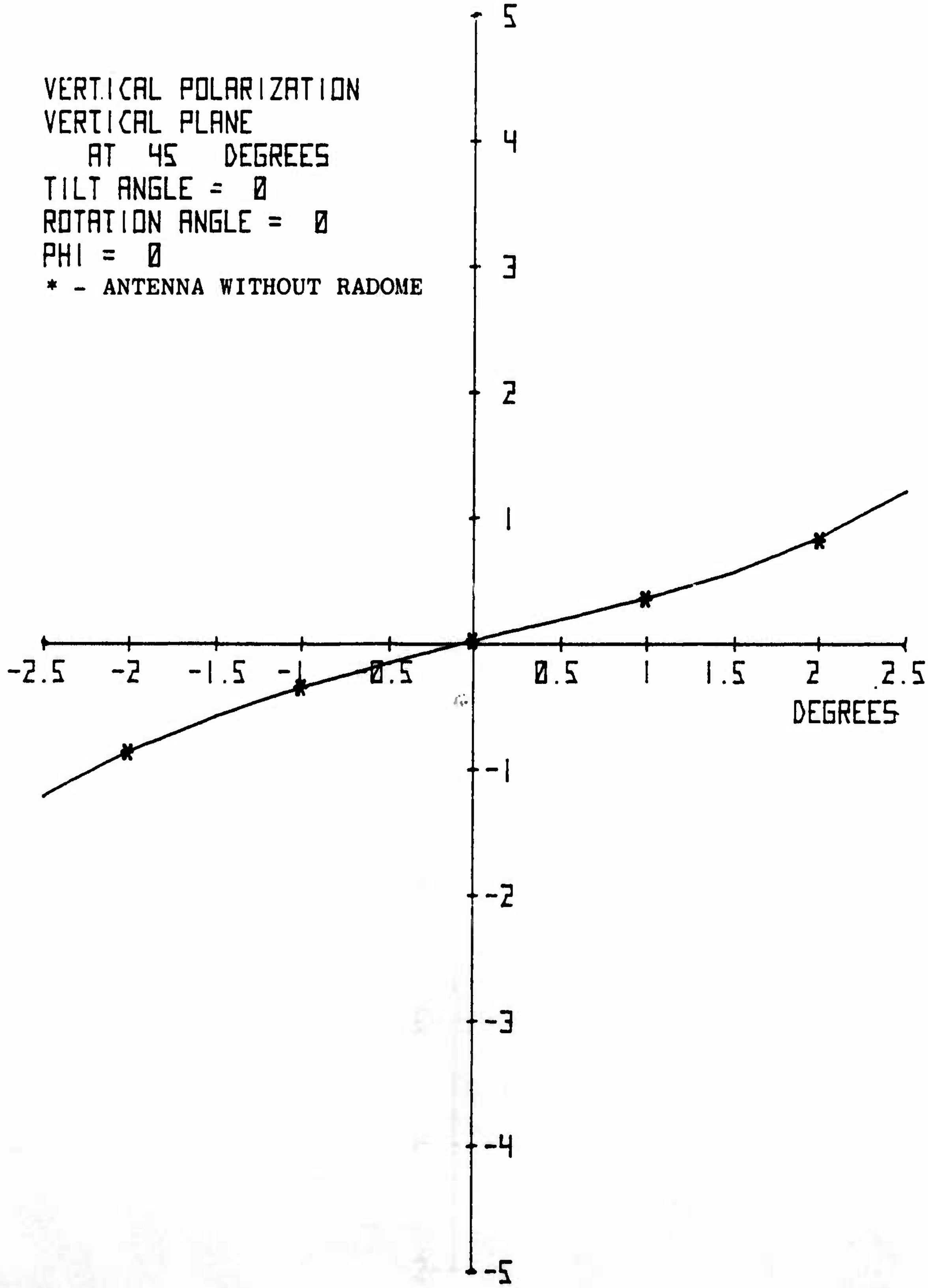


Figure 49. Off-Axis Angle Error Slope

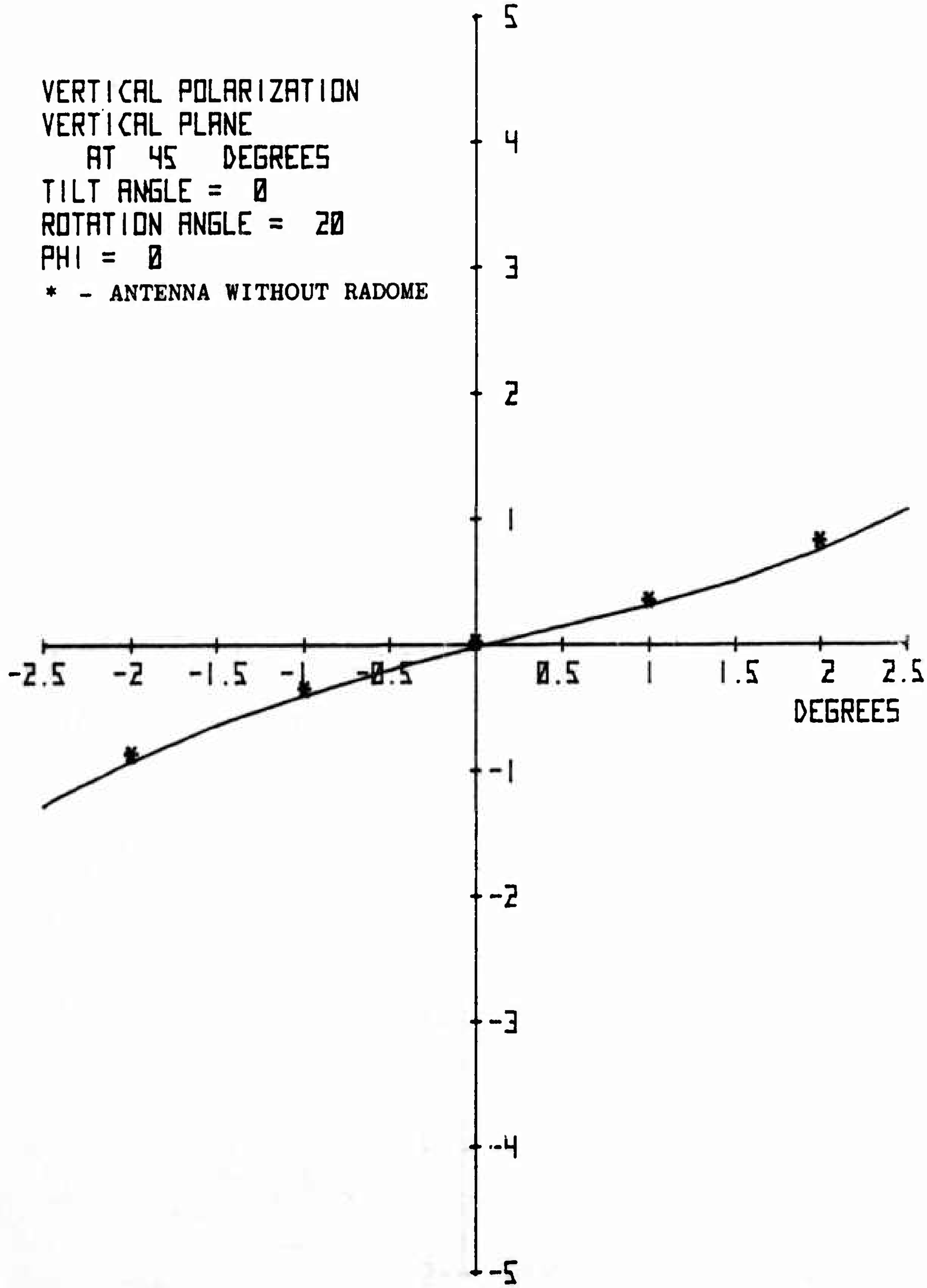


Figure 50. Off-Axis Angle Error Slope

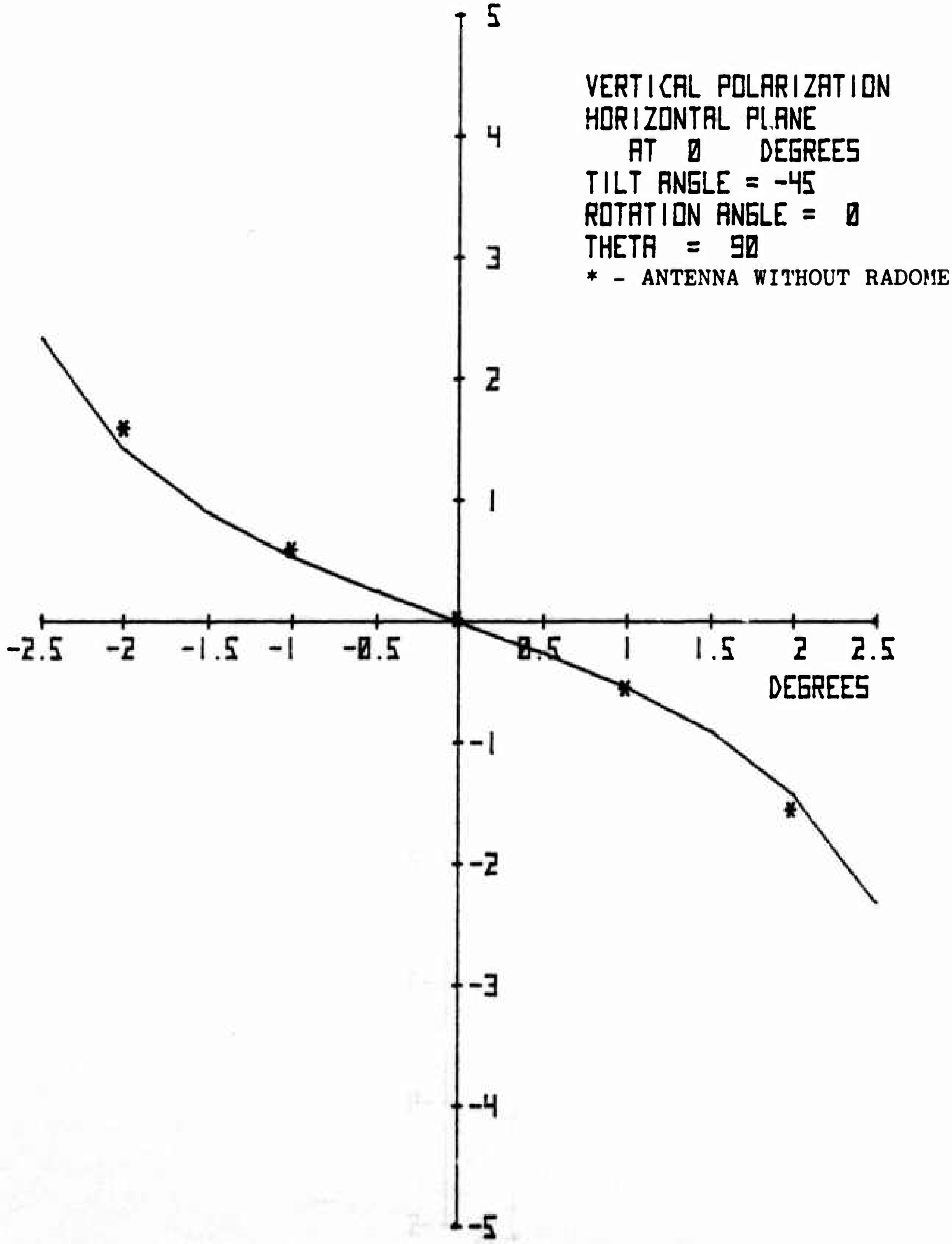


Figure 51. Off-Axis Angle Error Slope

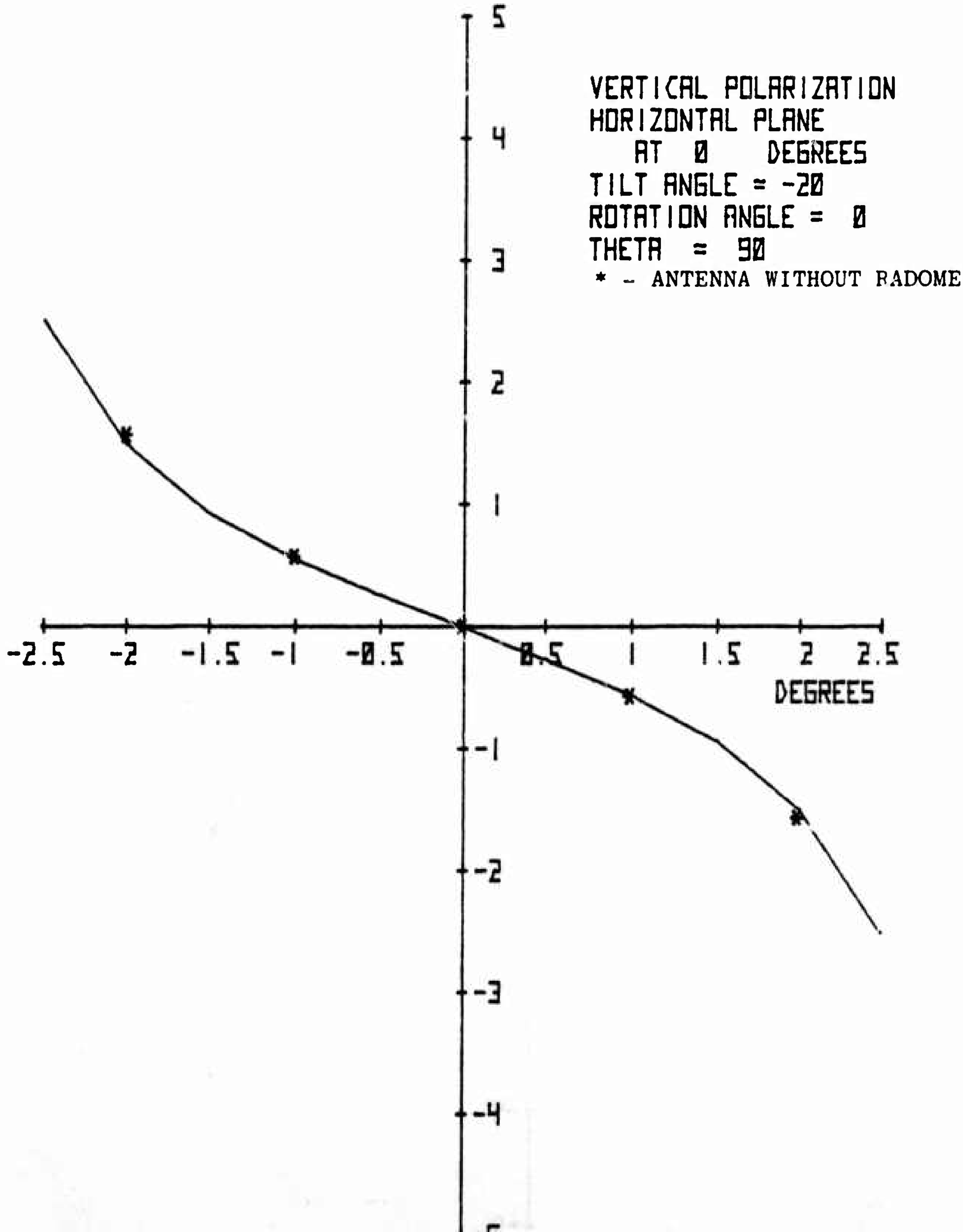


Figure 52. Off-Axis Angle Error Slope

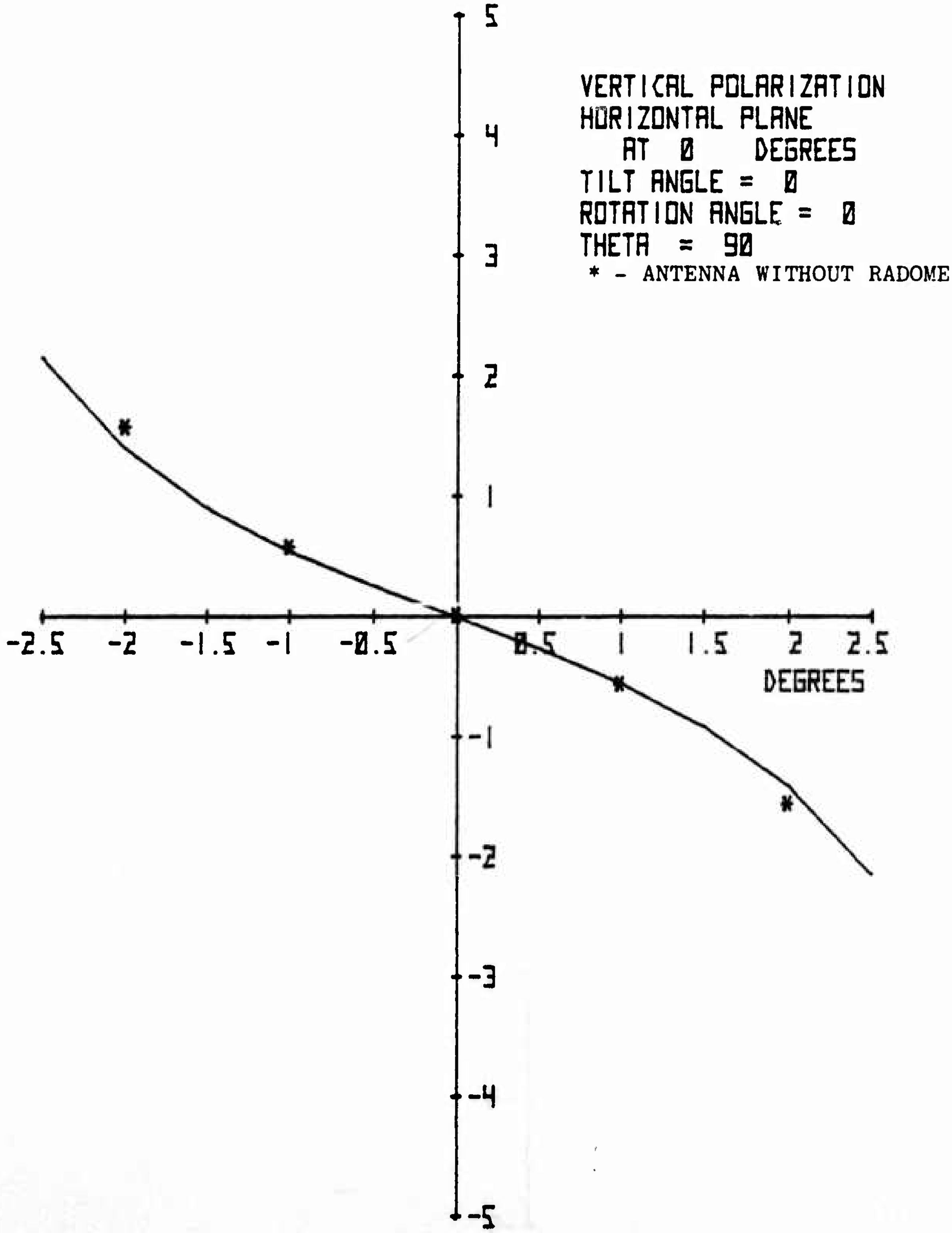


Figure 53. Off-Axis Angle Error Slope

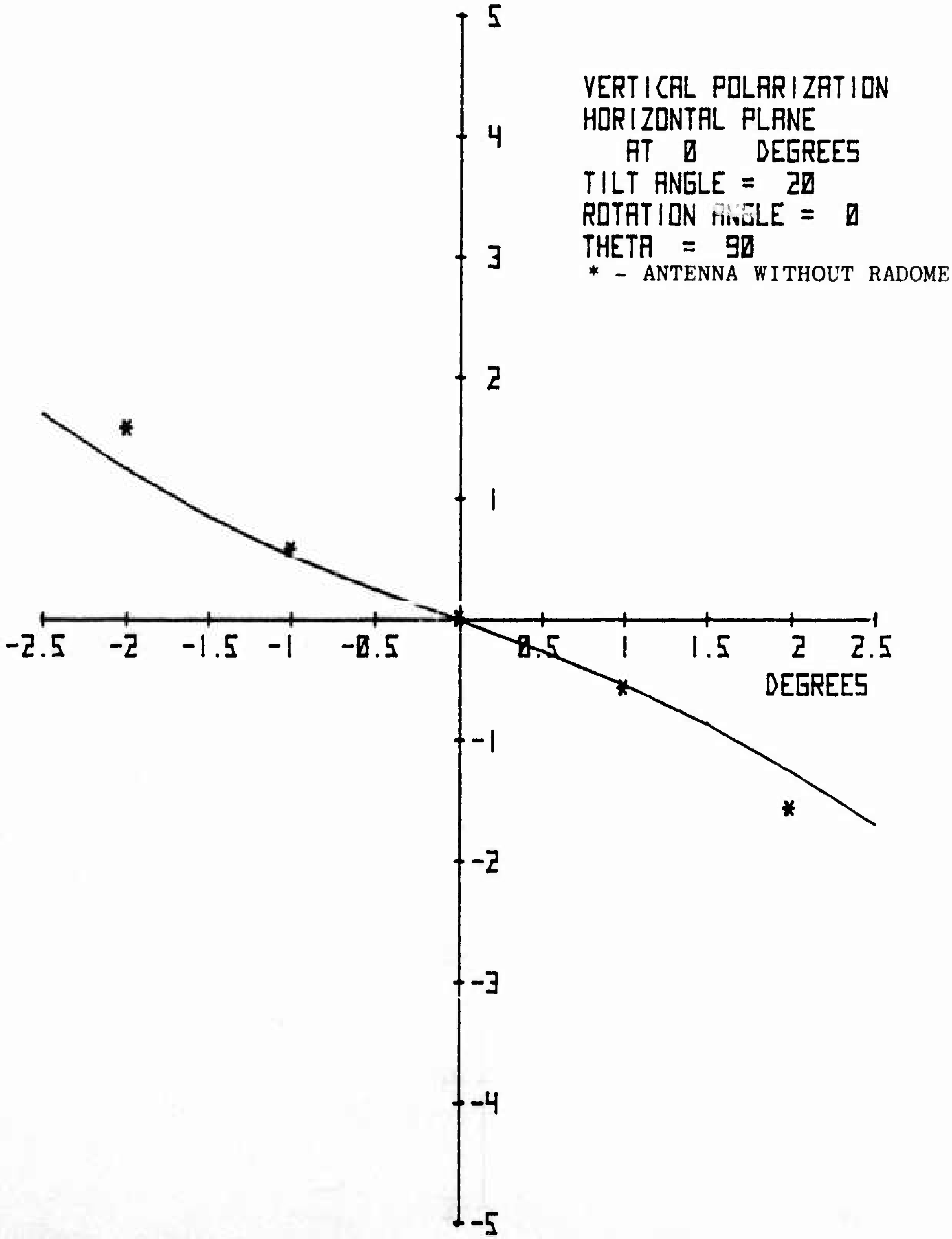


Figure 54. Off-Axis Angle Error Slope

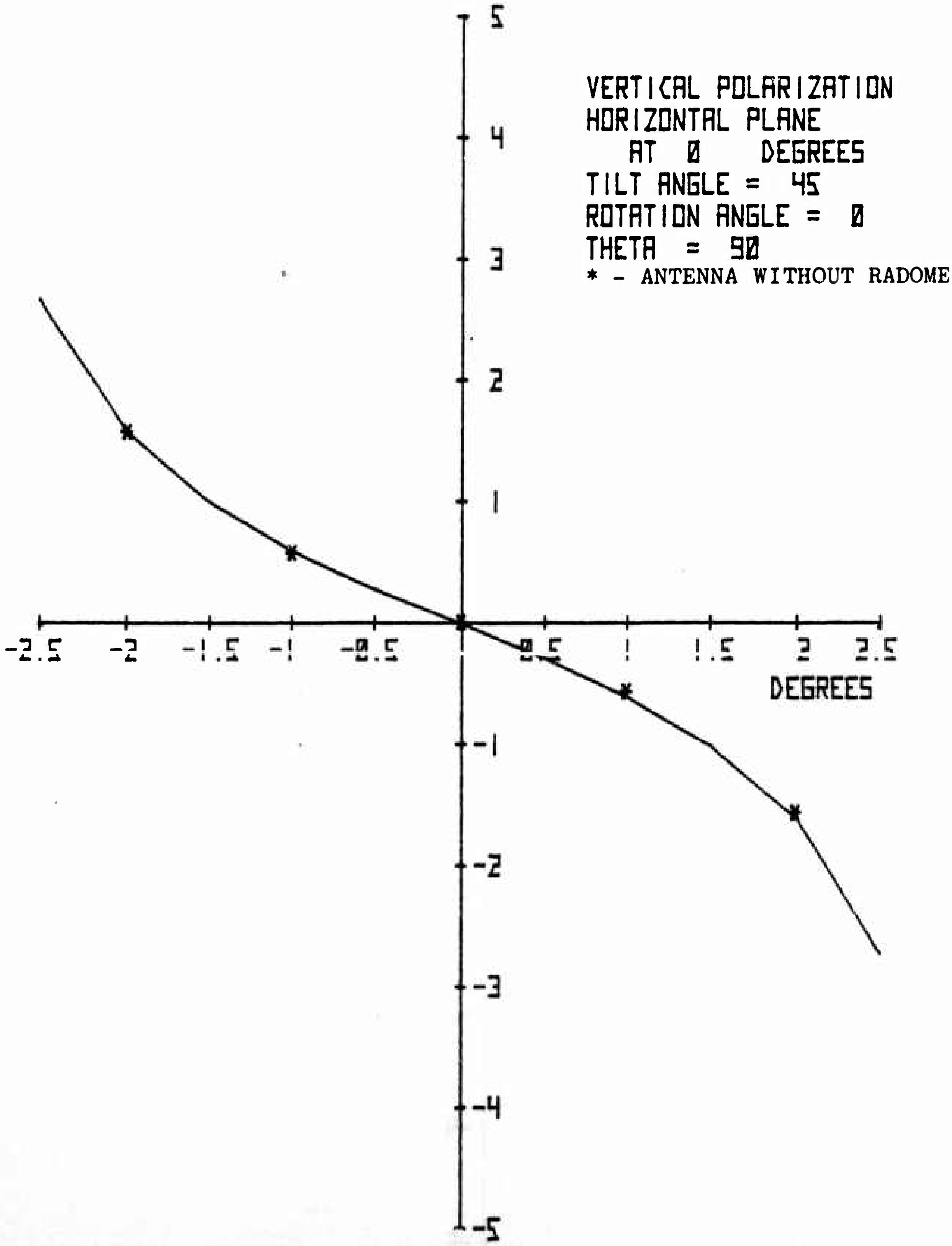


Figure 55. Off-Axis Angle Error Slope

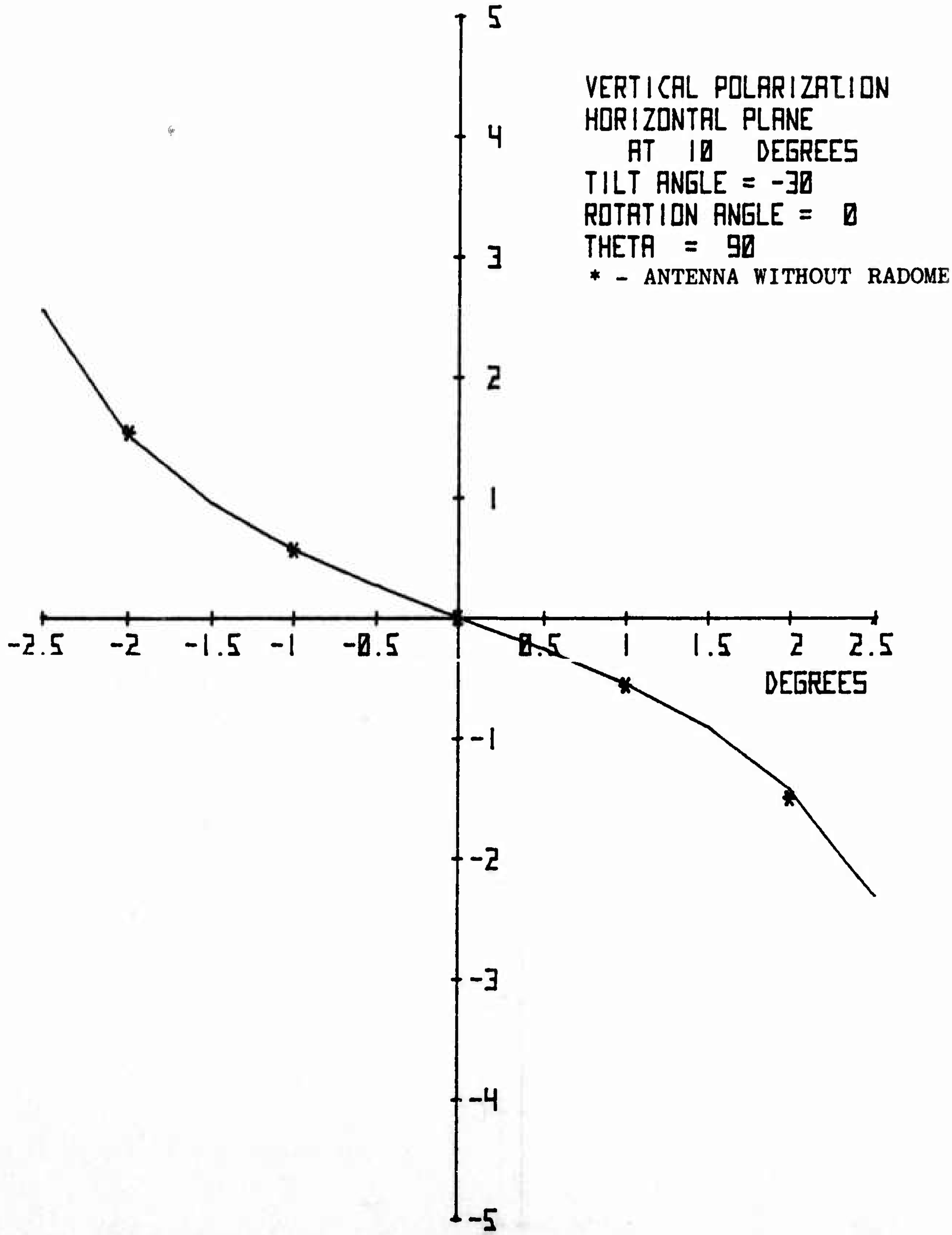


Figure 56. Off-Axis Angle Error Slope

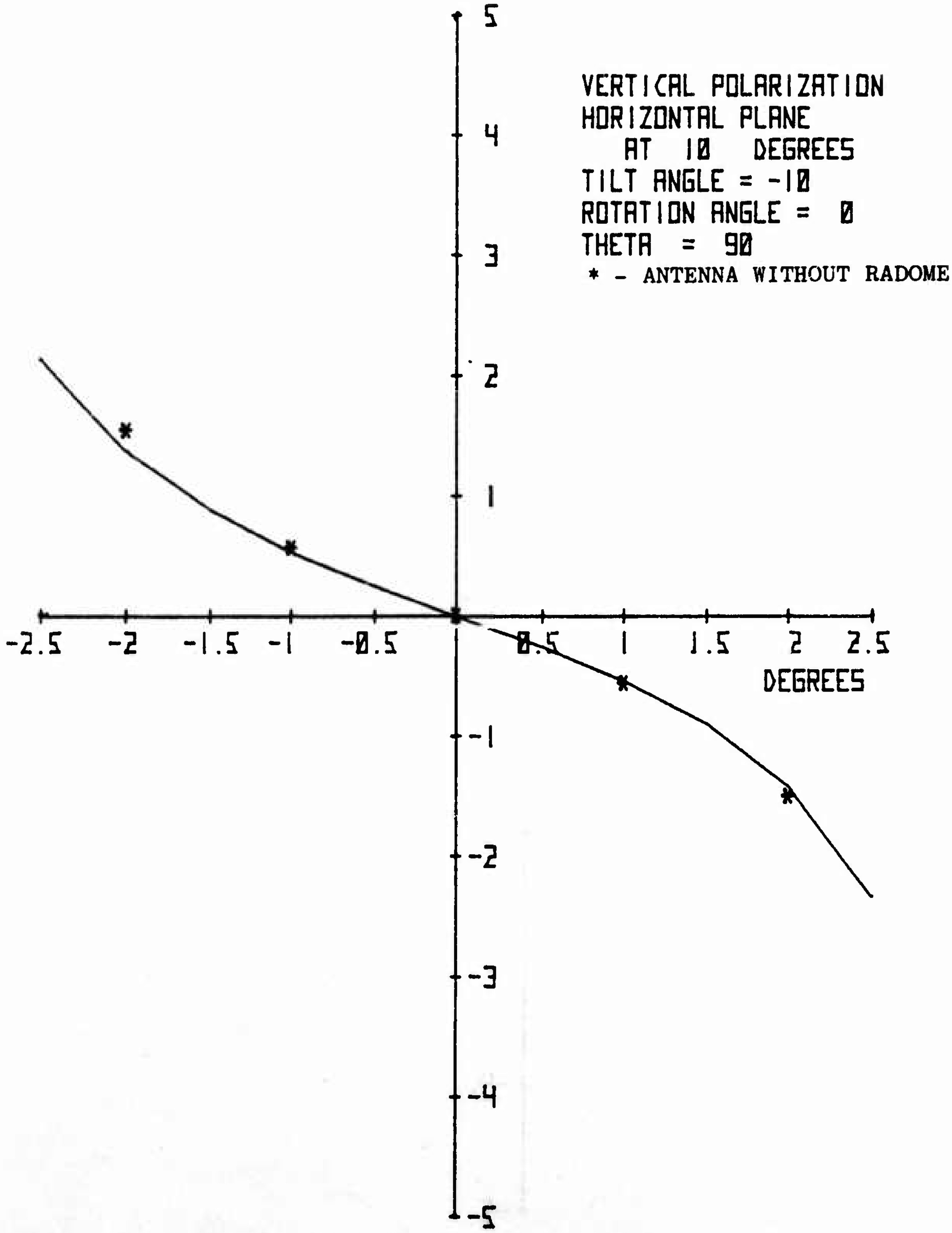


Figure 57. Off-Axis Angle Error Slope

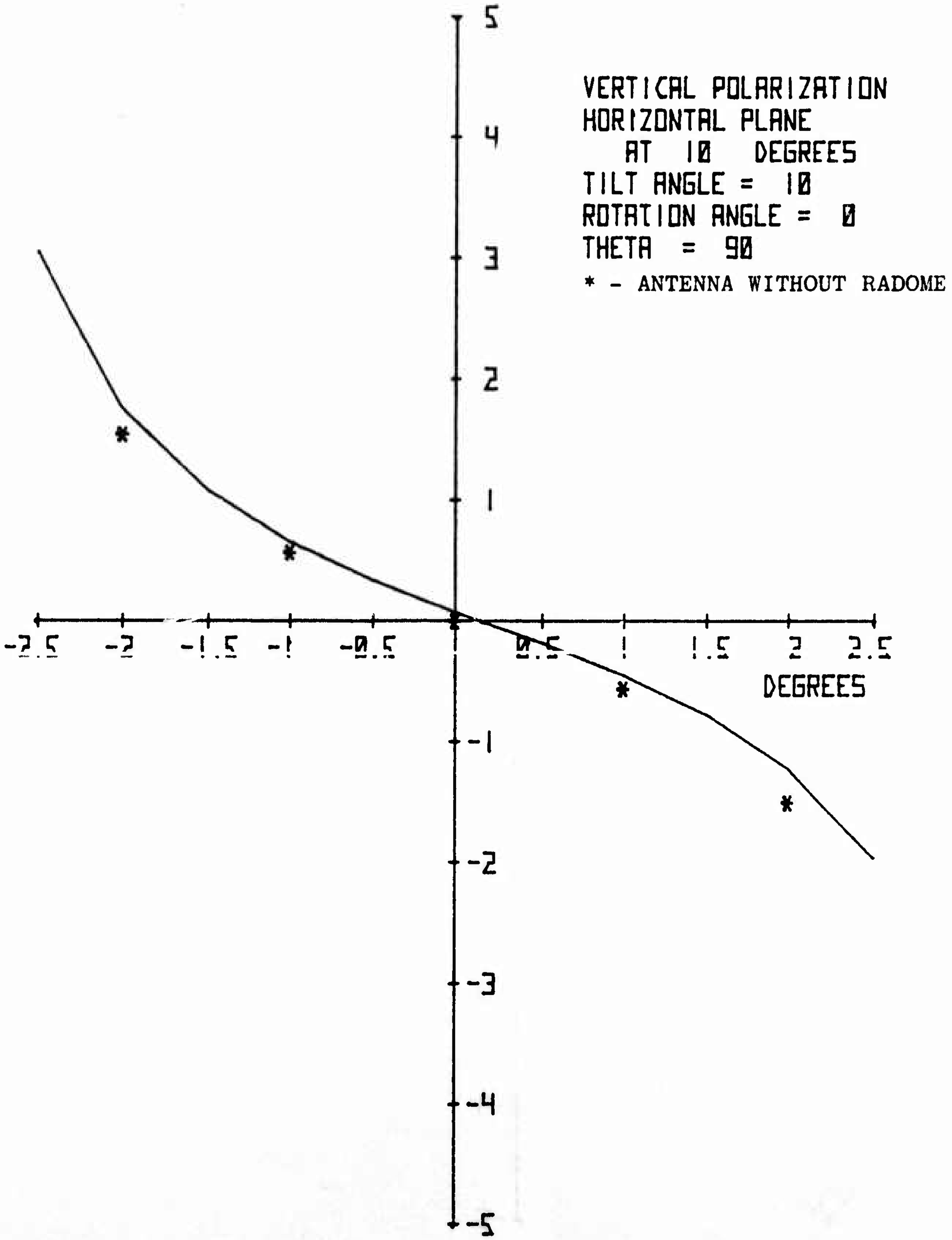


Figure 58. Off-Axis Angle Error Slope

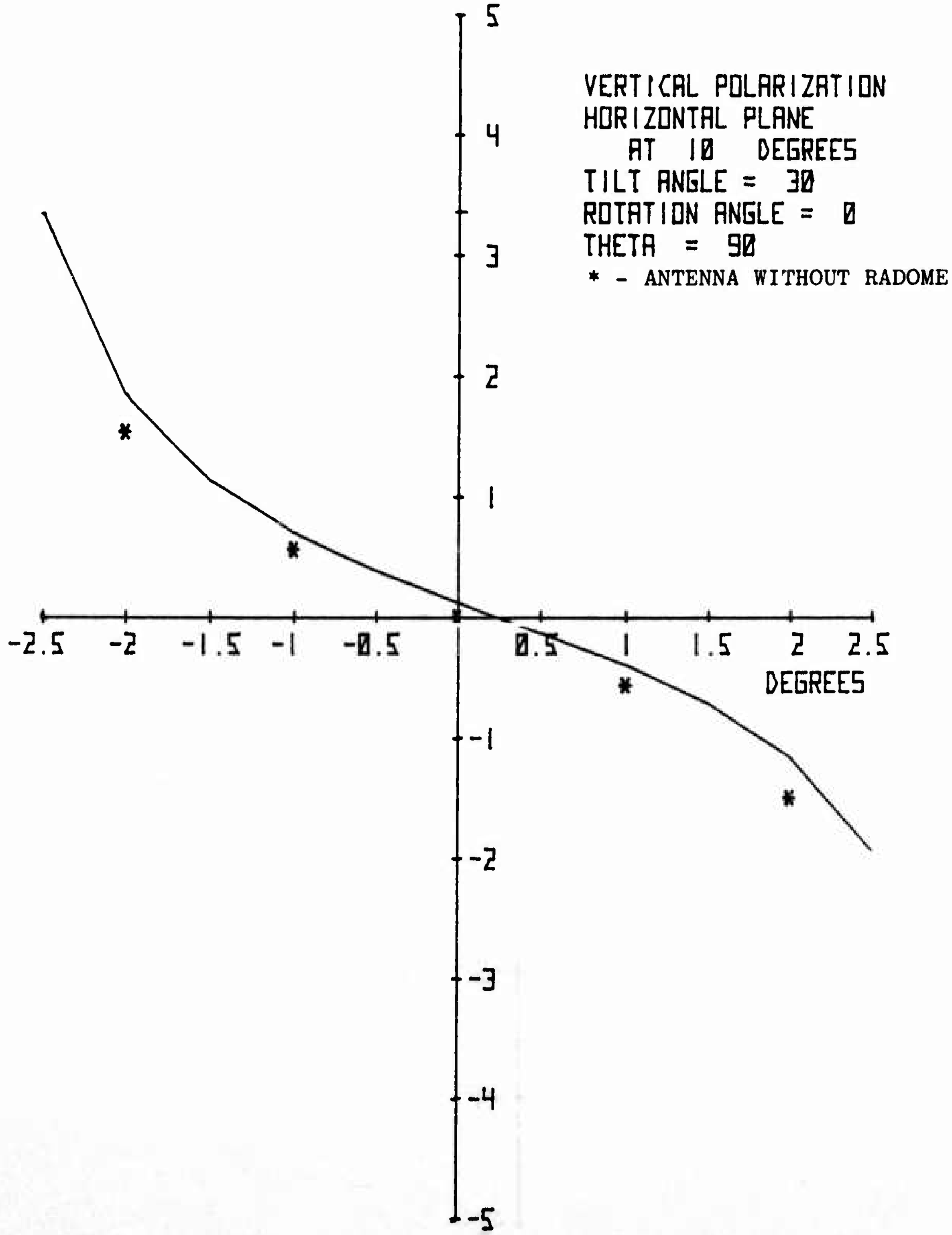


Figure 59. Off-Axis Angle Error Slope

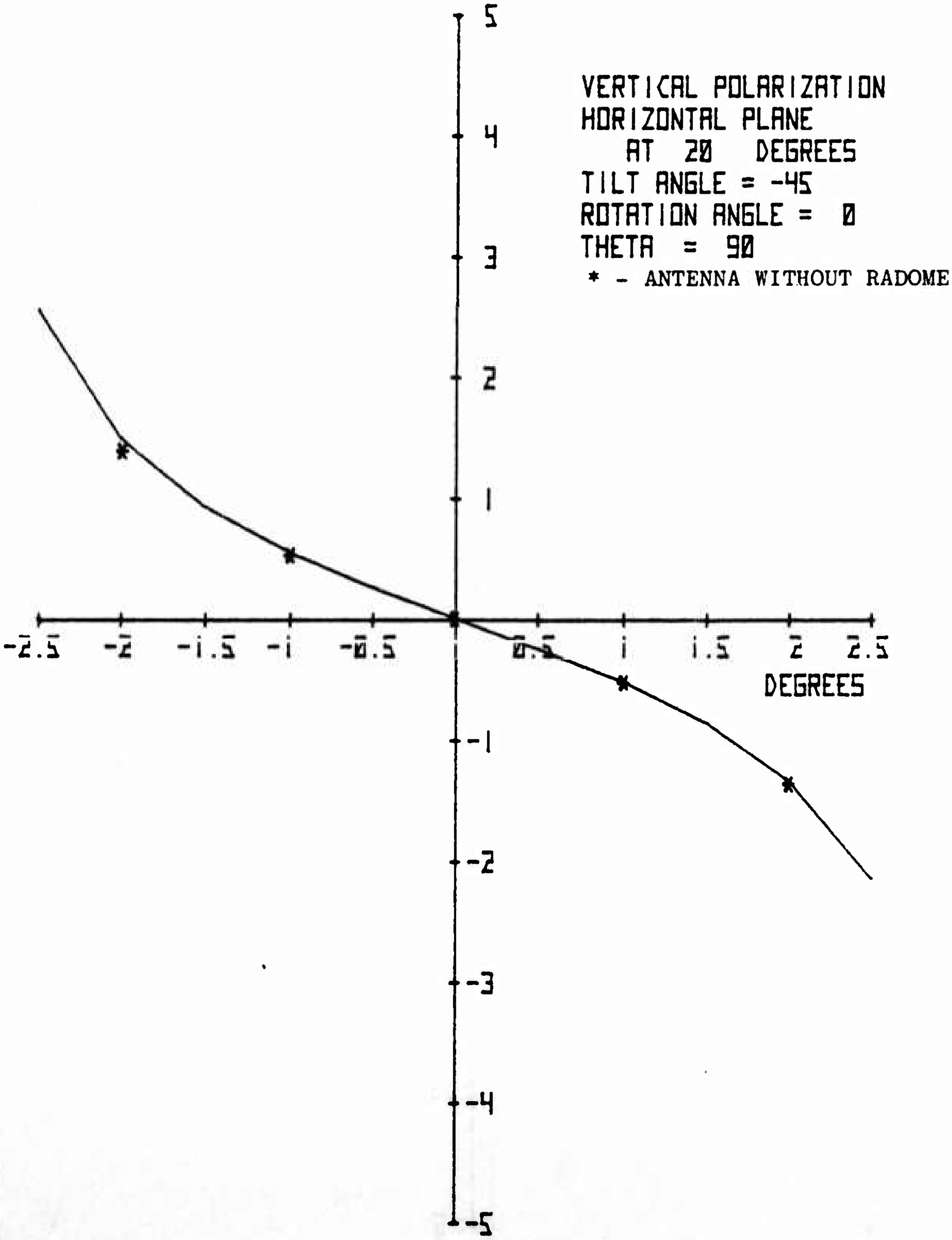


Figure 60. Off-Axis Angle Error Slope

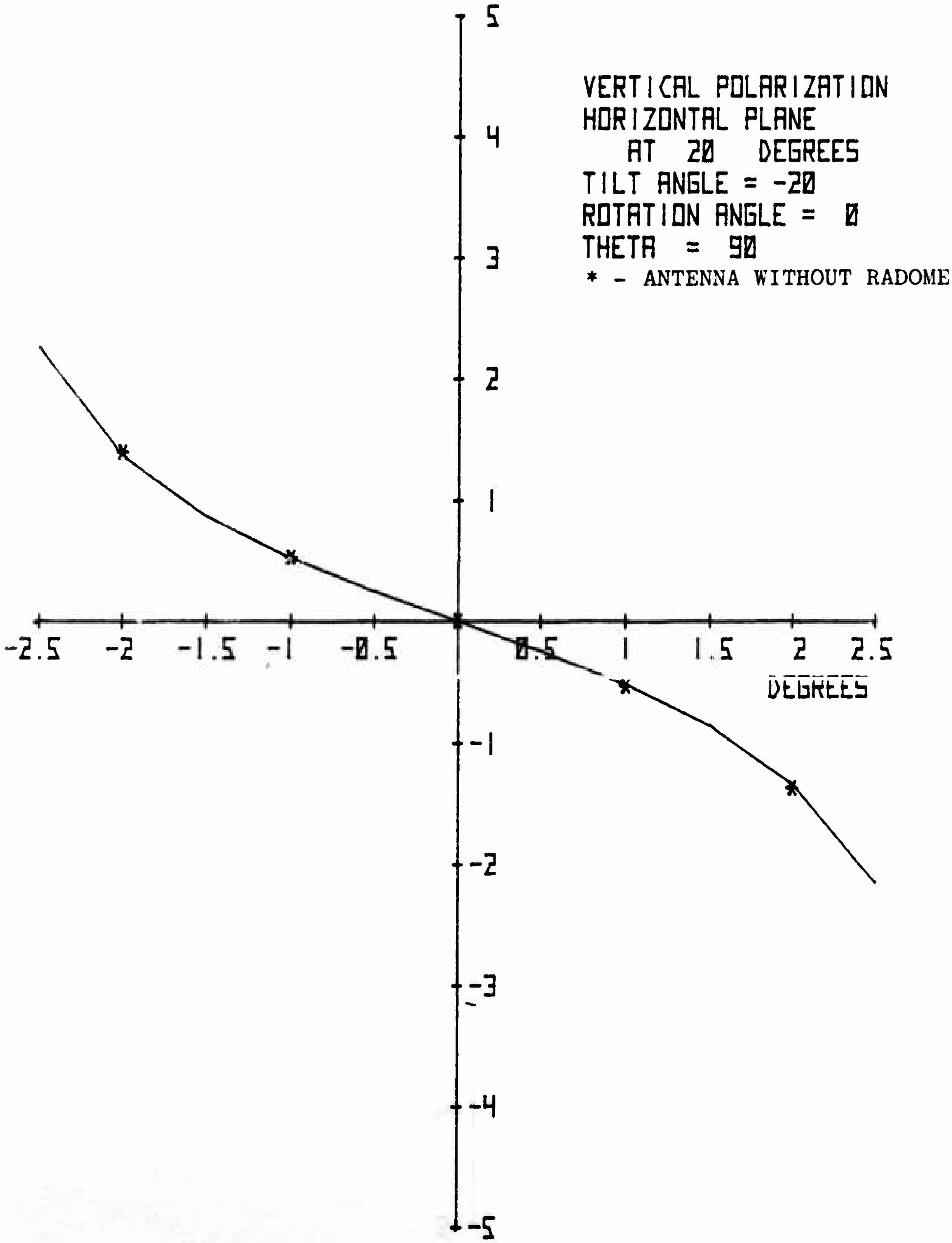


Figure 61. Off-Axis Angle Error Slope

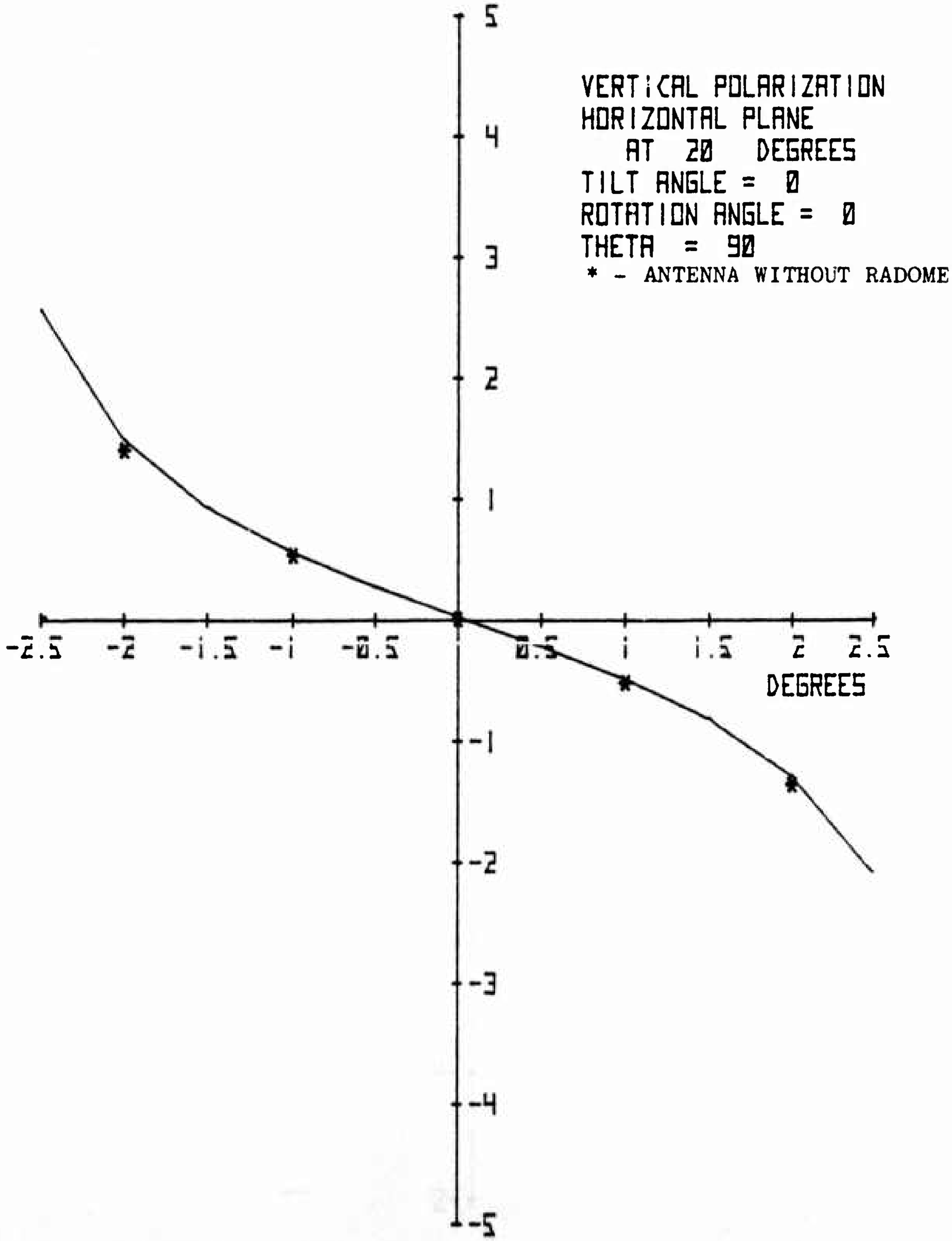


Figure 62. Off-Axis Angle Error Slope

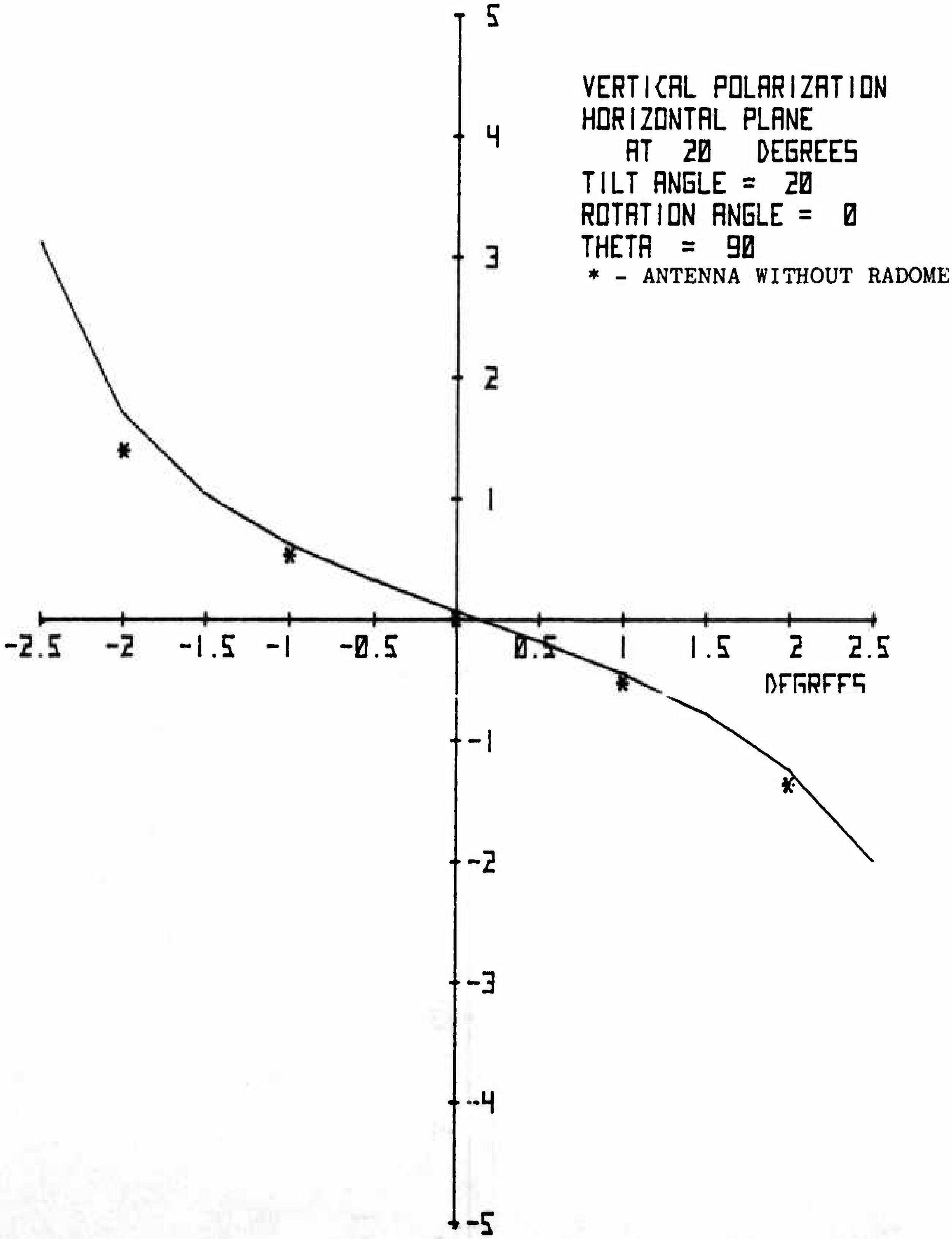


Figure 63. Off-Axis Angle Error Slope

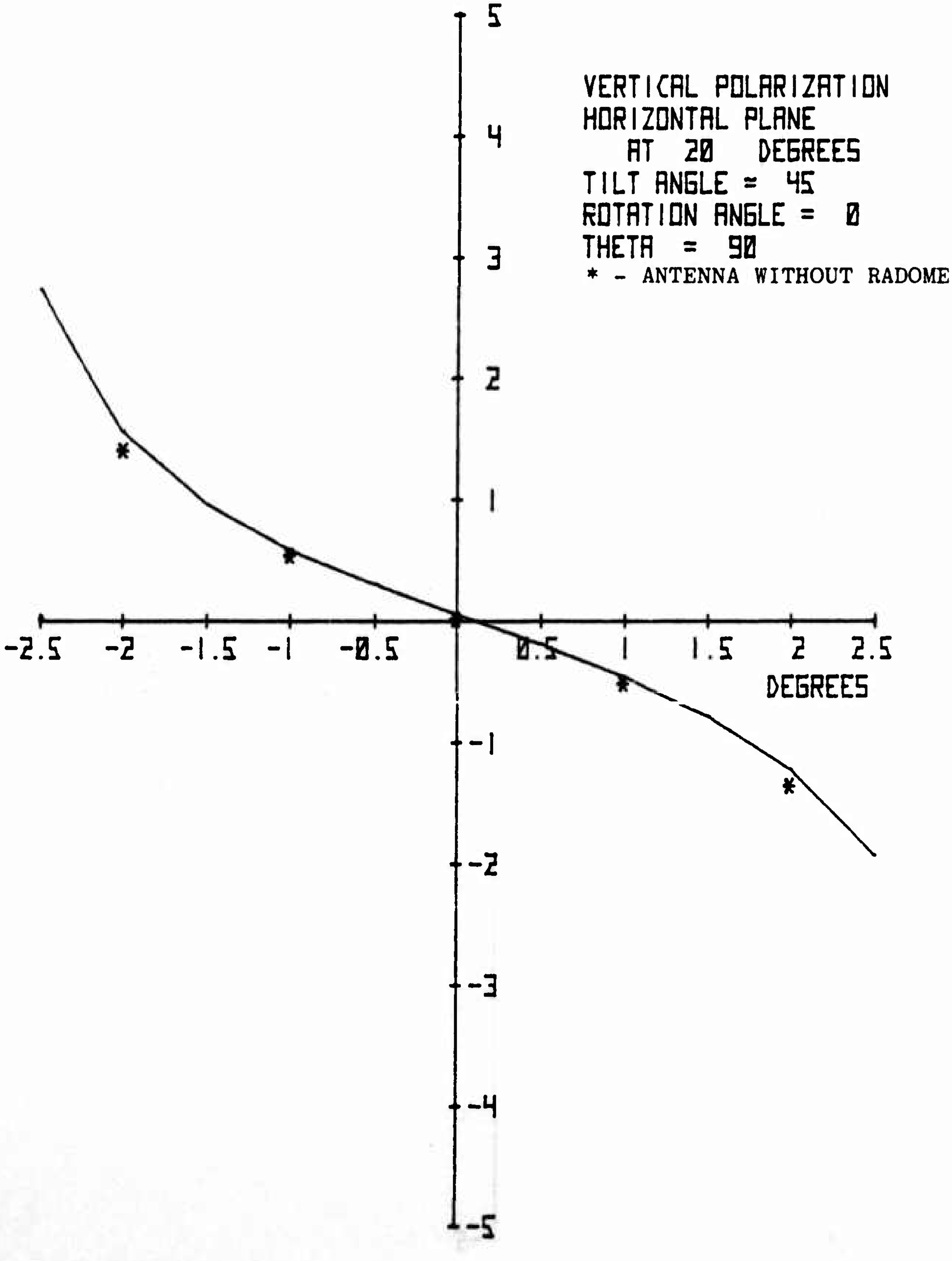


Figure 64. Off-Axis Angle Error Slope

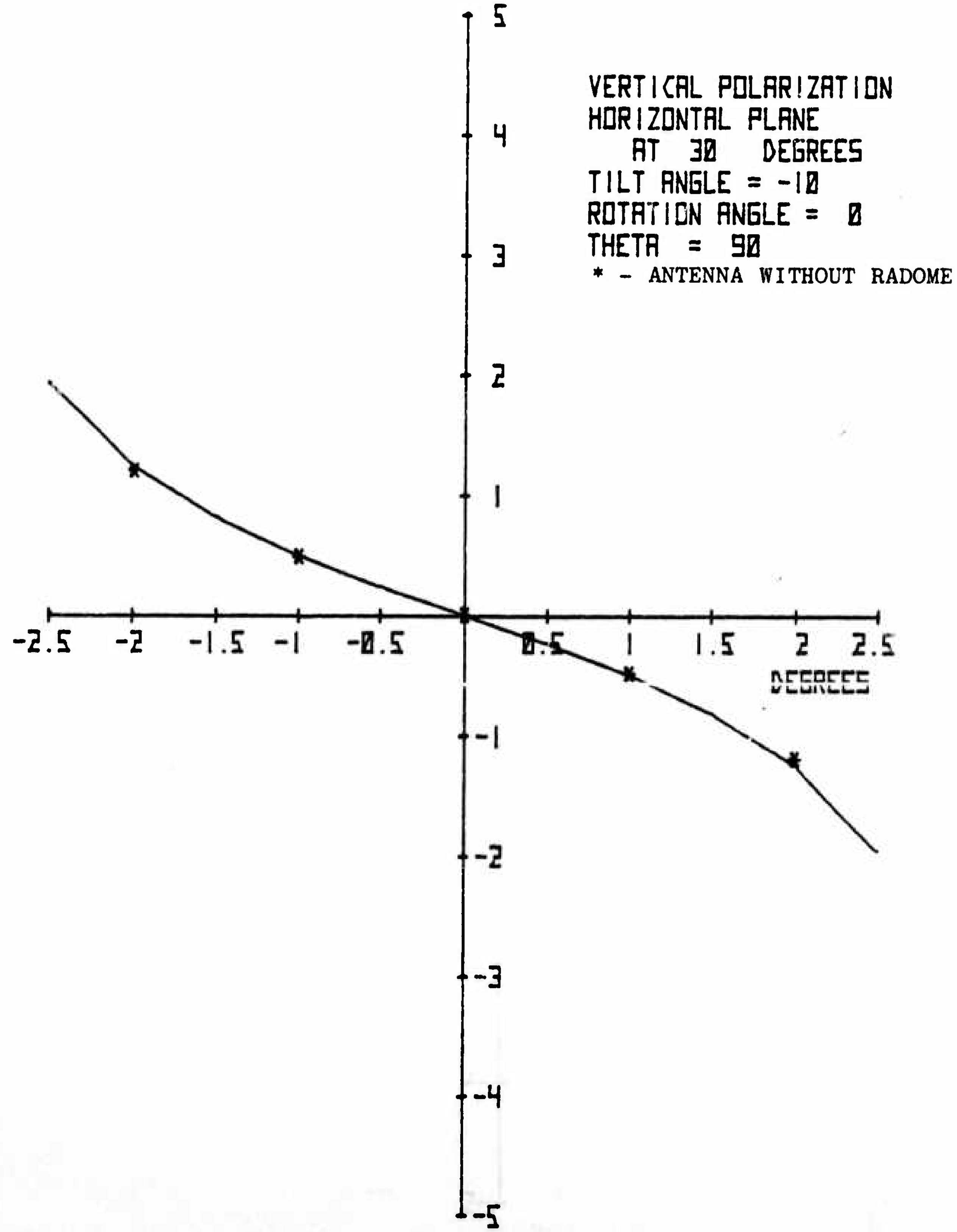


Figure 65. Off-Axis Angle Error Slope

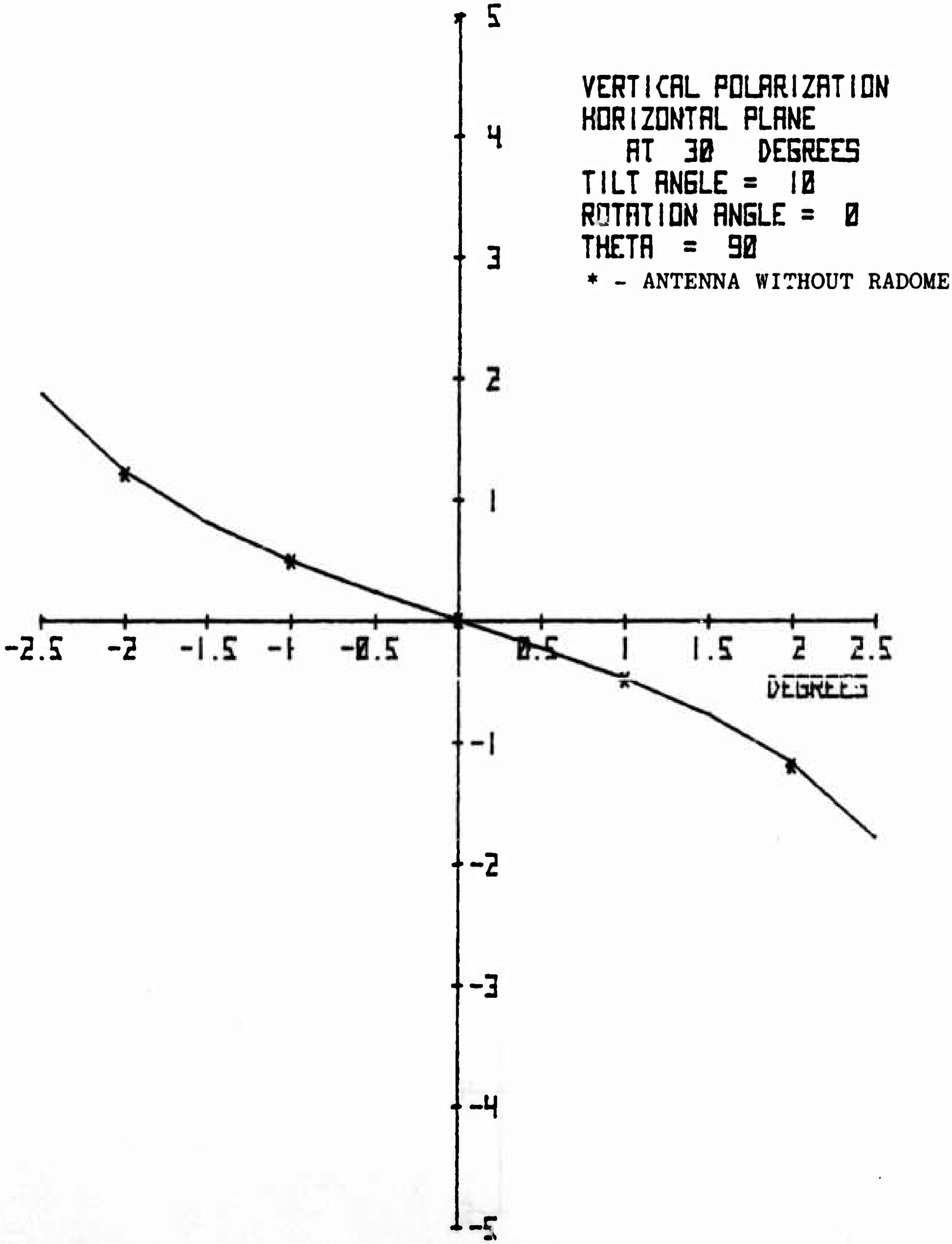


Figure 66. Off-Axis Angle Error Slope

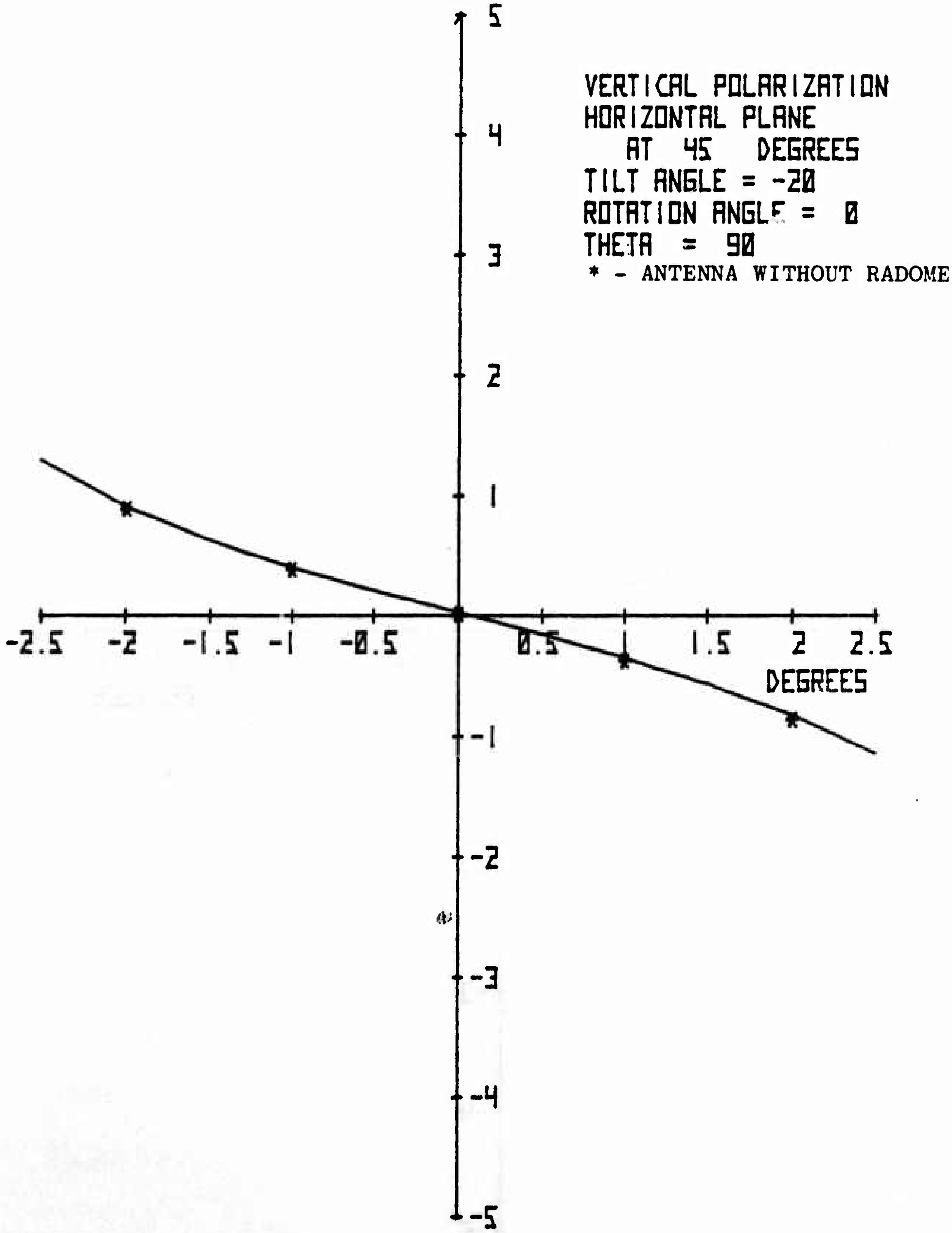


Figure 67. Off-Axis Angle Error Slope

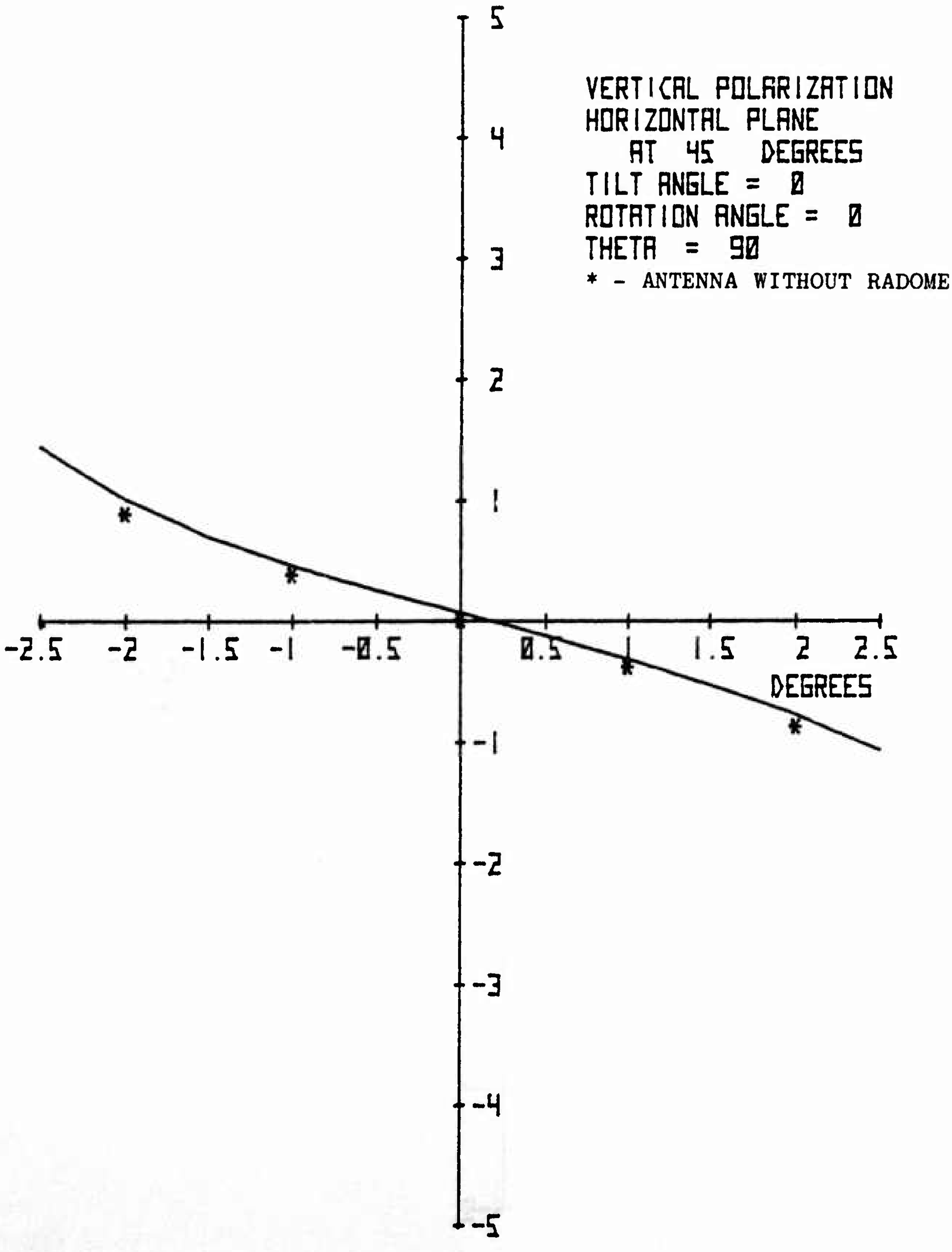


Figure 68. Off-Axis Angle Error Slope

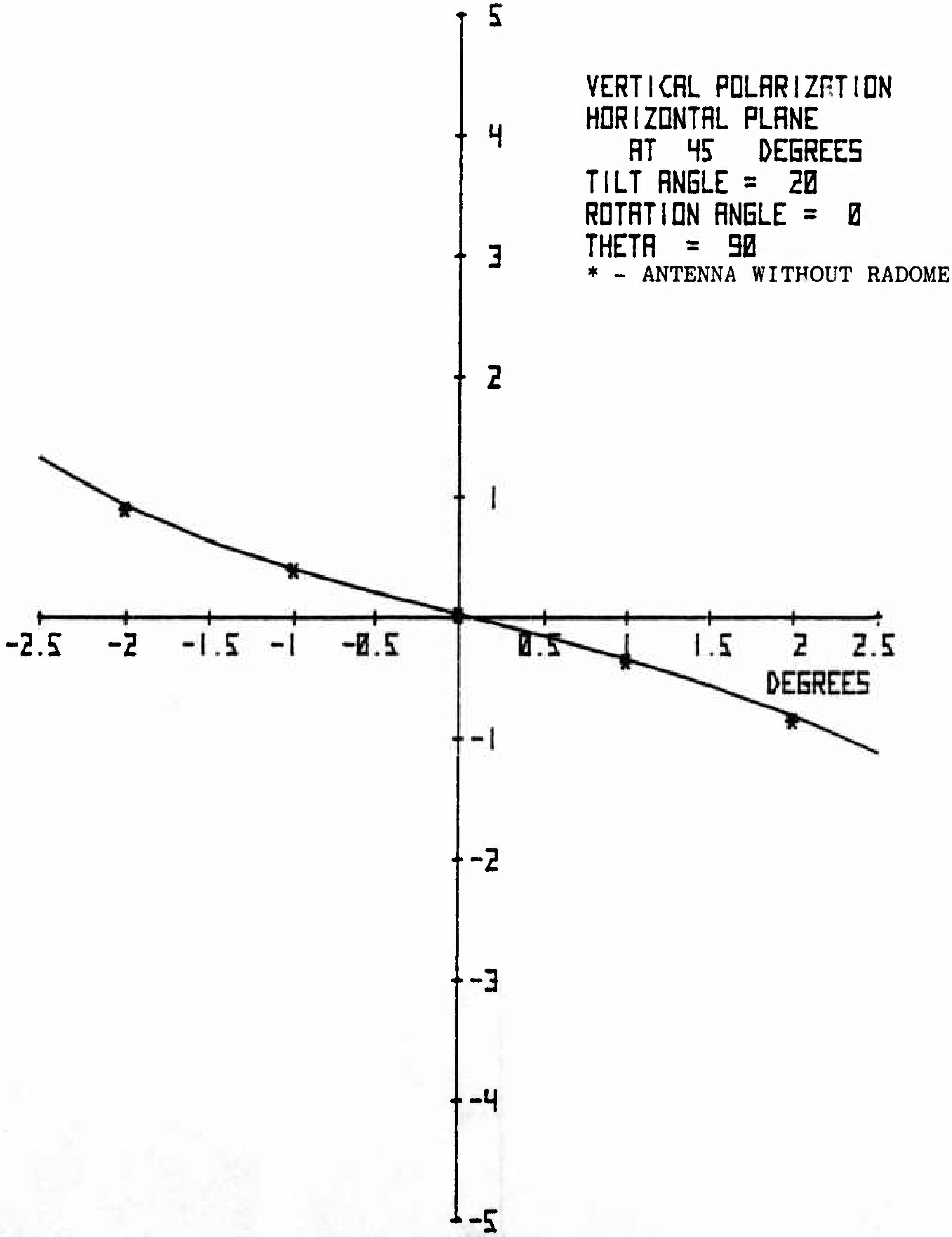


Figure 69. Off-Axis Angle Error Slope

6. ANTENNA/IMU RELATIVE MOTION

This study determined the magnitude of relative motion between the SPN/GEANS IMU and EAR antenna phase center. Relative motion values must be known in order to achieve desired radar performance characteristics. A portion of the relative motion is due to the type of mounting installation used. The preferred B-52 installation concept is too hard mount the antenna to the IMU on a common assembly and hard mount the assembly to airplane structure. Figure 70 is a sketch detailing this mounting assembly. Requirements defined by Westinghouse for relative motion were 100 μrads allowable in relative rotation and 4.0 x 10^{-4} ft. allowable in relative translation. Results of the study determined worse case relative rotation to be 53.1 μrads rms, which is well within the requirement. The worst case relative translation was 4.2 x 10^{-4} ft, which is slightly excessive. Two solutions are suggested to bring the translation motion within required limits.

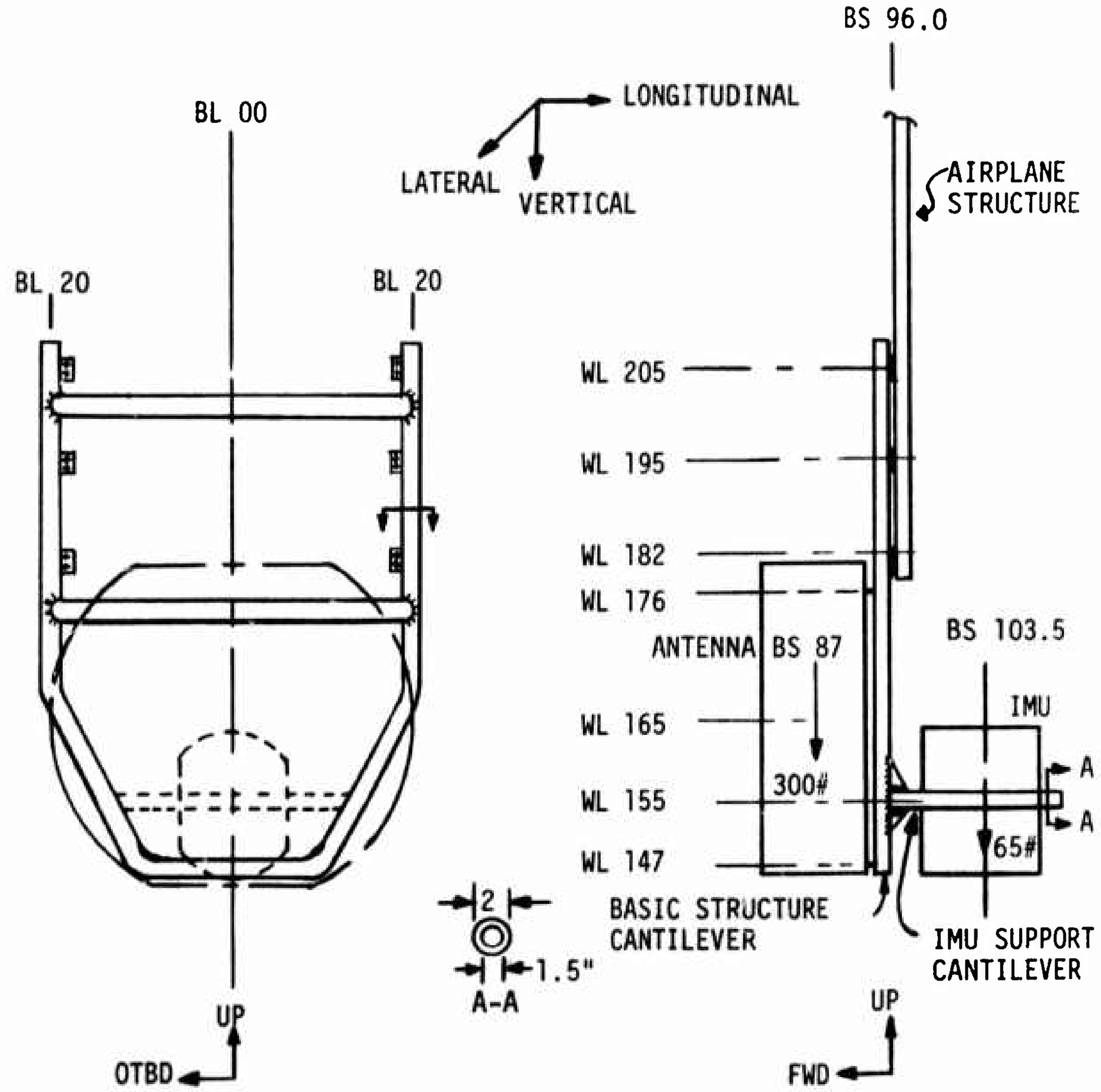
The electronically agile radar (EAR) requires motion compensation of the antenna phase center. An inertial motion unit (IMU or GEANS) is used to measure motion to be compensated. Because of inherent physical restraints the IMU must be located remote from the antenna phase center. This condition permits relative motion between the IMU and antenna phase center if the connecting structure is flexible. The problem is to determine if the magnitude of relative motion in a B-52 installation precludes achieving desired radar performance characteristics.

Two B-1 EAR installation concepts have been analyzed by Westinghouse (Ref. Antenna/IMU Mounting for Motion Compensation, EAR Memo #177). One concept combines the antenna and IMU on a common structural assembly and, in turn, mounts the assembly to airplane structure through vibration isolators. A second concept installs the antenna and IMU as separate installations with the antenna hard mounted and the IMU mounted on vibration isolators. As expected, the first concept results in significantly lower relative motion than the second concept. The comparative analyses serve to verify that when mounting the two units separately (IMU on isolators and the antenna hard mounted) most of the relative motion is induced through isolator flexibility. The analyses indicated that for the B-1 installation, either concept will maintain relative motion within acceptable limits.

Presently the preferred B-52 installation concept is to hard mount the antenna and IMU on a common structural assembly and, in turn, hard mount the assembly to airplane structure. The concept is shown in Figure 70. The present IMU design (Ref. Honeywell Dwg. DGG8099A) contains its own built-in isolation system; therefore, the IMU will be isolated while the antenna will be hard mounted. This installation corresponds to the second concept analyzed by Westinghouse. Additionally, the IMU isolation system used in the B-52 installation will be the same one that was in the Westinghouse relative motion analysis. Since it has been shown that most of the relative motion is induced through isolator flexibility, a reasonable estimate of isolator induced relative motion in the B-52 can be made directly from B-1 analysis results. In this analysis relative motion caused by both isolator and B-52 EAR mounting assembly flexibility will be analyzed.

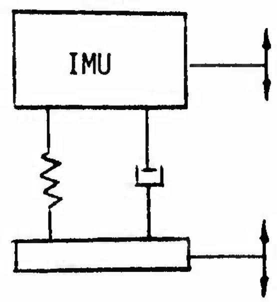
7. RELATIVE MOTION INDUCED BY ISOLATION FLEXIBILITY

Relative motion induced by isolator flexibility can be estimated by considering the relationship between response and excitation of the following base excited system.



NOTE: INFORMATION CONTAINED IN THIS FIGURE HAS BEEN SCALED FROM SK-W-AI-452 AND IS THEREFORE APPROXIMATE.

Figure 70. Mounting Assembly



$$W_{X_R}(f)_{B-1} = T^2(f)_{IMU}\, W_{X_B}(f)_{B-1}$$

where

$W_{X_R}(f)_{B-1}$ = Spectra of IMU response to B-1 excitation

$T(f)_{IMU}$ = Transfer function of IMU isolation system

$W_{X_B}(f)_{B-1}$ = Spectra of B-1 excitation

$$\therefore \quad T^2(f)_{IMU} = \frac{W_{X_R}(f)_{B-1}}{W_{X_B}(f)_{B-1}}$$

Since the IMU vibration isolation system used for the B-1 installation is the same one that will be used for the B-52 installation, the following relationship exists:

$$T^2(f)_{IMU} = \frac{W_{X_R}(f)_{B-52}}{W_{X_B}(f)_{B-52}}$$

$$\therefore \quad \frac{W_{X_R}(f)_{B-1}}{W_{X_B}(f)_{B-1}} = \frac{W_{X_R}(f)_{B-52}}{W_{X_B}(f)_{B-52}}$$

or

$$W_{X_R}(f)_{B-52} = \frac{\left[W_{X_B}(f)_{B-52}\right]\left[W_{X_R}(f)_{B-1}\right]}{W_{X_B}(f)_{B-1}}$$

further

$$\int_{f_1}^{f_2} W_{X_R}(f)_{B-52}\,df = \int_{f_1}^{f_2} \frac{\left[W_{X_B}(f)_{B-52}\right]\left[W_{X_R}(f)_{B-1}\right]df}{W_{X_B}(f)_{B-1}}$$

or

$$(X_R)_{B-52} = \left[\int_{f_1}^{f_2} \frac{\left[W_{X_B}(f)_{B-52}\right]\left[W_{X_R}(f)_{B-1}\right]df}{W_{X_B}(f)_{B-1}}\right]^{1/2}$$

Since the various response spectras ($W_{X_R}(f)_{B-1}$) are not known, the above integral cannot be evaluated. However, when considering that most of the relative motion is induced through isolator flexibility it is seen that the interval of integration is small. Therefore, a reasonable approximation of the integral can be made by replacing the spectras with their respective RMS values evaluated over the frequency range of the isolation system.

$$\therefore \quad (X_R)_{B-52} \approx \left[\frac{(X_B)_{B-52}}{(X_B)_{B-1}}\right] (X_R)_{B-1}$$

where

$(X_R)_{B-52}$ - B-52 IMU RMS response due to isolator motion

$(X_B)_{B-52}$ - B-52 IMU base RMS excitation

$(X_R)_{B-1}$ - B-1 IMU RMS response due to isolator motion

$(X_B)_{B-1}$ - B-1 IMU base RMS excitation

Using the above relationship, if B-52 and B-1 excitations are known, then B-52 IMU/ANTENNA isolator induced relative motions can be computed from B-1 IMU/ANTENNA relative motions.

RMS EXCITATIONS

For isolation system induced motions the excitation frequency band width will be taken to be the range of frequencies that the isolation system has a transmissibility greater than one. The upper frequency of the range is calculated as follows:

$$\text{Upper frequency} = (\sqrt{2})(40) \quad 56 \text{ Hz}$$

The lower frequency will be assumed to be zero. Therefore, the frequency range of interest if 0 - 56 Hz. Table XXI summarizes the RMS content of accelerations for the B-1 and B-52 in this frequency range.

TABLE XXI - RMS EXCITATIONS

Airplane	Gust Excitations g_{rms}	Random Structural Excitations g_{rms}
B-1	.795	.30
B-52	.381	.21

B-1 EAR RELATIVE MOTIONS

B-1 EAR relative motions are summarized in the following table.

TABLE XXII - B-1 EAR RELATIVE MOTIONS

Excitations*	**RMS RELATIVE MOTION	
	Translation Ft.	Rotation μRAD
Gust	4.0×10^{-4}	31.0
Random Structural	3.6×10^{-4}	68.0

* Rotational excitations presented for the B-1 are not known for B-52 and therefore will not be considered in this analysis. Also, sinusoidal excitation presented in the B-1 analysis will not be considered because it is less representative of structural excitations than the random excitation.

** Use of these values to determine isolator induced motion assumes that all motion reported in the Westinghouse analysis is a result of isolator flexibility.

B-52 EAR RELATIVE MOTIONS

B-52 EAR relative motions can now be calculated using the foregoing formula and data.

Motion induced by gust excitation becomes:

$$(X_R)_{B-52}(\text{TRANSLATION}) = (.381/.795)(4.0) = \underline{1.9 \times 10^{-4}\text{ Ft.}}$$

$$(X_R)_{B-52}(\text{ROTATION}) = (.381/.795)(31) = \underline{14.8\ \mu\text{RADS}}$$

Motion induced by random structural excitation becomes:

$$(X_R)_{B-52}(\text{TRANSLATION}) = (.21/.3)(3.6) = \underline{2.5 \times 10^{-4}\text{ Ft.}}$$

$$(X_R)_{B-52}(\text{ROTATION}) = (.21/.3)(68) = \underline{48\ \mu\text{RADS}}$$

8. RELATIVE MOTION INDUCED BY MOUNTING STRUCTURE ASSEMBLY

A first order approximation of the magnitude of relative motion resulting from B-52 mounting structure flexibility can be estimated by considering relative deflections of the cantilevers that make up the B-52 mounting structure. From inspection of Figure 70, it can be seen that the assembly is the most flexible about the lateral axis. As a result it will be assumed that rotations about both longitudinal and vertical axes are less than rotation about the lateral axis. Additionally, it will be assumed that the first mode of vibration is the principal contributor to relative motions.

Angular Relative Motion

Angular rotations from the fundamental bending modes of the basic structure cantilever and IMU support structure cantilever can be calculated as follows:

Assumptions: CG's of antenna and IMU located at geometric center of antenna and IMU respectively. It is further assumed that the antenna phase center is near antenna CG.

Effective length of the basic structure cantilever taken to be from WL 182.

Antenna/IMU combined load point = $\frac{(30)(300)+(40)(65)}{365} = 31.78$ In.

$\therefore$ Combined load point is at WL = 195 - 31.78 = 163.22.

Fundamental frequency of a cantilever beam = $\sqrt{\frac{3EI}{m\ell^3}}$

The basic structure cantilever has the following parameters:

$$E = 30 \times 10^6 \text{ Lbs/In}^2.$$

$$I = 2\left[\frac{\pi}{64}(d_o^4 - d_i^4)\right] = 2\left[\frac{3.14}{64}\left((2)^4 - (1.5)^4\right)\right] = 1.07 \text{ In}^4.$$

$$\ell = 182 - 163.22 = 18.78 \text{ In.}$$

$$m = \frac{365}{386} = .95 \text{ Lb-Sec}^2\text{/In.}$$

$$\omega_n = \sqrt{\frac{(3)(30\times10^6)(1.07)}{(.95)(18.73)^3}} = 124.21 \text{ Rads/Sec.}$$

$$f_n = 19.8 \text{ Hz.}$$

The IMU support structure cantilever has the following parameters:

$$E = 30 \times 10^6 \text{ Lbs/In}^2.$$

$$I = 1.07 \text{ In}^4.$$

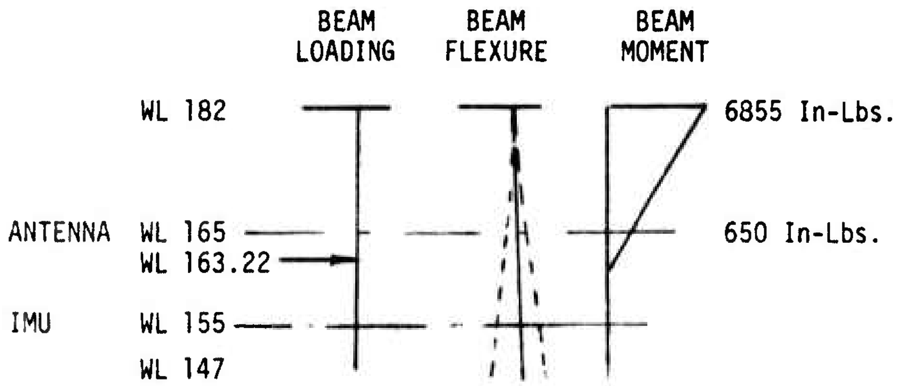
$$m = \frac{65}{386} = .17 \text{ Lbs-Sec}^2\text{/In.}$$

$$\ell \approx 103.5 - 96 = 7.5 \text{ In.}$$

$$\omega_n = \sqrt{\frac{(3)(30\times10^6)(1.07)}{(.17)(7.5)^3}} = 1158.76 \text{ Rads/Sec.}$$

$$f_n = 184.5 \text{ Hz.}$$

Since 19.8 Hz $\ll$ 184.5 Hz the IMU support structure can be considered essentially rigid when compared to the basic support structure. Angular rotation about the lateral axis occurring at the antenna CG may be calculated if it is assumed that the antenna also moves rigidly with the basic structure. The calculations are as follows:



$$\text{Antenna Rotation} = \frac{A_{M(ANT)}}{EI} = \frac{(650)(17)+(17/2)(6205)}{(30\times10^6)(1.07)} = .001987 \text{ Rads.}$$

$$\text{IMU Rotation} = \frac{A_{M(IMU)}}{EI} = \frac{(6855)(18.78/2)}{(30\times10^6)(1.07)} = .002001 \text{ Rads.}$$

Relative Antenna/IMU Rotation = .002001 - .001987 = 14.0 μRads.

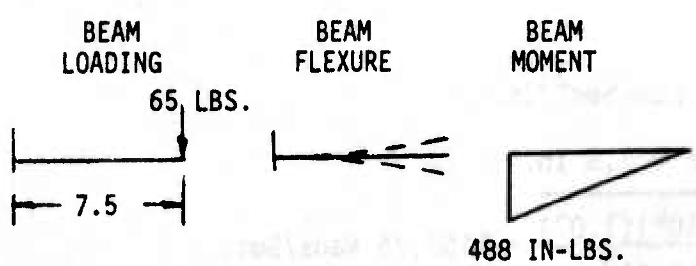
The above calculations are based on a one g acceleration field. From unpublished B-52 vibration data the acceleration power spectral level at this location and at 20 Hz for loaded structure is approximately .0003 g^2/Hz. Assuming a Q of 3 the mean square response becomes:

$$g^2 = \frac{\pi Q f_n W}{2} = \frac{(3.14)(3)(20)(.0002)}{2} \approx .0283$$

$$g_{rms} \approx .17$$

$\therefore$ Relative rotation = (14)(.17) $\approx$ 2.4 μRads rms.

Relative rotation induced from the IMU support cantilever can likewise be determined by the following calculations.



$$\text{IMU Rotation} = \frac{(488)(7.5/2)}{(30\times10^6)(1.07)} = .000057 \text{ Rads.}$$

It will be assumed that antenna rotation is zero at this frequency.

As before the calculations are based on a one g acceleration field. The acceleration power spectral level at this frequency is approximately .0001 g^2/Hz. Assuming a Q of 3 the mean square response becomes:

$$g^2 = \frac{(3.14)(3)(186)(.0001)}{2} = .0876$$

$$g_{rms} = .3$$

Relative motion = (.000057)(.3) = 17 μRads rms.

Translational Relative Motion

Relative translation will be calculated using standard cantilever beam deflection equations. Longitudinal deflections at antenna and IMU CG's can be calculated as follows:

$$\text{Antenna deflection} = \frac{P(\ell - X_A)^2}{6EI}(3b - \ell + X_A)$$

$$\text{IMU deflection} = \frac{Pb^2}{6EI}(3\ell - 3X_I - b)$$

where

P = 365 Lbs.

$E = 30.\text{x}10^6$ Lbs/In^2.

I = 1.07 In^4.

ℓ = 35.0 In.

b = 18.78 In.

X_A = 18.0 In.

X_I = 8.0 In.

$$\text{Antenna deflection} = \frac{(365)(35-18)^2}{(6)(30\text{x}10^6)(1.07)}[(3)(18.78) - 35 + 18] = .0215 \text{ In.}$$

$$\text{IMU deflection} = \frac{(365)(18.78)^2}{(6)(30\text{x}10^6)(1.07)}[(3)(35) - (3)(8) - 18.78] = .04158 \text{ In.}$$

Relative translation = .04158 - .0215 = .0201 In.

g_{rms} = .17

Relative translation = $\frac{(.17)(.0201)}{12}$ = 2.8 x 10^{-4} Ft.

Relative translation induced from the fundamental IMU support cantilever mode (185 Hz) can likewise be determine by the following calculations.

$$\text{IMU deflection} = \frac{P\ell^3}{3EI}$$

$P = 65$ Lbs.

$\ell = 7.5$ In.

$E = 30.\times 10^6$ Lbs/In^2.

$I = 1.07$ In^4.

$$\text{IMU deflection} = \frac{(65)(7.5)^3}{(3)(30\times 10^6)(1.07)} = 0.00028 \text{ In.}$$

$g_{rms} \approx .3$

Assuming that antenna phase center translation is zero at this frequency, the relative translation becomes:

$$\text{Relative translation} = \frac{(.3)(.00028)}{12} = 0.07 \times 10^{-4} \text{ Ft. rms.}$$

Total rms relative rotation due to airplane structure

$$= \sqrt{(2.4)^2 + (17)^2} = 17.2 \ \mu\text{Rads rms.}$$

Total rms relative translation due to airplane structure

$$\approx 2.8 \times 10^{-4} \text{ Ft. rms.}$$

9. TOTAL RELATIVE MOTION OF B-52 EAR INSTALLATION

Summing isolator and airplane structural contributions, total B-52 relative rotations and translations become:

$$\text{Total rms relative rotations} = \sqrt{(14.8)^2+(48)^2+(17.2)^2} = \underline{\underline{53.1}} \text{ Rads rms.}$$

$$\text{Total rms relative translations} = \sqrt{(1.9)^2+(2.5)^2+(2.8)^2} = 4.2\times 10^{-4} \text{ Ft. rms.}$$

The allowable relative rotation is 100 μrads. Worst case relative rotations of the B-52 installation (53.1 μrads rms) is well within the requirement. The allowable relative translation is 4.0×10^{-4} ft. Worst case relative translation of the B-52 installation (4.2×10^{-4} ft. rms) slightly exceeds the requirement.

Relative translation can be reduced within the requirements by either (1) placing the IMU higher up on the basic installation structure, or (2) by adding a support at the lower end of the basic installation structure. It is recommended that the most feasible of these design changes be incorporated so that relative translations will be reduced within acceptable limits.

10. B-52 FLIGHT CHARACTERISTICS DATA FOR TF/TA IMPLEMENTATION

B-52 flight characteristics data provided for the Westinghouse TF/TA simulation program are contained in this section. The data are presented through definitions, equations, charts, graphs, and schematic diagrams. The data describe longitudinal non-linear aero coefficients for translation in a vertical plane, rotation in the pitch axis, elevator control surface characteristics, pertinent geometric, weight, and inertial data, control column force and displacement data, and pertinent performance and flight envelope data.

a. Aerodynamic Characteristics - Lift Force Coefficient

The dimensionless aerodynamic lift force coefficient is given in terms of its significant components by the equation below.

At a given flight condition:

$$C_L = C_{L_{BASIC}} + \Delta\left(C_L\right)_{\alpha_W = o} + \Delta\left(\frac{dC_L}{d\alpha}\right)_{ELASTIC} \cdot \alpha_{wing} + \frac{dC_L}{dn_z} \cdot n_z$$

$$+ \frac{dC_L}{d\hat{\dot{\alpha}}}\left(\frac{\dot{\alpha}\bar{c}}{2v}\right) + \frac{dC_L}{d\hat{q}}\left(\frac{q\bar{c}}{2v}\right) + \frac{dC_L}{d\Lambda} \cdot \Lambda + K_e \cdot \frac{dC_L}{d\delta_e} \cdot \delta_e$$

$C_{L_{BASIC}}$ = Basic lift coefficient for the right airplane at α = 0°, in free air and with the landing gear retracted.

$\Delta\left(C_L\right)_{\alpha_W = o}$ = Change in basic lift coefficient at α_{wing} = 0° due to aeroelasticity.

$\Delta\left(\frac{dC_L}{d\alpha}\right)_{ELASTIC}$ = Change in slope basic lift coefficient with angle of attack due to the aeroelasticity.

$\frac{dC_L}{dn_z}$ = Change in basic lift coefficient due to aeroelastic inertia relief caused by normal load factor, n_z. (For wings level steady state flight, n_z = 1.0.)

$\frac{dC_L}{d\hat{\dot{\alpha}}}$ = Change in basic lift coefficient due to nondimensionalized rate of change of angle of attack.

$\frac{dC_L}{d\hat{q}}$ = Change in basic lift coefficient due to non-dimensionalized pitch rate.

$\frac{dC_L}{ds}$ = Change in basic lift coefficient due to change in stabilizer angle from $s = 0°$.

$K_e \cdot \frac{dC_L}{d\delta_e}$ = Change in basic lift coefficient due to change in elevator angle from $\delta_e = 0°$.

The lift coefficient data are depicted in Figures 71 through 85. Drag coefficient data are shown in Figure 101.

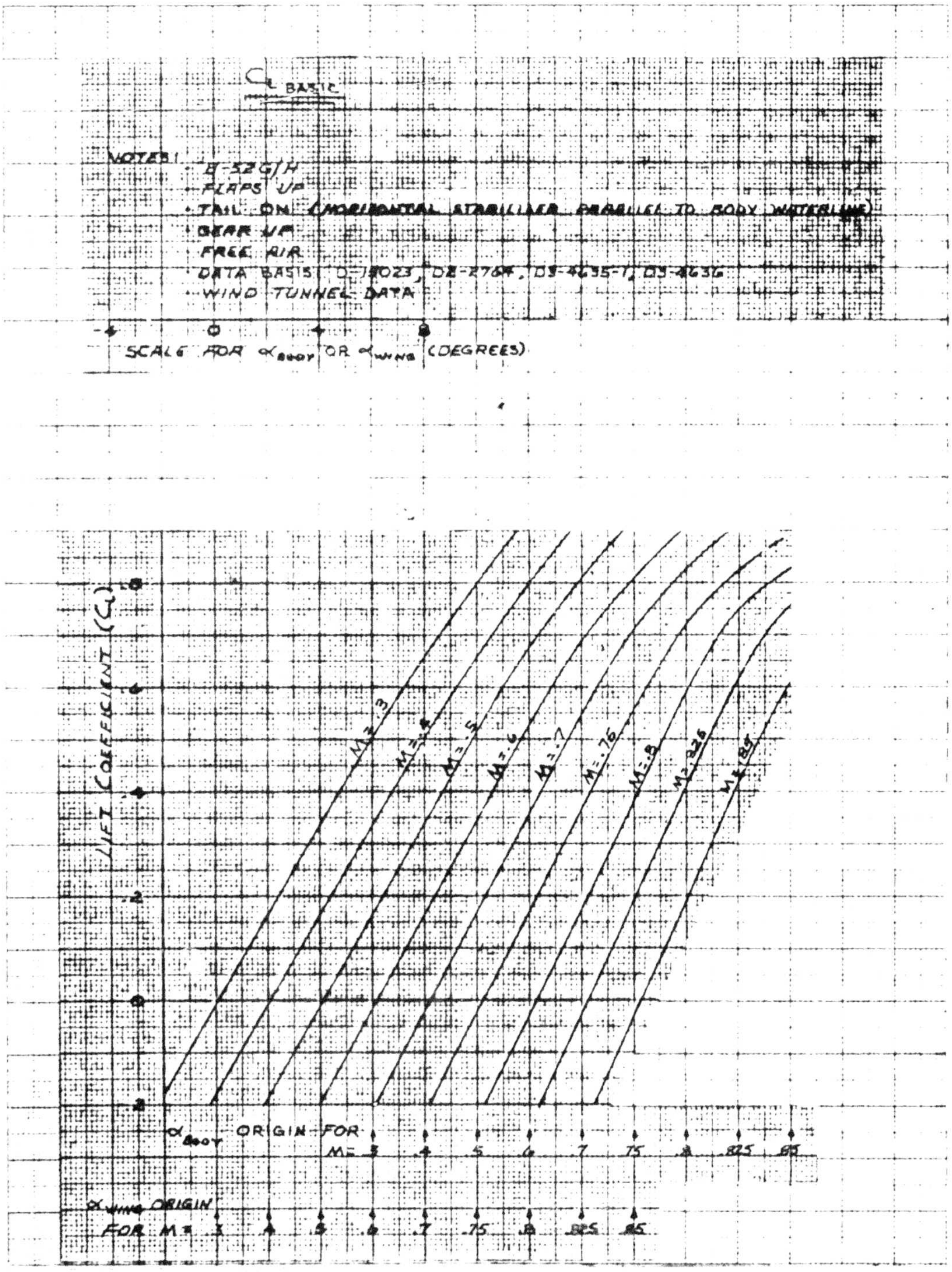


Figure 71. Lift Coefficient - Effect of Angle of Attack on Basic Centerline

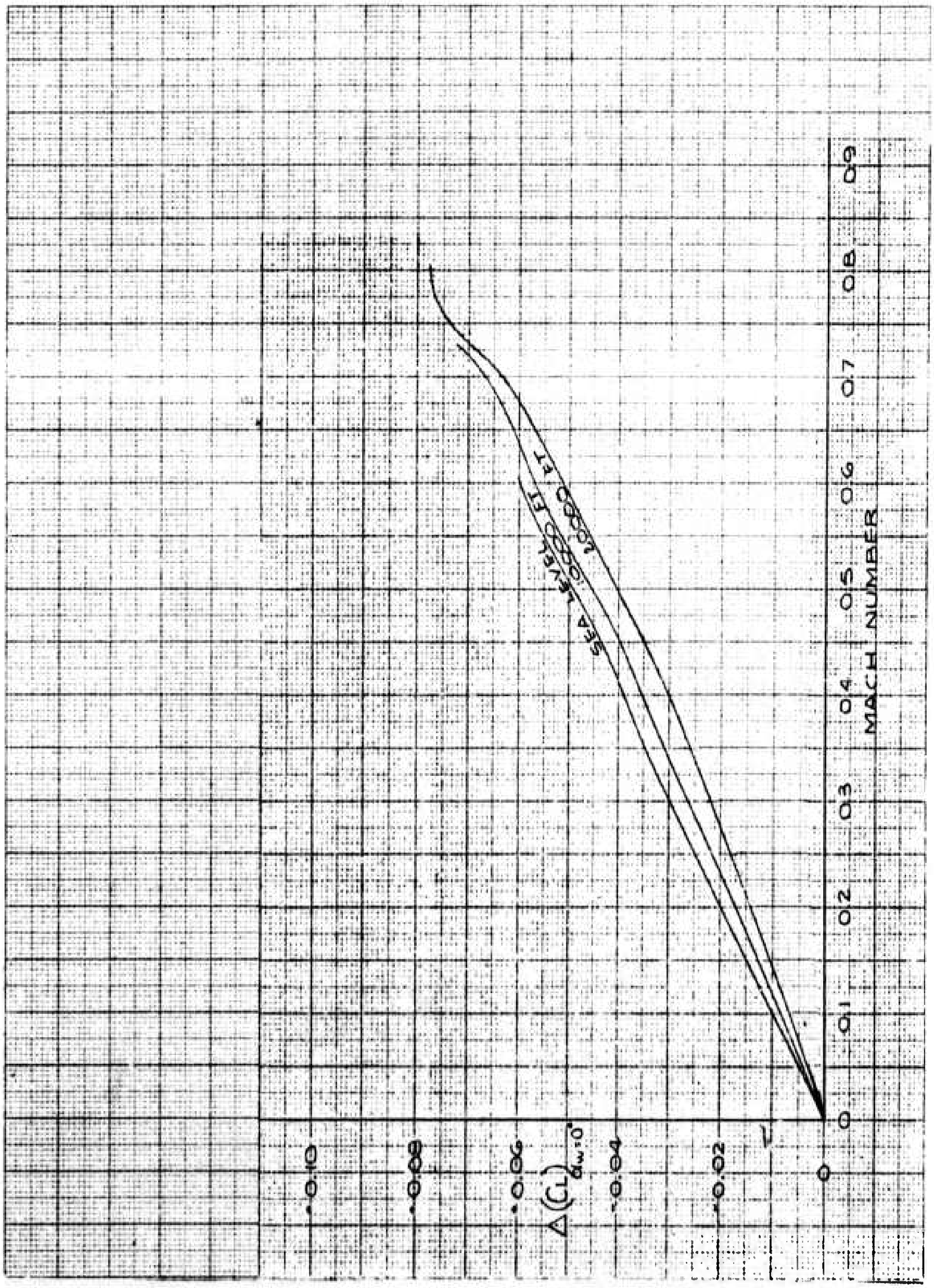


Figure 72. Lift Coefficient - Aeroelastic Effect on $\Delta(C_L)$ awing = 0°

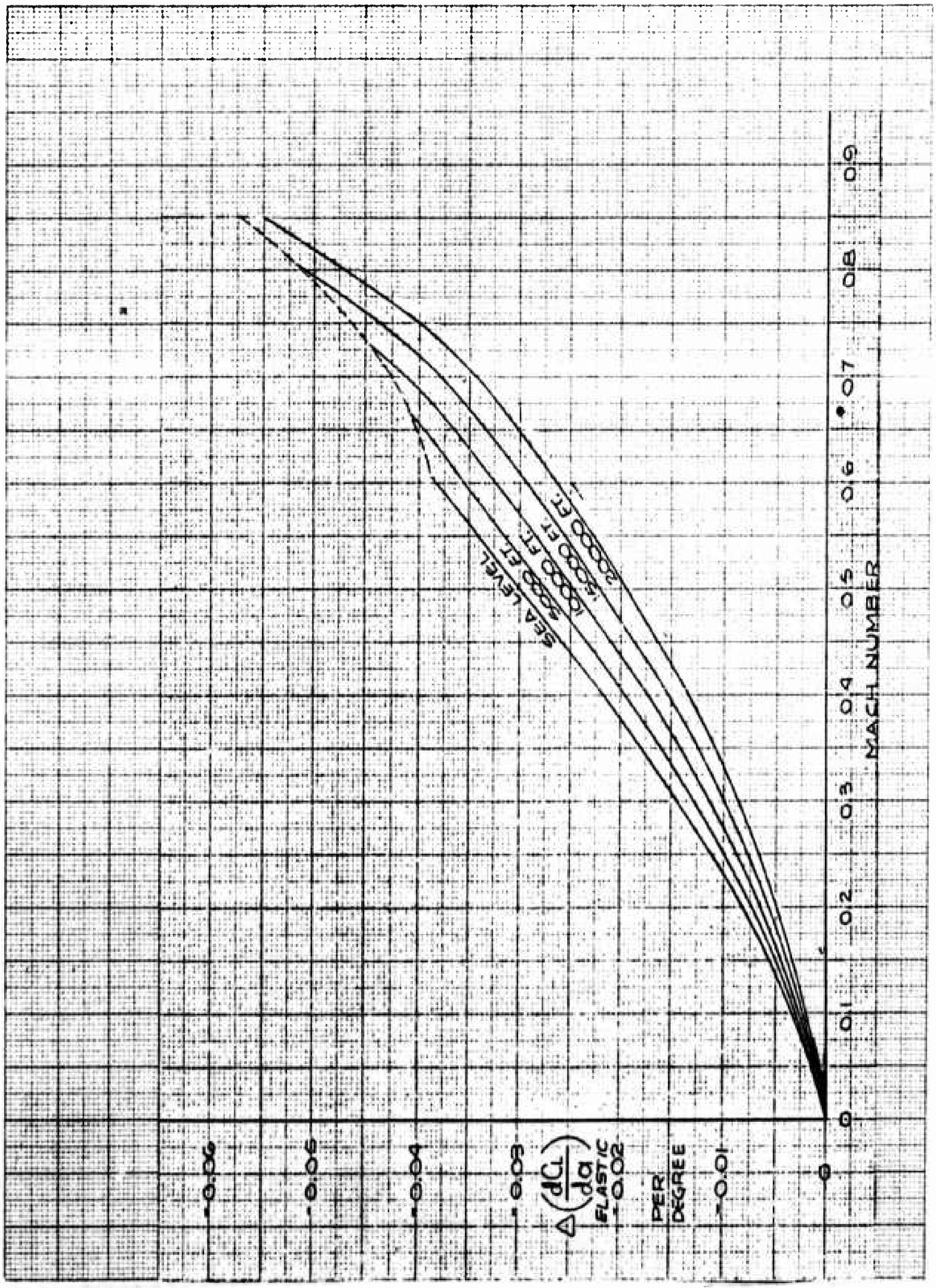


Figure 73. Lift Coefficient - Aeroelastic Effect on $\Delta(dC_L/da)$

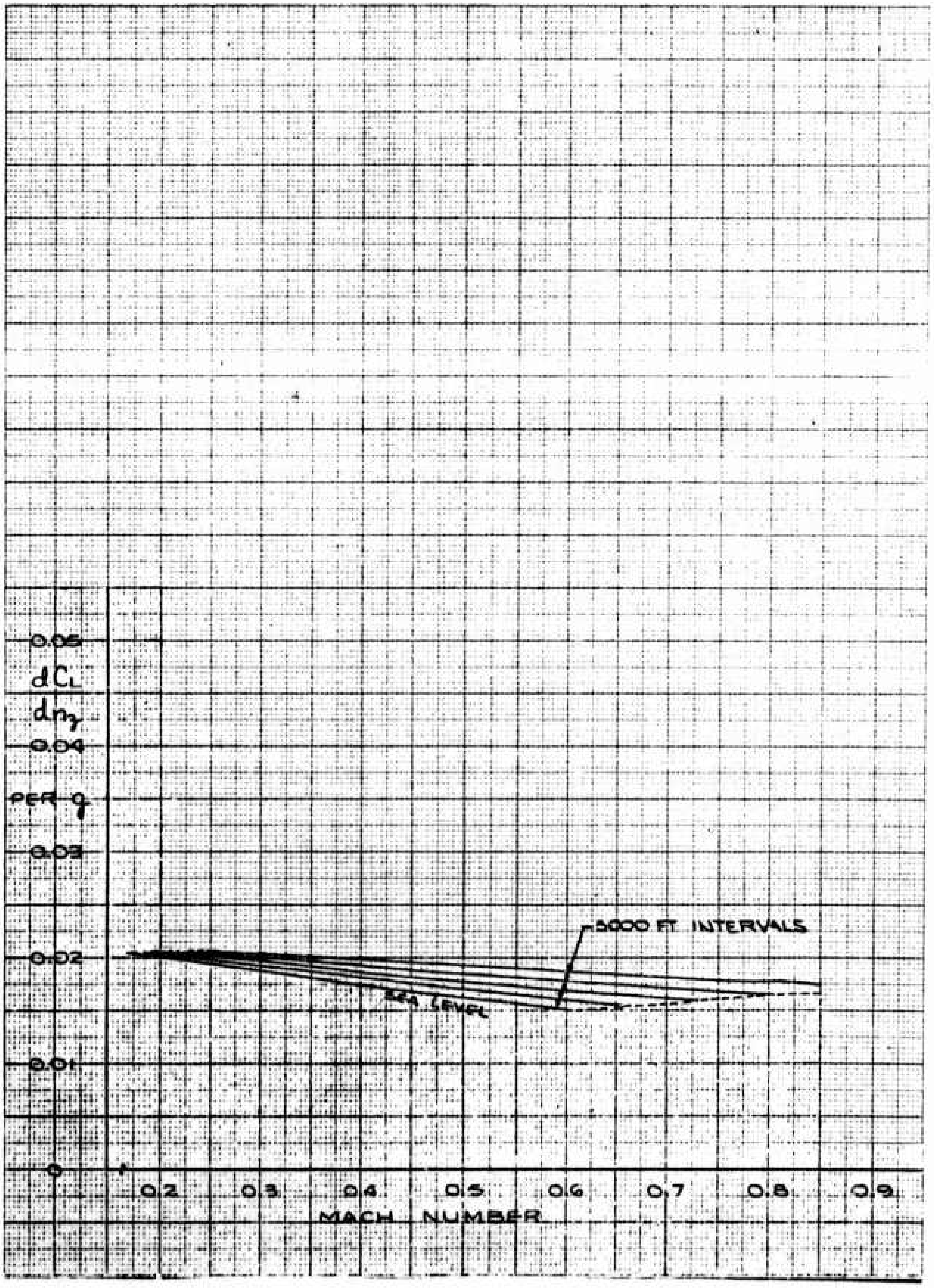


Figure 74. Lift Coefficient - Effect of Normal Load Factor @ 200,000 Lbs.

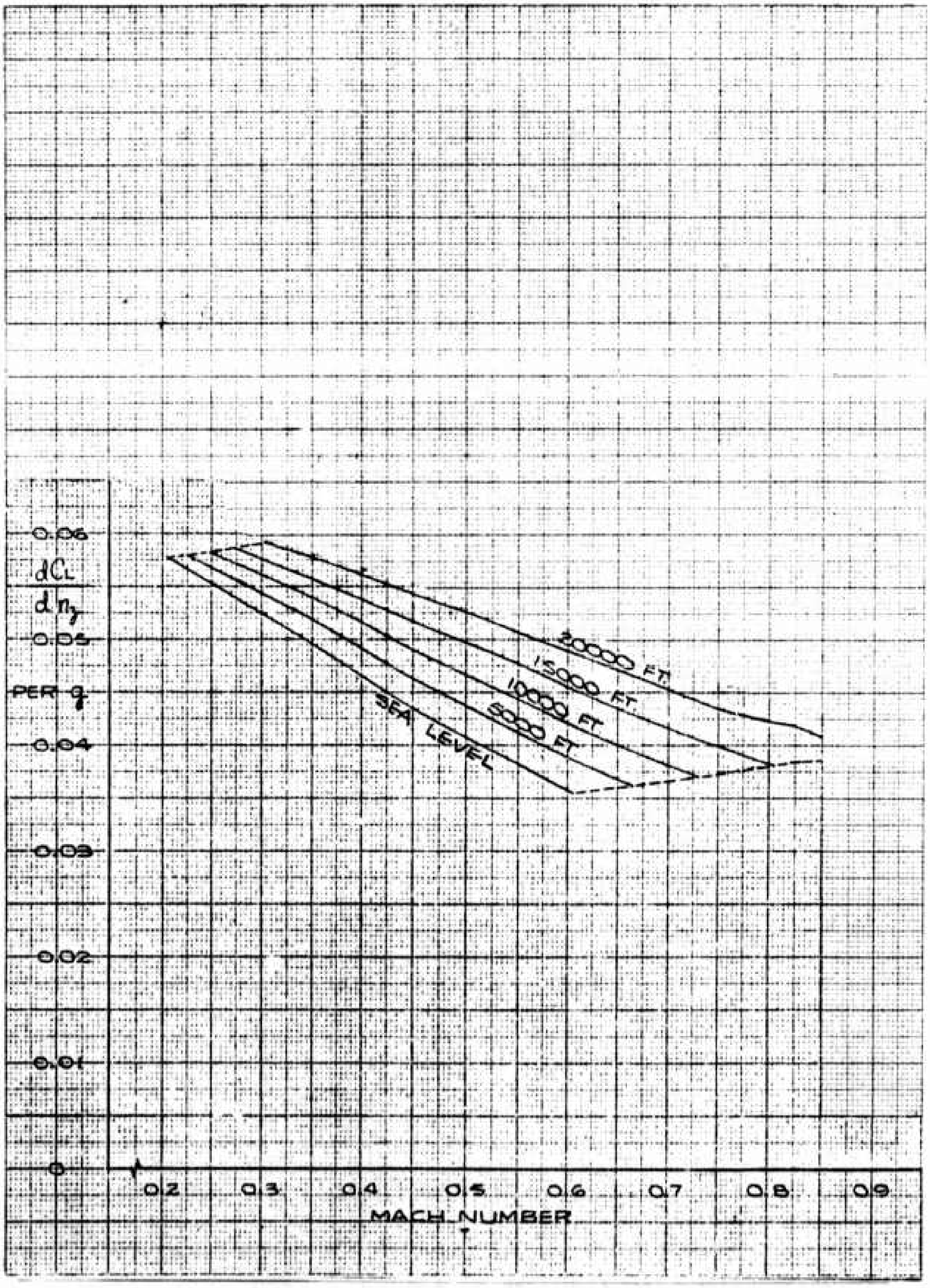


Figure 75. Lift Coefficient - Effect of Normal Load Factor @ 300,000 Lbs.

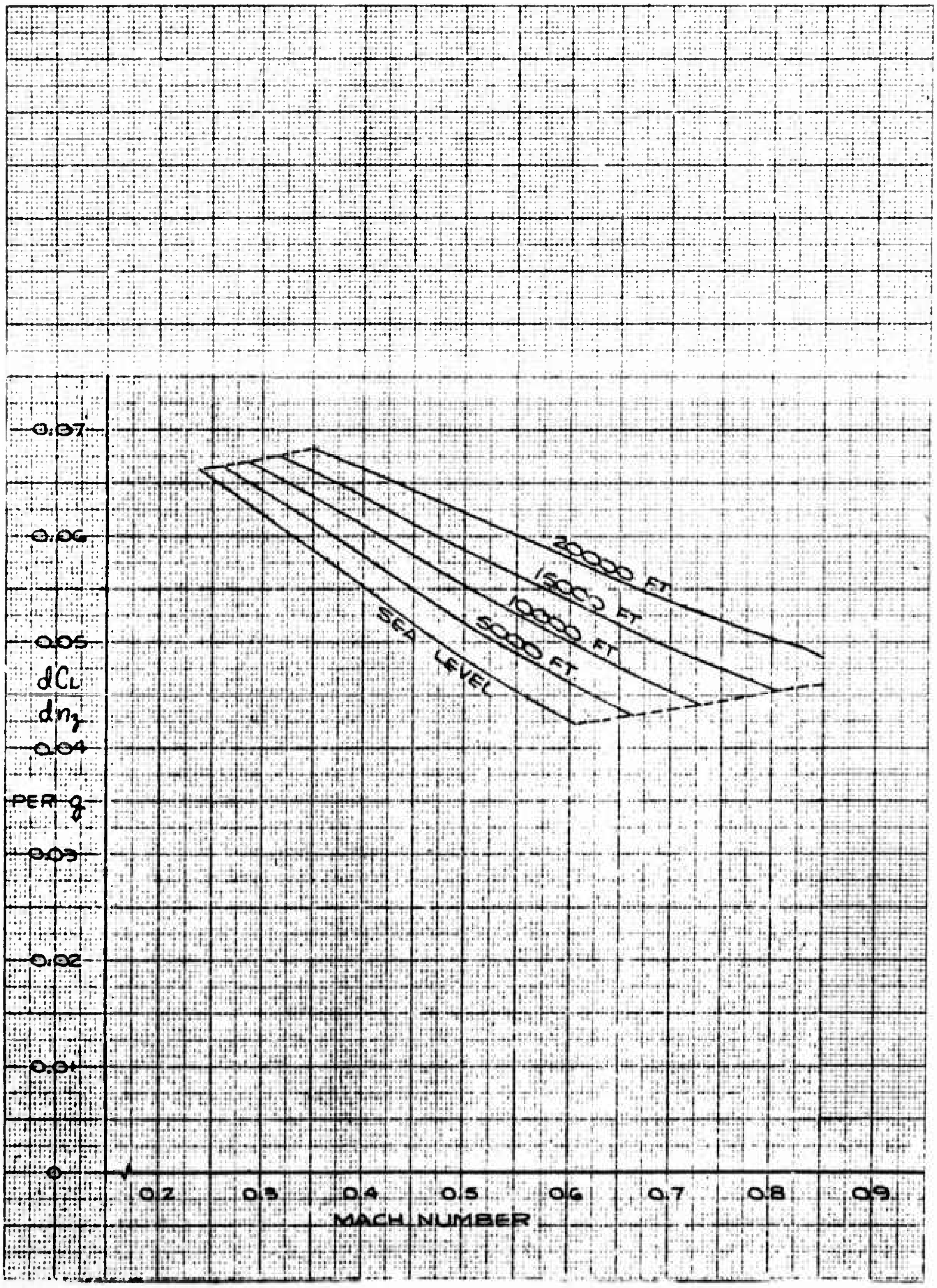


Figure 76. Lift Coefficient - Effect of Normal Load Factor @ 400,000 Lbs.

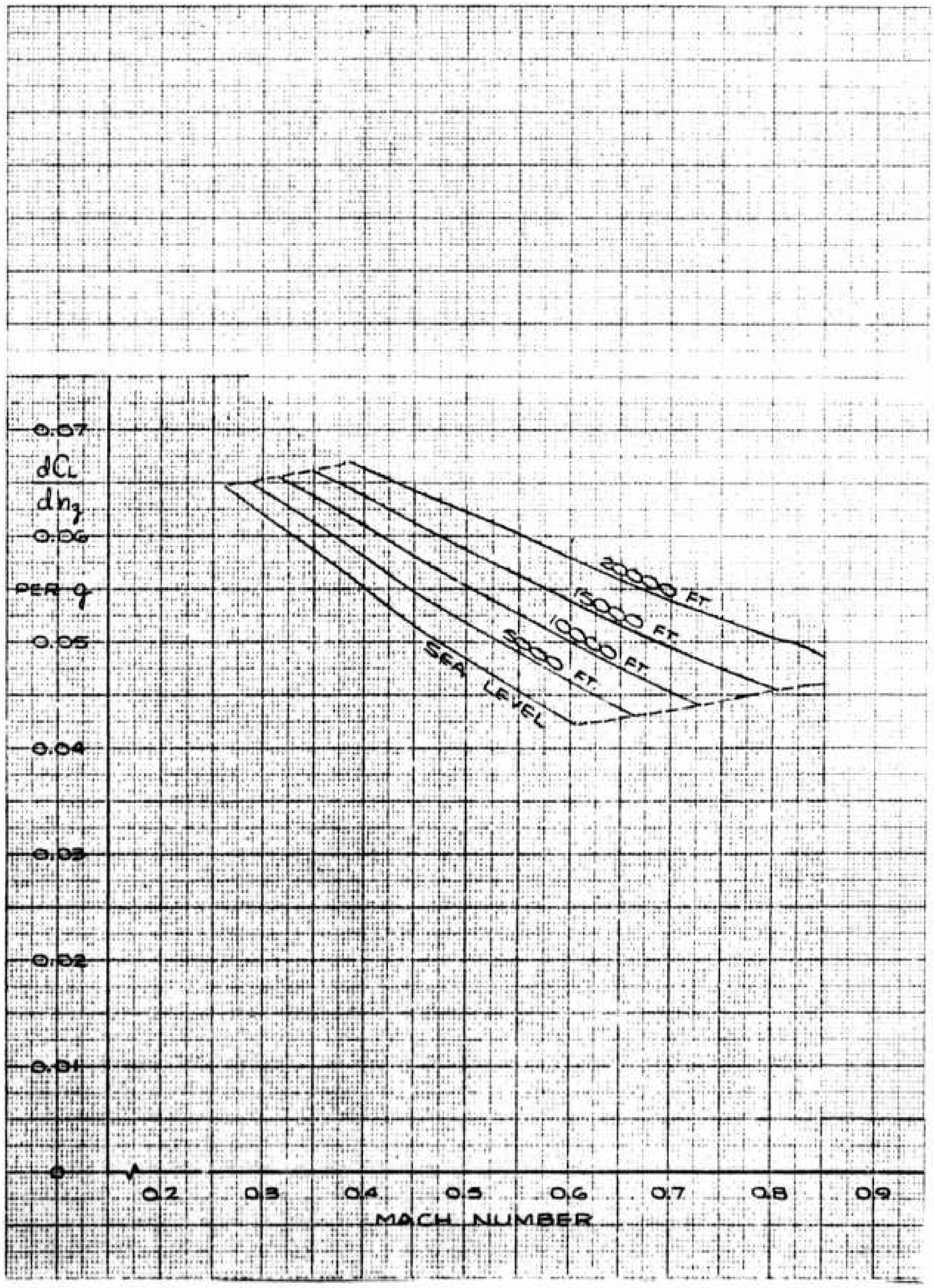


Figure 77. Lift Coefficient - Effect of Normal Load Factor @ 488,000 Lbs.

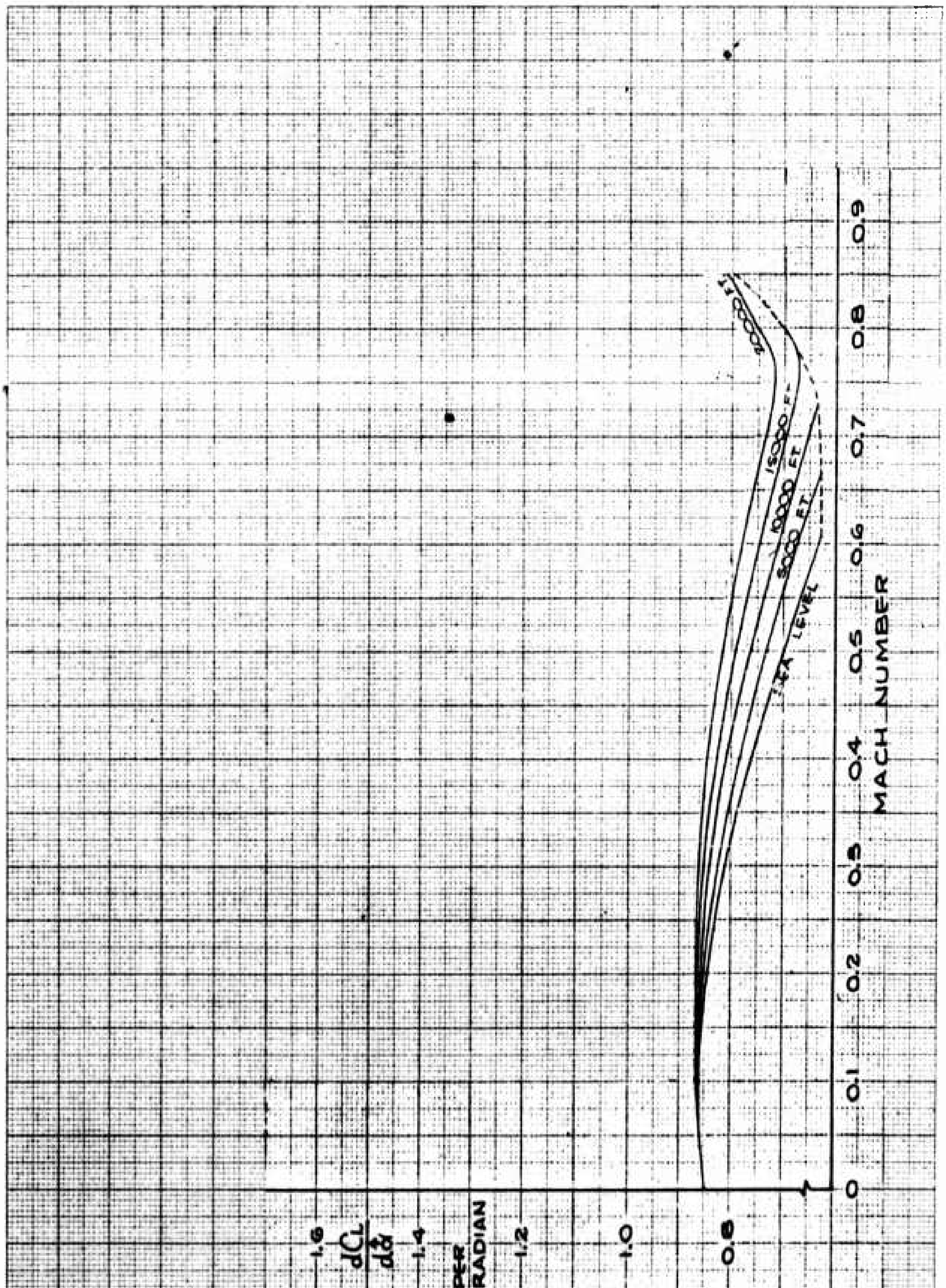


Figure 78. Lift Coefficient - Effect of $\hat{\alpha}$

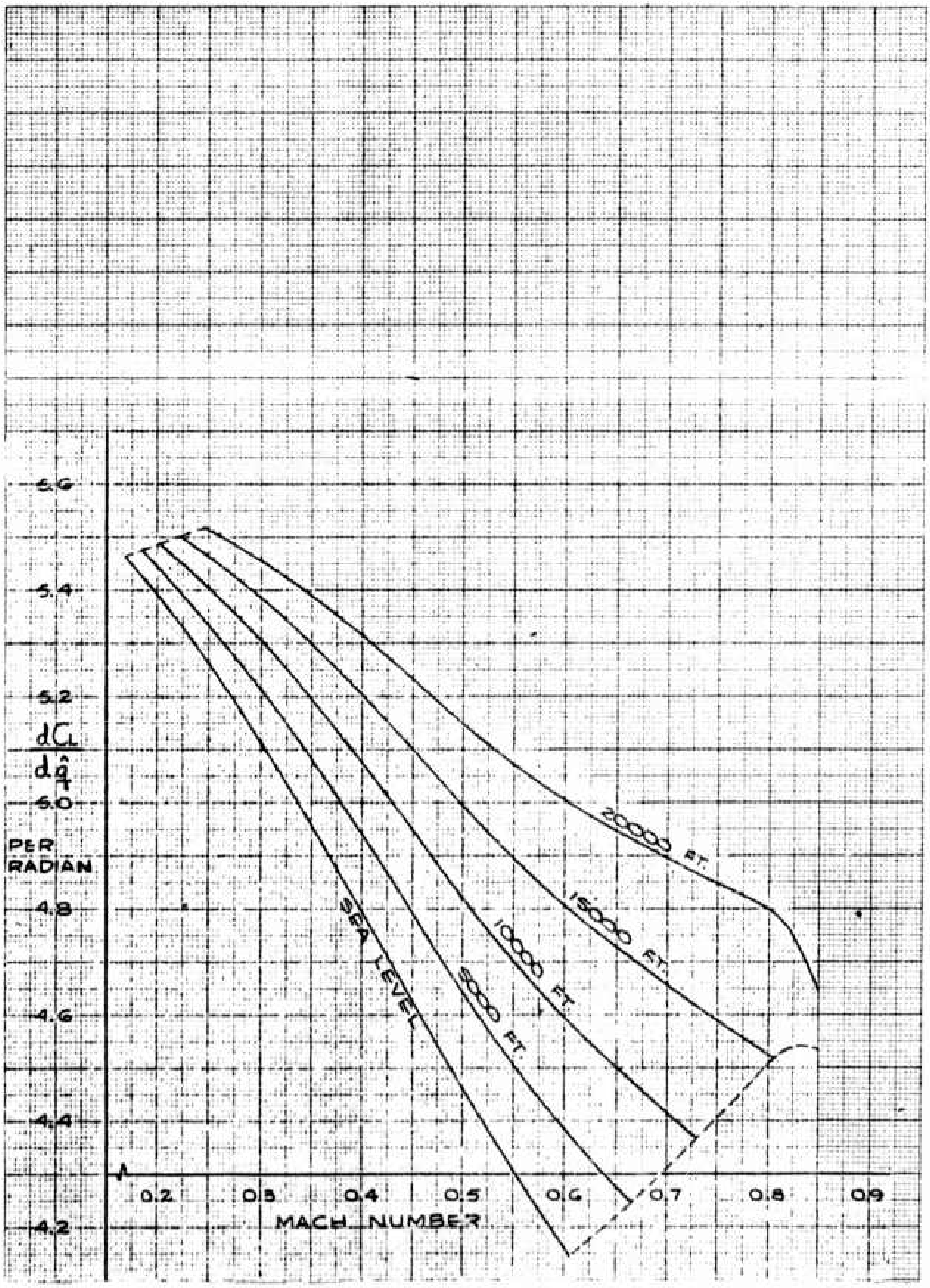


Figure 79. Lift Coefficient - Effect of $\hat{q}$ @ 200,000 Lbs.

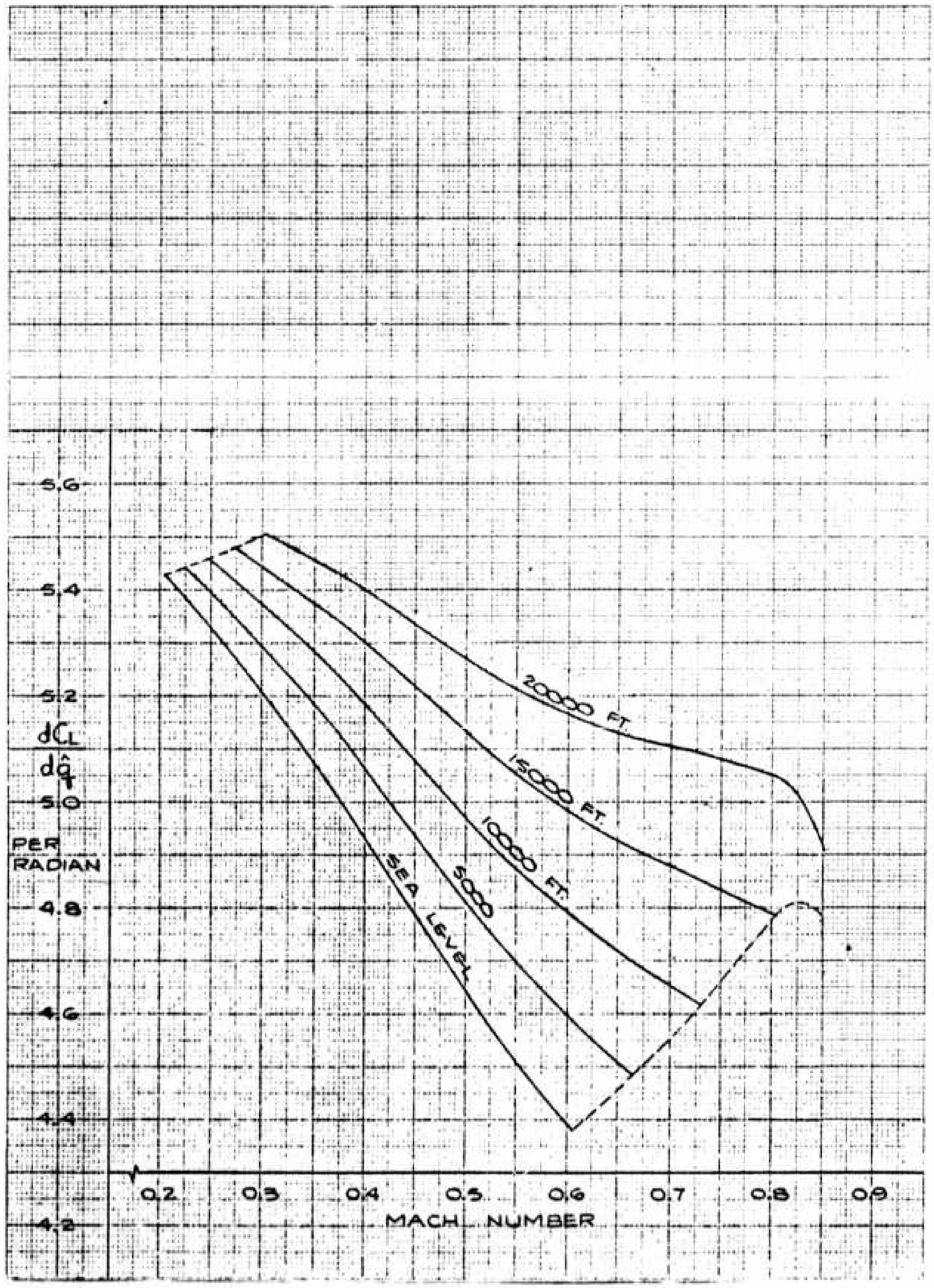


Figure 80. Lift Coefficient - Effect of $\hat{q}$ @ 300,000 Lbs.

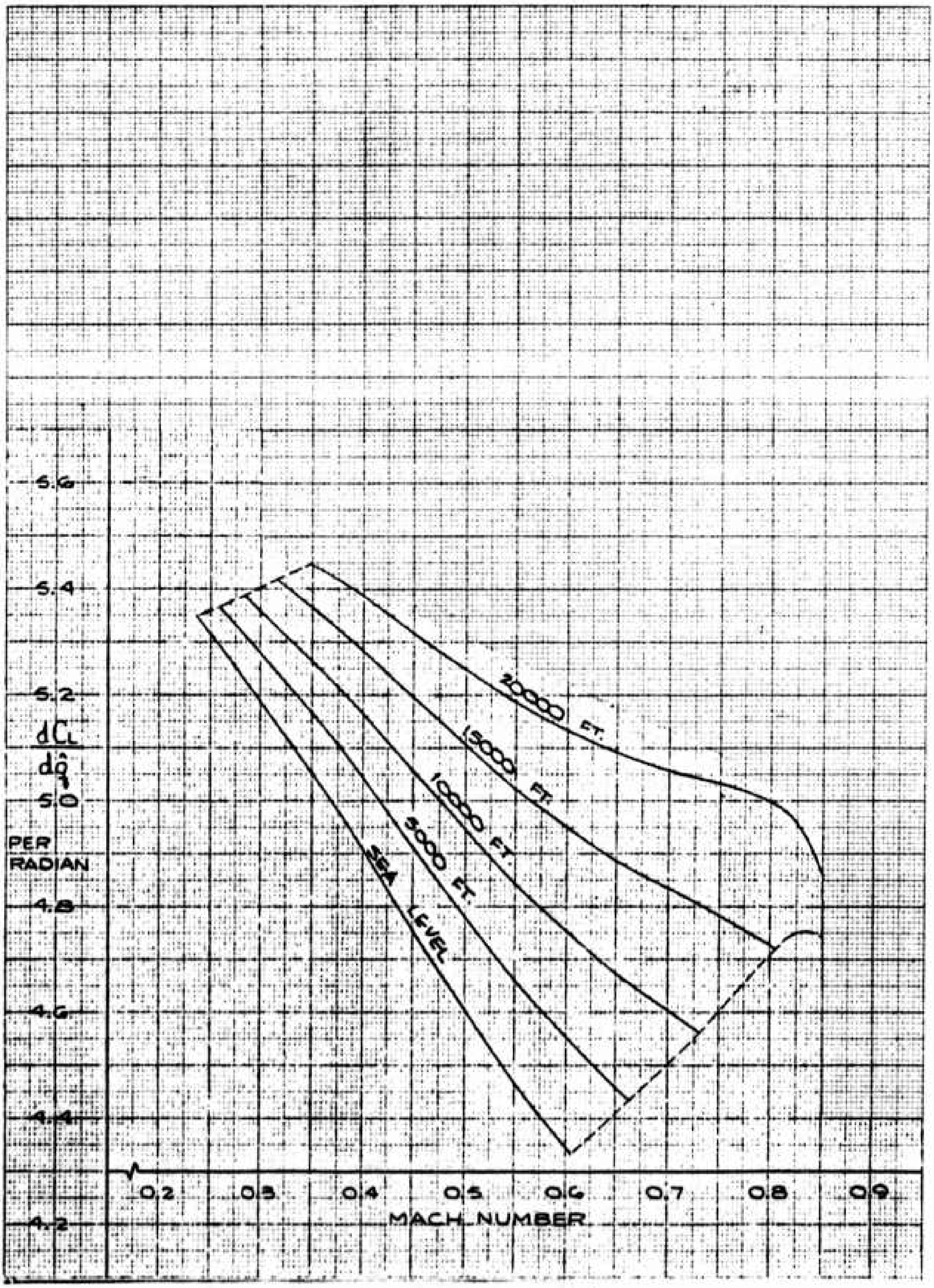


Figure 81. Lift Coefficient - Effect of $\hat{q}$ @ 400,000 Lbs.

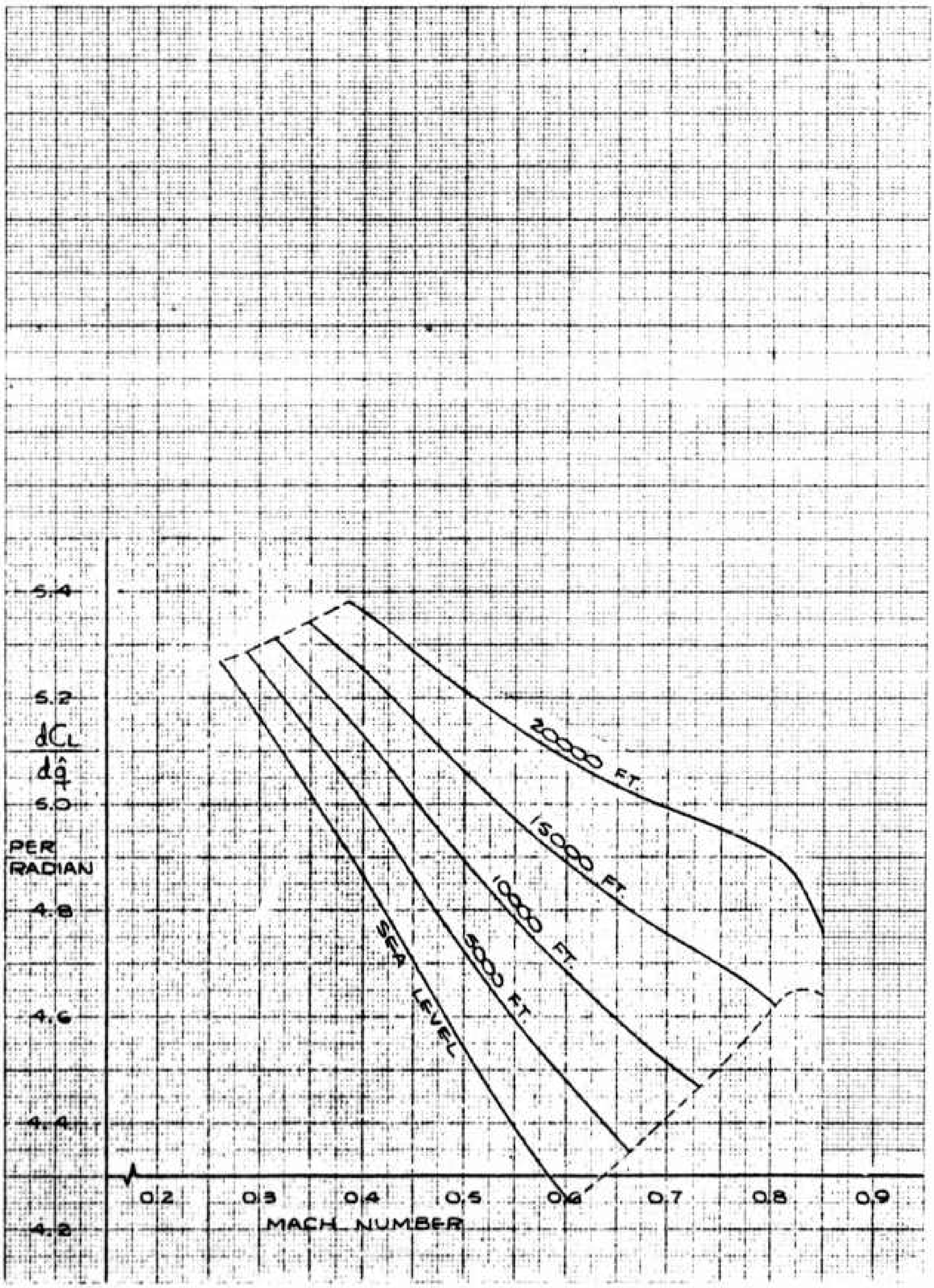


Figure 82. Lift Coefficient - Effect of $\hat{q}$ @ 488,000 Lbs.

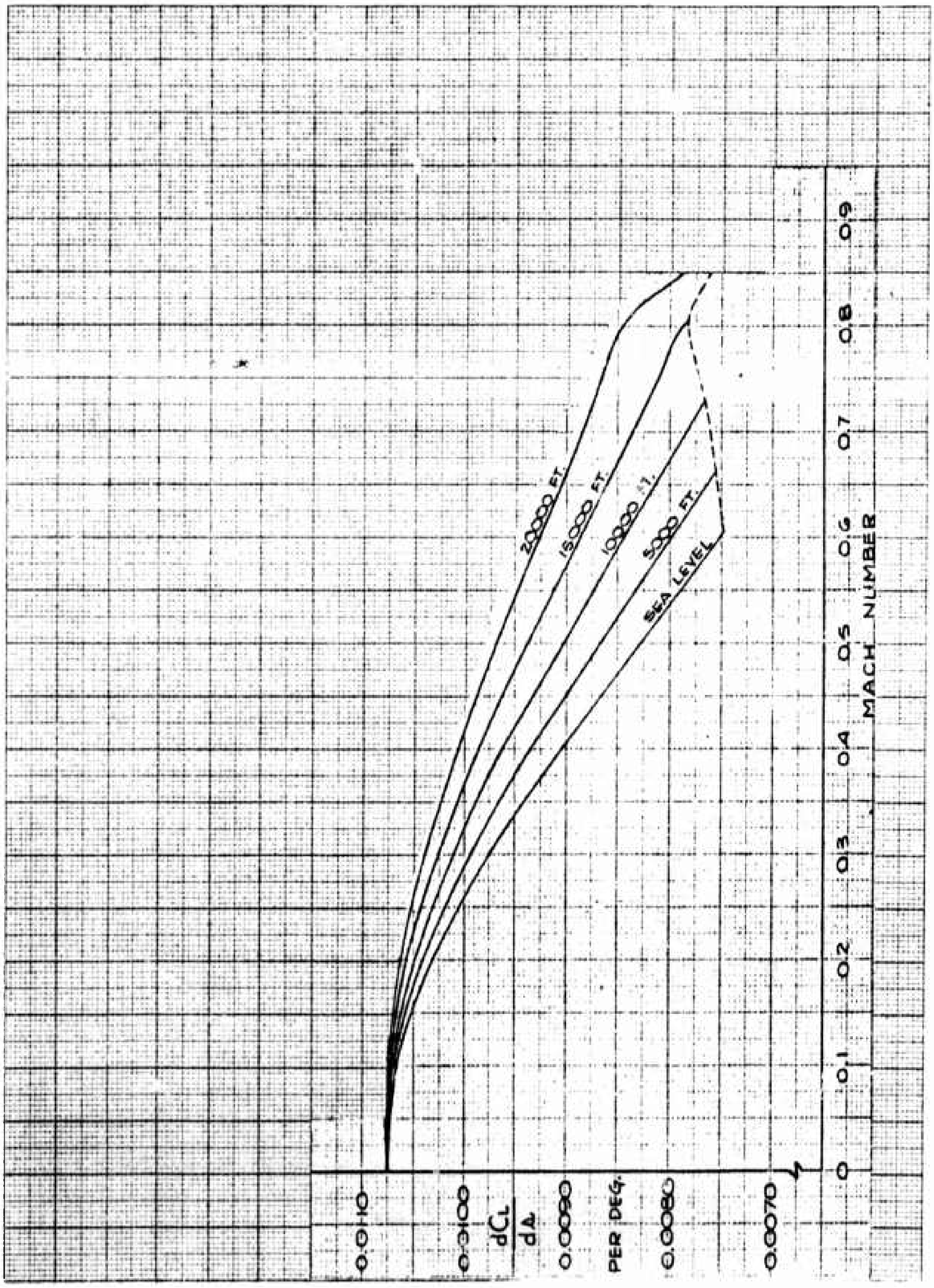


Figure 83. Lift Coefficient - Effect of Stabilizer

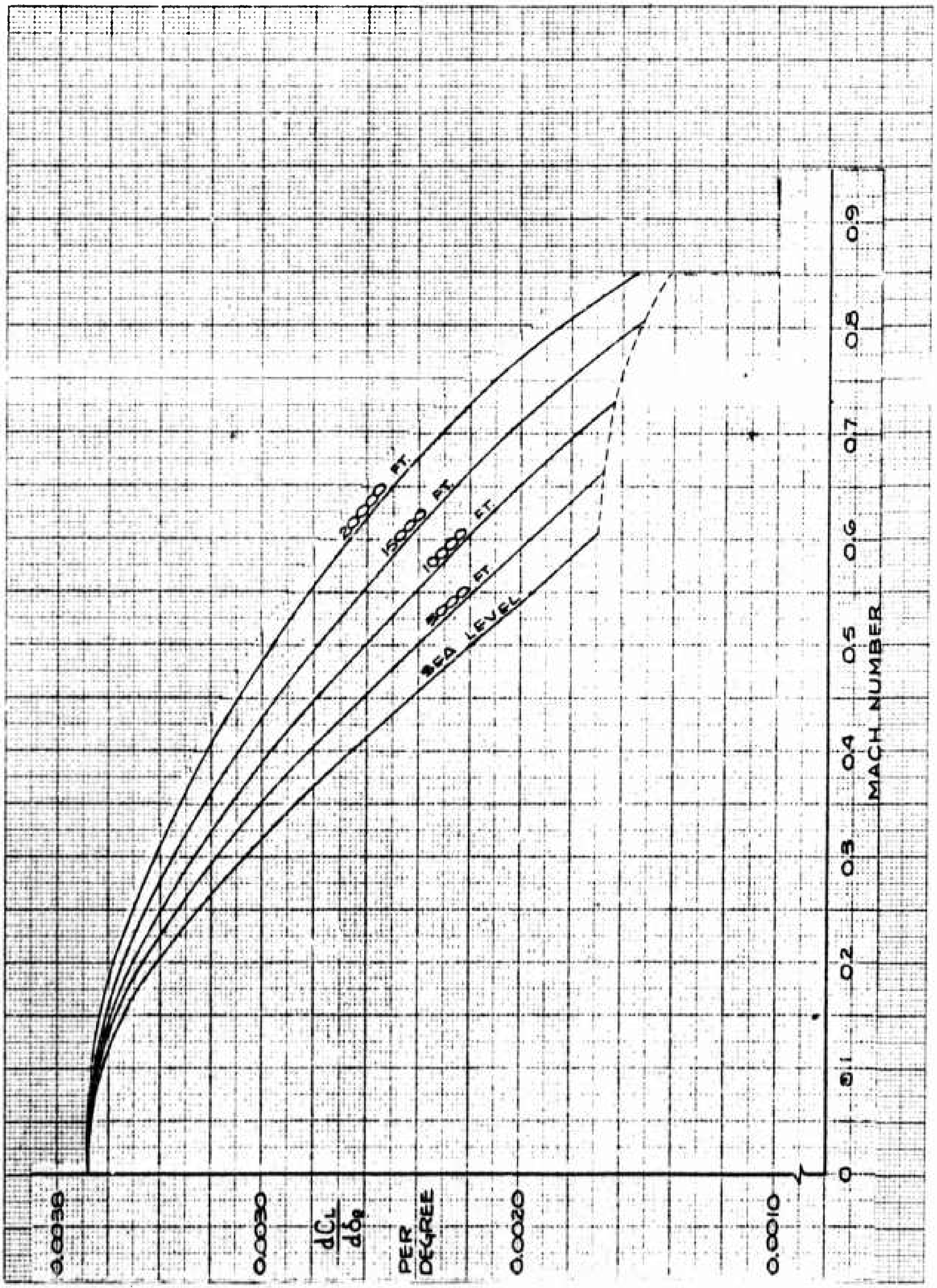


Figure 84. Lift Coefficient - Effect of Elevator

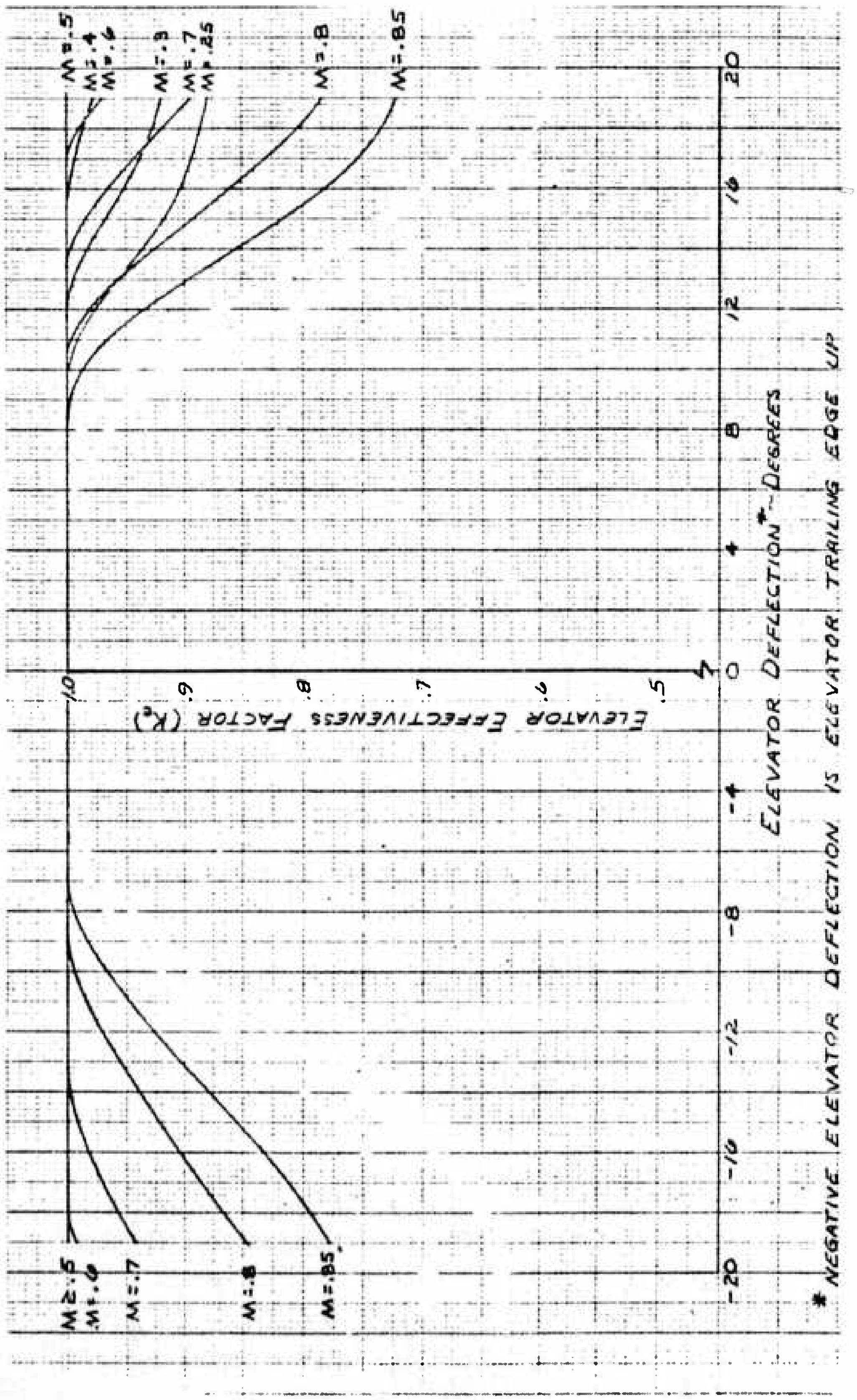


Figure 85. Elevator - Effectiveness Factor

b. Pitching Moment Coefficient

The dimensionless aerodynamic pitching moment coefficient is given in terms of its significant components by the equation below.

At a given flight condition:

$$C_m = C_{m_{0.25_{BASIC}}} + \Delta\left(C_{m_{0.25}}\right)_{\alpha_W = 0} + \Delta\left(\frac{dC_{m_{0.25}}}{d\alpha}\right)_{ELASTIC} \cdot \alpha_{wing}$$

$$+ \frac{dC_{m_{0.25}}}{dn_z} \cdot n_z + C_L\left(c.g. - 0.25\right) + \frac{dC_{m_{0.25}}}{d\hat{\dot{\alpha}}}\left(\frac{\dot{\alpha}\bar{c}}{2V}\right)$$

$$+ \frac{dC_{m_{0.25}}}{d\hat{q}}\left(\frac{q\bar{c}}{2V}\right) + \frac{dC_{m_{0.25}}}{d\mathcal{A}} \cdot \mathcal{A} + K_e \frac{dC_{m_{0.25}}}{d\delta_e} \cdot \delta_e$$

$C_{m_{0.25_{BASIC}}}$ = Basic pitching moment coefficient for the rigid airplane at $\alpha_W = 0°$, in free air and with the landing gear retracted.

$\Delta\left(C_{m_{0.25}}\right)_{\alpha_W = 0}$ = Change in basic pitching moment coefficient at $\alpha_{wing} = 0°$ due to aeroelasticity.

$\Delta\left(\frac{dC_{m_{0.25}}}{d\alpha}\right)_{ELASTIC}$ = Change in slope of pitching moment coefficient with angle of attack due to aeroelasticity.

$\frac{dC_{m_{0.25}}}{dn_z}$ = Change in basic pitching moment coefficient due to aeroelastic inertia relief caused by normal load factor, n_z. (For wings level, steady flight, $n_z = 1.0$).

C_L (c.g. -0.25) = Change in basic pitching moment coefficient due to center of gravity variation from 25% M.A.C.

$$\frac{dC_{m_{0.25}}}{d\hat{\dot{\alpha}}}$$ = Change in basic pitching moment coefficient due to non-dimensionalized rate of change of angle of attack.

$$\frac{dC_{m_{0.25}}}{d\hat{q}}$$ = Change in basic pitching moment coefficient due to non-dimensionalized pitch rate.

$$\frac{dC_{m_{0.25}}}{d i_s}$$ = Change in basic pitching moment coefficient due to change in stabilizer angle from i_s = 0°.

$$K_e \frac{dC_{m_{0.25}}}{d\delta_e}$$ = Change in the basic pitching moment coefficient due to change in elevator angle from δ_e = 0°.

Figures 86 through 100 describe the pitching moment data.

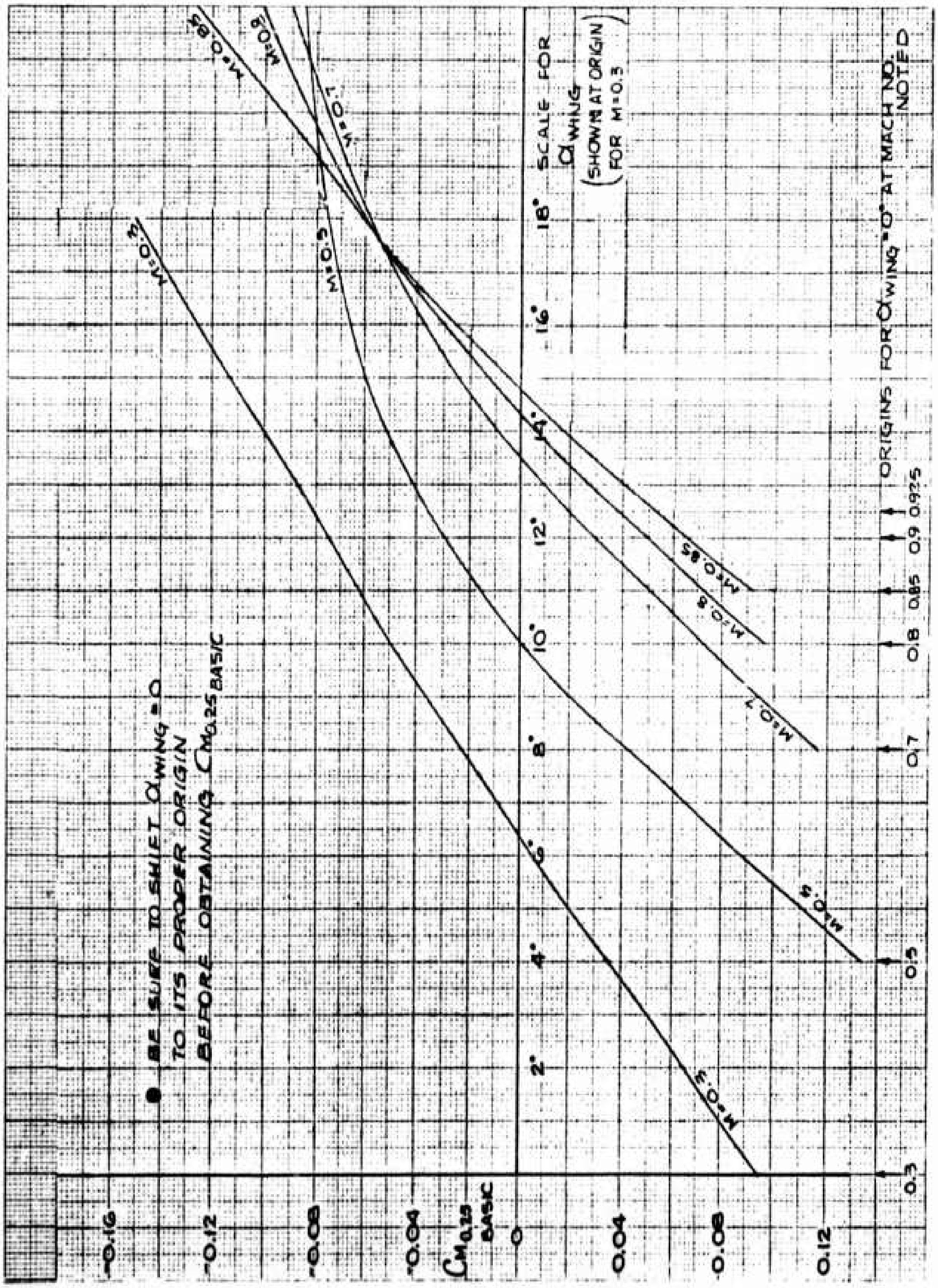


Figure 86. Pitching Moment Coefficient - Effect of Angle of Attach on $C_{M0.25}$ Basic

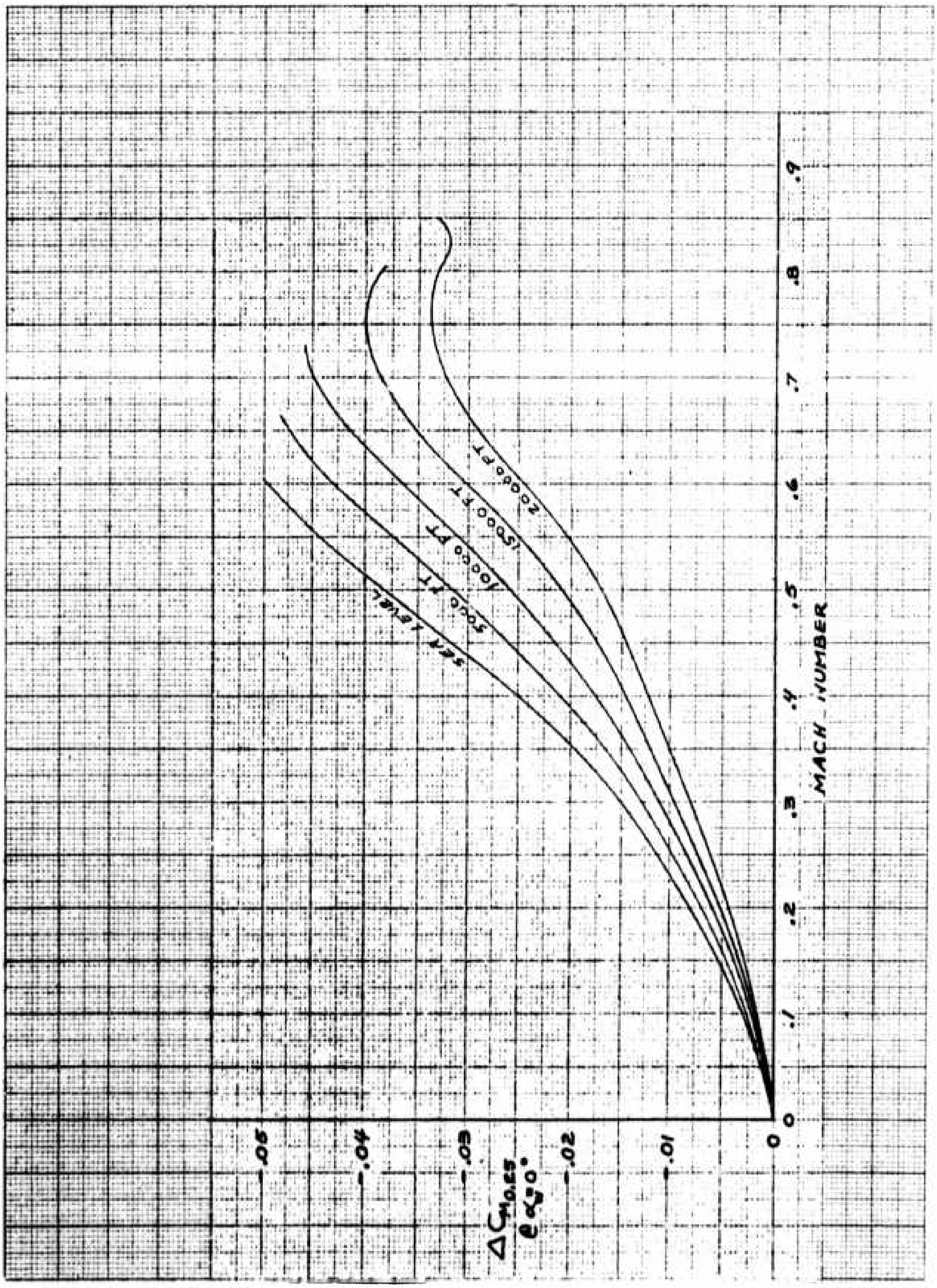


Figure 87. Pitching Moment Coefficient - Aeroelastic Effect on $\Delta C_{MO.25}$ @ α_W = 0°

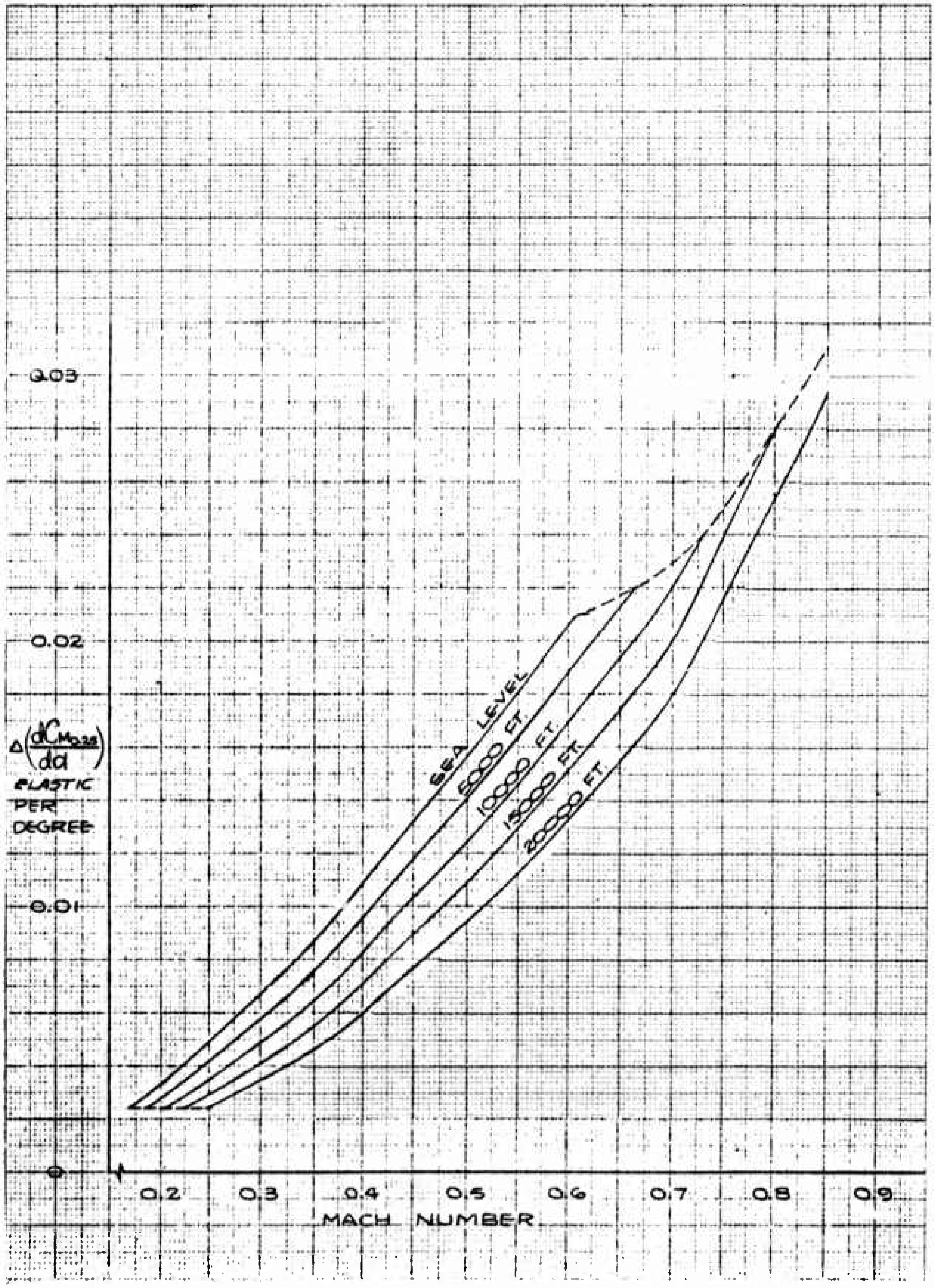


Figure 88. Pitching Moment Coefficient - Aeroelastic Effect on $\Delta(dC_{M0.25}/d\alpha)$

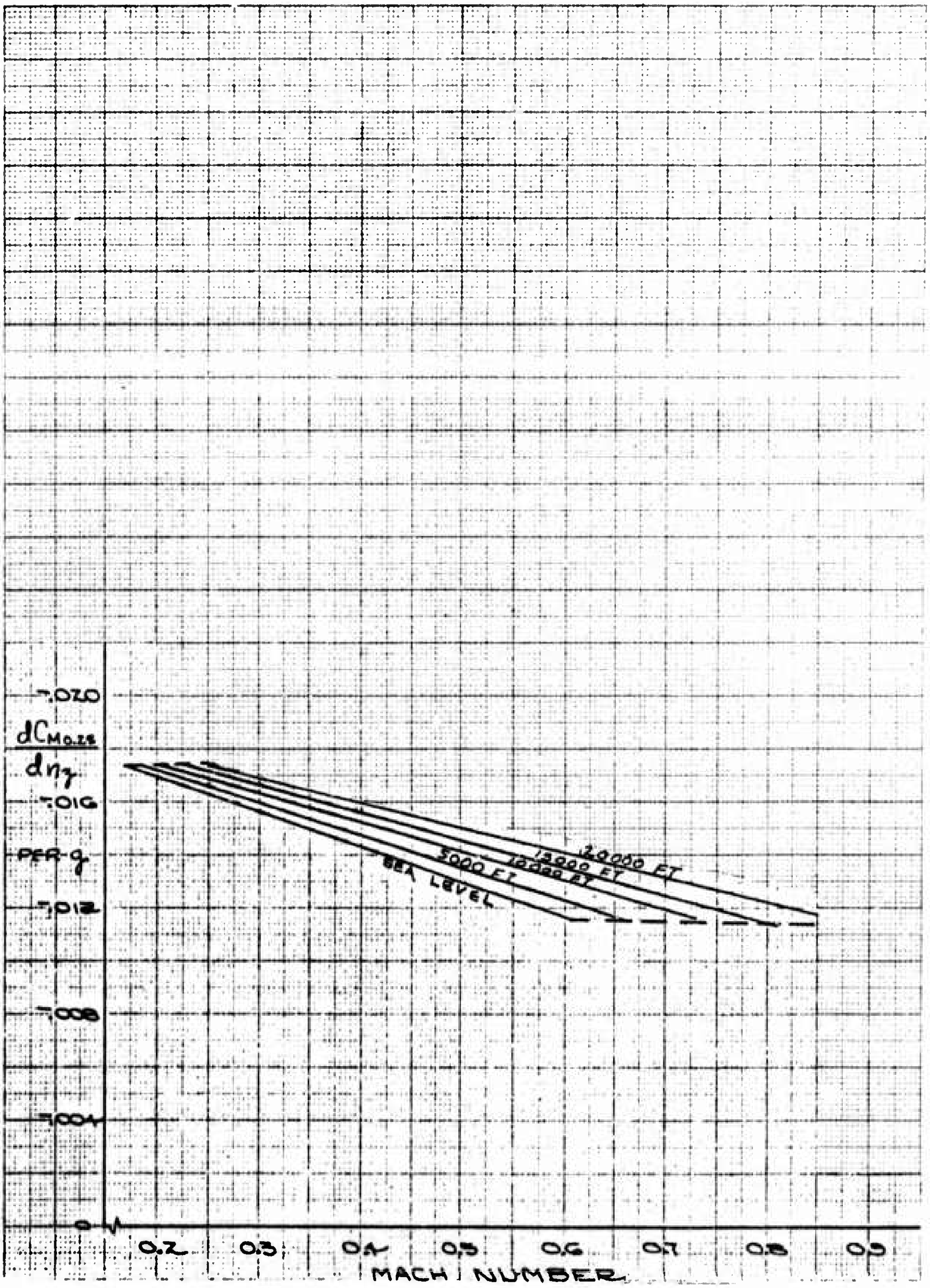


Figure 89. Pitching Moment Coefficient - Effect of Normal Load Factor - G.W. 200,000 Lbs.

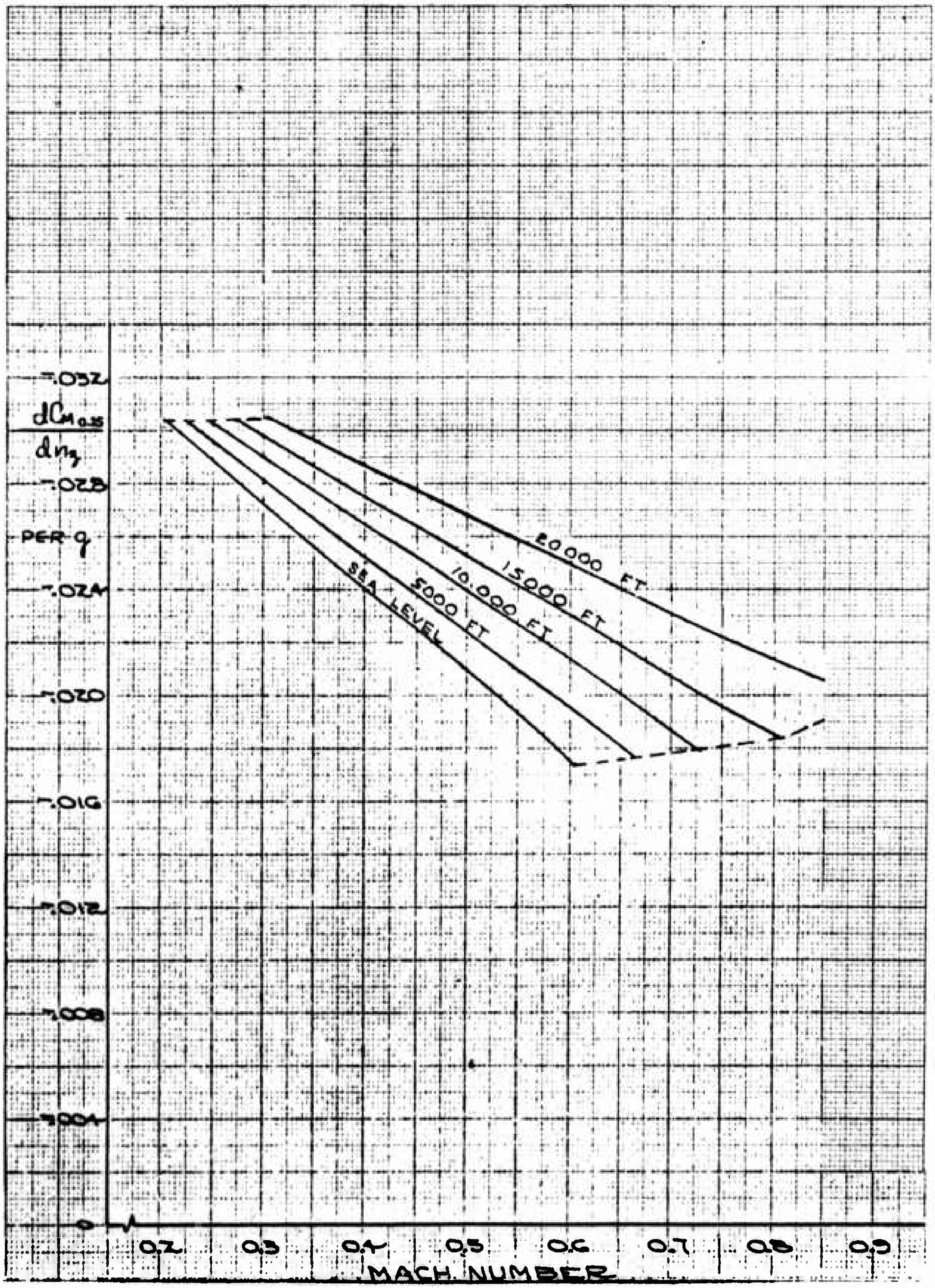


Figure 90. Pitching Moment Coefficient - Effect of Normal Load Factor - G.W. 300,000 Lbs.

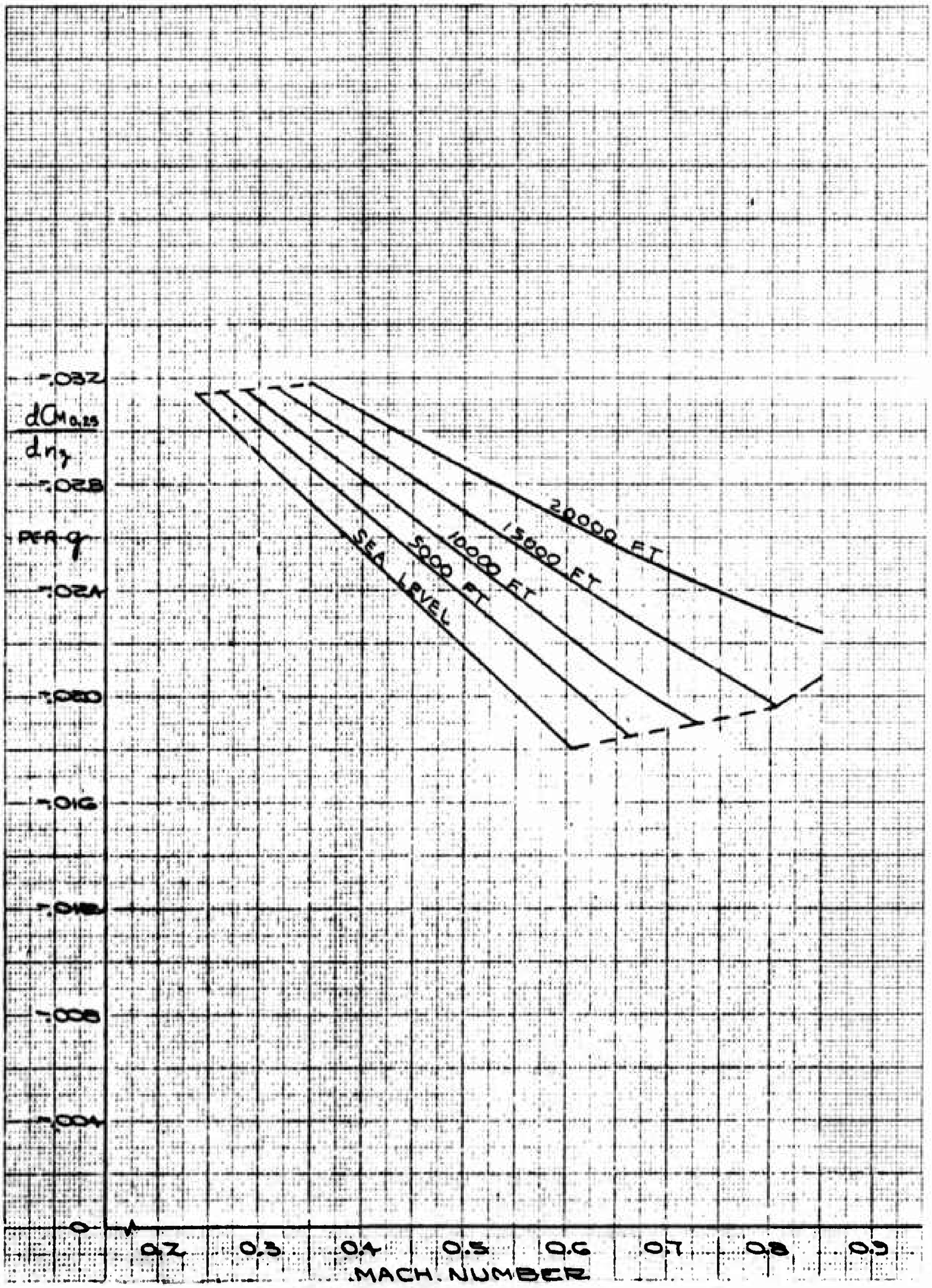


Figure 91. Pitching Moment Coefficient - Effect of Normal Load Factor - G.W. 400,000 Lbs.

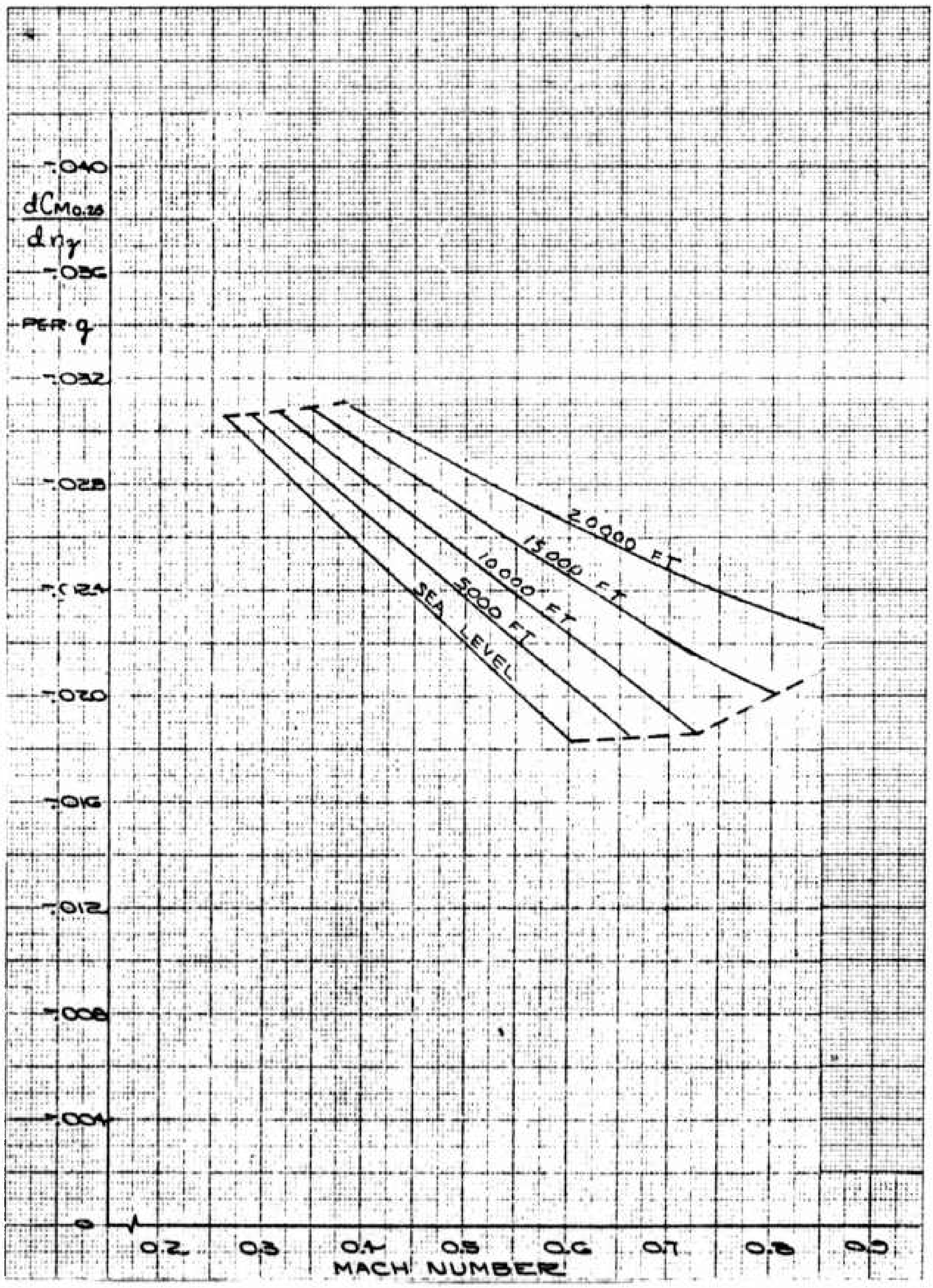


Figure 92. Pitching Moment Coefficient - Effect of Normal Load Factor - G.W. 488,000 Lbs.

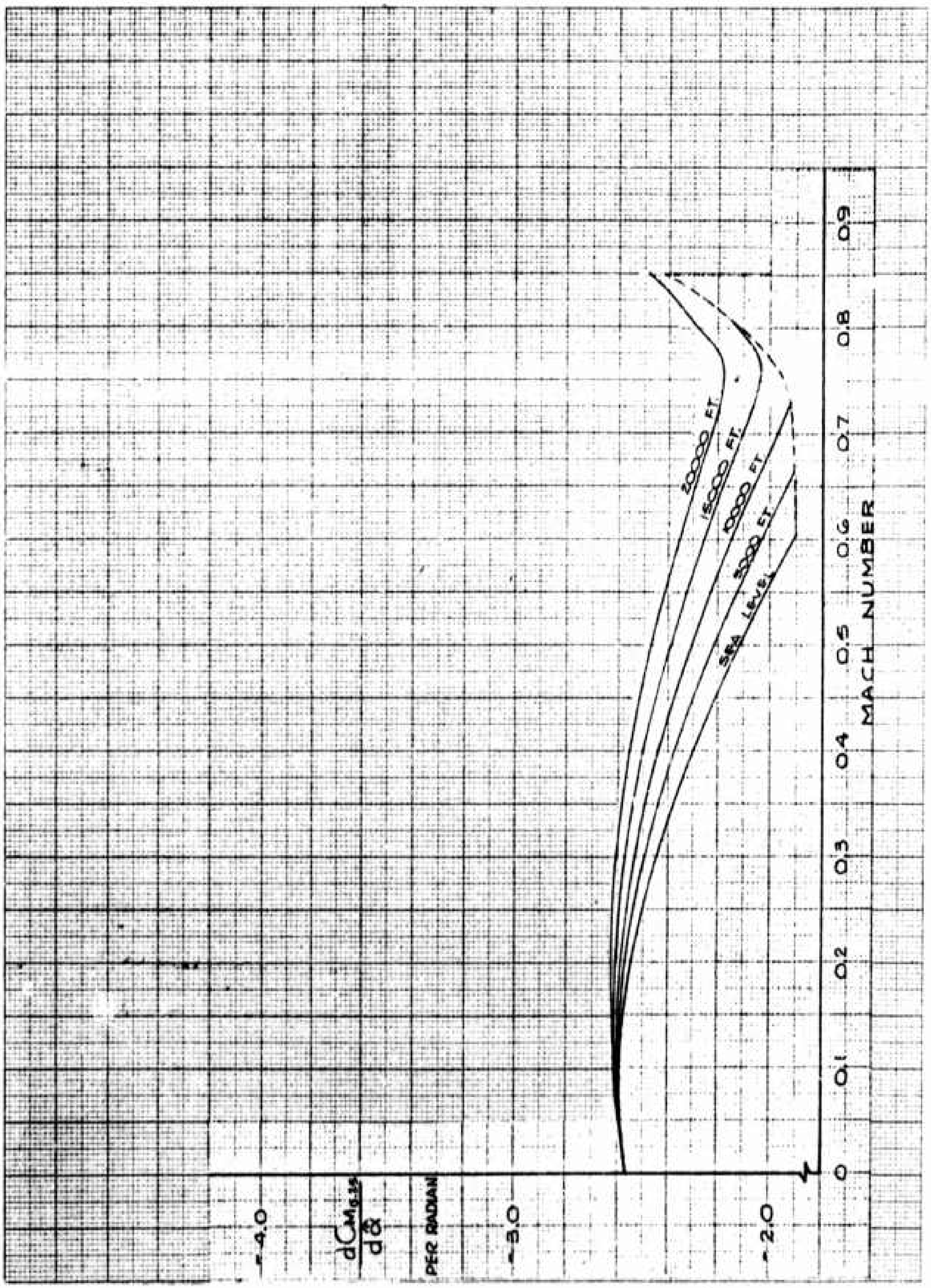


Figure 93. Pitching Moment Coefficient - Effect of $\hat{\alpha}$

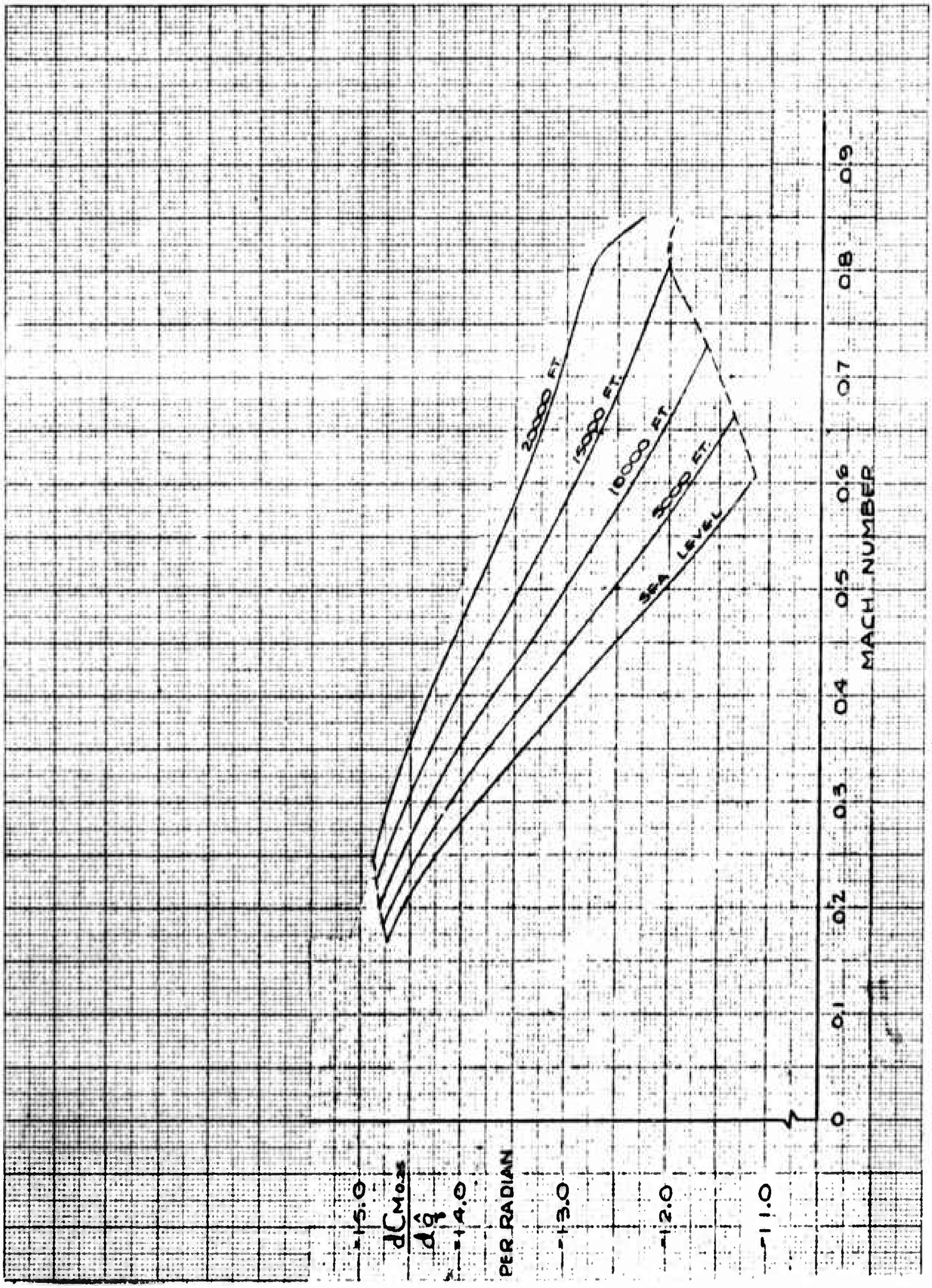


Figure 94. Pitching Moment Coefficient - Effect of $\hat{q}$ @ 200,000 Lbs.

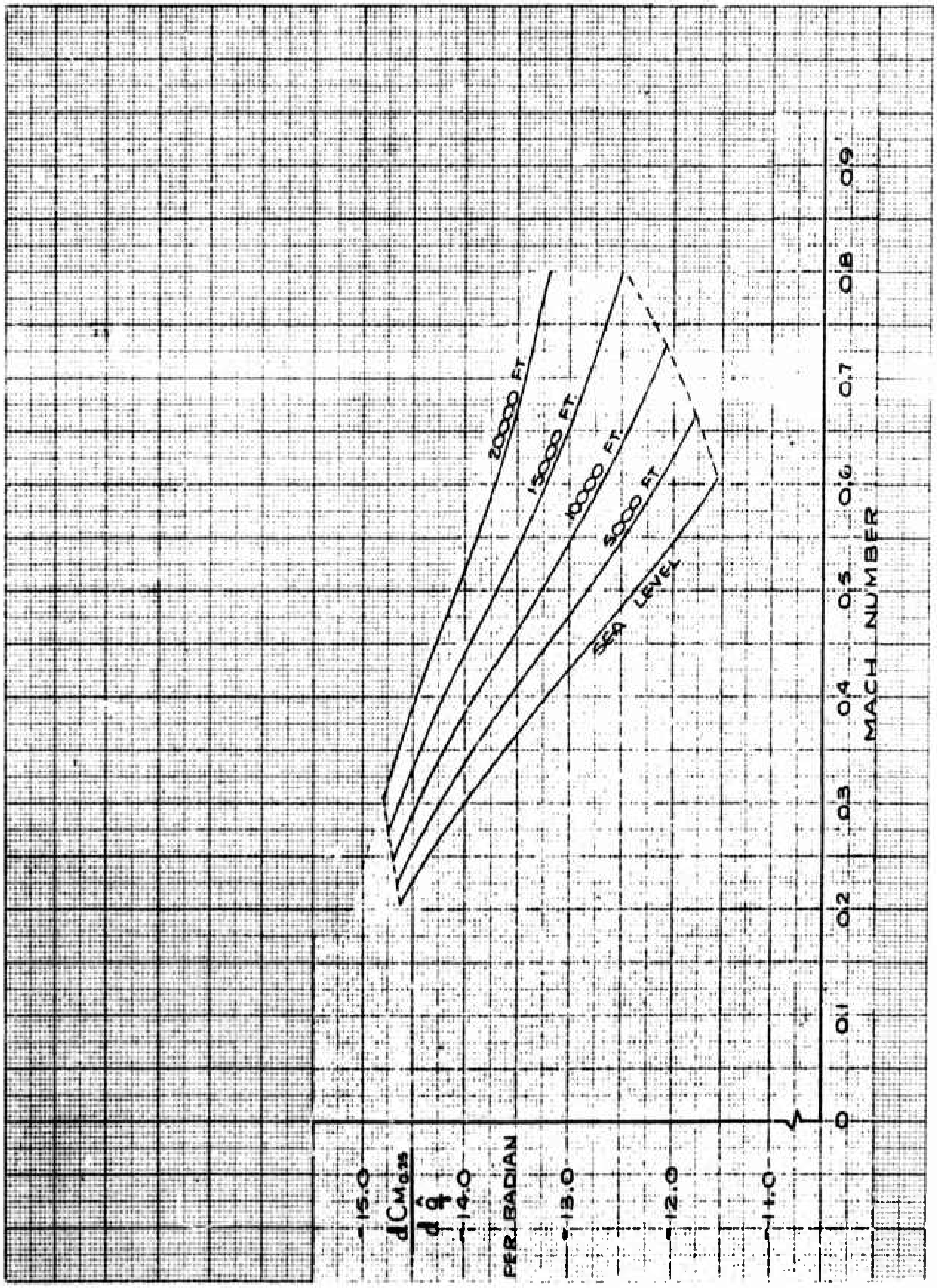


Figure 95. Piching Moment Coefficient - Effect of $\hat{q}$ @ 300,000 Lbs.

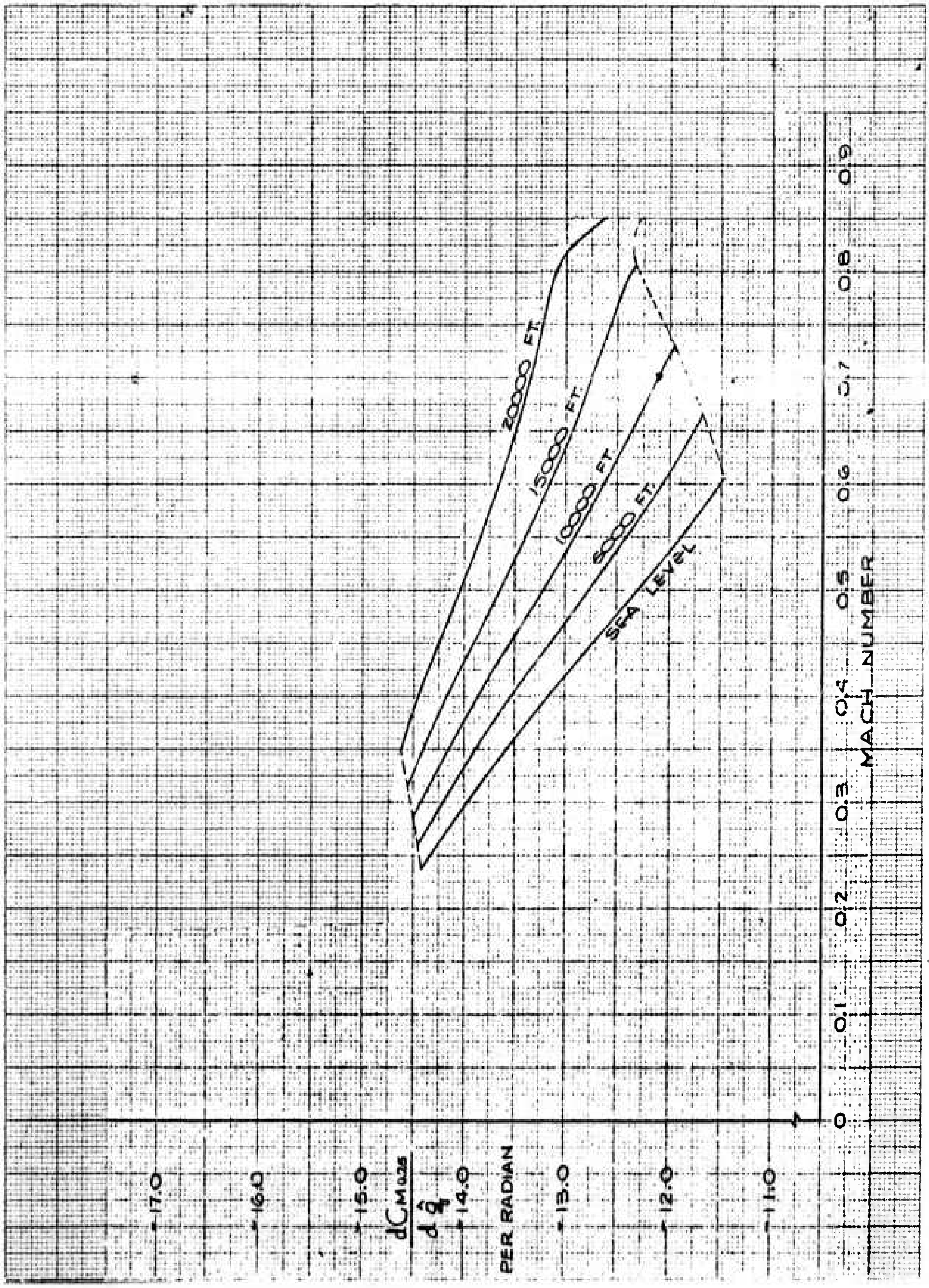


Figure 96. Pitching Moment Coefficient - Effect of $\hat{q}$ @ 400,000 Lbs.

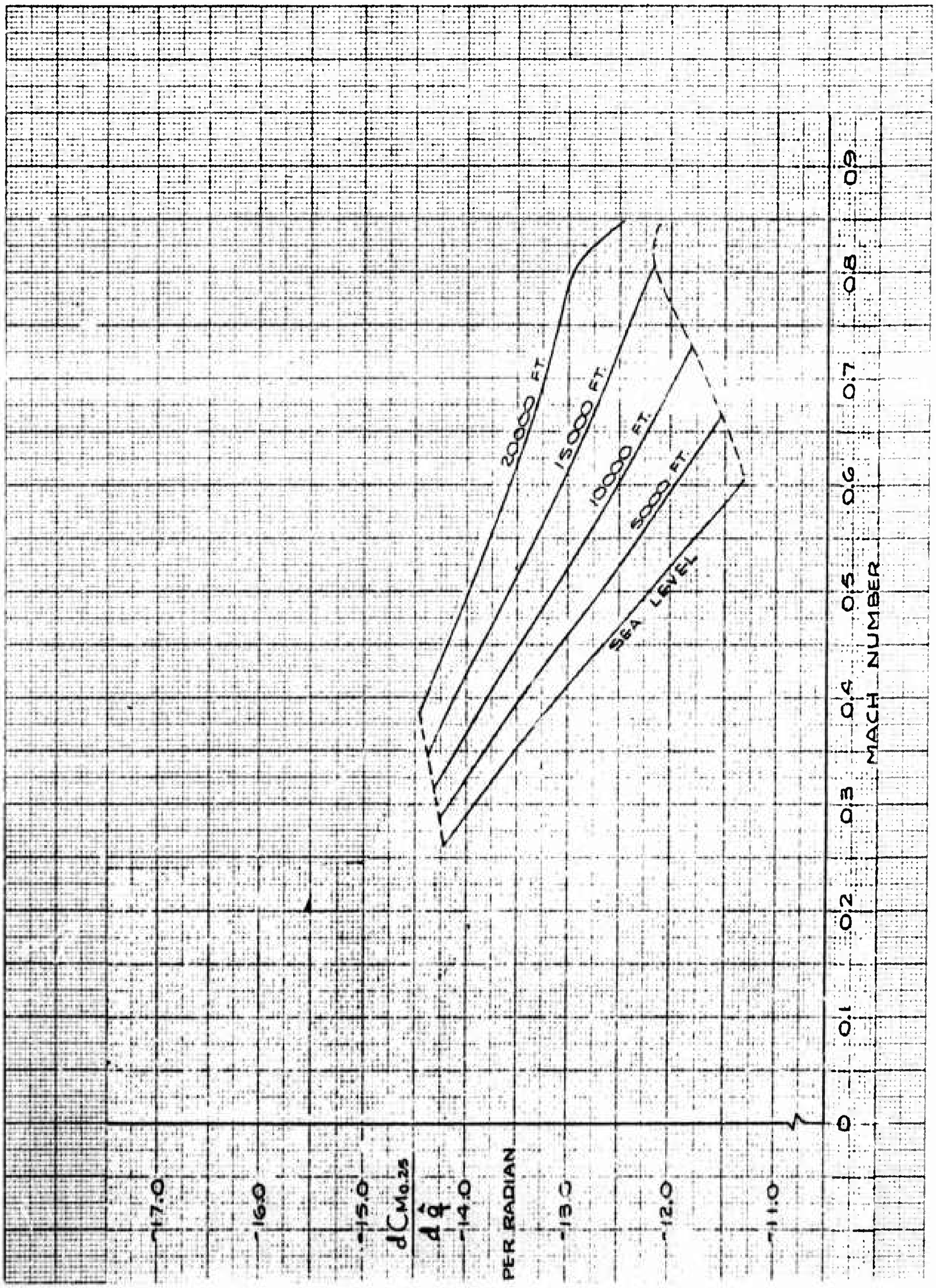


Figure 97. Pitching Moment Coefficient - Effect of $\hat{q}$ @ 488,000 Lbs.

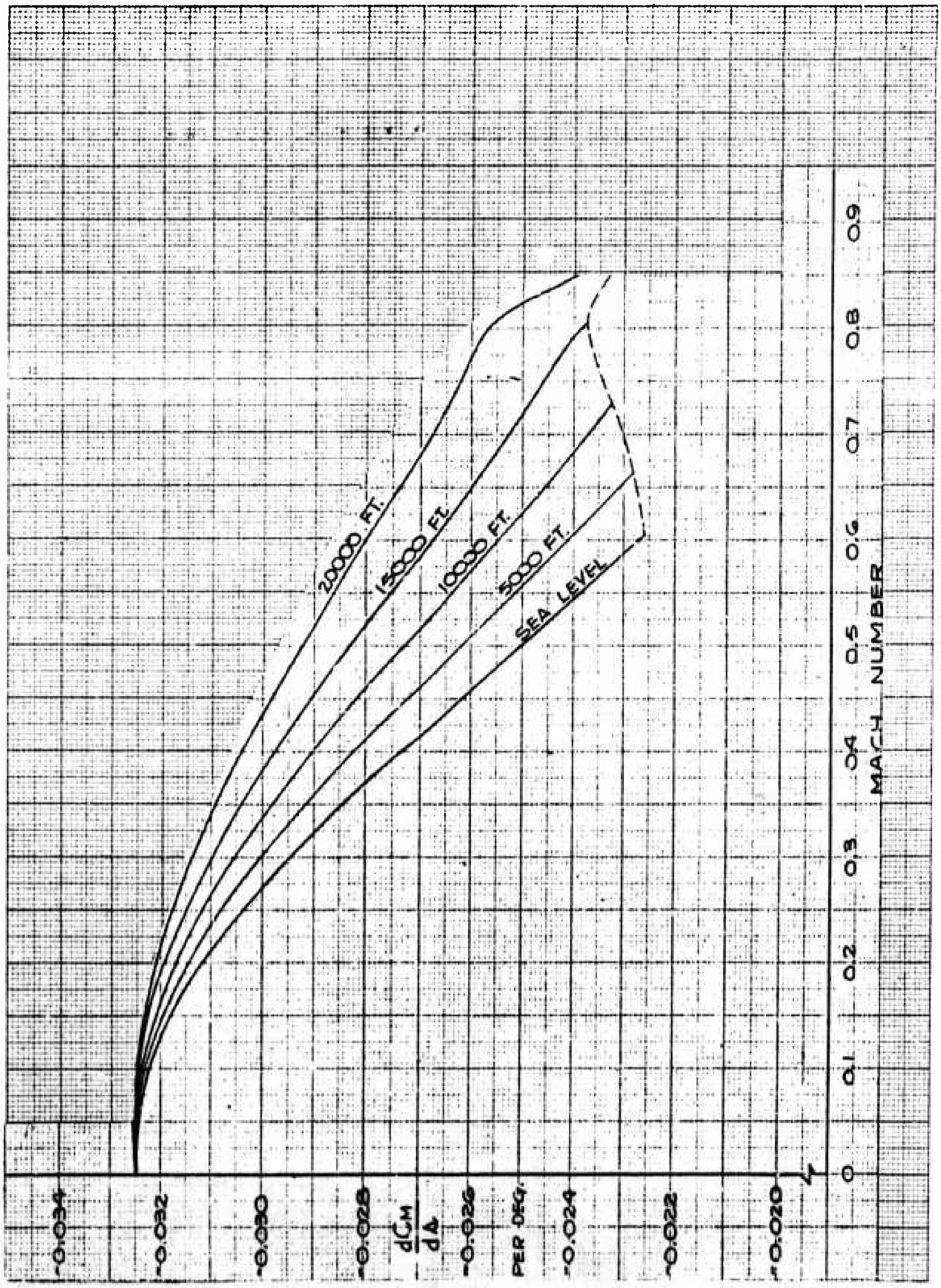


Figure 98. Pitching Moment Coefficient - Aeroelastic Effect On Stabilizer Effectiveness

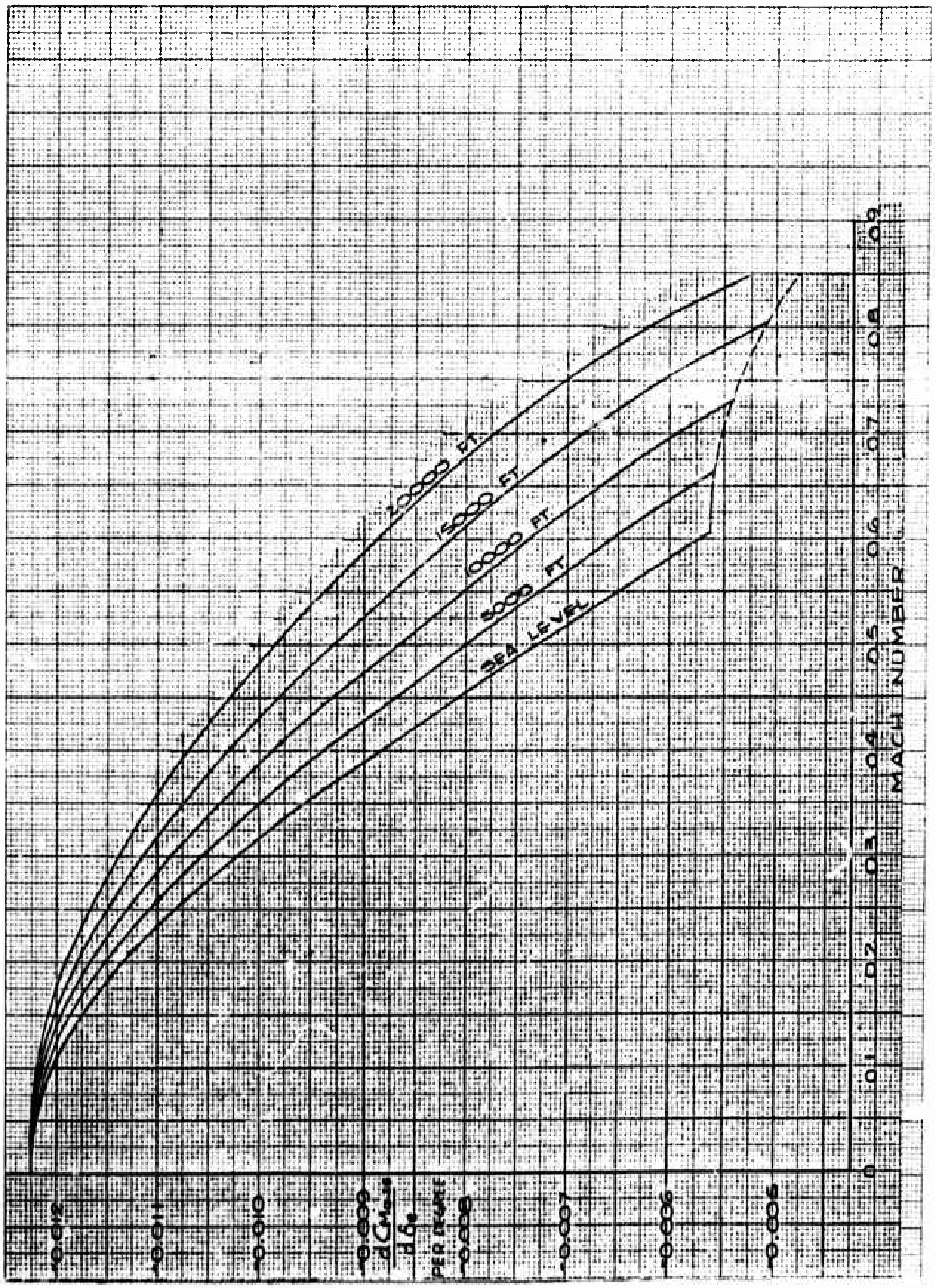


Figure 99. Pitching Moment Coefficient - Aeroelastic Effect On Elevator Effectiveness

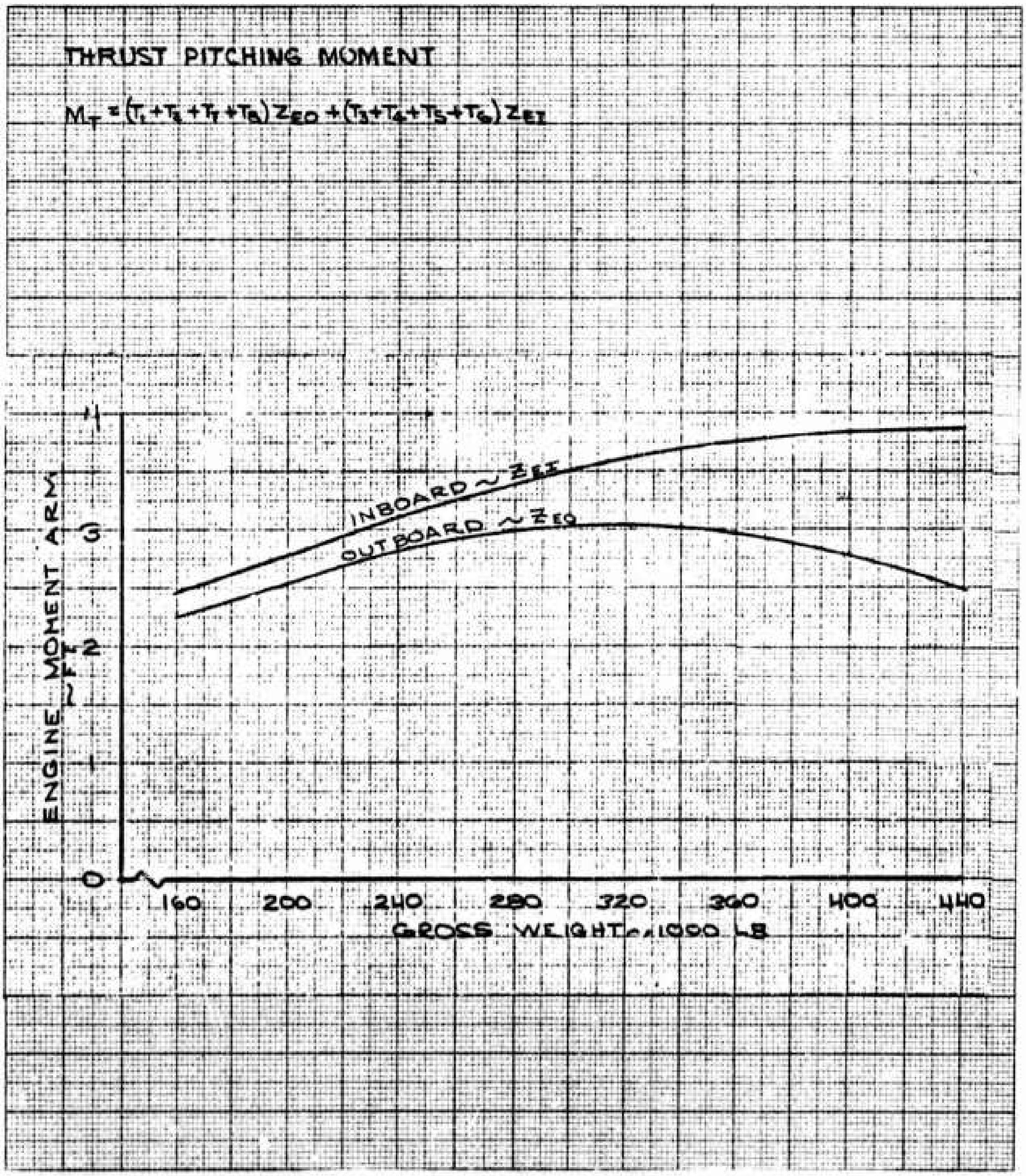


Figure 100. Engine Pitching Moment - Arm - Flaps Up

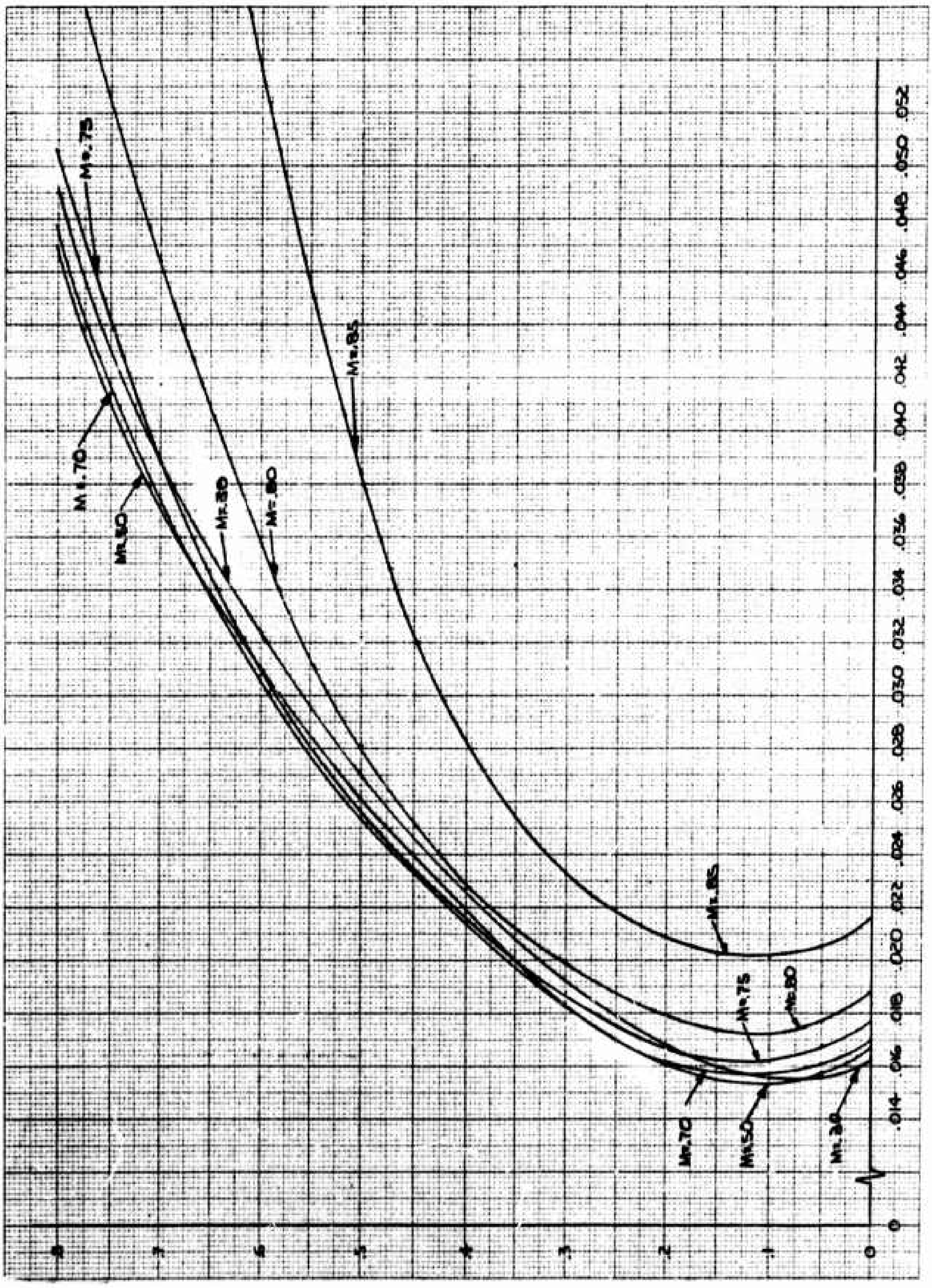


Figure 101. Drag Coefficient Effect of Mach Number

In-Flight Limits

The values given below are considered to be the maximum in-flight limits of the airplane.

a. Minimum Angular Velocities

1)	Pitch rate	6.5°/sec.
2)	Yaw rate	5°/sec.
3)	Roll rate	40°/sec.

b. Maximum Angular Accelerations

1)	Pitch acceleration	$30°/sec.^2$
2)	Yaw acceleration	$12°/sec^2$
3)	Roll acceleration	$45°/sec^2$

c. Maximum rates of climb and dive

1) Maximum <u>steady</u> rate of climb = 9000 ft/min.
2) Maximum rate of climb = 28,000 ft/min.
   (Zoon climb)
3) Maximum rate of descent = 32,000 ft/min.

d. Maximum Normal Acceleration

2g at Gross Weights above 325,000 lb. and at all weights flaps down.

e. Maximum forward velocity and acceleration

1) Maximum speed = 554 knots true at 20,000 ft.
2) Maximum acceleration = $16.8\ ft/sec.^2$

f. Maximum rate of change in angle of attack = 6.5°/sec.

g. Maximum Pitch Angle

1) Positive = 30°
2) Negative = 30°

Tables XXIII and XXIV provide geometrical and moment of inertia data.

TABLE XXIII (Sheet 1 of 2)

MOMENT OF INERTIA TABLE

CONFIGURATION

ARMAMENT- CHAFF, AMMO, FLARES, POD ROCKETS

GROSS WEIGHT	Iyy $\times 10^6$	CENTER OF GRAVITY
491000.	32843.	28.2
486000.	32776.	28.6
448000.	32616.	27.4
421000.	32502.	28.2
373000.	32116.	27.7
361000.	32011.	23.0
349000.	31950.	22.3
260000.	31374.	25.6
242000.	31302.	24.2
222000.	31238.	21.1
217000.	31234.	18.7
184000.	31078.	19.3

TABLE XXIII (Sheet 2 of 2)

MOMENT OF INERTIA TABLE

CONFIGURATION - AGM-69 MISSILES

ARMAMENT - CHAFF, AMMO, FLARES, (12) AGM-69 MISSILES
(2) AGM-69 PYLONS

GROSS WEIGHT	$I_{YY} \times 10^6$	CENTER OF GRAVITY
511000.	32690.	27.3
503000.	32646.	27.8
466000.	32458.	26.5
430000.	32158.	26.1
388000.	31926.	23.0
339000.	31596.	24.3
330000.	31538.	20.9
290000.	31290.	22.0
273000.	31266.	17.2
259000.	31178.	17.3
251000.	31150.	18.4
231000.	31078.	19.2

TABLE XXIV

GEOMETRIC DATA

AIRFRAME GEOMETRIC DATA

WING

TOTAL AREA (INCLUDING FUSELAGE AREA)	4000 FT^2
ASPECT RATIO	8.55
TAPER RATIO	.4
DIHEDRAL	2.5°
INCIDENCE ANGLE	6°
℄ THRUST INCIDENCE TO WING CHORD PLANE	-2.5°
MAC	275.5 IN
TWIST	0
SPAN	185 FT

11. ENGINE PERFORMANCE DATA

Figure 102 provides an index to the performance data.

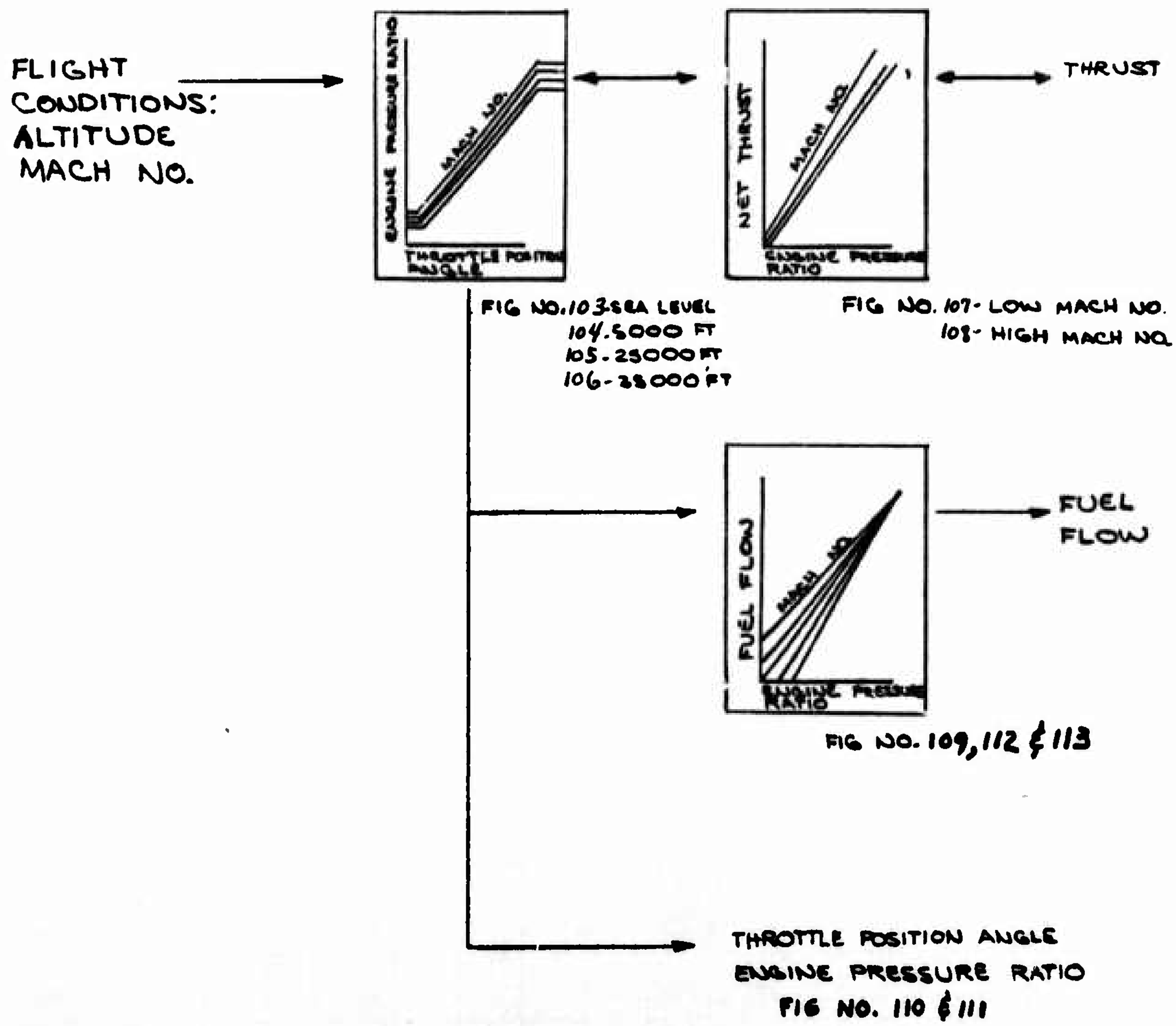


Figure 102. Engine Performance Data Index

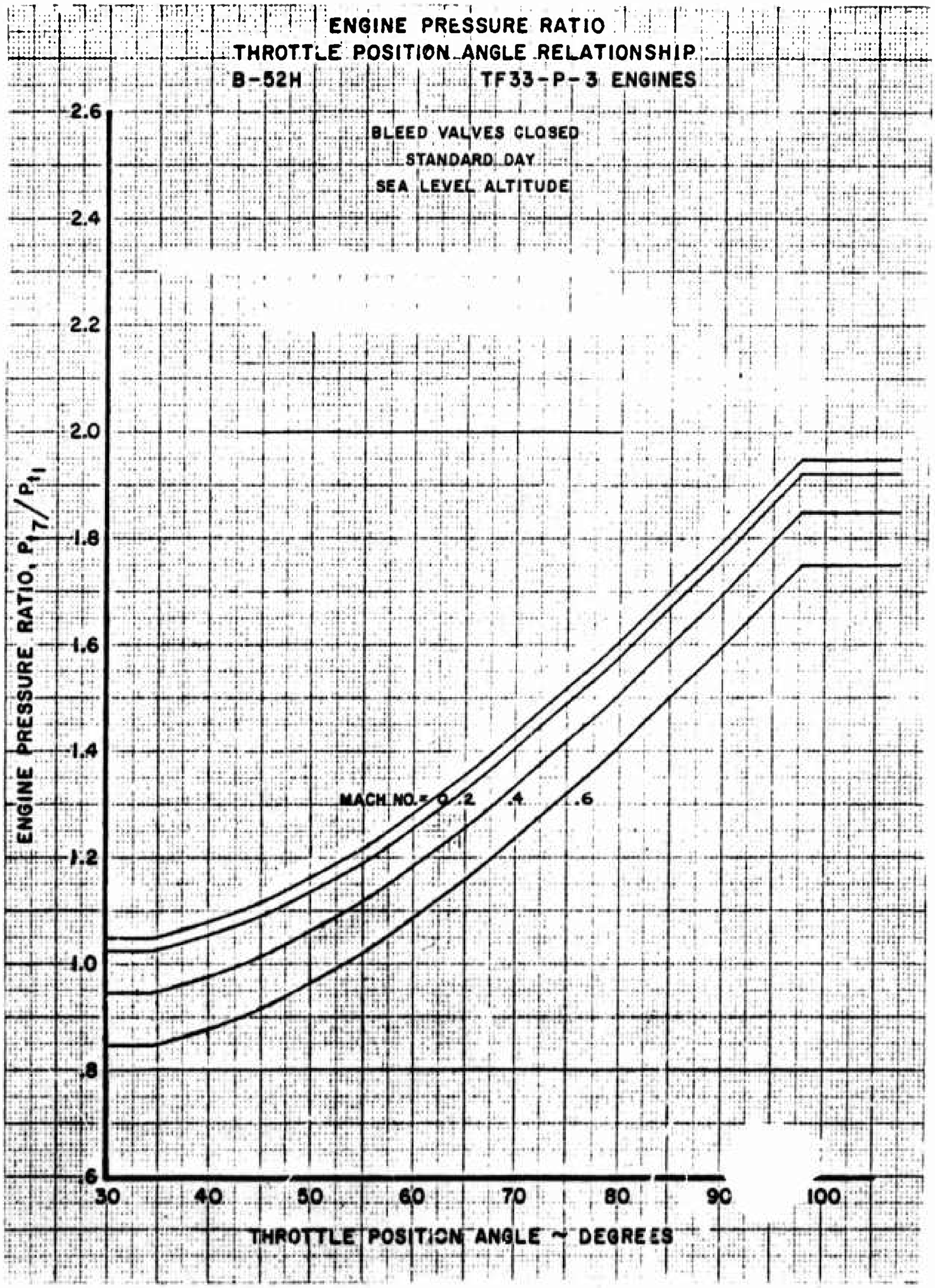


Figure 103. B-52H Engine Pressure Ratio Versus Throttle Position Angle at Sea Level Altitude

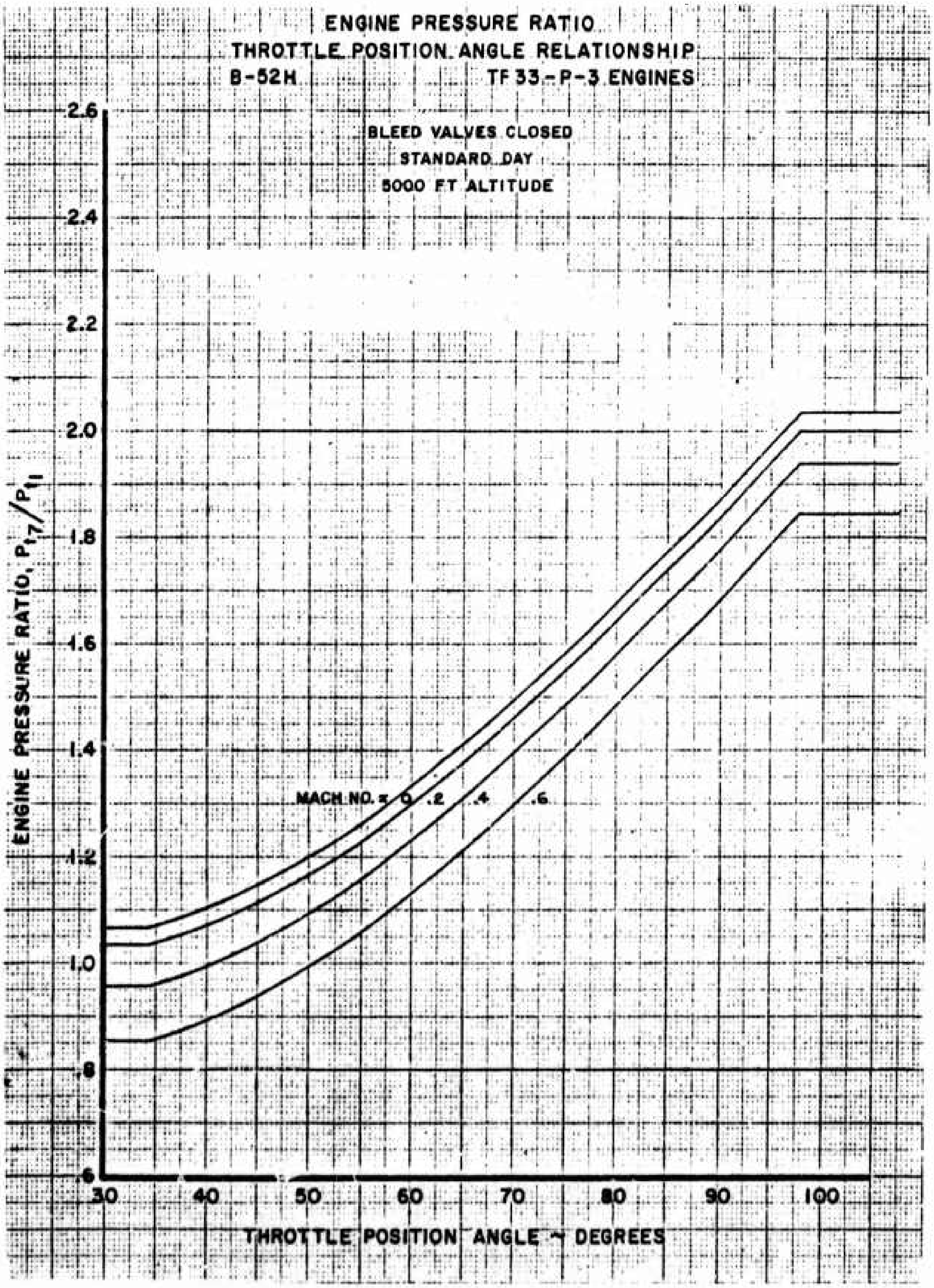


Figure 104. B-52H Engine Pressure Ratio Versus Throttle Position Angle at 5000 Ft. Altitude

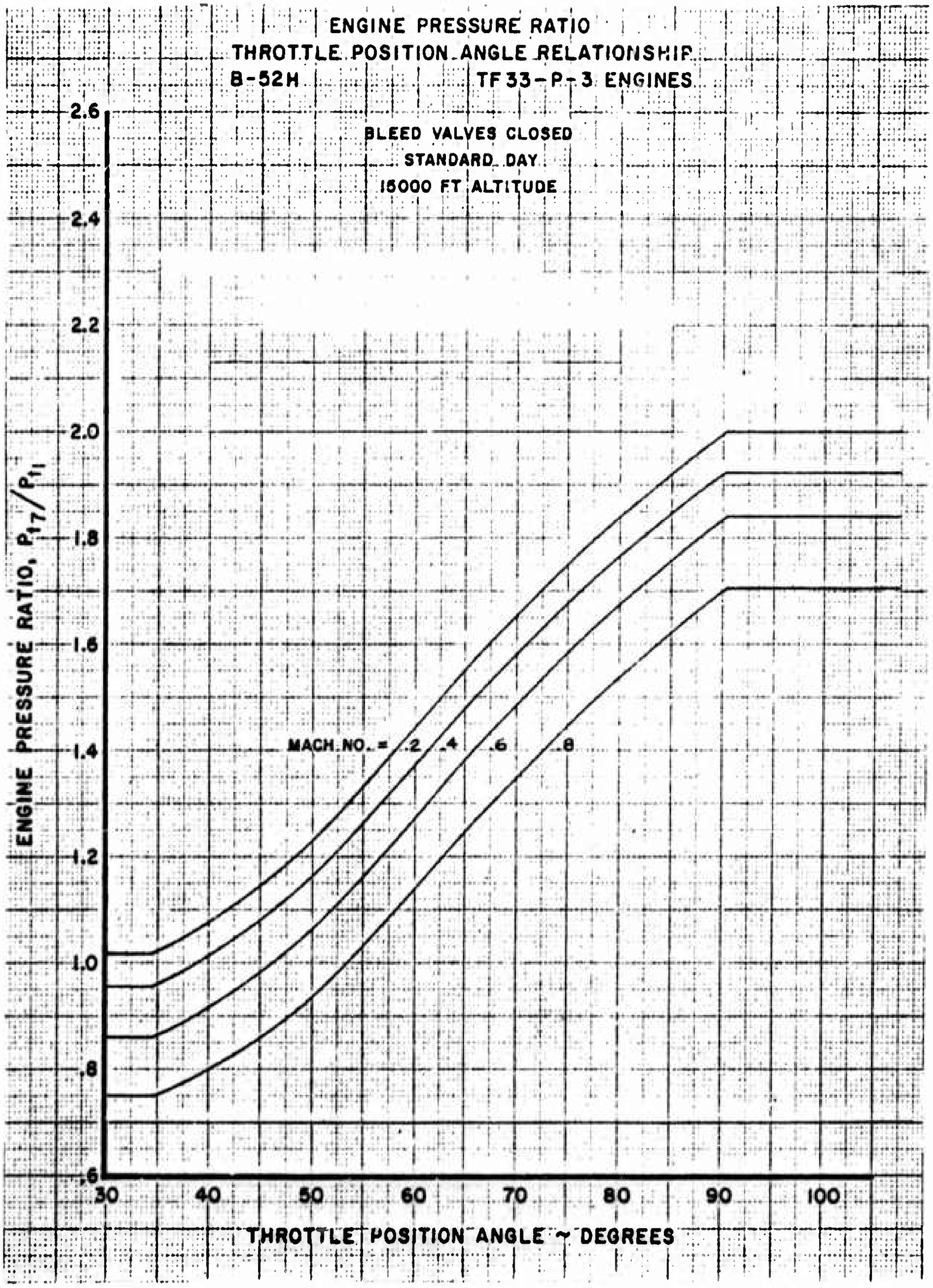


Figure 105. B-52H Engine Pressure Ratio Versus Throttle Position Angle at 15,000 Ft. Altitude

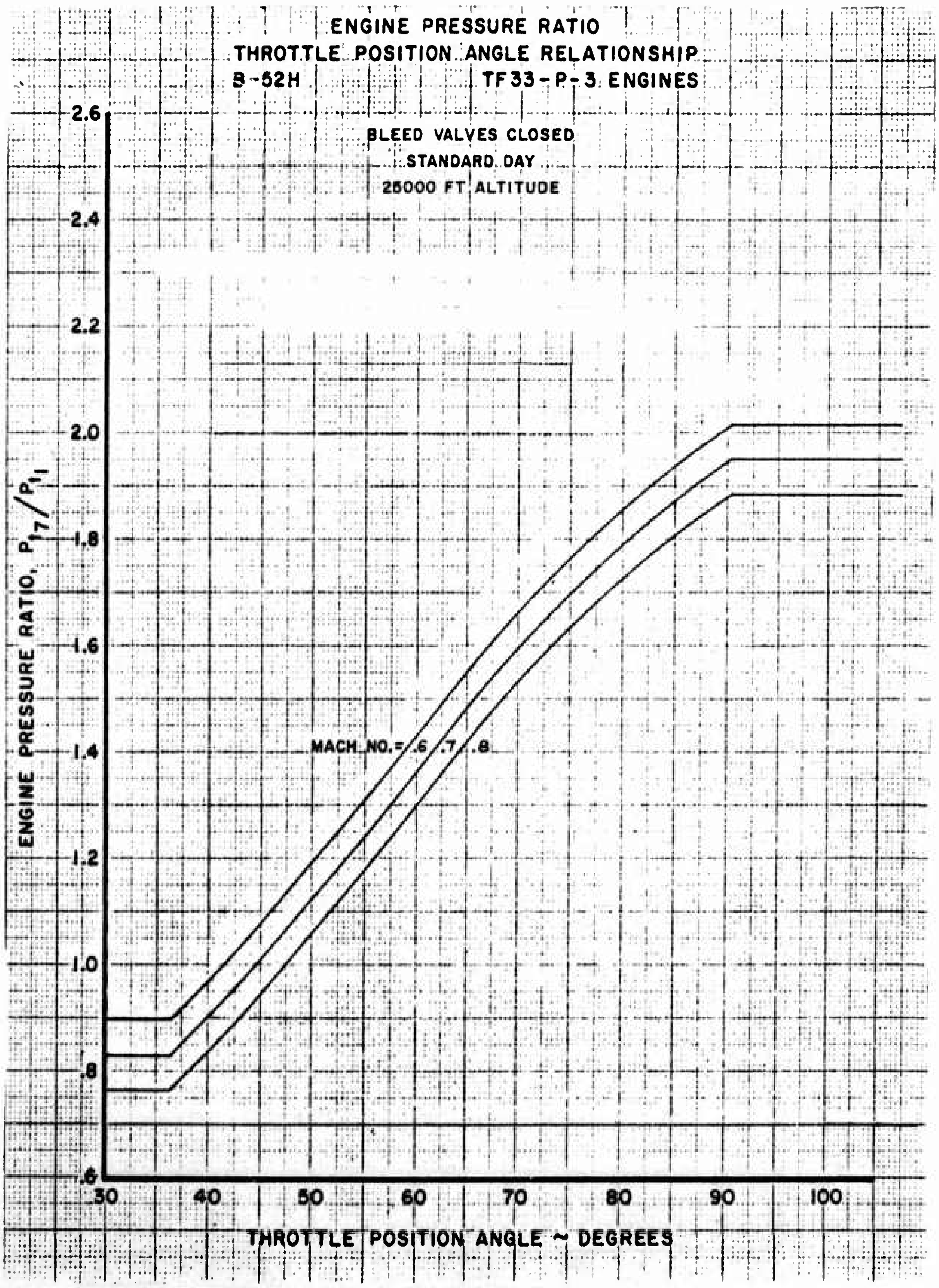


Figure 106. B-52H Engine Pressure Ratio Versus Throttle Position Angle at 25,000 Ft. Altitude

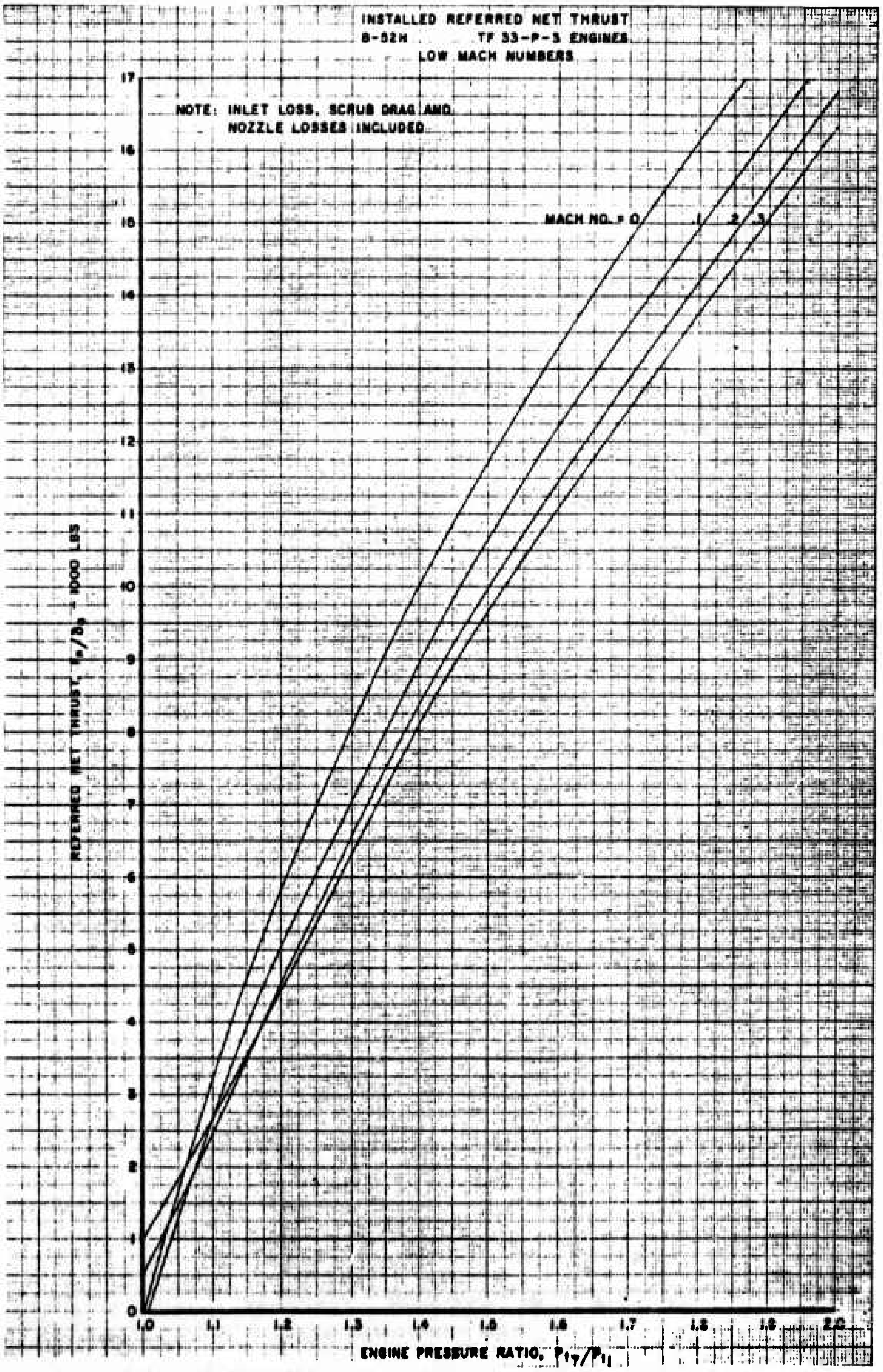


Figure 107. Net Thrust - Low Mach Number

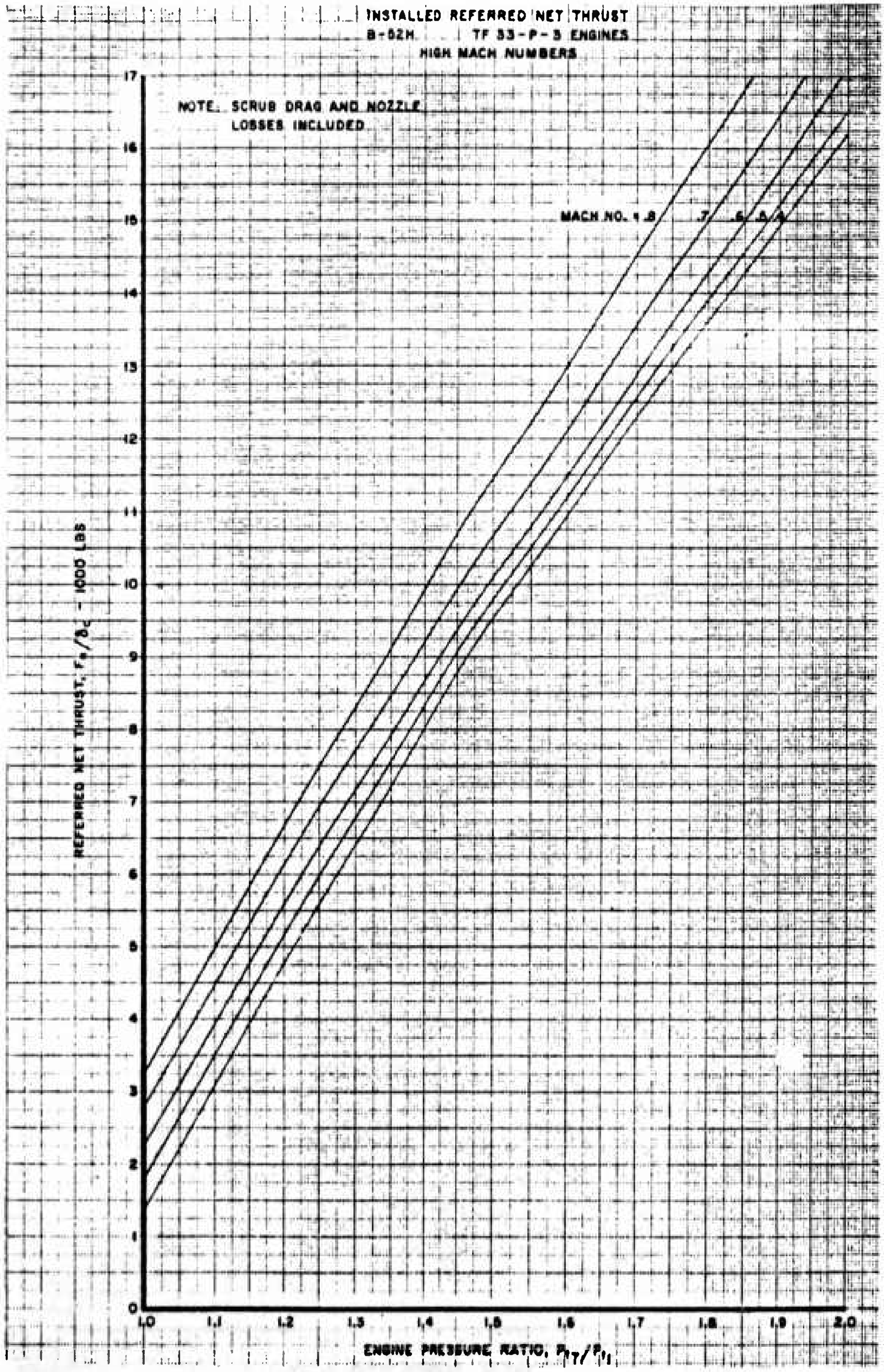


Figure 108. Net Thrust - High Mach Number

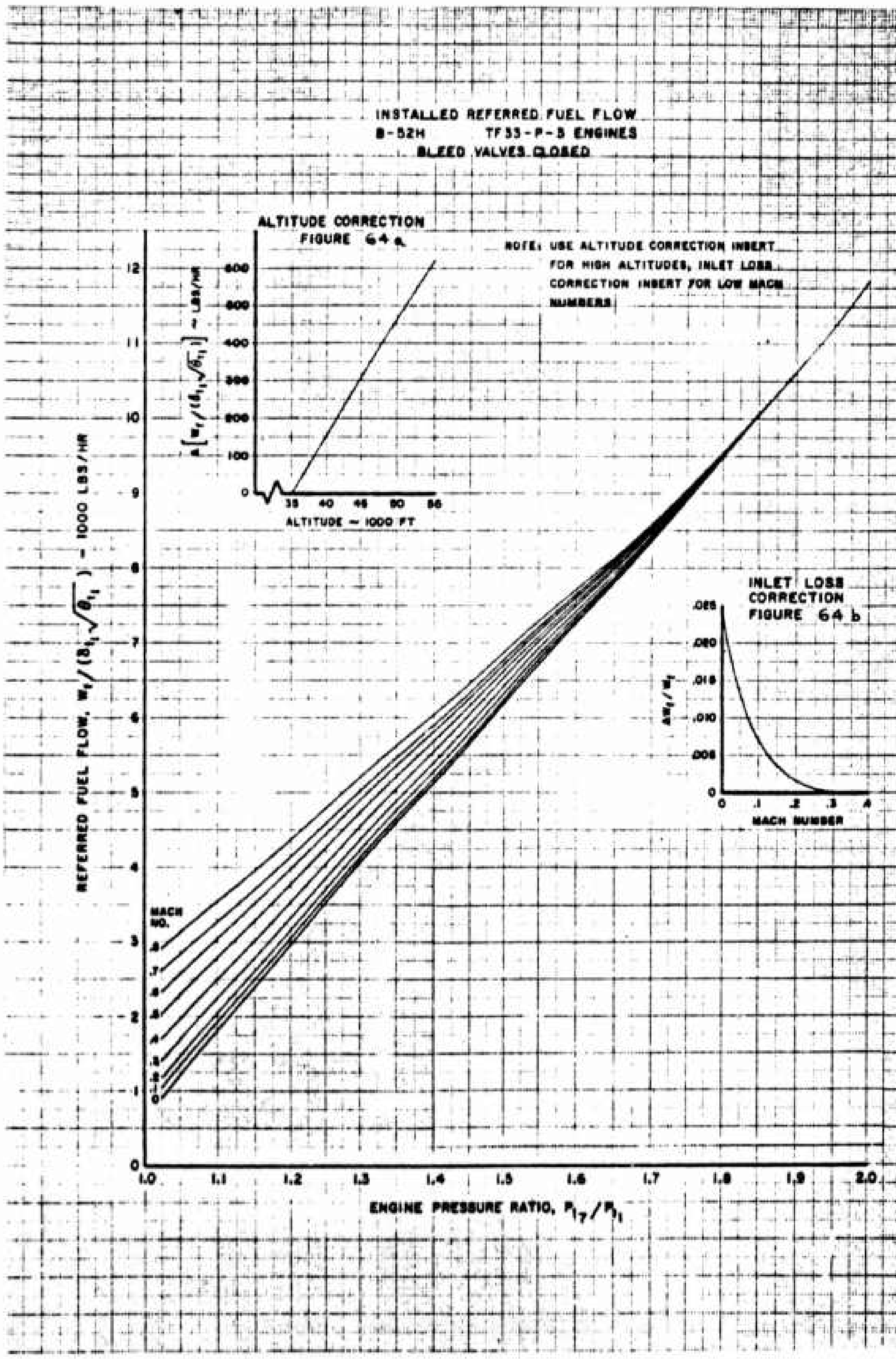


Figure 109. Fuel Flow - Bleed Valves Closed

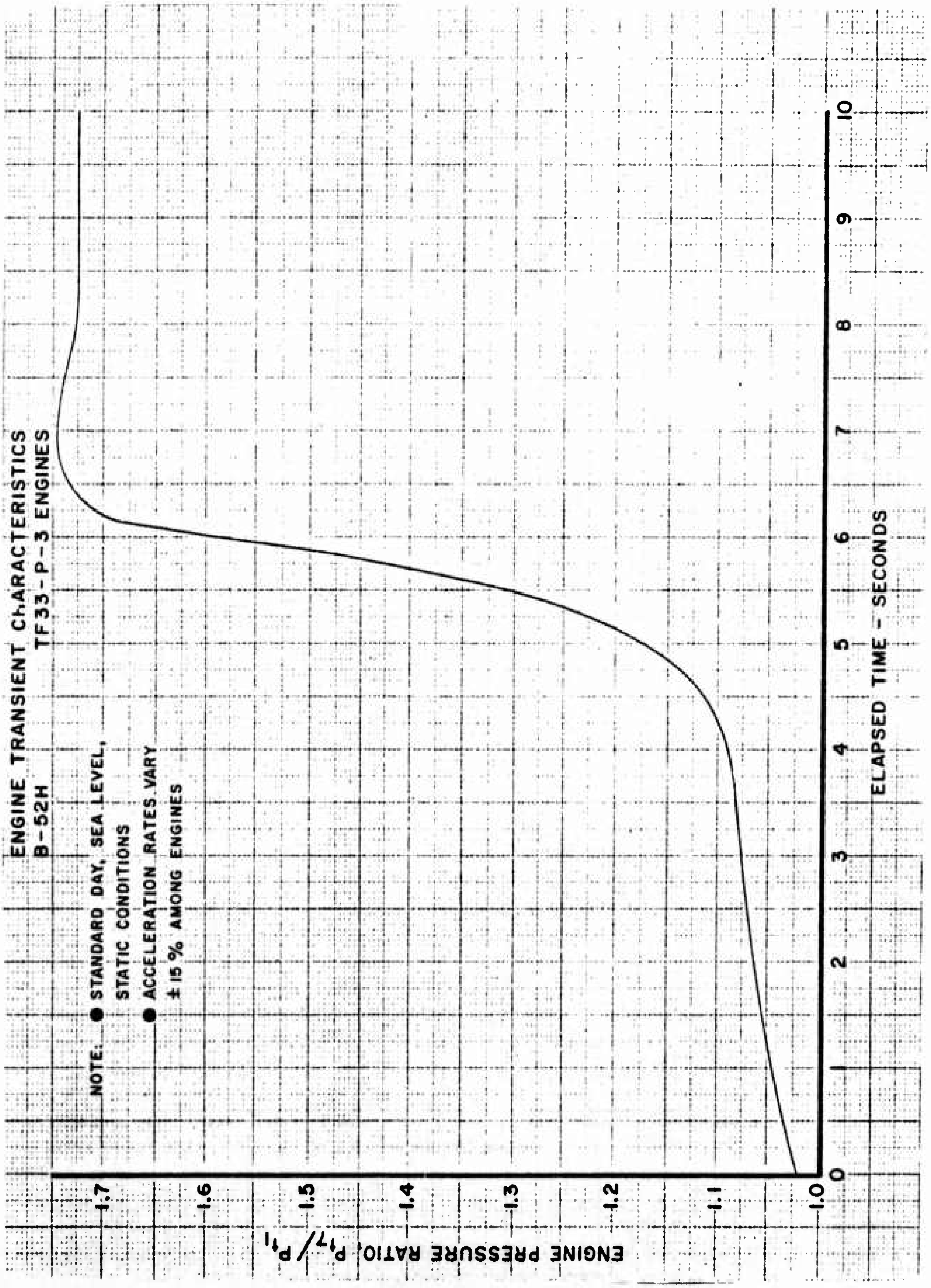


Figure 110. B-52H Engine Pressure Ratio Acceleration Characteristics From Idle to Military Power

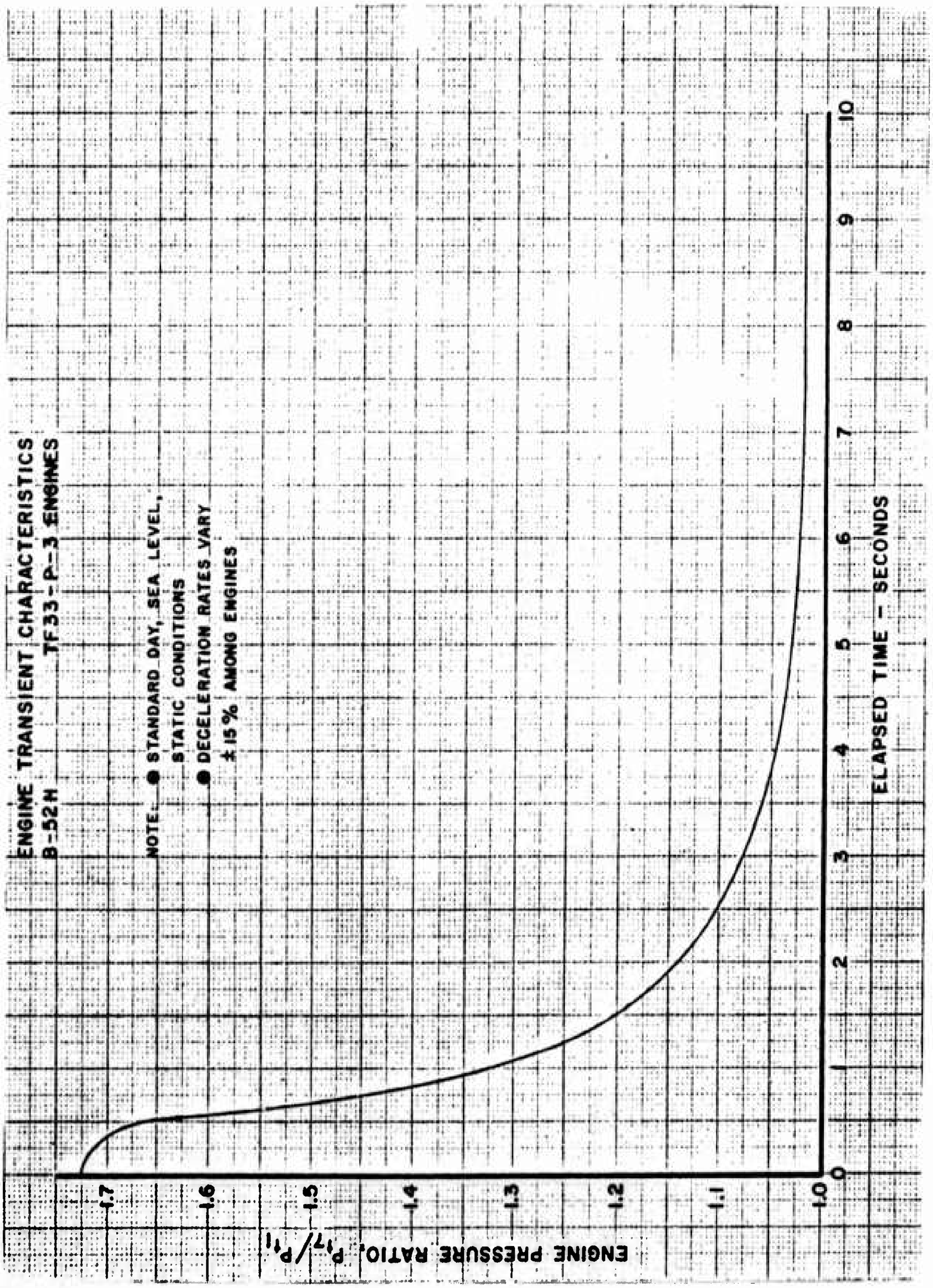


Figure 111. B-52H Engine Pressure Ratio Acceleration Characteristics From Military to Idle Power

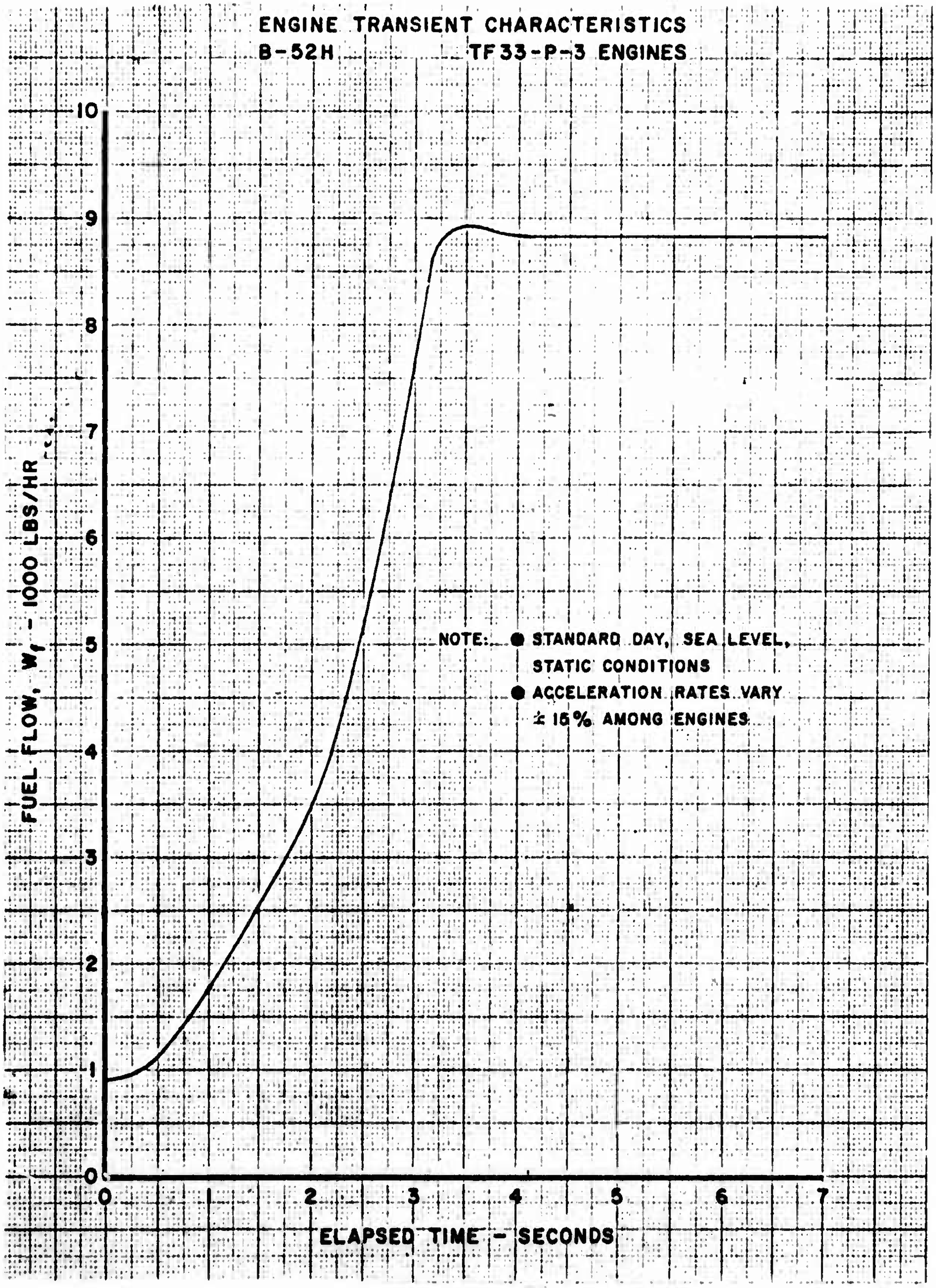


Figure 112. B-52H Fuel Flow Acceleration Characteristics From Idle to Military Power

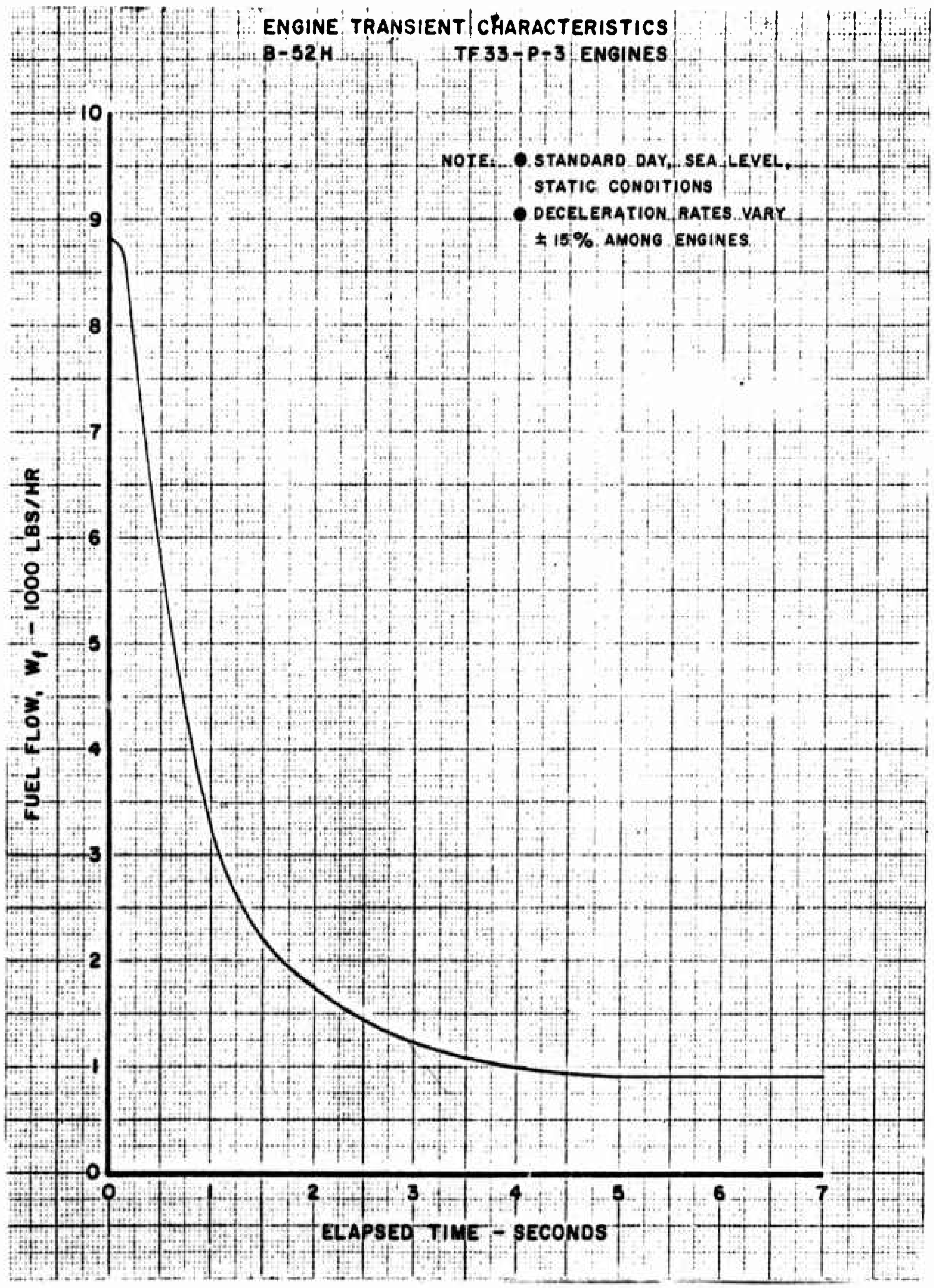


Figure 113. B-52H Fuel Flow Deceleration Characteristics From Military to Idle Power

12. B-52 LONGITUDINAL FLIGHT CONTROL SYSTEM

The longitudinal flight control system consists of the following elements: manual control system, autopilot (AFCS), stability augmentation system (SAS), and hydraulically powered control surfaces (elevator and stabilizer). The elevator drive system is depicted in Figure 114. Note that the autopilot is in parallel with the manual control system while the SAS provides a series input to the elevator.

a. Manual Control System

1) Elevator

Salient features of the elevator system are presented below. Additional details of the elevator control system/autopilot interface are shown in Figure 125.

2) Control Column - Elevator Displacement Relationship

Under no load conditions (zero cable stretch), fore and aft control column deflections of 14.23 degrees generate elevator deflections of $\pm$ 19 degrees. Stops exist at 17.16 degrees of column and $\pm$19 degrees of elevator.

3) Elevator Actuator Force and Rate Limits

The elevator actuators are force limited at $\pm$ 8,200 pounds, and an 80 degree/second no load elevator rate limit exists. The corresponding inflight elevator deflection limits versus airspeed and altitude are defined by Figure 115 for normal operation with full hydraulic power.

4) Elevator Actuator Dynamics

Within the linear range, below actuator rate and position limits, the following transfer function describes elevator actuator response to mechanical input commands.

$$\frac{\Delta \text{Elevator}}{\Delta \text{Elevator Command}} = \frac{(36.5)(17,200)}{(S+38.5)(S^2+34.5\ S+17,200)}\ \frac{\text{deg}}{\text{deg}}$$

5) Elevator Hysteresis

Elevator hysteresis about the hinge line is estimated as 0.1 degrees.

6) Stabilizer

The trim system is discussed in the following paragraphs. Autopilot automatic trim is discussed in Section IX.14.

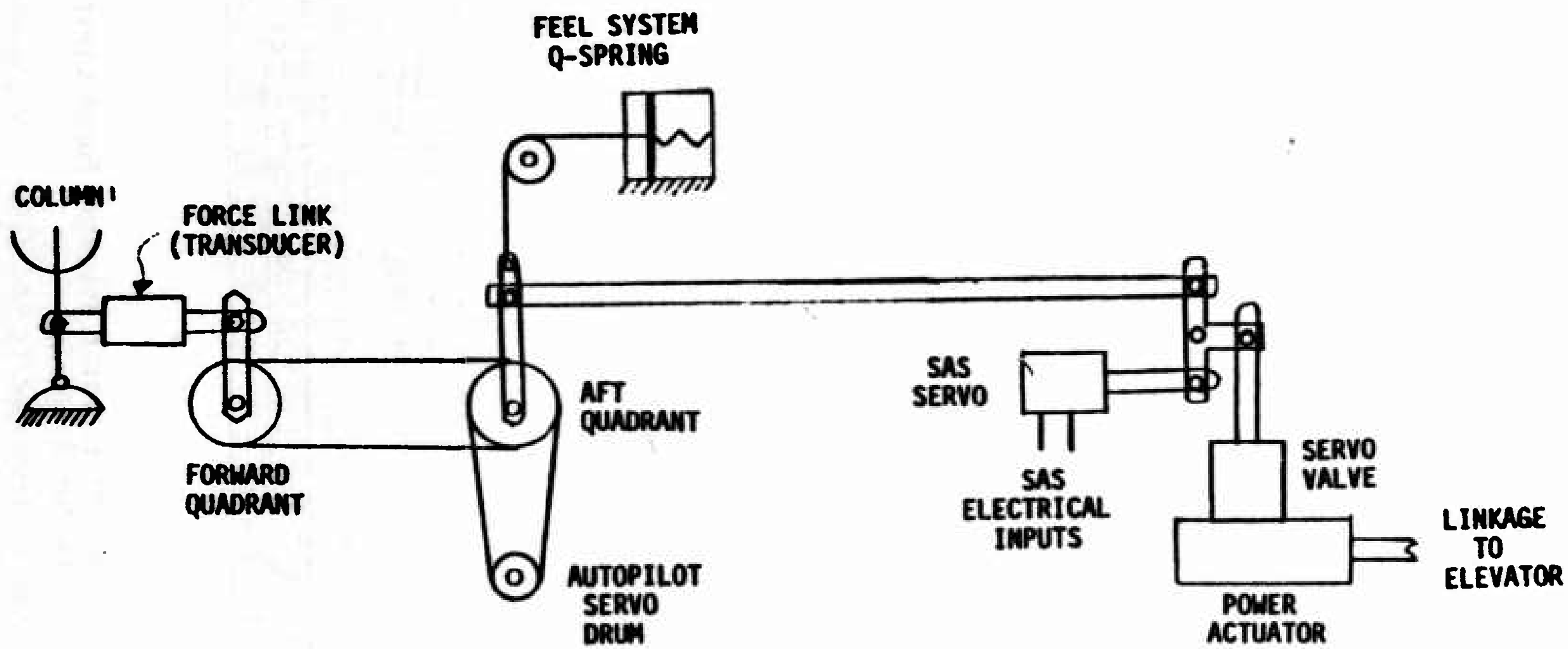


Figure 114. Elevator Drive System (Simplified)

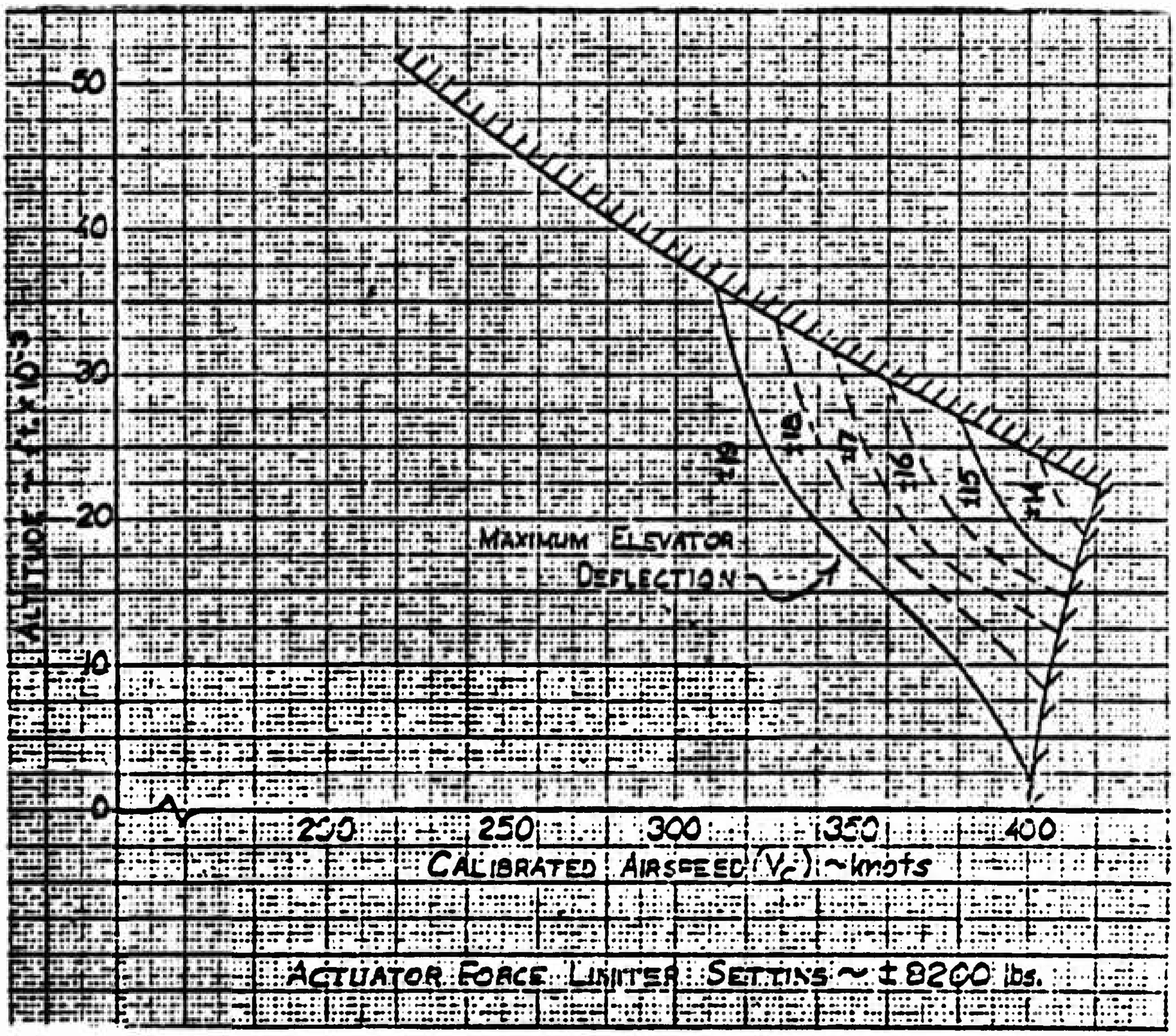


Figure 115. Effect of Elevator Actuator Force Limit (Elevator Blowdown)

7) Trim Wheel - Stabilizer Displacement Relationship

With the trim wheel rotated to the aft stop, the pilot's and copilot's trim indicators indicate 4 degrees (4 "units") nose up and the stabilizer moves to the 4 degree leading edge down position. With the trim wheel rotated to the forward stop, the pilot's and copilot's trim indicators indicate 9 degrees (9 "units") nose down and the stabilizer leading edge deflects up to the 9 degree position. The nominal force required to rotate the trim wheel is estimated at 7.5 pounds applied at the periphery of the wheel and the maximum trim wheel force permitted is limited to 10 pounds. One revolution of the trim wheel yields 1 degree of stabilizer deflection.

The trim wheel may be used to overpower the autopilot stabilizer trim servo or the stabilizer electric trim motor. The overpower force required at the periphery of the trim wheel to stop the autopilot trim servo is 28 to 38 pounds and the force required to overpower the electrical trim motor is 60 to 75 pounds.

8) Stabilizer Electrical Trim

With the aft AC power box stabilizer trim actuator circuit breaker and the pilot's trim control circuit breaker closed and the aisle control stand stabilizer trim cutout switch in the normal position, the stabilizer may be trimmed using one of the two stabilizer trim switches. Electrical operation of the trim system also results in rotation of the manual trim wheels.

9) Stabilizer Trim Rate Limit

Stabilizer rate limits are set by hydraulic flow limits and electrical input rate limits into the followup system. Electrical input into the followup system is equivalent to approximately 0.65 degrees of stabilizer per second. However, stabilizer rate limits are essentially set by hydraulic flow rates under most conditions, and may range from 0.5 to 0.75 deg/sec depending on engine thrust setting.

10) Stabilizer Trim Circuit Breakers, Cutout and Force Switches

Multiple means of disabling stabilizer trim are provided. A trim control circuit breaker marked "Trim Contr" is located on the pilot's circuit breaker panel. The trim function of the trim buttons is also disabled any time the autopilot elevator servo cutout switch is "in" and the autopilot servos are engaged. Stabilizer trim may also be disabled by breaking the trim control circuits with the stabilizer trim cutout switch. Manual control of stabilizer trim is not affected by either the trim control circuit breaker or the trim cutout switch.

Force switches, installed within the control column linkage, also provide the pilots with a means to immediately interrupt stabilizer trim movement. When pilot column force above a threshold of 24 to 36 pounds is applied, power to the trim motor clutch on the side yielding trim opposing the column force is interrupted. Trim remains available in the direction compatible with applied column force through use of the trim buttons. Normal two direction electrical trim operation returns upon relaxation of column force to below the threshold level. In addition, with this column force applied, and with the elevator servo opposing the applied force, the autopilot disengages. The autopilot then remains disengaged until manually re-engaged. Force switch operation does not affect manual trim operation through the trim wheel/cable linkage.

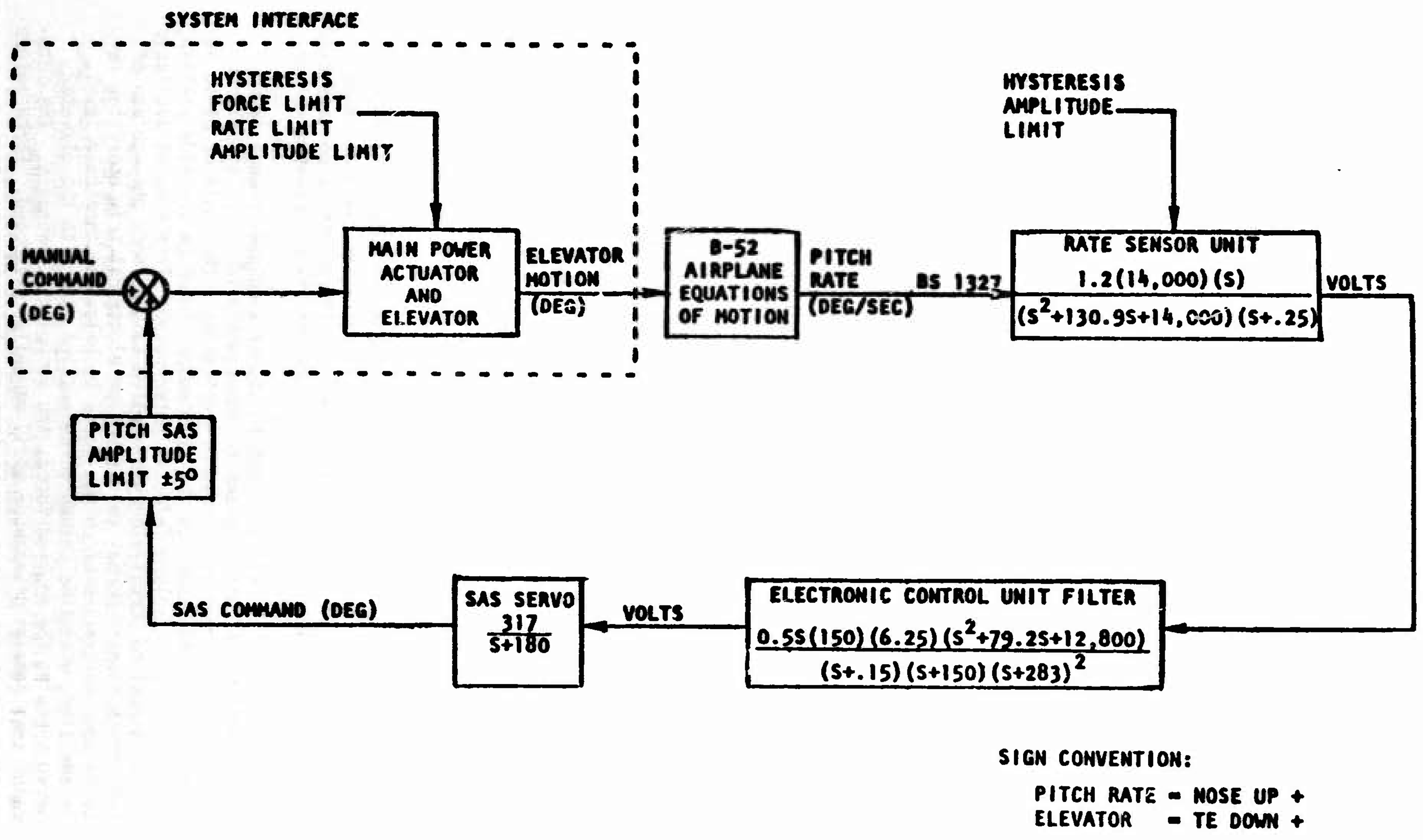


Figure 116. Pitch SAS Block Diagram

13. PITCH STABILITY AUGMENTATION SYSTEM (SAS)

a. Pitch SAS Functional Description

The B-52 SAS provided the contemporary stability augmentation function. Pitch rate is fed back to the elevator through a washout filter to augment short period and structure mode damping.

Triple redundant signal paths with failure monitoring are used, providing fail operation capability. Indicators also provide warning if two associated channel failure of the pitch SAS occur and the SAS disengages.

A SAS control panel is located on the pilot's forward instrument panel. Engage/Disengage switches are provided. These switches are automatically returned to the disengaged position when released by system failure logic circuitry. The SAS receives power through four DC circuit breakers and six AC circuit breakers.

b. Pitch SAS Analytical Description

Figure 116 defines the pitch SAS block diagram and associated linear transfer functions for system elements. The elevator actuator-surface is part of the Manual Control System.

14. AUTOMATIC FLIGHT CONTROL SYSTEM (PITCH AXIS)

This section contains an abridged description of the autopilot pitch axis. The synchronization, stabilization, and low level modes of operation are included, along with the pitch safety functions and automatic trim functions. Details of the interlock system and the air refuel, altitude hold, and automatic approach modes have been deleted.

Air Force T.O. 5A1-2-28-32, A/A 42G-11 Automatic Flight Control System Field Maintenance Instructions, contains a complete description of the AFCS components and theory of operation.

a. Synchronization Mode

The autopilot is in the synchronization mode any time the Pilot switch is on and the Servo switch is in the disengaged position. In this mode, the autopilot servo drum is declutched from the autopilot servo motor and the autopilot does not drive the manual flight control system. The pitch controller follows the airplane pitch attitude to provide smooth engagement when the stabilization mode is engaged.

b. Stabilization Mode

The flight stabilization or attitude hold mode is selected by turning on the Pilot switch and engaging the Servo switch. In this mode the conventional attitude hold function is provided and "hands off" flying is possible. Maneuvering may be accomplished through the Flight Controller while in this mode. Movement of the contol column is restrained by the autopilot servo.

This mode is not suitable for low level flying because precision pitch axis control cannot be obtained using the Flight Controller Pitch Knob located on the aisle stand. At low altitudes, the stabilization mode should be engaged only long enough to ascertain that the autopilot is functioning properly, then the low level mode should be engaged.

c. Low Level Mode

The low level mode may be engaged, providing the flight stabilization mode has been previously selected. This system essentially allows the pilot to perturbate the reference airplane attitude being held by the flight stabilization mode, through use of the control column. Force transducers (links) sense control column force and drive the autopilot giving a "power steering" effect. Automatic "g" limiting gradually decreases this effect as normal acceleration levels increase. Returning the Low Level switch to OFF will disengage the autopilot (i.e., return to synchronization mode).

d. Safety Functions

Two pitch safety functions are provided: the pitch overpower function and the safety monitor function. The figure below illustrates the pitch overpower function, which is operative in all modes. The servo control output and the output of the pilot's force links are compared to determine if the autopilot is aiding or opposing the manual command at the control column. If the autopilot is opposing the manual command and a control column force of over 30 pounds is being applied, the autopilot will be disengaged.

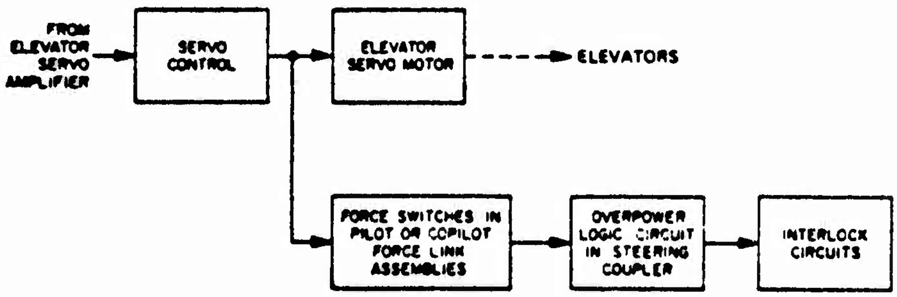


Figure 117. Pitch Overpower Function

The safety monitor function, as shown in Figure 118 protects against excessive pitch angular acceleration due to a malfunction or error in manual control. This safety monitor function operates in all modes except synchronization, low level, and aerial refuel. Using one forward and one aft accelerometer, an angular acceleration signal is derived and compared to the servo control output. If excessive angular acceleration is sensed, and if the autopilot servo control is not operating to alleviate this acceleration, the autopilot will be disengaged when preset limits are exceeded. An excessive servo control command in the absence of a corresponding derived pitch angular acceleration can also result in disengagement.

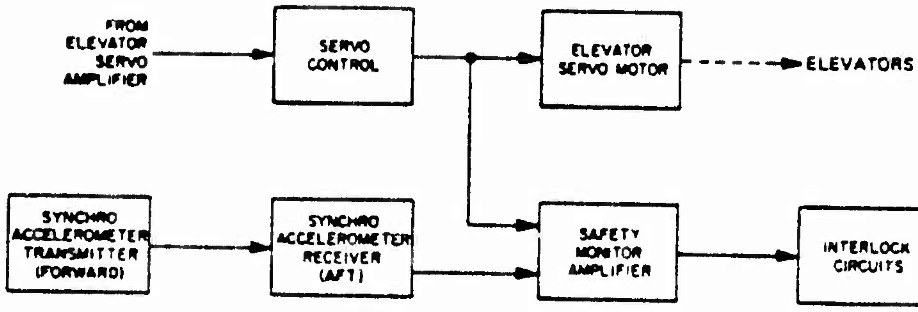


Figure 118. Safety Monitor Function

15. DETAILED SYSTEM DESCRIPTION

a. Command Control Section

An abridged representation of the pitch axis command control section is presented in Figure 119. Physical signal representations (400 Hz AC, 3 wire synchro, and synchro shaft position) have not been preserved.

Mode selection is performed with the three position switch. The interlock system restricts mode switching to the sequences indicated in the mode transition diagram. In the low level mode, the pitch knob is clutched to a detent switch and any attempt to rotate the pitch knob will disengage the autopilot.

The command control section accepts inputs from the following units: pitch attitude gyro, flight controller pitch knob, and the steering coupler (see Figure 127). The data is processed and used to drive the pitch servo amplifier (see Figure 122).

A simplified representation of the system in the engaged configuration is shown in Figure 120. The output of the pitch command synchro and the pitch integrator is summed by the pitch integrator synchro to form the pitch reference attitude. This reference attitude is introduced as a command to the pitch followup loop (first order lag) by the pitch followup synchro. The output of the pitch followup loop (shaft position) is introduced as a command to the stabilization loop through the elevator synchro. Additional commands may be introduced to the pitch followup loop through the summing junction as indicated. In the low level mode, the steering coupler provides commands to the points labeled "INPUTS".

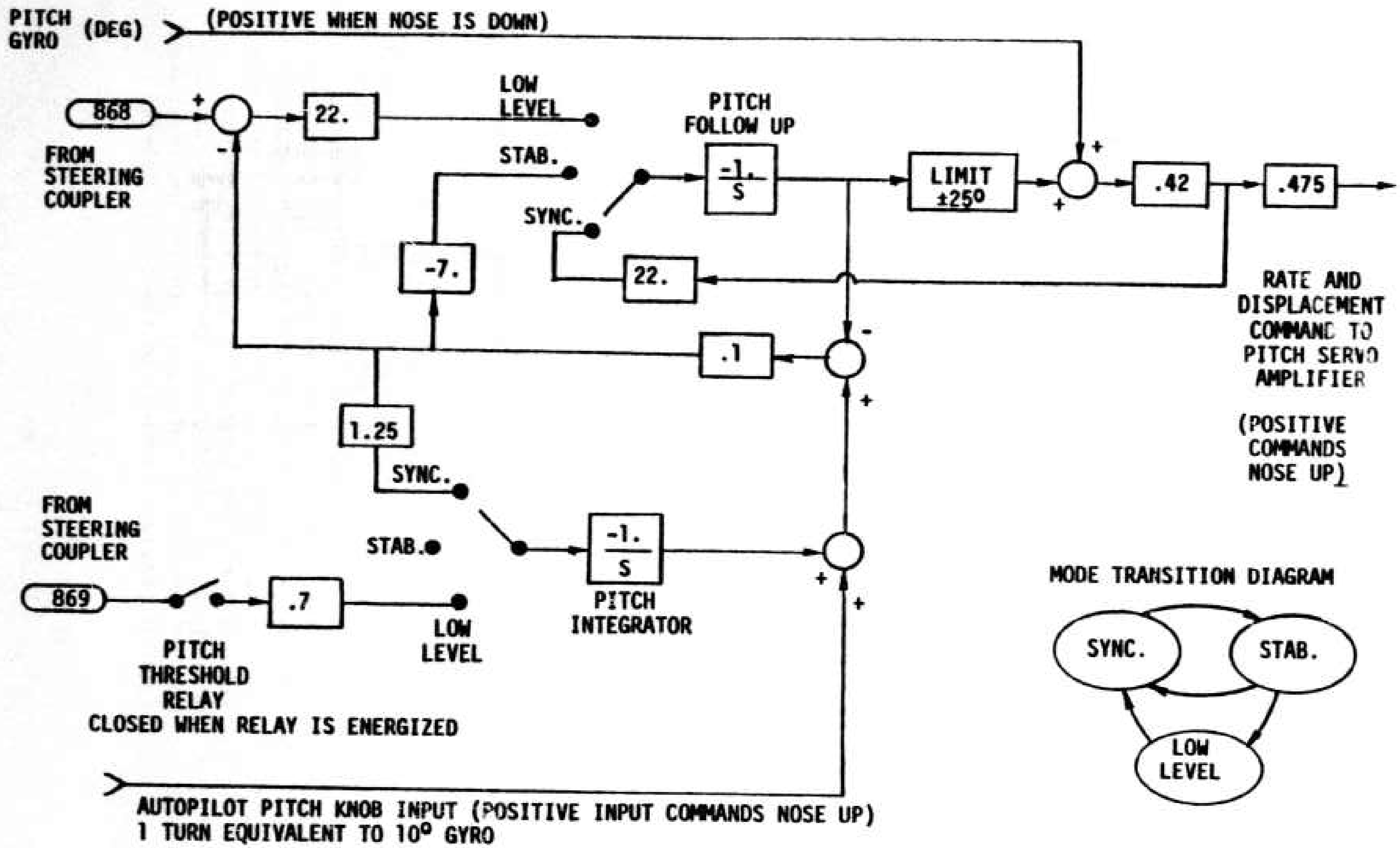


Figure 119. Pitch Axis Command Control Block Diagram (Synchronization, Stabilization, and Low Level Modes Only)

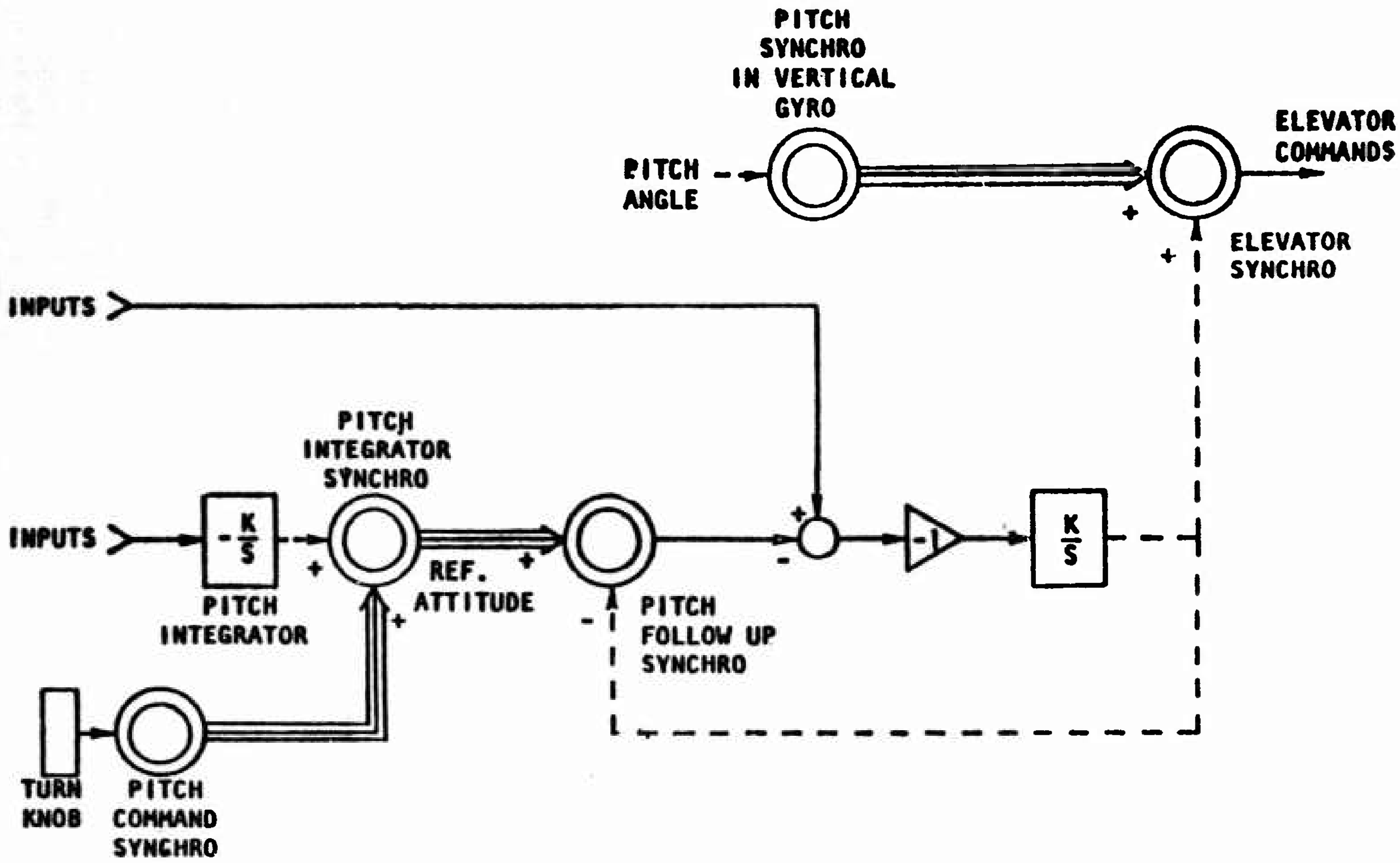


Figure 120. Simplified Pitch Configuration When Engaged

In the disengaged configuration (synchronization mode) the command sequence is reversed, as illustrated in Figure 121. A given pitch angle sensed by the vertical gyro drives the pitch followup shaft in an equal and opposite direction. This then commands the pitch followup shaft in an equal and opposite direction. This then commands the pitch integrator (which now has a feedback loop closed around it). In the steady state the pitch integrator shaft rotation will be synchronized with the aircraft pitch attitude. Thus, when the system is engaged, the reference attitude (pitch integrator synchro output) corresponds to the aircraft attitude and engage transients are thereby minimized.

b. Servo Amplifier

A block diagram of the pitch servo amplifier is presented in Figure 122. Four basic input signals drive the servo amplifier. The rate command from the command control section is demodulated to DC and passed through a derived rate network. The displacement command from the command control section is summed with a position feedback signal developed by the repeatback synchro. These 400 Hz AC signals are transformer coupled to a signal derived from the servo control output voltage. Tap switches on the amplifier determine relative gains and the derived rate filter break frequencies.

The DC and AC signals are summed and amplified in the mixer and power amplifier stages of the servo amplifier. Plate and cathode voltages are 400 Hz AC and the mixer and power amplifier stages conduct on alternating half cycles producing a half wave rectified output. For DC inputs the output waveshape has a sine form while for 400 Hz AC inputs the output waveshape has a sine squared form. The gains labeled Form Factor account for the changes in RMS output as a function of the input waveform. The waveshape of the servo controller output may vary and an average form factor has been used.

The power amplifier stage is subject to saturation as indicated on Figure 122.

The servo amplifier output provides field excitation to the servo control unit, a constant speed motor-generator power amplifier.

c. Servo Control, Servo Motor and Drive Assembly

The servo control and servo motor and drive assembly are shown in a general form in Figure 123. Figure 124 defines the servo control hysteresis. The servo control armature resistance has been combined with the load resistance to form the total resistance R_T. Table XXV defines the motor constants and gear ratios for each axis.

When the servo clutch is engaged, the motor back drives the manual flight control system and the feel system. The servo motor delivers torque to the mechanical system through the servo drum. Servo drum rate affects delivered torque through the motor back EMF coupling represented by K_E.

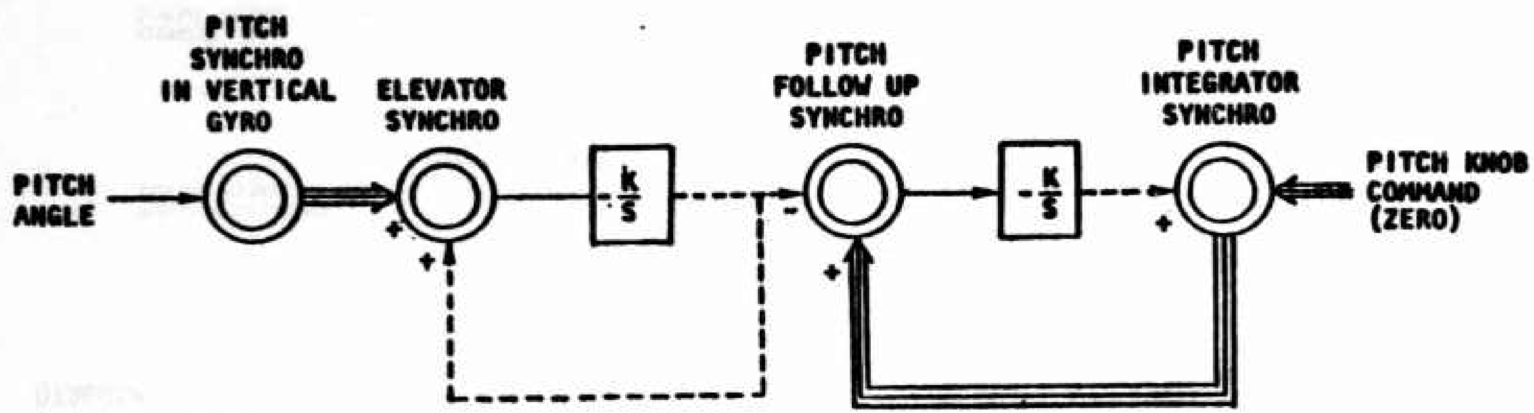


Figure 121. Simplified Pitch Synchronization Configuration

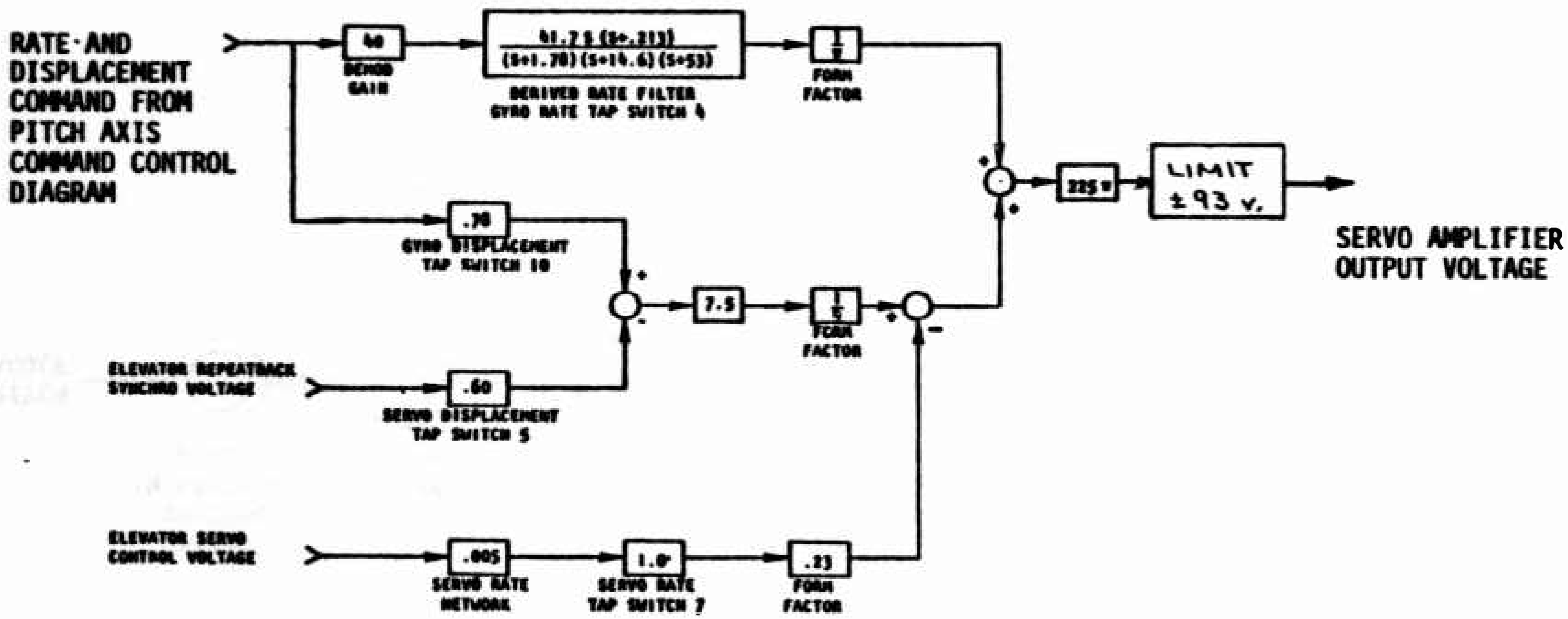


Figure 122. Pitch Servo Amplifier Block Diagram

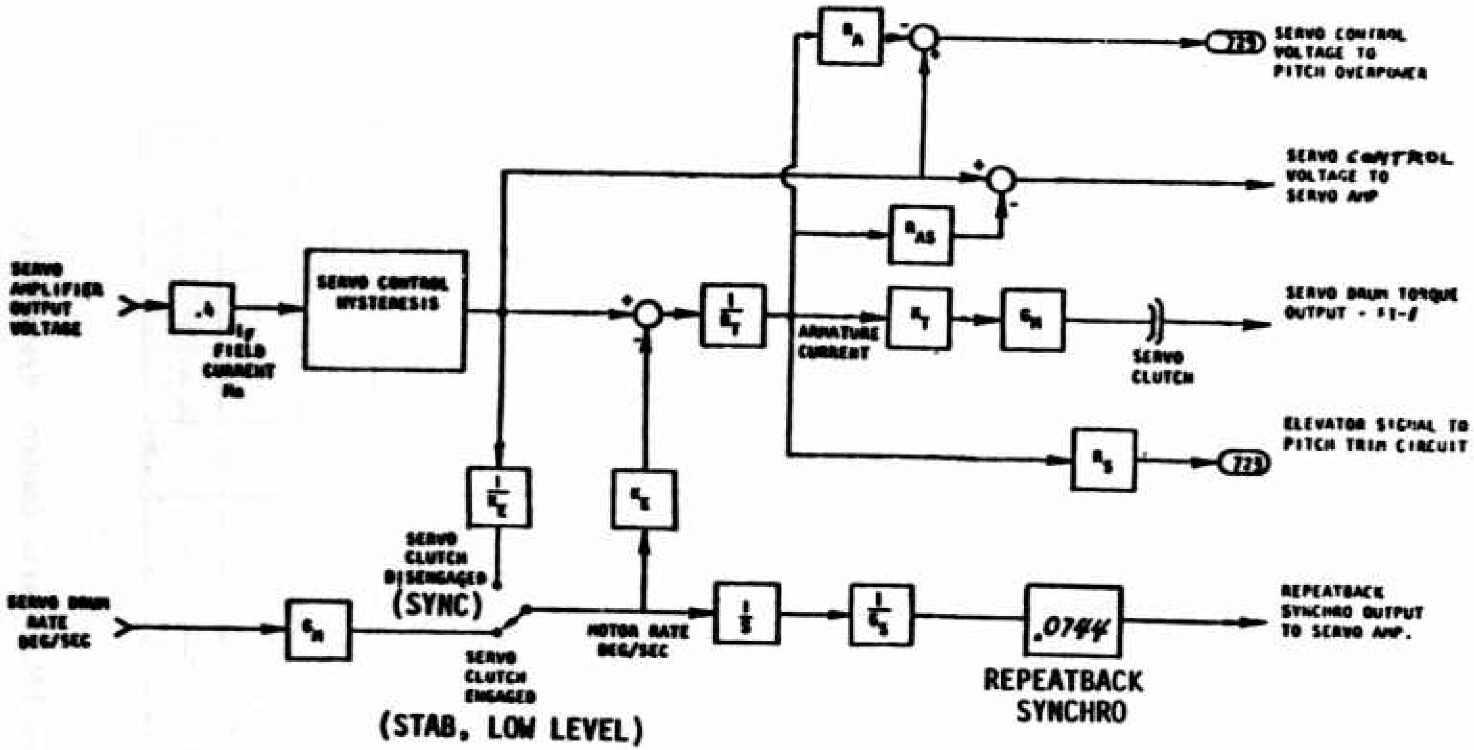


Figure 123. Servo Control and Servo Motor and Drive Assembly Block Diagram

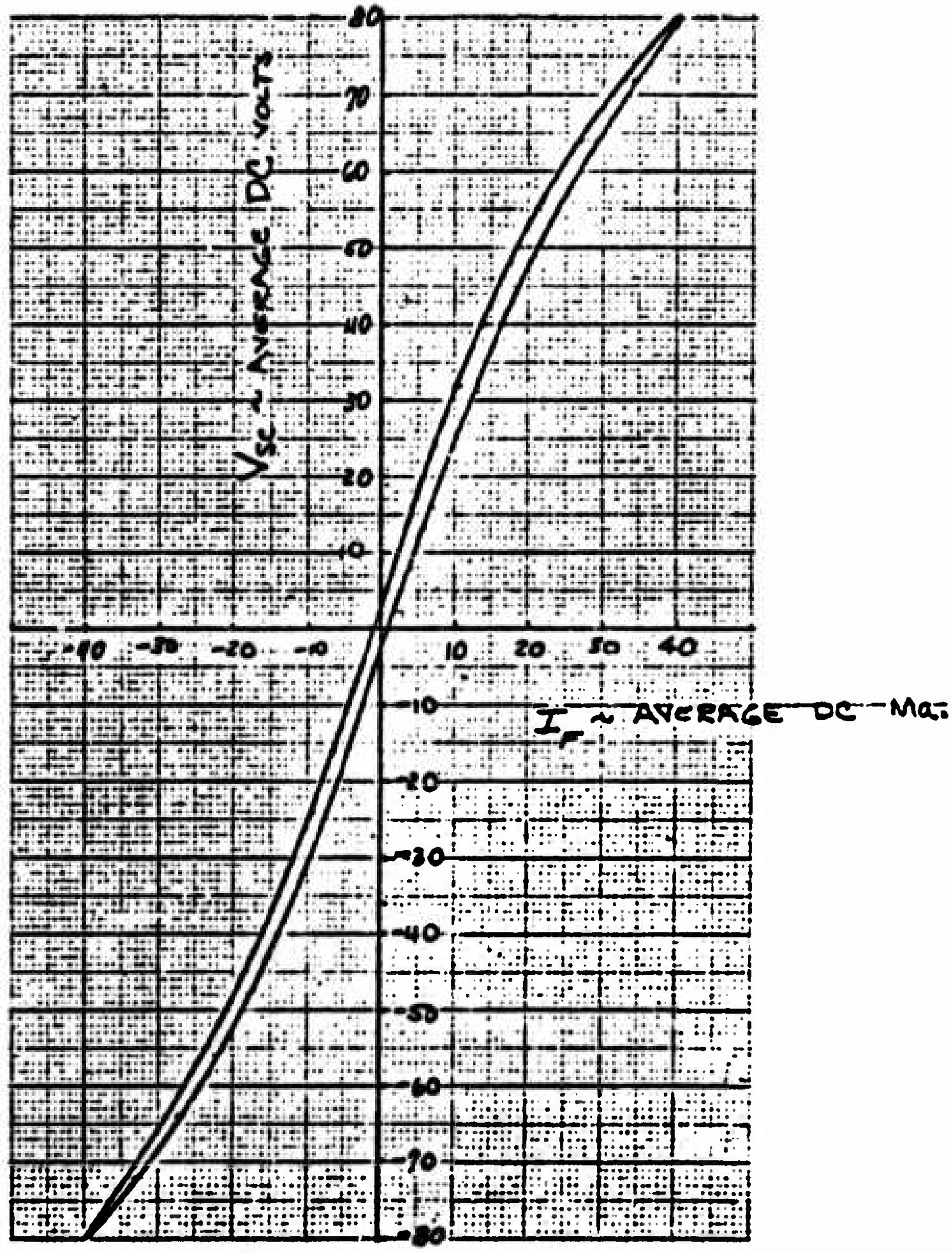


Figure 124. Servo Control Hysteresis

TABLE XXV

ELEVATOR SERVO MOTOR AND DRIVE ASSEMBLY CONSTANTS

SYMBOL	DESCRIPTION	VALUE	UNITS
K_T	Motor Torque Constant	.0887	$\frac{FT-\#}{AMP}$
K_E	Motor Back EMF Constant	2.55×10^{-3}	$\frac{VOLTS}{DEG/SEC}$
G_M	Power Gear Ratio	180	-
G_S	Synchro Gear Ratio	1125	-
R_A	Servo Control Armature Resistance	12	Ohms
R_{AM}	Motor Armature Resistance	5	Ohms
R_L	Line Resistance	0.7	Ohms
R_S	Series Resistance	10	Ohms
R_{AS}	$R_A + R_S$	22	Ohms
R_T	$R_A + R_{AM} + R_L + R_S$	27.7	Ohms

The motor is also geared to a repeatback synchro which provides position feedback to the servo amplifier. Note that the repeatback synchro is always driven by the motor and cannot be declutched.

During sycnrhonization the autopilot is declutched from the control system and the servo motor is virtually unloaded. Under this condition, motor rate is approximately proportional to motor input voltage.

The voltage applied to the servo motor is fed back to the servo amplifier. A signal proportional to servo armature current is used to control the automatic pitch trim system. This signal is sensed across a ten ohm series resistor.

d. Interface to Primary Flight Control System

Autopilot servo drum torque output is transmitted via cable and summed with pilot input and feel system torques to drive the manual flight control system. Figure 125 illustrates the mechanical linkages and dynamic relationships for the pitch axis. Figure 126 defines the feel system forces applied to the aft quadrant.

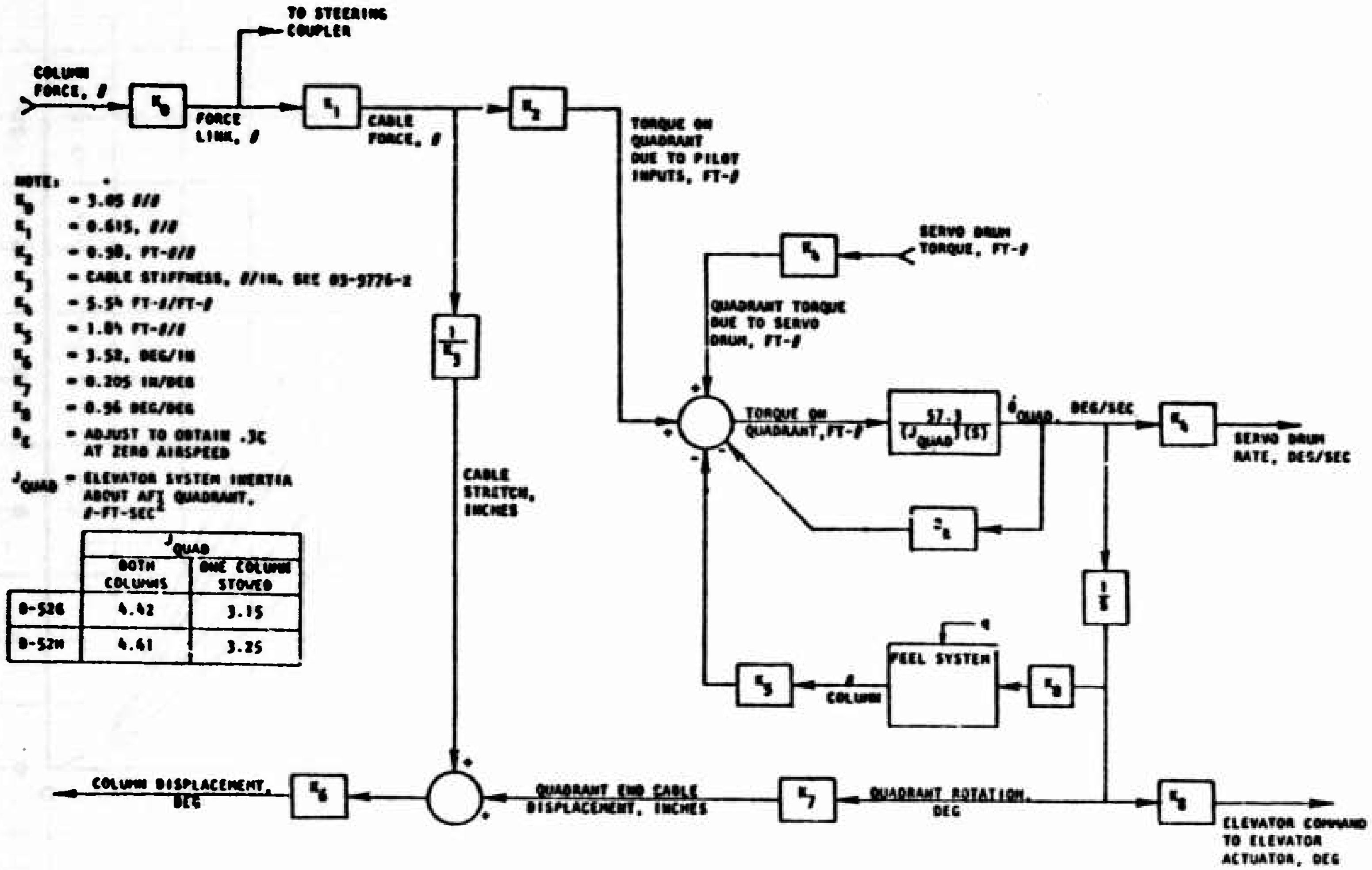
e. Pitch Axis Steering and G-Limiting

Figure 127 defines the pitch steering and g-limiting transfer function. Pilot/copilot commands through the control columns and aircraft acceleration detected by the pitch accelerometer provide input commands. After processing, the signals are mixed to provide control surface rate and position commands to the amplifier assembly.

The pitch and acceleration channels each provide threshold circuitry to energize the pitch threshold relay, K8587. This relay is energized if either or both thresholds are exceeded. In the energized state, this relay enables the pitch integrator servo to respond to steering signals as shown on Figure 119 and disables the pitch trim servo as shown on Figure 133. Pitch steering electronic gain and deadzone are defined on Figure 128. Figure 129 defines the accelerometer gradient. The acceleration gain and deadzone provided by the steering coupler electronics are defined on Figure 130.

f. Pitch Overpower Control

Pitch overpower control is defined on Figure 131. Overpower switches are actuated through the pilot/copilot control columns to enable overpower control. The elevator servo control voltage is then processed through the steering coupler electronics to determine if the elevator servo voltage is aiding or opposing the control column command. If the elevator servo voltage is opposing and exceeds the overpower threshold, the overpower relay is energized. Energizing the overpower relay disengages the servo switch and return the aircraft to manual flight control (synchronization mode).



	J_{QUAD} BOTH COLUMNS	J_{QUAD} ONE COLUMN STOWED
B-52G	4.42	3.15
B-52H	4.41	3.25

Figure 125. Elevator Control System/Autopilot Interface Diagram

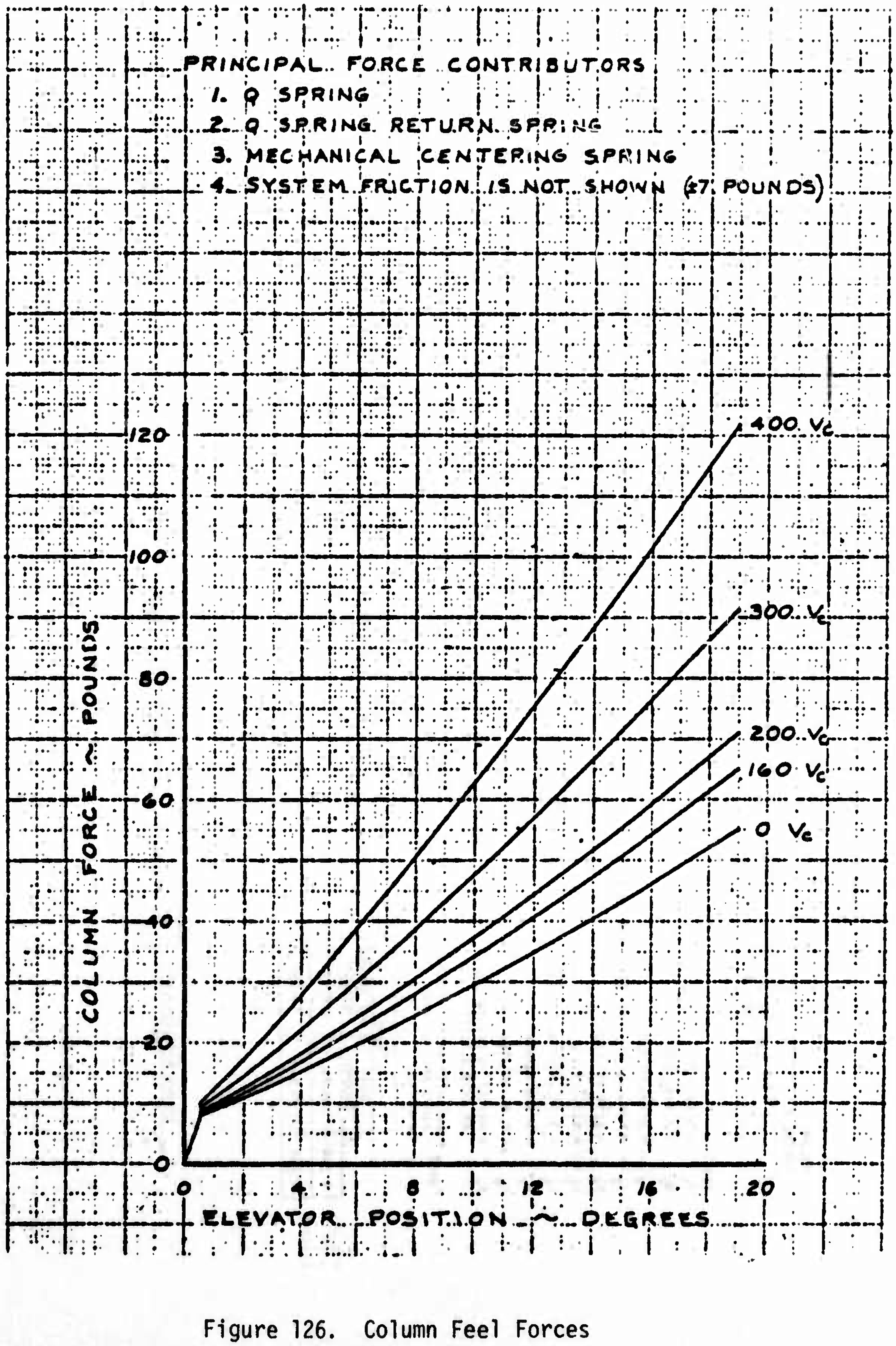


Figure 126. Column Feel Forces

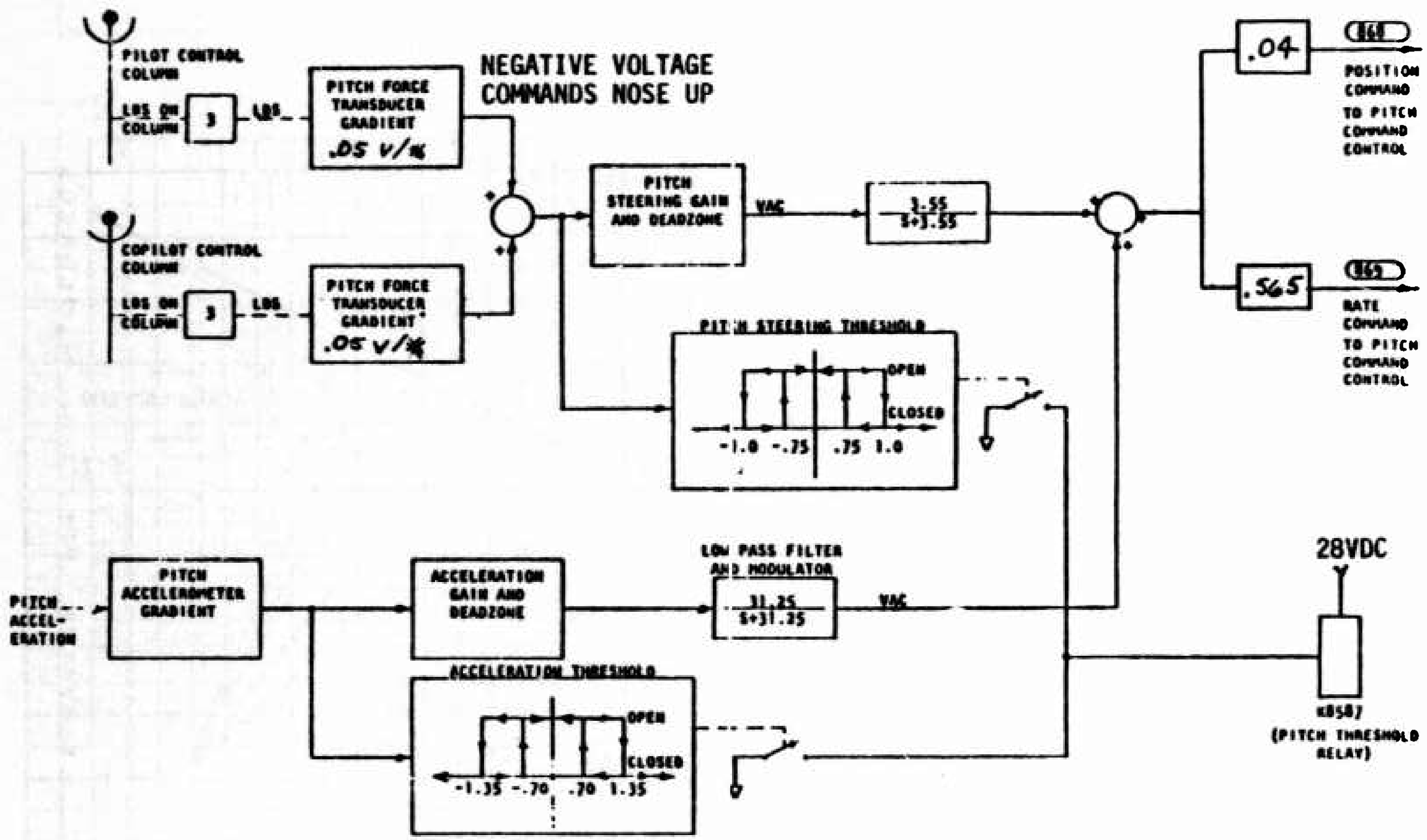


Figure 127. Pitch Steering Coupler Block Diagram

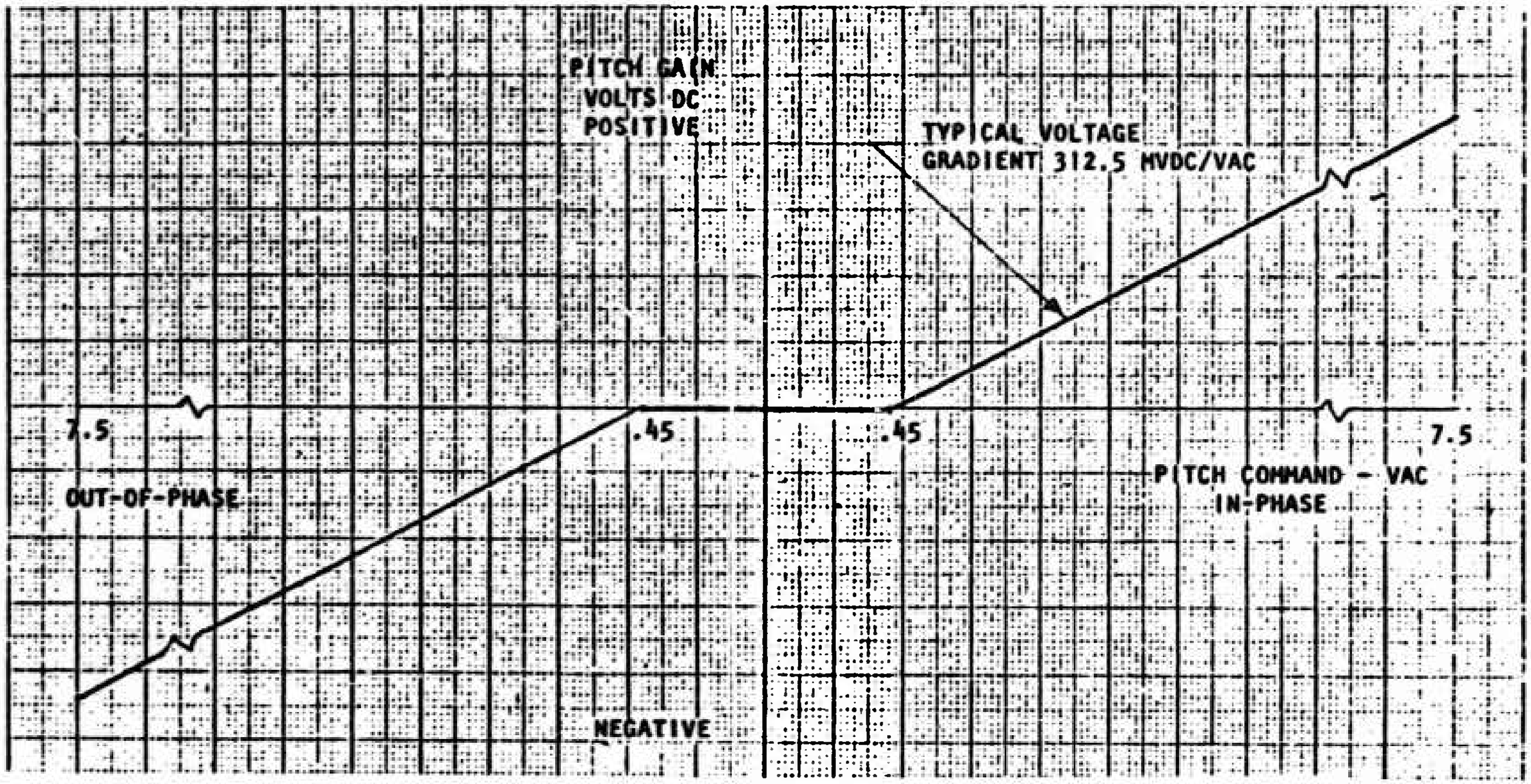


Figure 128. Pitch Steering Gain and Deadzone

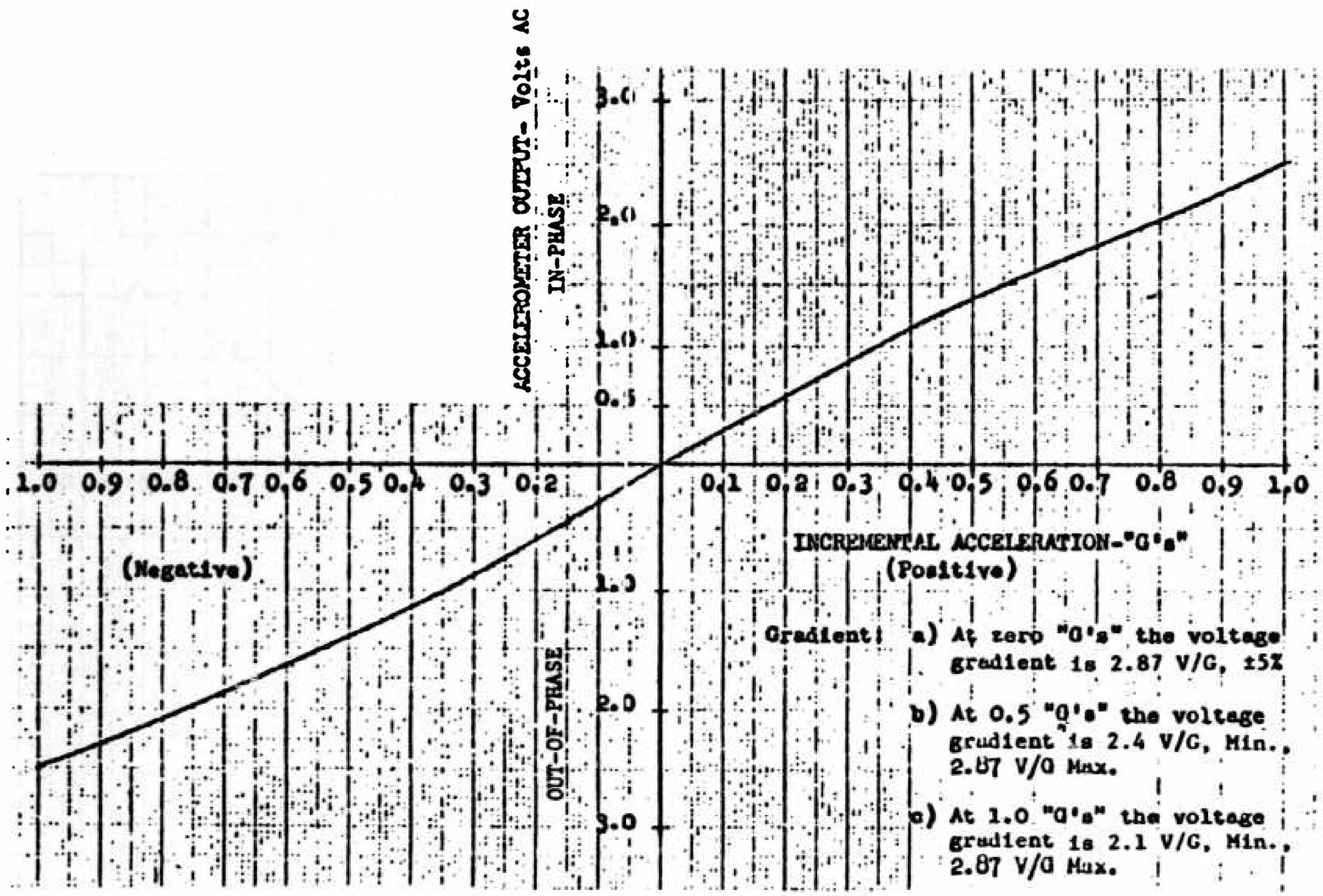


Figure 129. Pitch Accelerometer Gradient

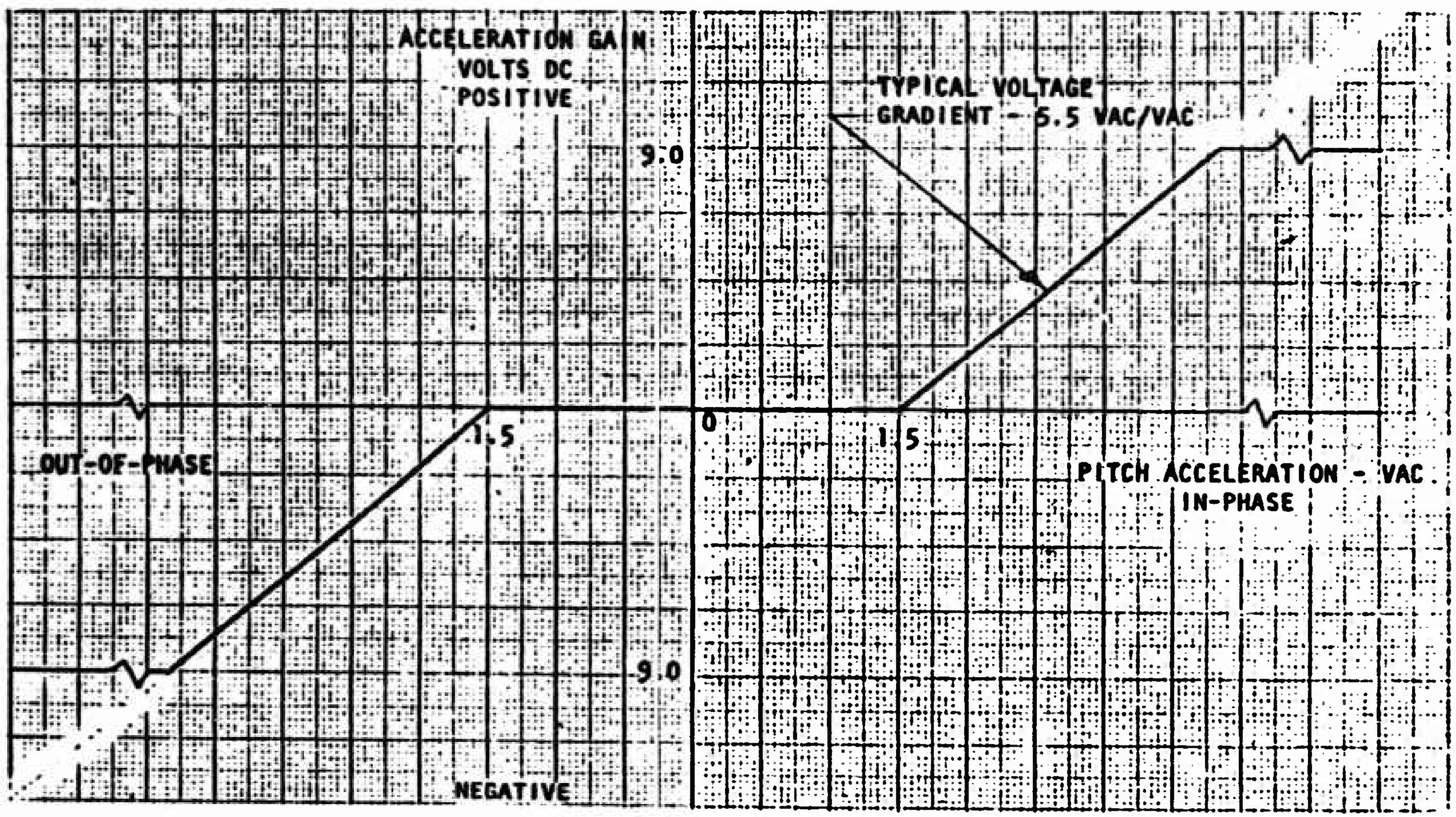


Figure 130. Acceleration Gain and Deadzone

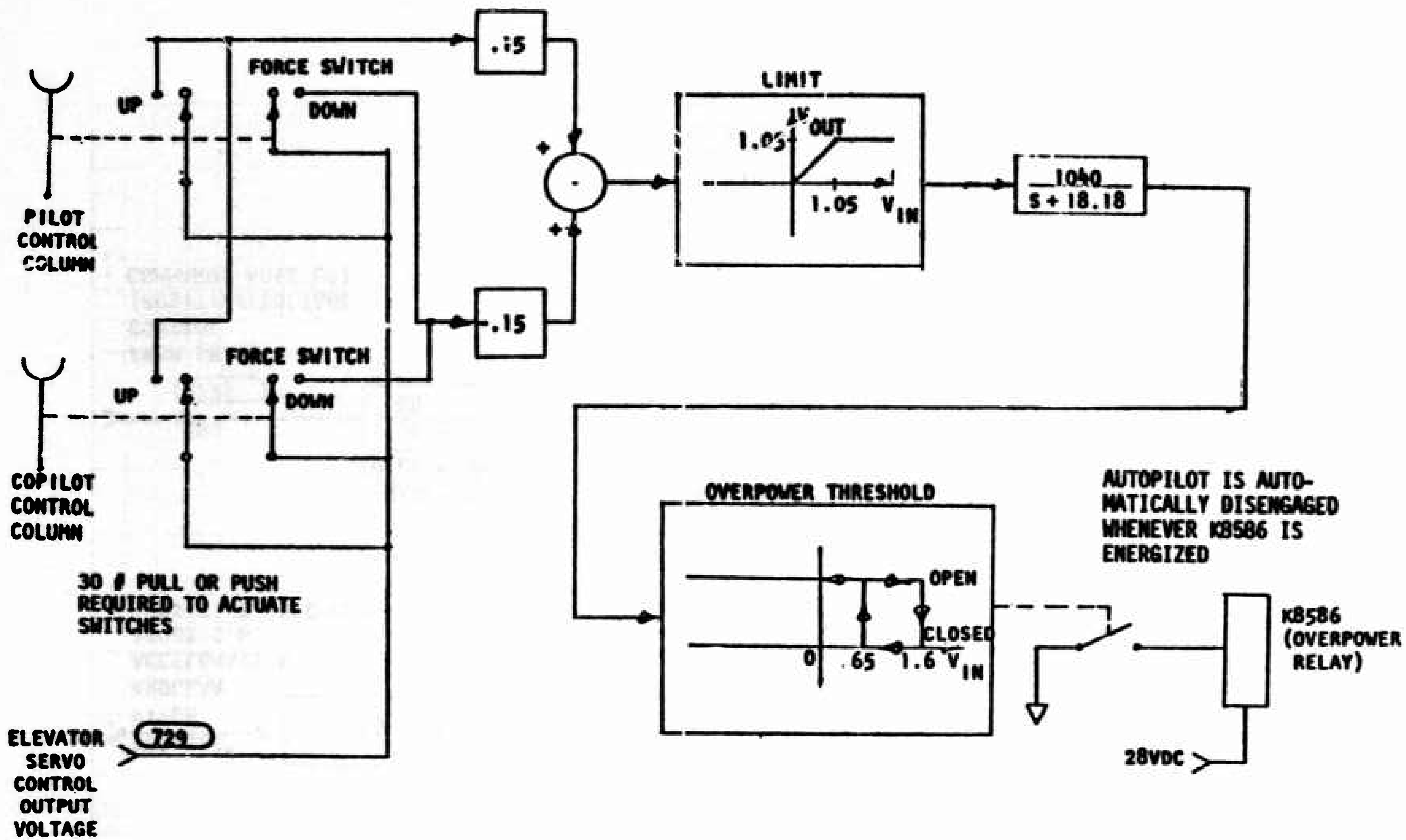


Figure 131. Pitch Overpower Block Diagram - Steering Coupler

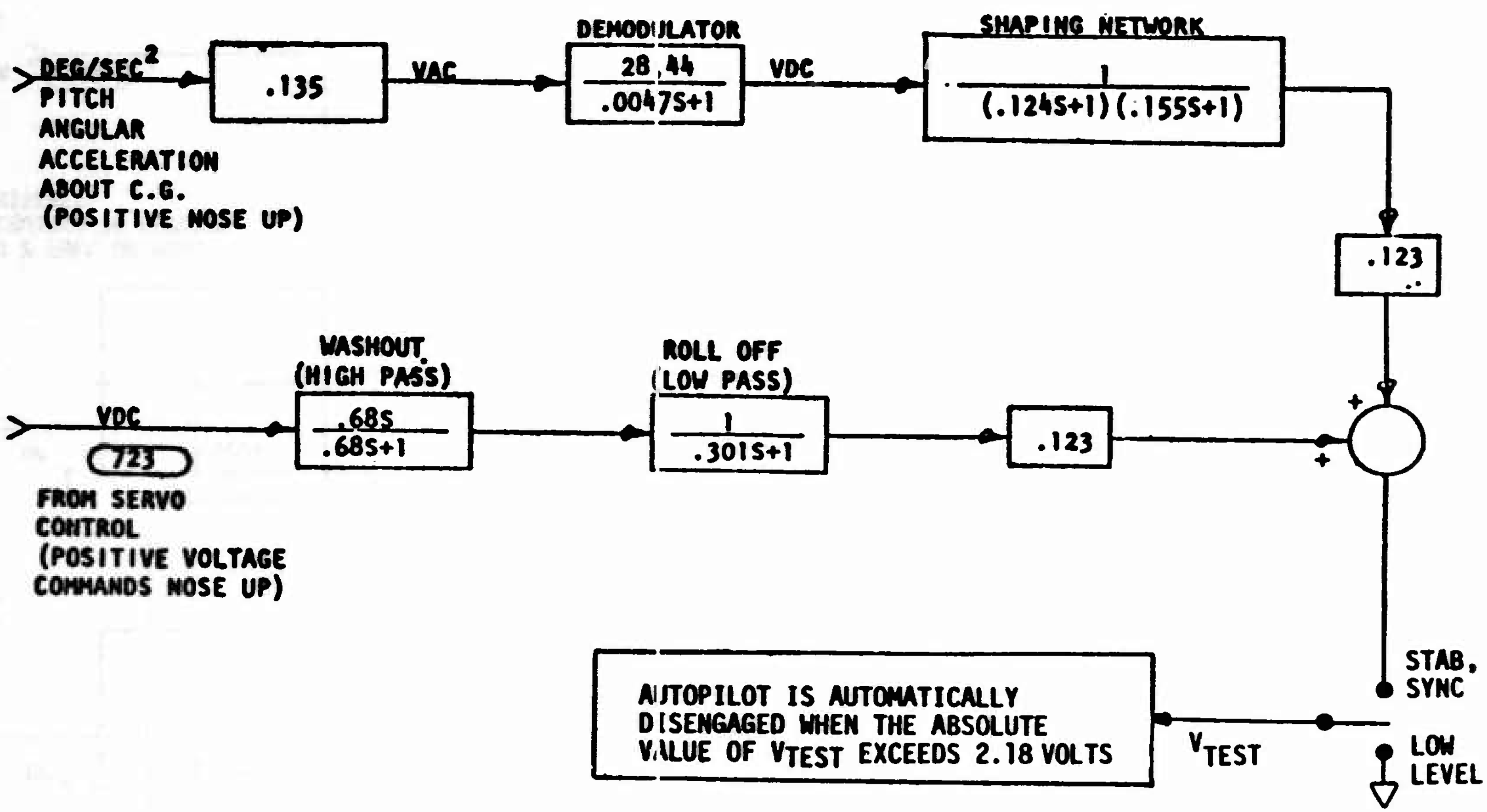


Figure 132. Safety Monitor Amplifier Block Diagram

g. Safety Monitor Amplifier

The Safety Monitor Amplifier is defined by Figure 132. Aircraft pitch angular acceleration and servo control output voltage are monitored to energize and/or de-energize automatic cutoff relays.

If the signal difference between the angular acceleration signal path and the servo control signal path exceeds the limit of ±2.18 volts, the AFCS will automatically disengage. The safety monitor is bypassed in the low level mode.

h. Automatic Pitch Trim Control System

The AFCS automatic pitch trim function is defined by Figure 133. When the appropriate interlock conditions are satisfied, power will be available at terminal 839. A servo controller voltage of sufficient magnitude at terminal 723 will energize the current sensitive automatic trim control relay. Depending on the polarity of the amplifier input, either trim power relay K4606 or K4607 will be energized to provide a signal path for the servo motor armature. Autopilot trim servo displacement provides a command to the stabilizer actuator for stabilizer trim.

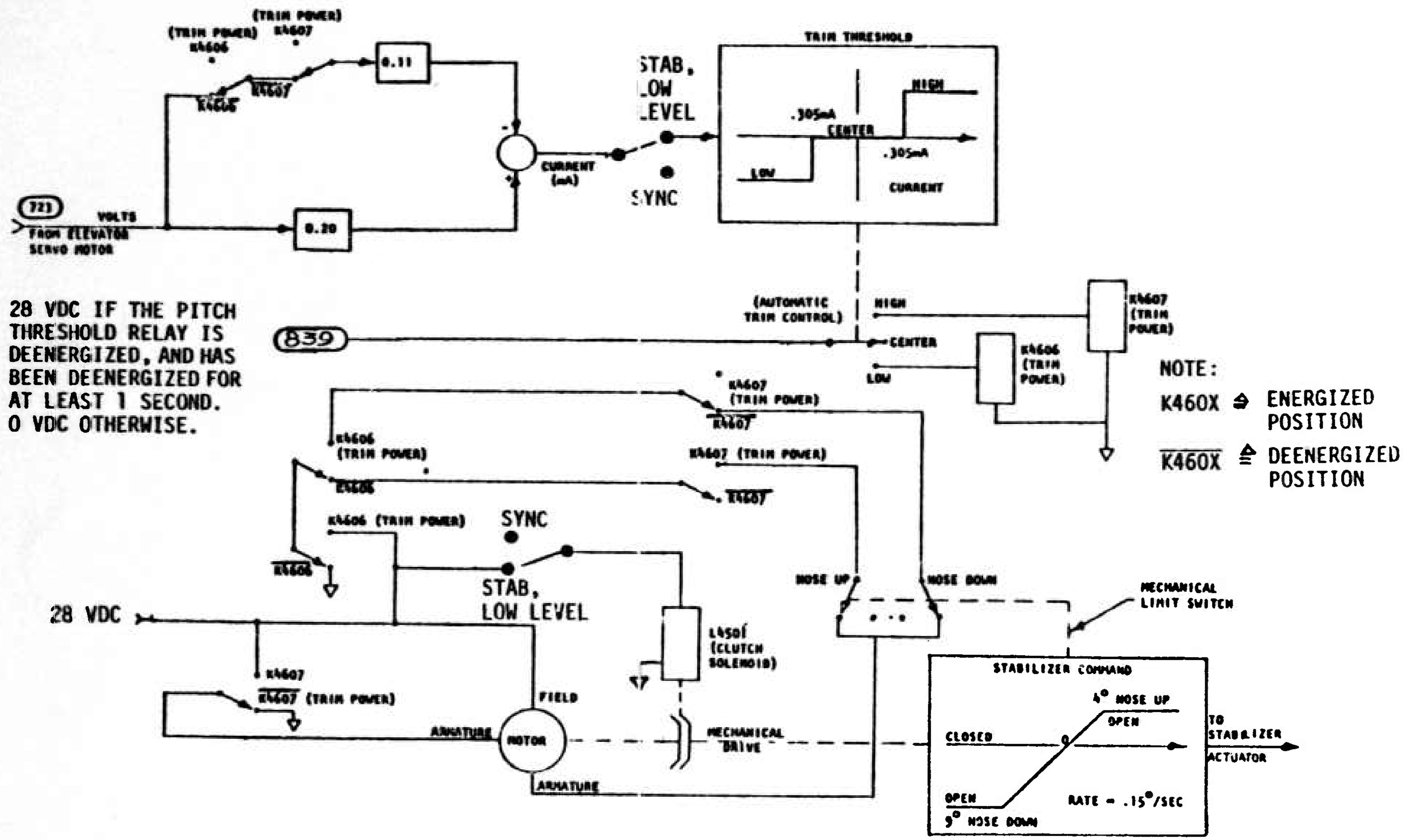


Figure 133. Pitch Trim Control Block Diagram

APPENDIX

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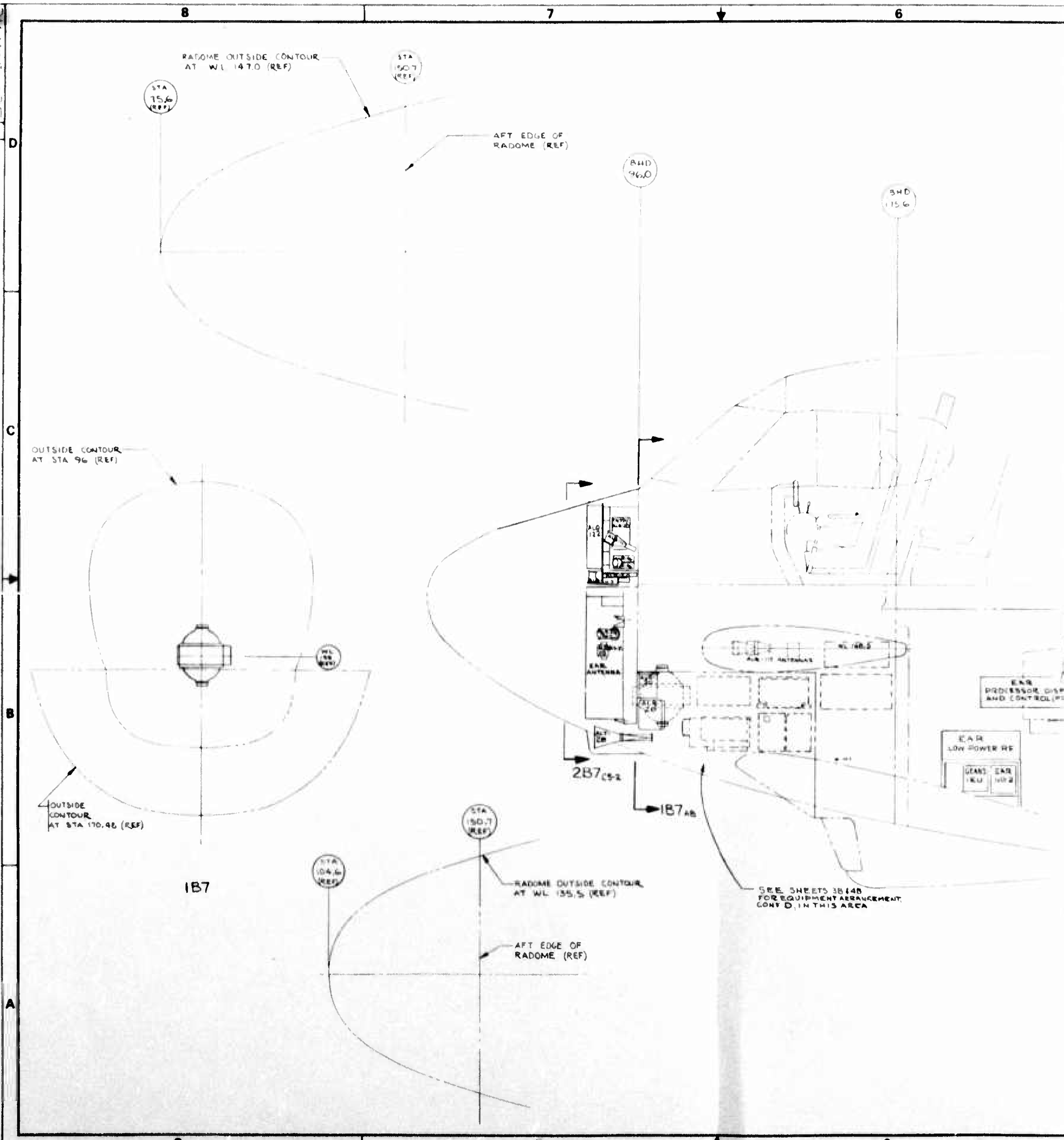

8
7
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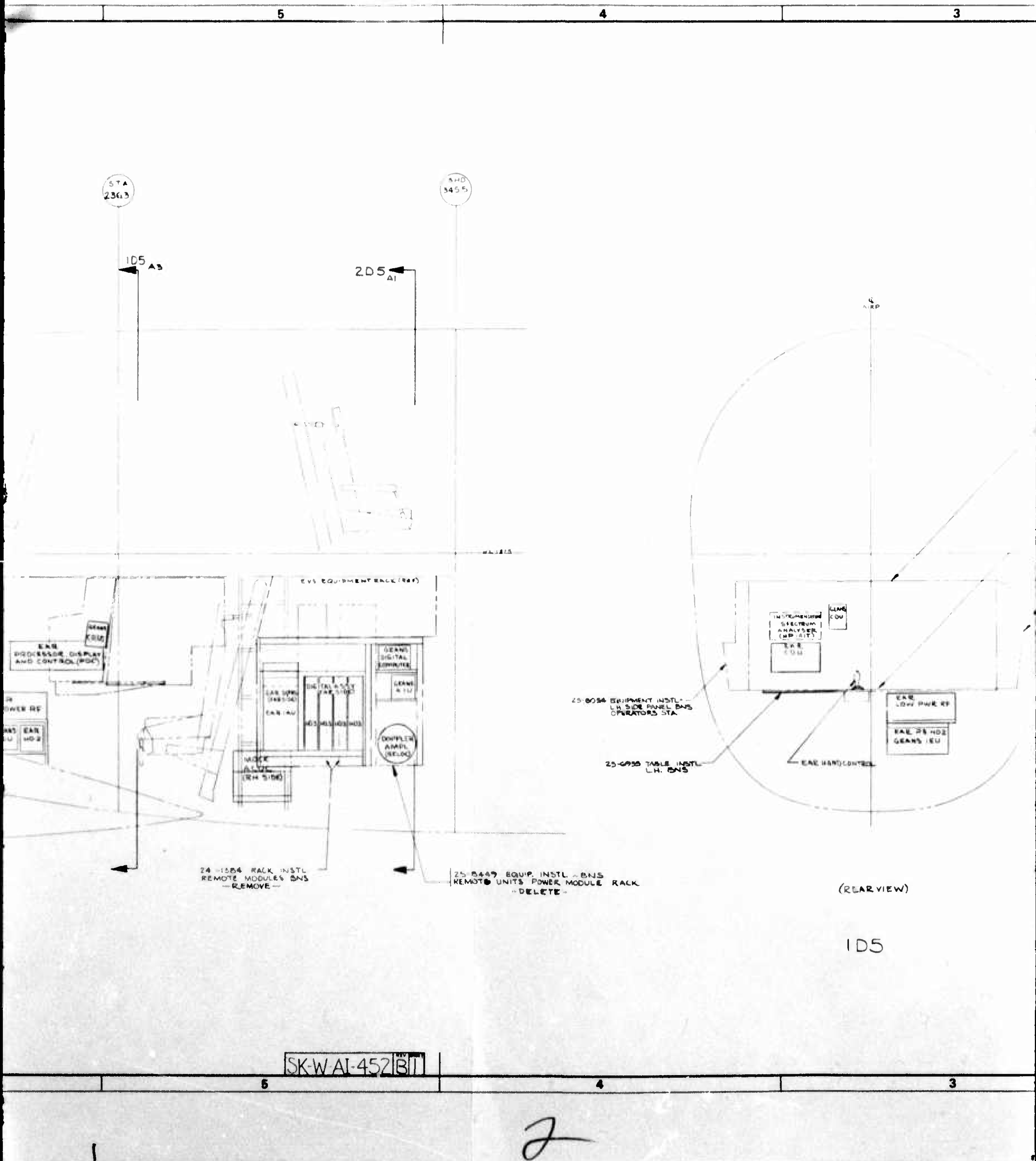
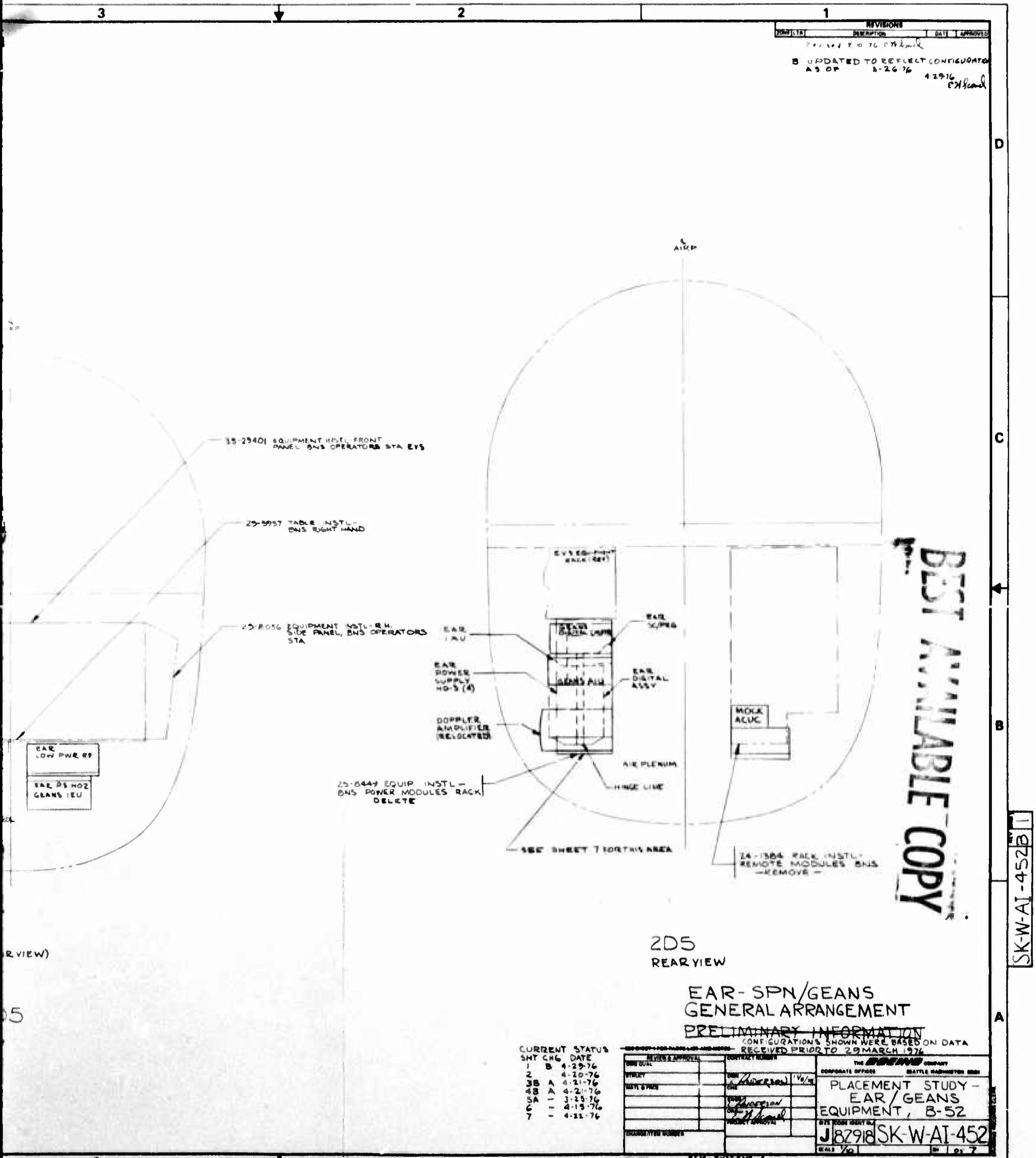
A
RADOME OUTSIDE CONTOUR AT WL 147.0 (REF)
STA 156 (REF)
STA 150.7 (REF)
AFT EDGE OF RADOME (REF)
BHD 96.0
BHD 115.6
OUTSIDE CONTOUR AT STA 96 (REF)
OUTSIDE CONTOUR AT STA 170.48 (REF)
1B7
2B7
1B7
EAR ANTENNA
EAR PROCESSOR DISP. AND CONTROL
EAR LOW POWER RF
STA 104.6 (REF)
STA 150.7 (REF)
RADOME OUTSIDE CONTOUR AT WL 135.5 (REF)
AFT EDGE OF RADOME (REF)
SEE SHEETS 3B & 4B FOR EQUIPMENT ARRANGEMENT, CONT D, IN THIS AREA

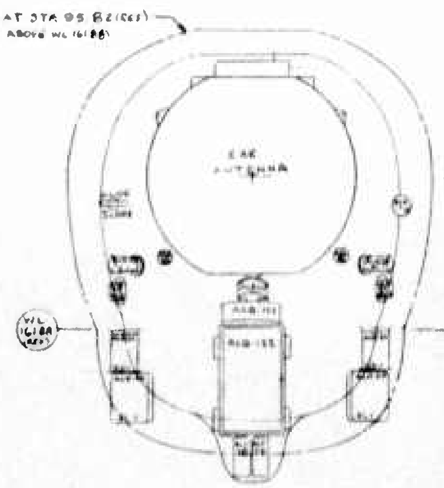


5
4
3
STA 2363
BHD 3455
1D5 A3
2D5 A1
EVS EQUIPMENT RACK (REF)
EAR PROCESSOR, DISPLAY AND CONTROL (PDC)
GEANS DIGITAL COMPUTER
DOPPLER AMPL (RELOC)
24-1584 RACK INSTL REMOTE MODULES BNS -REMOVE-
25-8449 EQUIP. INSTL.-BNS REMOTE UNITS POWER MODULE RACK -DELETE-
INSTRUMENTATION SPECTRUM ANALYSER (HP 141T)
EAR CDU
25-8054 EQUIPMENT INSTL.-L.H. SIDE PANEL BNS OPERATORS STA
25-6958 TABLE INSTL. L.H. BNS
EAR HANDCONTROL
EAR LOW PWR RF
EAR PS HO2 GEANS IEU
(REAR VIEW)
1D5
SK-W-AI-452 BT
5
4
3
2


REVISIONS
B UPDATED TO REFLECT CONFIGURATION AS OF 3-26-76
4-29-76
AIRP
35-29401 EQUIPMENT INSTL FRONT PANEL BNS OPERATORS STA EVS
25-5957 TABLE INSTL - BNS RIGHT HAND
25-8036 EQUIPMENT INSTL - R.H. SIDE PANEL, BNS OPERATORS STA
EAR IAU
EAR POWER SUPPLY HD-5 (4)
DOPPLER AMPLIFIER (RELOCATED)
EAR SC/PEG
EAR DIGITAL ASSY
GEANS AIU
MOLK ACUC
AIR PLENUM
HINGE LINE
25-8449 EQUIP INSTL - BNS POWER MODULES RACK DELETE
SEE SHEET 7 FOR THIS AREA
24-1384 RACK INSTL - REMOTE MODULES BNS - REMOVE -
EAR LOW PWR RF
EAR PS HO2 GEANS IEU
R VIEW)
2D5
REAR VIEW
EAR-SPN/GEANS
GENERAL ARRANGEMENT
~~PRELIMINARY INFORMATION~~
CONFIGURATIONS SHOWN WERE BASED ON DATA RECEIVED PRIOR TO 29 MARCH 1976
CURRENT STATUS
SHT CHG DATE
1 B 4-29-76
2 4-20-76
3B A 4-21-76
4B A 4-21-76
5A - 3-25-76
6 - 4-15-76
7 - 4-22-76
THE BOEING COMPANY
PLACEMENT STUDY - EAR / GEANS EQUIPMENT, B-52
J 82918 SK-W-AI-452
SCALE 1/10
SH 1 OF 7
3-71530 ECM SYSTEMS & AVIONICS INSTL
SK-W-AI-452 B

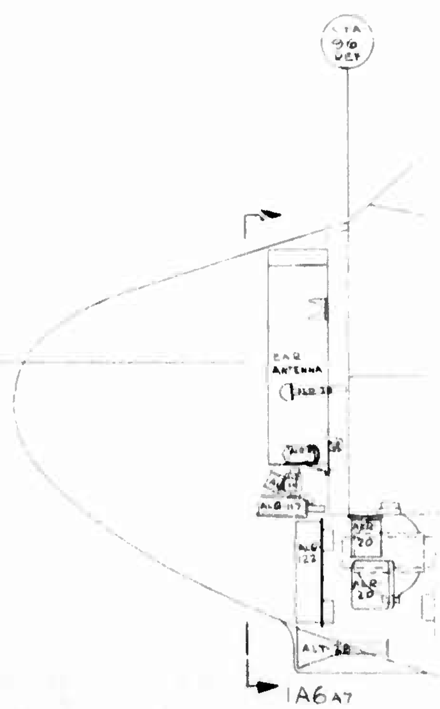
3

SK-W-A1-4-25



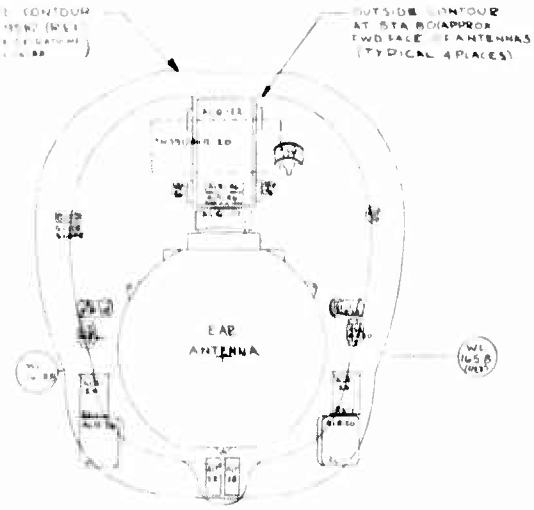
CONFIGURATION 4A

SECT 1A6

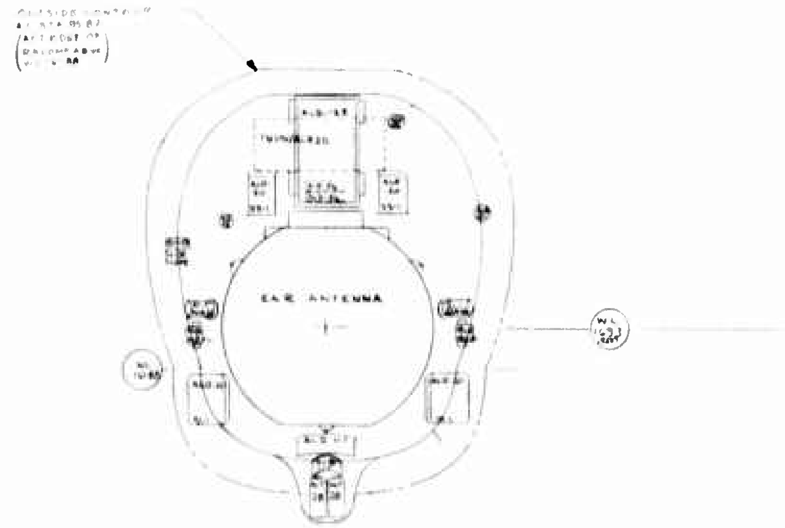


CONFIG

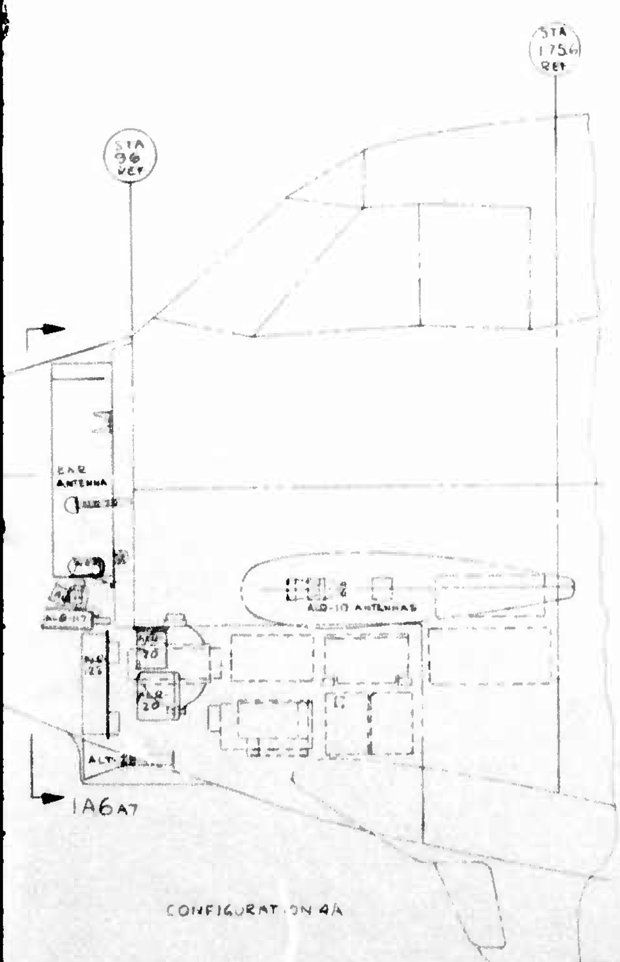
5 4 3



CONFIGURATION 1A
SECT 2B7-1
(SEE SHEET 5A)

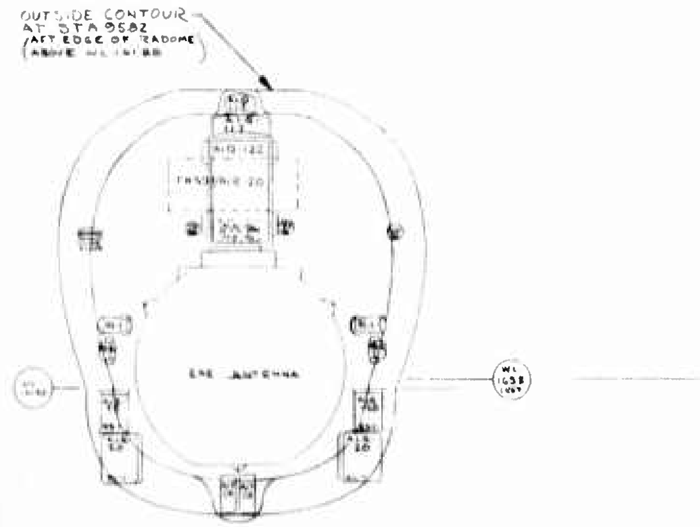


CONFIGURATION 3A
SECT 1C2



1A6A7

CONFIGURATION 4A

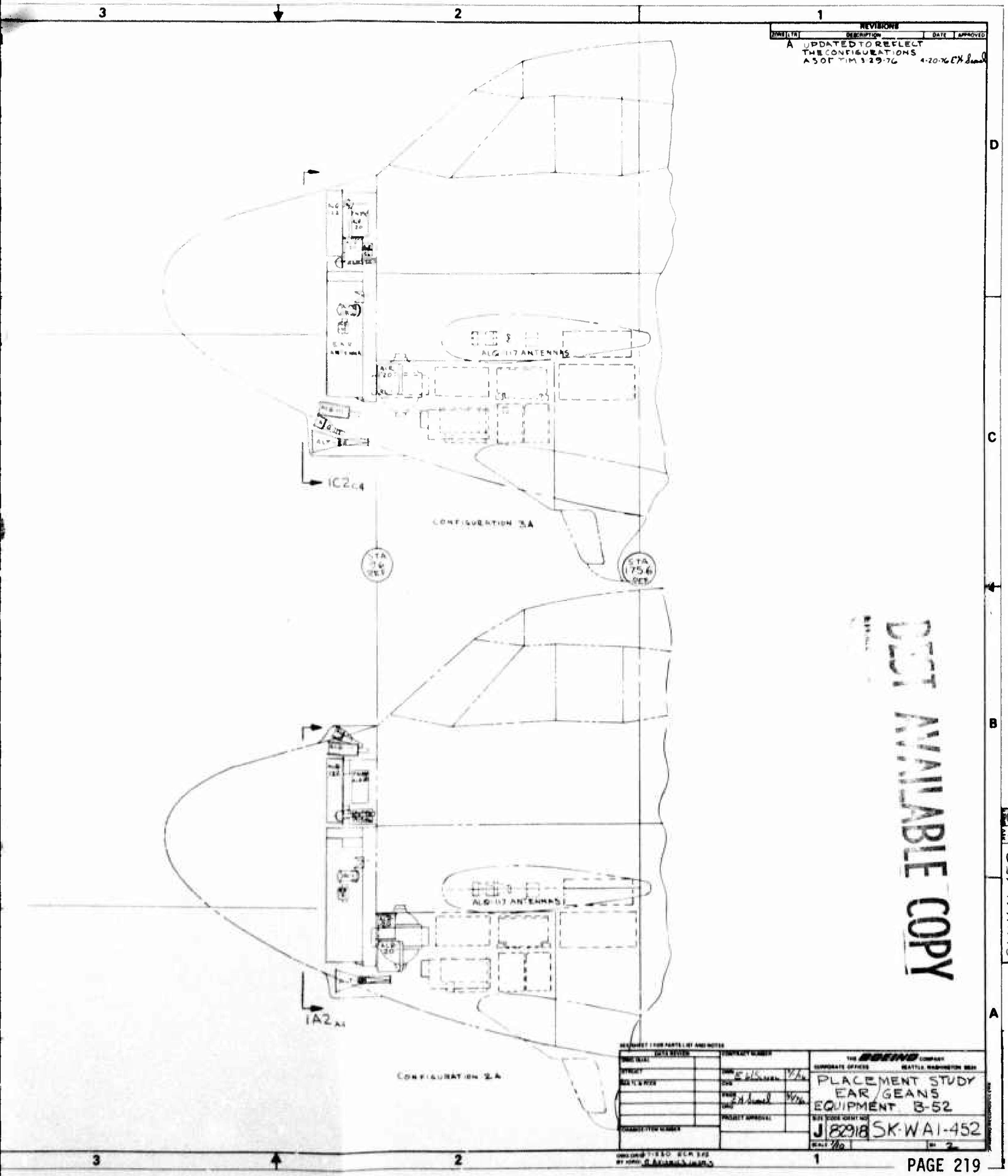


CONFIGURATION 2A
SECT 1A2

SK-W-AI-452 A 2

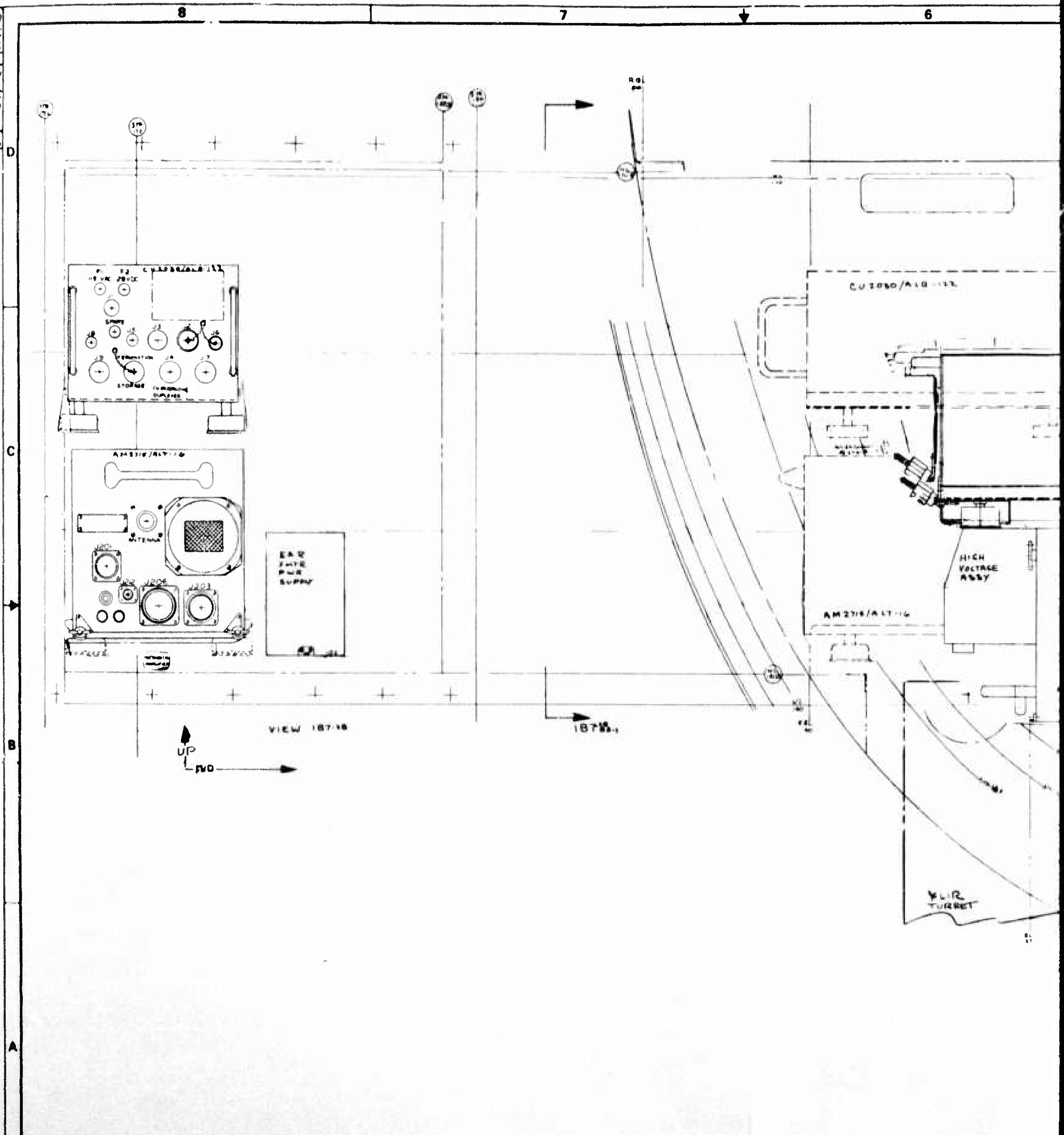
5 4 3

2

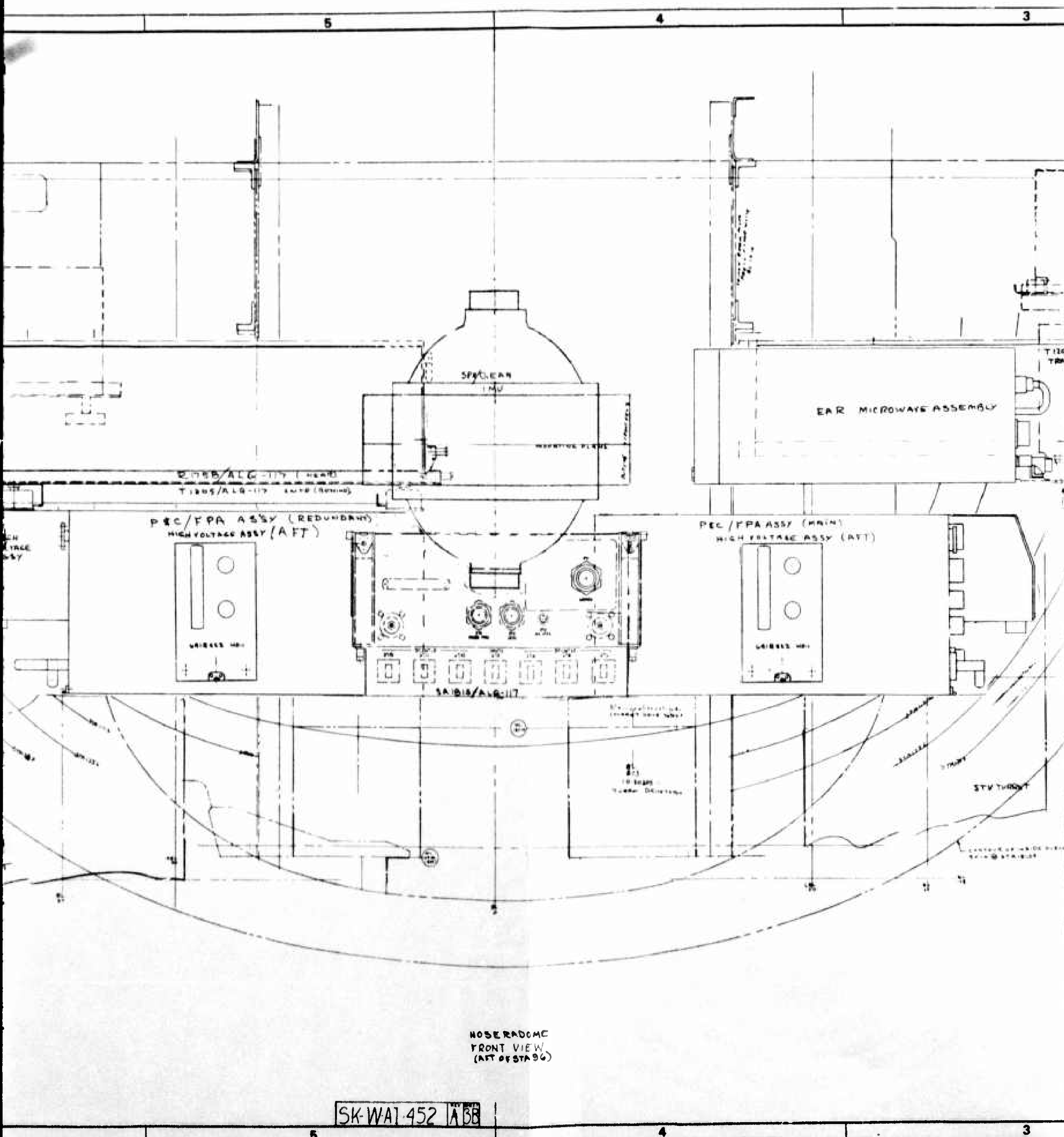

REVISIONS
A UPDATED TO REFLECT THE CONFIGURATIONS AS OF TIM 3-29-76 4-20-76
ALQ-117 ANTENNAS
CONFIGURATION 3A
STA 176.6 DET
IC2
ALQ-117 ANTENNAS
CONFIGURATION 2A
IA2
THE BOEING COMPANY
PLACEMENT STUDY EAR/GEANS EQUIPMENT B-52
J 82918 SK-WA1-452
SK-WA1-452 A2

3

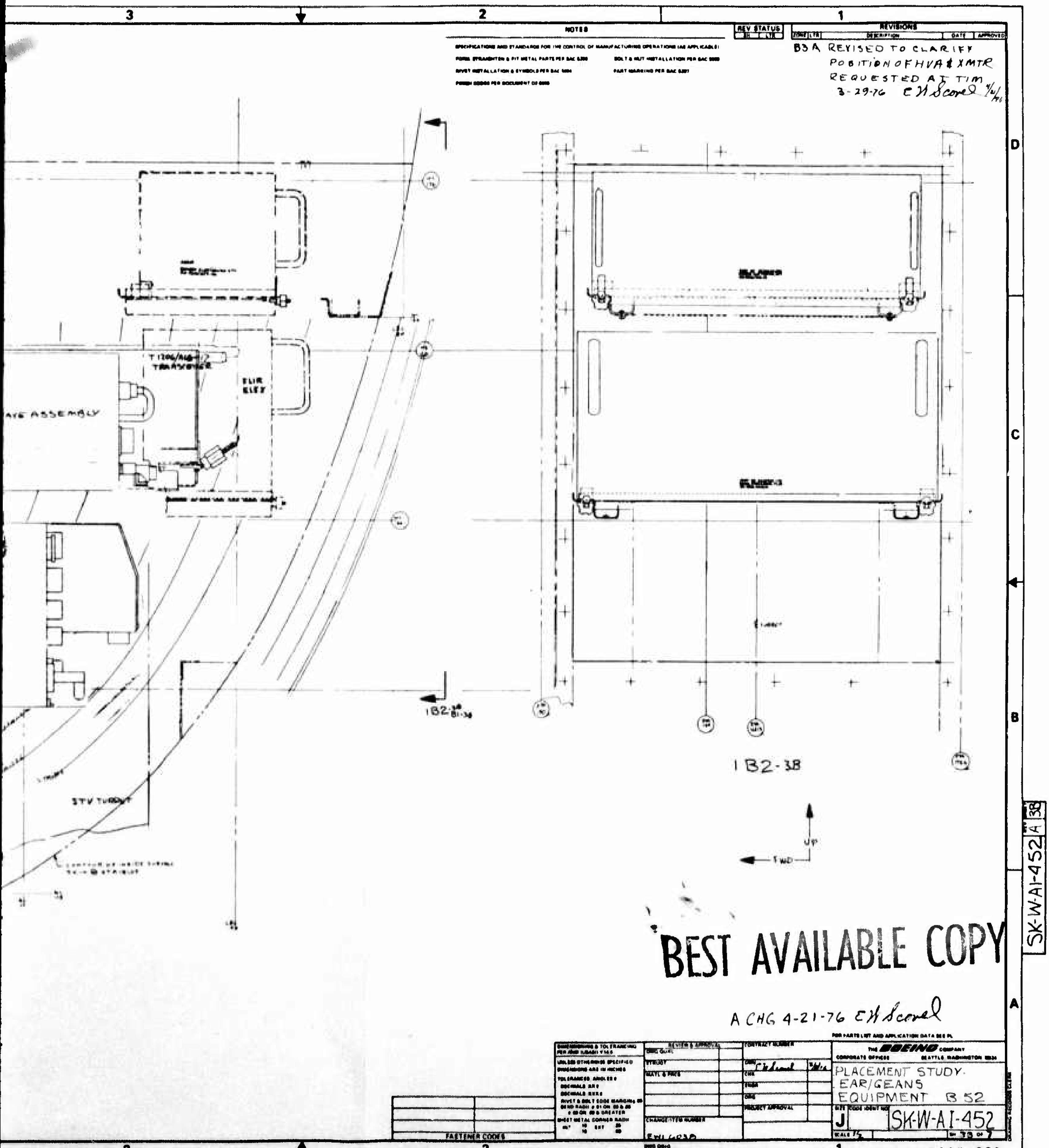
SK-W-A1-452



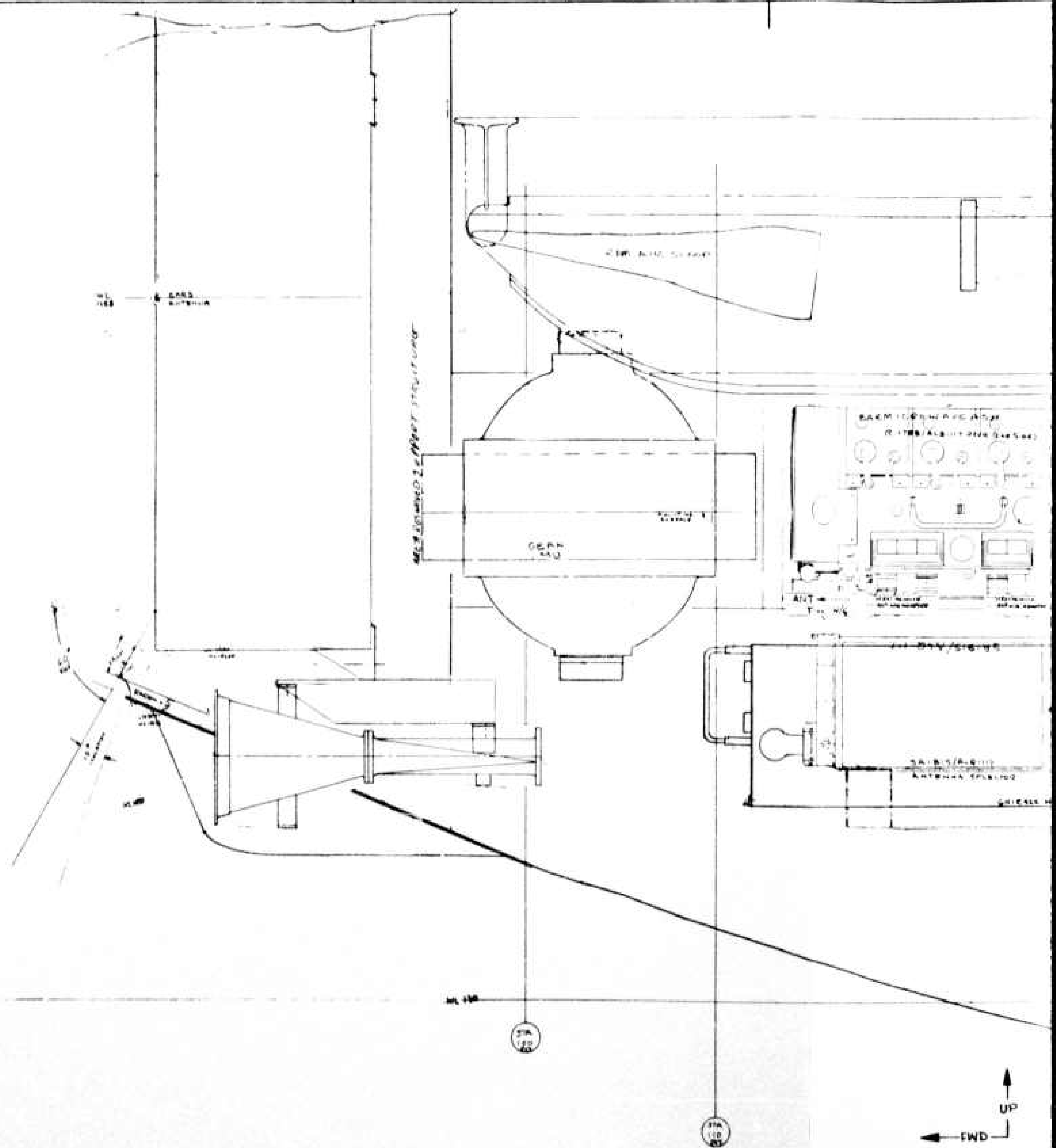
8
7
6
D
C
B
A
ANTENNA
J201
J203
EAR
XMTR
PWR
SUPPLY
HIGH
VOLTAGE
ASSY
UP
FWD
VIEW
FLIR
TURRET



5
4
3
SPHERICAL IMU
EAR MICROWAVE ASSEMBLY
T1205/ALQ-117
P&C/FPA ASS'Y (REDUNDANT)
HIGH VOLTAGE ASSY (AFT)
PEC/FPA ASSY (MAIN)
HIGH VOLTAGE ASSY (AFT)
SA1818/ALQ-117
STV TURRET
NOSE RADOME
FRONT VIEW
(AFT OF STA 96)
SK-WA1-452
5
4
3


3
2
1
NOTES
REV STATUS
REVISIONS
DESCRIPTION
DATE
APPROVED
B3 A REVISED TO CLARIFY POSITION OF HVA & XMTR REQUESTED AT TIM 3-29-76 EH Scorel 4/1/76
T 1206/ALQ-117 TRANSCEIVER
FLIR ELEX
AYE ASSEMBLY
STV TURRET
1B2-38
1 B2-38
UP
FWD
D
C
B
A
A CHG 4-21-76 EH Scorel
FOR PARTS LIST AND APPLICATION DATA SEE PL
THE BOEING COMPANY
SEATTLE, WASHINGTON
REVIEW & APPROVAL
CONTRACT NUMBER
PROJECT APPROVAL
FASTENER CODES
PLACEMENT STUDY.
EAR/GEANS
EQUIPMENT B 52
J
SK-W-AI-452
SK-W-AI-452 A 38

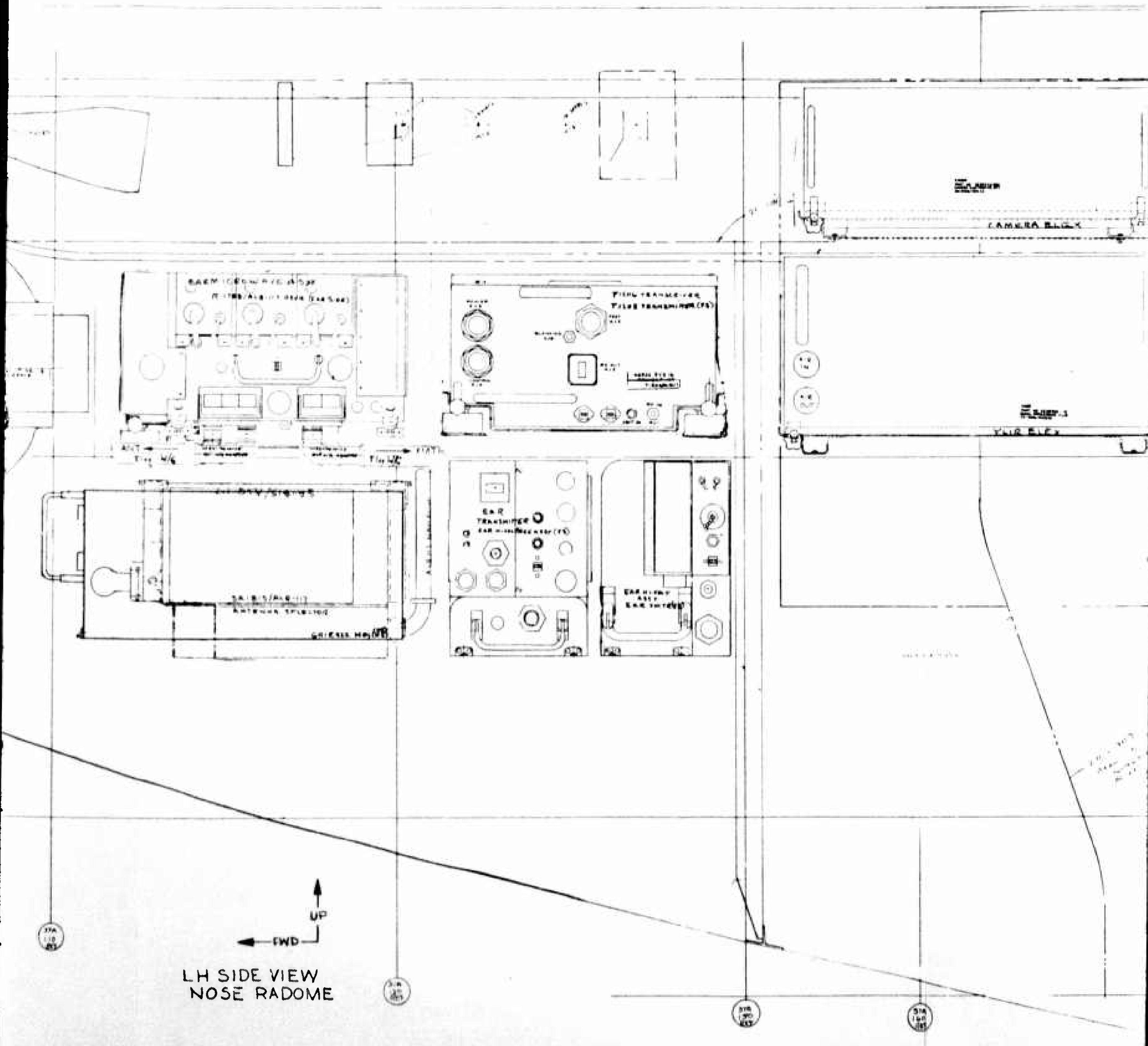
3



LH SIDE VIEW
NOSE RADOME

SK-W-A1-452 A 48

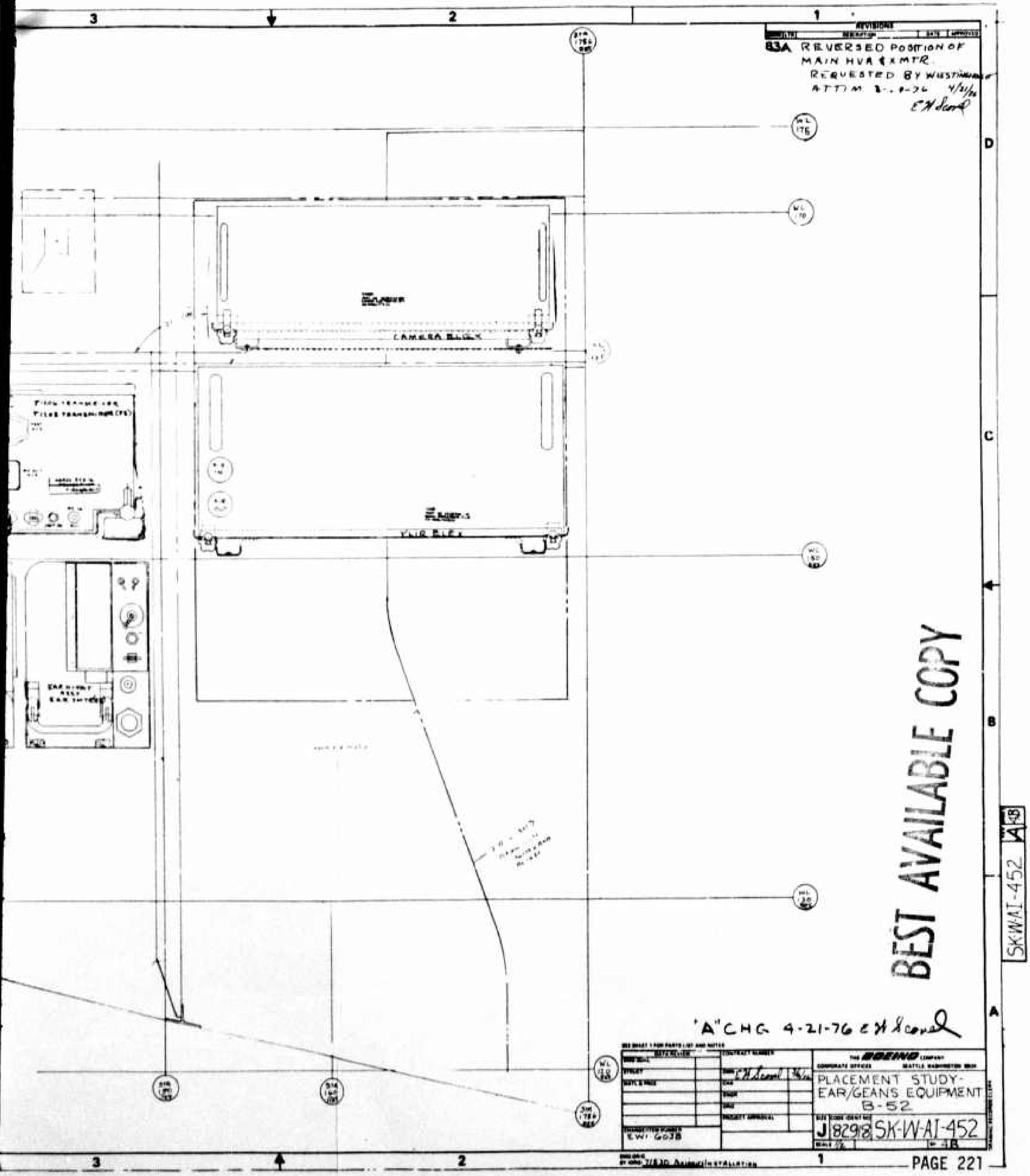
4	3	2



LH SIDE VIEW
NOSE RADOME

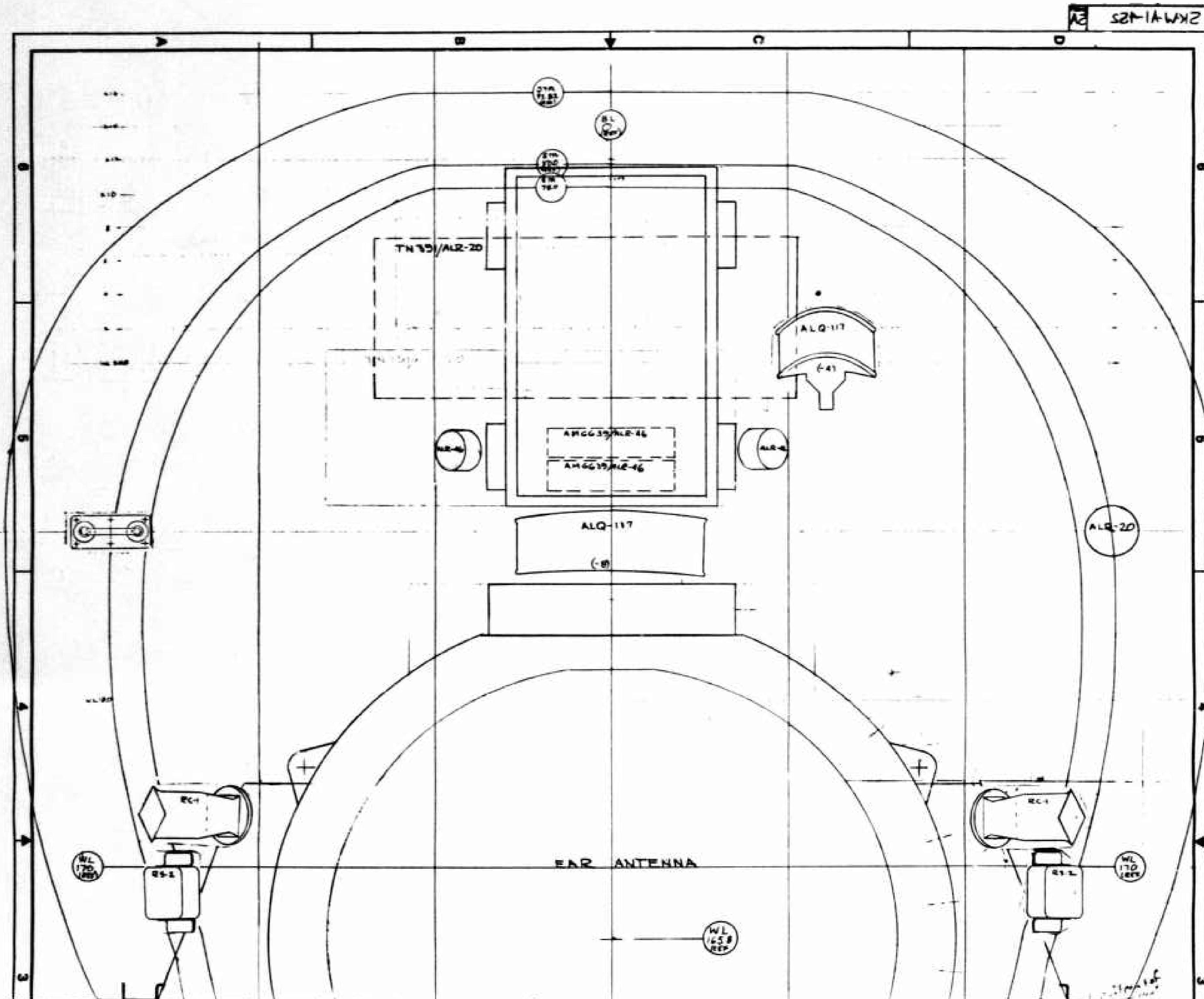
452 A 4B

4	3	2

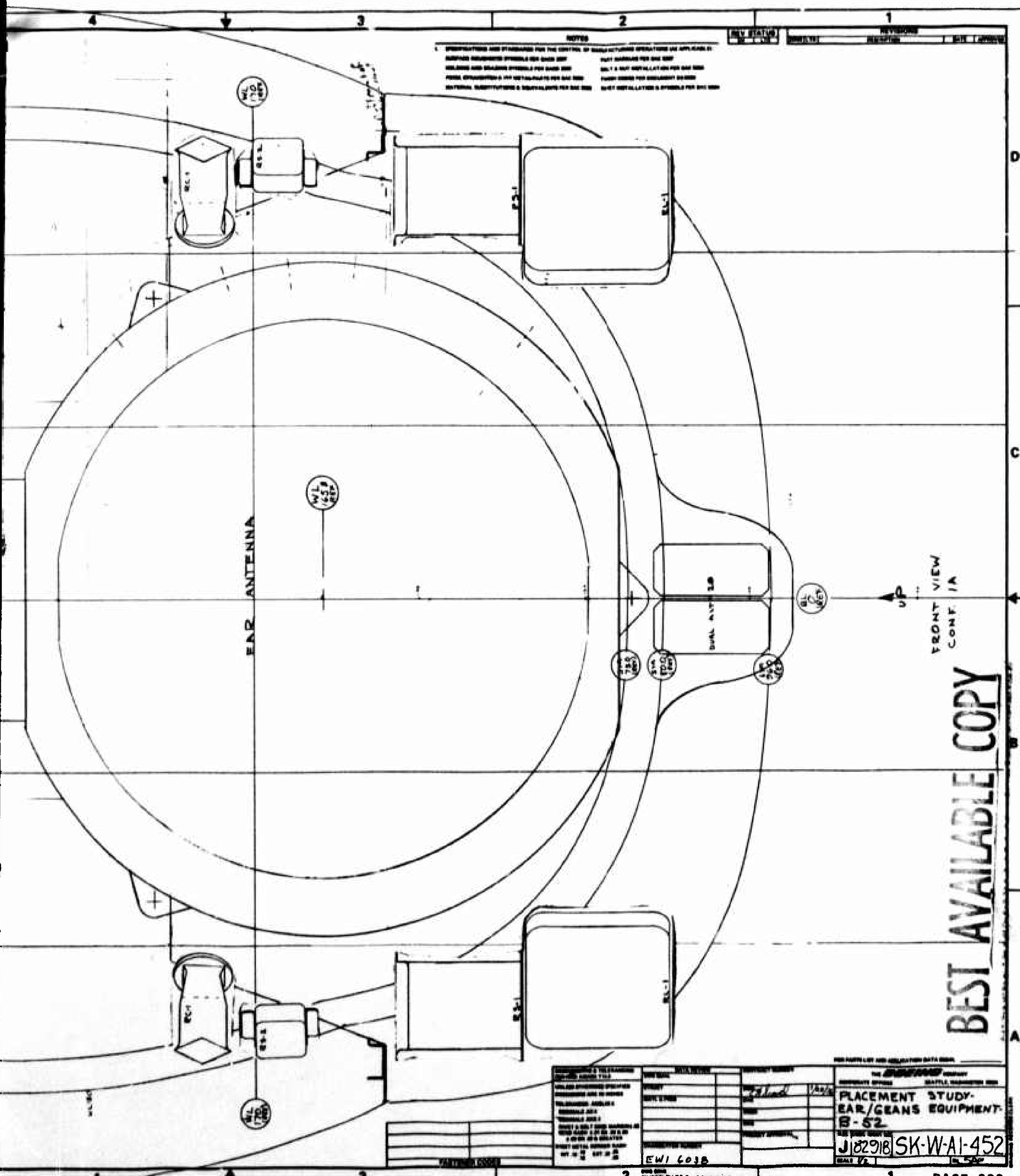


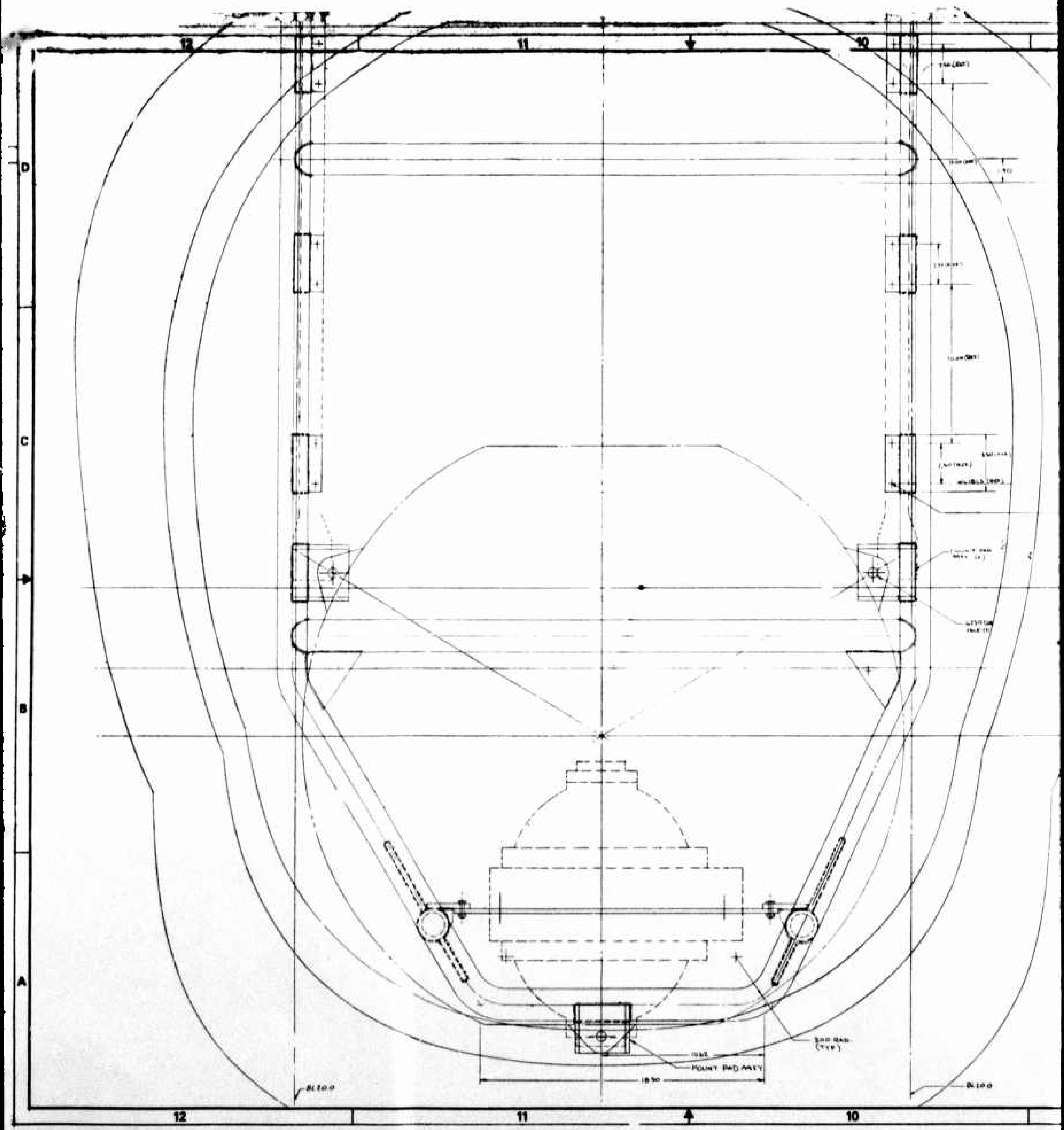
REVISIONS
REVERSED POSITION OF MAIN HVR & XMTR
CAMERA BLOCK
FLIR BLOCK
'A' CHG 4-21-76
BEST AVAILABLE COPY
SK-W-AI-452
THE BOEING COMPANY
PLACEMENT STUDY-
EAR/GEANS EQUIPMENT
B-52
SK-W-AI-452

3

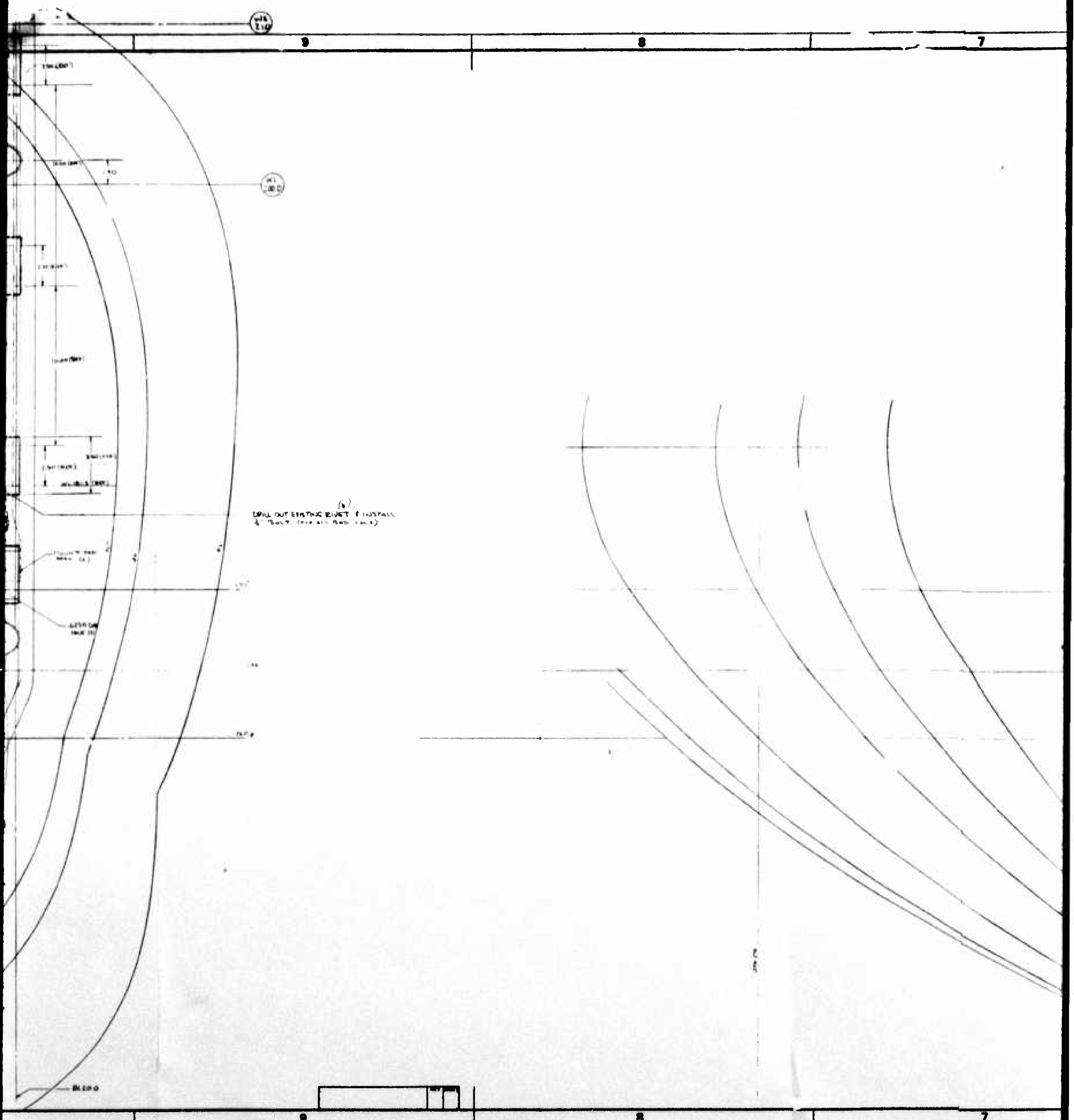


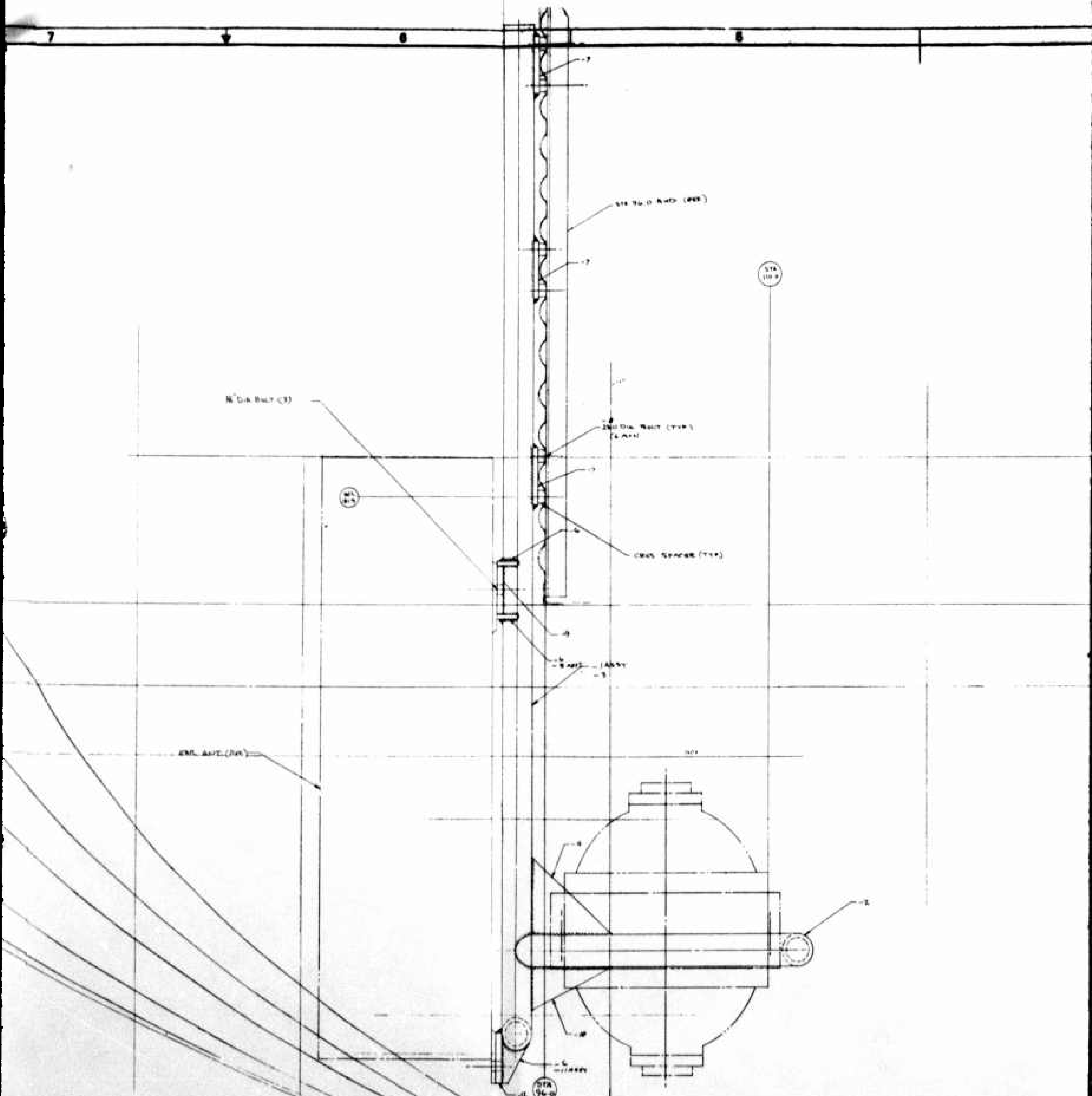
TN 391/ALR-20
ALQ-117
(-41)
AMG639/ALR-46
AMG639/ALR-46
ALR-46
ALQ-117
(-8)
ALR-20
EAR ANTENNA
RC-1
RS-2
WL 170 REF
WL 165.8 REF
A
B
C
D
6
5
4
3


NOTES
REV STATUS
REVISIONS
EAR ANTENNA
DUAL ALT
UP
FRONT VIEW
CONF. IA
BEST AVAILABLE COPY
SK-W-A1-452
5A
FASTENER CODES
EWI 6038
PLACEMENT STUDY-
EAR/GEANS EQUIPMENT-
B-52
J 82918 SK-W-A1-452



12
11
10
D
C
B
A
MOUNT PAD ASSY
1830
BL20.0
BL10.0


9
8
7
DRILL OUT EXISTING RIVET & INSTALL
BOLT (TYP ALL BAD LOCS)
BLEND
9
8
7



3

4 | 3 | 2

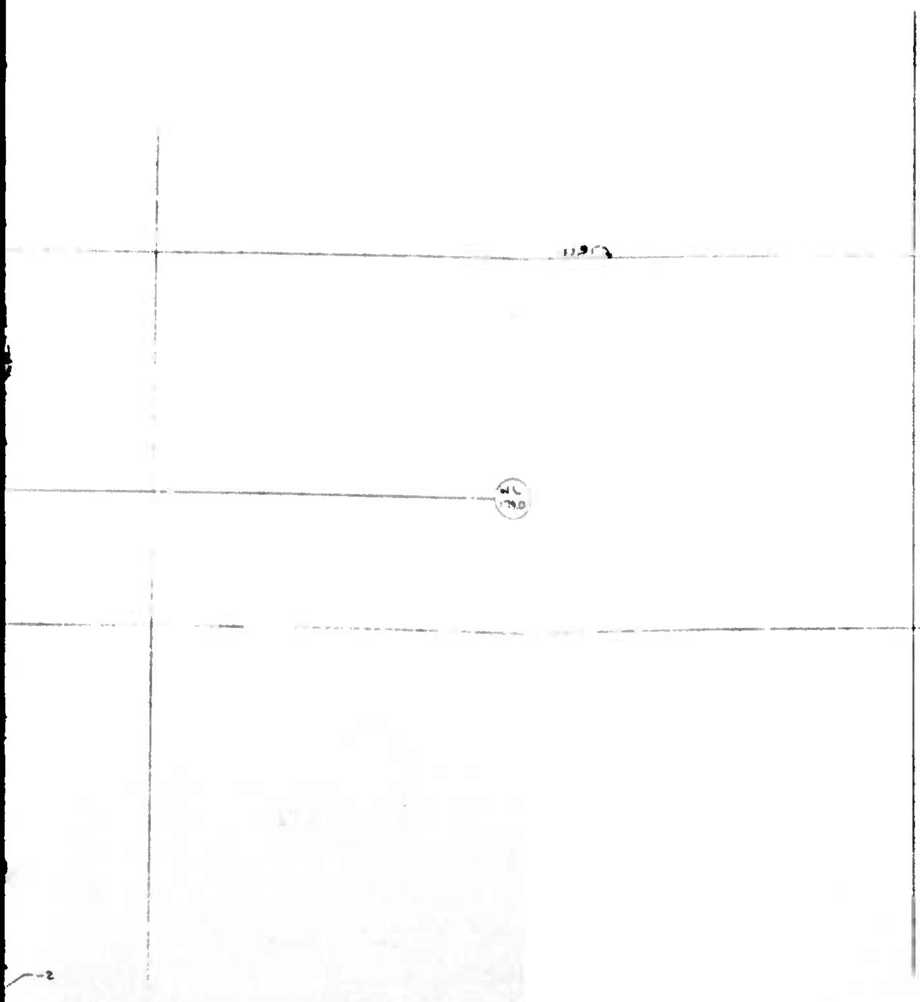
NOTES

SPECIFICATIONS AND STANDARDS FOR THE CONTROL OF MANUFACTURING

FORM, STRAIGHTEN & FIT METAL PARTS PER GAC 0000

RIVET INSTALLATION & SYMBOLS PER GAC 0000

FINISH CODES PER DOCUMENT 00-0000



QTY	QTY	QTY	QTY	QTY	QTY	FIND NUMBER	PART OR IDENTIFYING NUMBER	NOMENCLATURE
							-12	PAD
							-11	MOUNT
							-10	GUS
							-9	PA
							-8	SPA
							-7	PA
							-6	GUS
							-5	MOU
							-4	GUS
							-3	FRA
							-2	FRA
							-1	FRA

APPLICATION

FASTENER CODE

4 | 3 | 2

NOTES

SPECIFICATIONS OR STANDARDS FOR THE CONTROL OF MANUFACTURING OPERATIONS (AS APPLICABLE)

FORM, STRAIGHTEN & FIT METAL PARTS PER BAC [illegible] — BOLT & NUT INSTALLATION PER BAC [illegible]

RIVET INSTALLATION & SYMBOLS PER BAC [illegible] — PART MARKING PER BAC [illegible]

FINISH CODES PER [illegible]

REV STATUS | REVISIONS

BEST AVAILABLE COPY

PART OR IDENTIFYING NUMBER	NOMENCLATURE OR DESCRIPTION	MATERIAL AND SPECIFICATION
-12	PAD	CRES PLATE
-11	MOUNT PAD ASSY	—
-10	GUSSET	CRES PLATE
-9	PAD	CRES PLATE
-8	SPACER	CRES BAR
-7	PAD	CRES PLATE
-6	GUSSET	CRES PLATE
-5	MOUNT PAD ASSY	—
-4	GUSSET	CRES PLATE
-3	FRAME	CRES TUBING - 2.00 OD X .12 WALL PER ASTM [illegible]
-2	FRAME	CRES TUBING - 2.00 OD X .12 WALL PER ASTM [illegible]
-1	FRAME ASSY	—

PARTS LIST

EWI-GO3B

THE BOEING COMPANY

G. ADRIAN

PLACEMENT STUDY - EAR/GEANS EQUIPMENT B-52

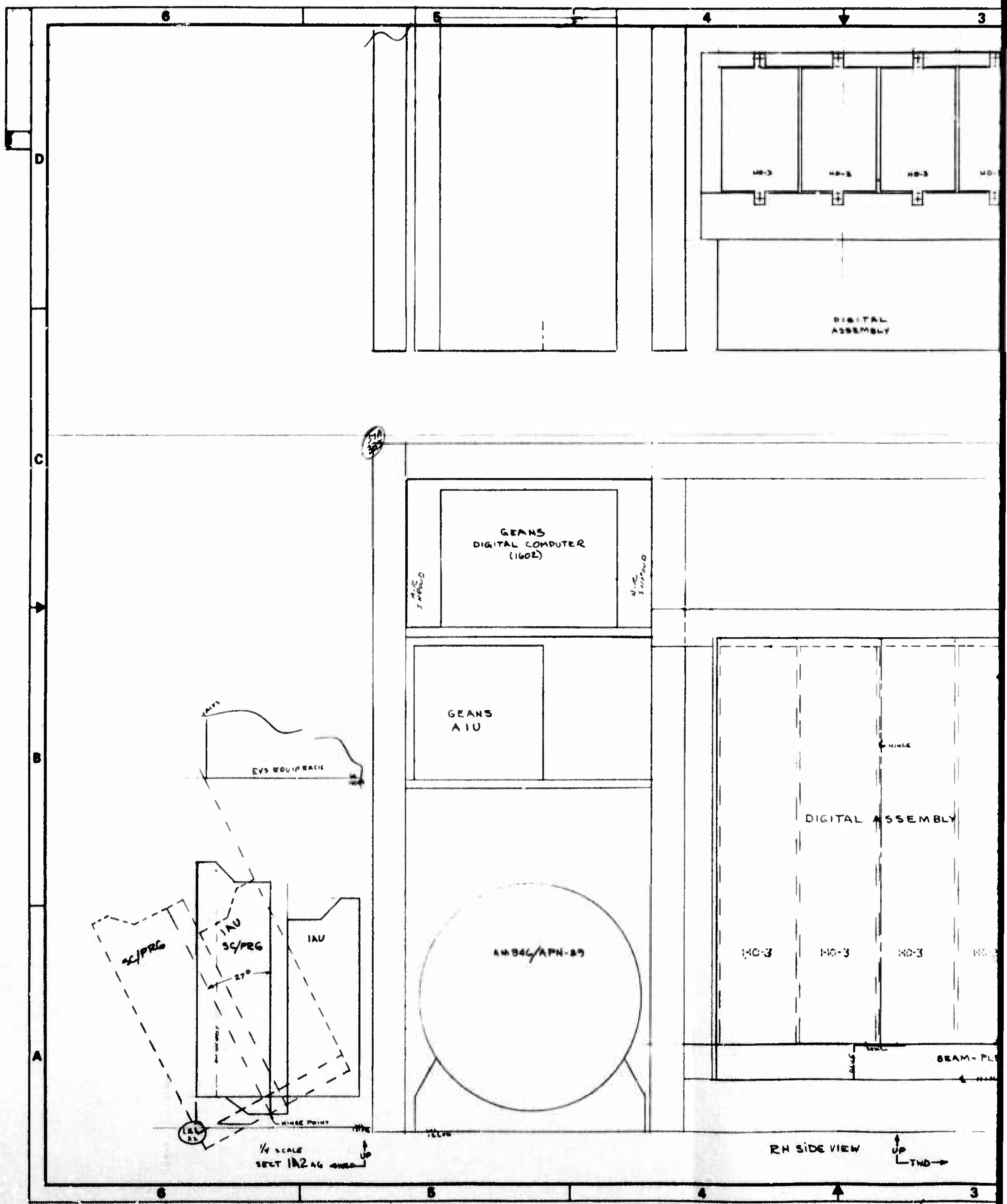
J 82918 SK-W-AI-452

SCALE 1/2 — SH 6

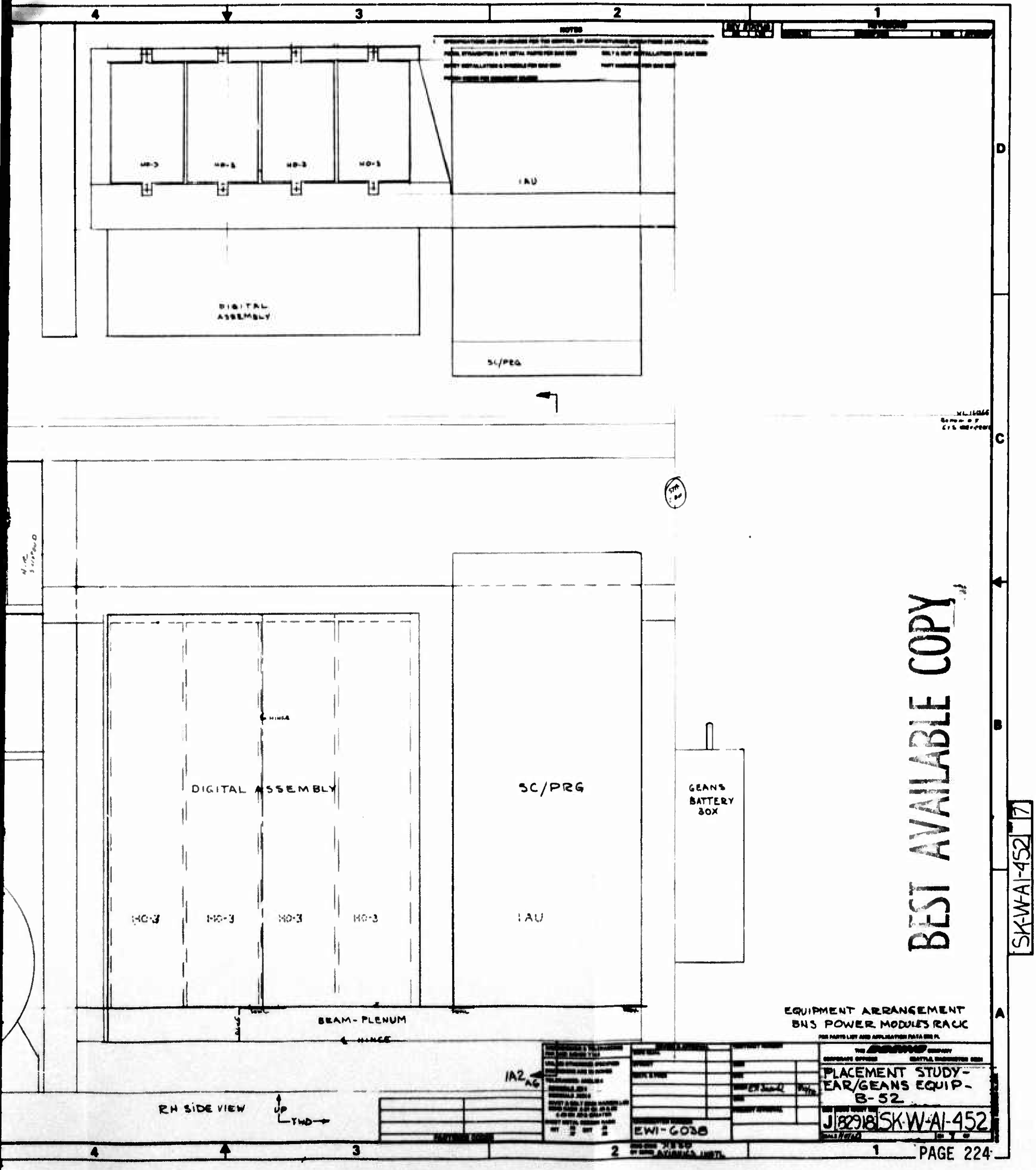
71530

SK-W-AI-452 6

5



6
5
4
3
D
C
B
A
DIGITAL ASSEMBLY
GEANS DIGITAL COMPUTER (1602)
GEANS AIU
DIGITAL ASSEMBLY
HO-3
IAU
SC/PRG
IAU SC/PRG
27°
HINGE POINT
1/4 SCALE
RH SIDE VIEW
UP
FWD



HO-3
HO-3
HO-3
HO-3
IAU
DIGITAL ASSEMBLY
SC/PEG
DIGITAL ASSEMBLY
SC/PRG
GEANS BATTERY BOX
HO-3
HO-3
HO-3
HO-3
IAU
HINGE
BEAM - PLENUM
HINGE
RH SIDE VIEW
UP
FWD
IA2
EQUIPMENT ARRANGEMENT
BNS POWER MODULES RACK
PLACEMENT STUDY-
EAR/GEANS EQUIP-
B-52
J 82918 SK-W-AI-452
EWI-G038
BEST AVAILABLE COPY
SK-W-AI-452 17