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PART II - AIRCRAFT PRELIMINARY DESIGN REPORT V/STOL JET OPERATIONS RESEARCH AIRPLANE DESIGN STUDY - SECTIONS VI TO XI

by

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ABSTRACT

This study is presented in two parts:

Part I - Aircraft Preliminary Design Report V/STOL Jet Operations Research Airplane Design Study - Sections I to V (NASA CR-66421)

Part II - Aircraft Preliminary Design Report V/STOL Jet Operations Research Airplane Design Study - Sections VI to XI (NASA CR-66422)

These reports provide a preliminary design description of a new and modified V/STOL jet operations research airplane. This preliminary design was conducted by Fairchild-Hiller Corporation, Republic Aviation Division during Part III of Contract NAS 1-6778, "Jet V/STOL Operations Research Airplane Design Study."

TABLE OF CONTENTS

Section		Page
I	INTRODUCTION	1-1
II	AIRCRAFT DESCRIPTION AND SPECIFICATION	2-1
	A. Description	2-1
	1. Background	2-1
	2. Configuration Approach	2-1
	3. Configuration Comparison	2-2
	B. Specifications	2-5
III	PERFORMANCE DATA	3-1
	A. Introduction	3-1
	B. Tabulated Data	3-1
	1. Aircraft Dimensional Data	3-1
	2. Weights	3-4
	C. Aerodynamic Data	3-6
	1. Drag	3-6
	2. Lift	3-16
	3. References	3-43
	D. Power Required	3-44
	1. Method	3-44
	2. Graphical Presentation	3-44
	E. Power Available	3-44
	F. Performance Data: Conventional Flight	3-62
	1. Fuel Flow Data	3-62
	2. Take-Off Calculation	3-62
	3. Rate of Climb	3-67
	4. Speeds	3-86
	5. Ceilings	3-92
	6. Landings	3-96
	7. Cruise Control	3-101
	8. Dive Recovery Data	3-106
	9. Missions	3-122

TABLE OF CONTENTS (Cont'd)

Section			Page
	G.	Performance Data: Hover and Transition Flight	3-125
		1. Hover	3-125
		2. Glide Path Control at Constant Speed	3-134
		3. Acceleration: Hover to Aerodynamic Flight	3-134
		4. Deceleration: Aerodynamic Flight to Hover	3-137
IV	PROPULSION AND THERMODYNAMIC DATA		4-1
	A.	Installed Engine Performance	4-1
		1. Ambient Temperature	4-1
		2. Induction System Total Pressure Recovery	4-2
		3. Compressor Bleed Air Extraction	4-4
		4. Horsepower Extraction	4-4
		5. Diverter Valve Losses	4-4
		6. Engine Cooling	4-5
		7. Exhaust Tailpipe	4-5
		8. Vectoring Nozzle	4-5
		9. Sensitivity Analysis	4-6
		10. References	4-6
	B.	Reaction Control System	4-44
		1. System Design	4-44
		2. System Performance	4-46
		3. References	4-47
	C.	Vectoring System	4-53
		1. General	4-53
		2. Types of Thrust Vectoring Systems Considered	4-53
		3. Final System Selection	4-53
	D.	Environmental Control System	4-56
		1. Function	4-56
		2. Airconditioner Selection and Operation	4-56
		3. ECS Modifications for Crew Safety	4-57
		4. Competitive Air Conditioners Studied	4-58

TABLE OF CONTENTS (Cont'd)

Section				Page
		5.	Aircraft Heat Load Analysis	4-59
		6.	"Off-the-Shelf" ACM Survey	4-60
		7.	ACM Auxiliary Provisions	4-62
		8.	ECS Ground Operation	4-62
		9.	Additions to Existing ACM AirConditioner	4-63
	E.	Engine Exhaust Effects		4-68
		1.	Engine Inlet Hot Gas Injestion	4-68
		2.	Ground Flow Fields	4-69
		3.	Computed Estimate of Ground Flow Field	4-69
		4.	Tire Heating	4-71
		5.	Skin External Heating	4-72
		6.	Internal Structural Heating	4-74
		7.	References	4-76
	F.	Lift Engine Air Start		4-84
	G.	Fuel System		4-91
	H.	Thrust Reverser Installation		4-93
	I.	Fire Detection and Extinguishing System		4-95
		1.	Fire Detection System	4-95
		2.	Fire Extinguishing System	4-95
V	WEIGHT AND BALANCE			5-1
			CONCEPT C	
	A.	Weight, Balance, and Inertia Summary		5-1
	B.	Group Weight Statement		5-3
	C.	Detail Work Statement		5-9
	D.	Center-of-Gravity Travel Envelope		5-45
	E.	Weight Growth		5-47
		1.	Effect of Weight Growth on Hover Endurance	5-47
		2.	Effect of Weight Growth at Constant Hover Endurance	5-47

TABLE OF CONTENTS (Cont'd)

Section		Page
	CONCEPT J	
	A. Weight, Balance, and Moments of Inertia	5-50
	1. Summary Group Weight Statement	5-50
	2. Group Weight Statement	5-50
	3. Center-of-Gravity Travel Envelope	5-50
	4. Moment of Inertia	5-50
	B. Growth Analysis	5-54
	C. Weight Substantiation	5-54
	1. T-39 Breakdown	5-54
	2. VTO Controls Weight Substantiation	5-54
VI	STABILITY AND CONTROL, AND FLYING QUALITIES	6-1
	A. Stability and Control	6-1
	1. Basic Airframe Stability	6-1
	2. Damping	6-25
	3. Trim Characteristics	6-25
	4. Methods of Obtaining Control	6-51
	B. Flying Qualities	6-60
	1. Stability Augmentation System (SAS)	6-60
	2. Variable Stability System (VSS)	6-73
	3. Flight Characteristics Simulation	6-91
	4. Flexibility of Operation	6-114
VII	STRUCTURES	7-1
	CONCEPT C	
	A. Structural Design	7-1
	B. Landing Gear Design	7-2
	C. Materials	7-3
	D. Flight Loads	7-12
	1. General	7-12
	2. Symmetric Conditions	7-12
	3. Asymmetric Conditions	7-13

TABLE OF CONTENTS (Cont'd)

Section			Page
	E.	Ground Loads	7-13
	F.	Acoustic Environment	7-23
	G.	Structural Temperatures	7-29
	H.	Wing Analysis	7-31
	I.	Horizontal Tail Analysis	7-48
	J.	Vertical Tail Analysis	7-55
	K.	Fuselage Analysis	7-63
	L.	Landing Gear Analysis	7-74
	M.	Fatigue	7-81
	N.	Flutter	7-85
		1. General	7-85
		2. Wing Analysis	7-85
		3. Horizontal Tail Analysis	7-85
		4. Vertical Tail Analysis	7-86
		CONCEPT J	
	A.	Flight Loads	7-91
	B.	Ground Loads	7-91
VIII	ELECTRICAL		8-1
	A.	General	8-1
	B.	Communications System	8-1
		1. VHF Transceiver	8-1
		2. Audio Amplifier (Intercom)	8-7
		3. Air Traffic Control (ATC) Transponder	8-7
	C.	Navigation Subsystem	8-9
		1. Compass System	8-9
		2. VHF Omnidirectional Range (VOR) and Localizer and Glide Slope Receiver	8-10
		3. Marker Beacon	8-11
		4. Distance Measuring Equipment (DME)	8-12

TABLE OF CONTENTS (Cont'd)

Section		Page
	D. Antenna Subsystem	8-13
	E. Electrical Power Generation	8-13
	1. A.C. System	8-17
	2. D.C. System	8-17
	F. Aircraft Sensors and Displays	8-21
IX	HYDRAULIC-MECHANICAL SUBSYSTEMS	9-1
	A. Flight Control System	9-1
	1. Primary Flight Controls	9-1
	2. Secondary Flight Controls	9-5
	B. Hydraulic System	9-7
	1. General Description	9-7
	2. Pumps	9-8
	3. Operating Temperature	9-8
	4. Reservoirs	9-9
	5. Landing Gear System	9-9
	6. Wheel Brake System	9-9
	7. Nose Wheel Steering System	9-10
	8. Fuel Boost Pump System	9-10
	9. Lines and Fittings	9-10
	10. Firewall Shut-Off Valves	9-11
	11. Service Couplings	9-11
	12. Concept "J" Hydraulic Systems	9-11
X	COCKPIT ARRANGEMENT	10-1
	A. Crew Station Design	10-1
	1. General	10-1
	2. Geometric Configuration	10-2
	3. Arrangement of Controls and Displays	10-2
	4. External Vision	10-6
	5. Escape Provisions	10-7
	6. Cabin Environment	10-7

TABLE OF CONTENTS (Cont'd)

Section			Page
	B.	Controls and Displays	10-10
		1. Flight Control	10-10
		2. Propulsion, Induction, and Vectoring	10-19
		3. Fuel System	10-25
		4. Landing and Deceleration	10-26
		5. Electrical Power and Lighting	10-28
		6. Communications and Navigation	10-30
		7. Environment	10-33
		8. Warning and Caution Signalling	10-34
		9. Seat and Escape	10-36
	C.	Crew Accommodations	10-37
		1. Air Conditioning System	10-37
		2. Breathing Oxygen System	10-37
		3. Escape System	10-38
		4. Relief Provisions	10-43
		5. Stowage Provisions	10-45
		6. Blind Flight Simulation Provisions	10-45
		7. Noise Protection	10-46
		8. Cabin Lighting	10-46
XI	QUALITY ASSURANCE AND RELIABILITY		11-1
	A.	Quality Assurance Philosophy	11-1
		1. Organization	11-1
		2. Procurement	11-1
	B.	Reliability Investigations	11-2
		1. Reliability Block Diagram and Mathematical Model	11-3
		2. Prediction	11-3
		3. Trade-Off Studies	11-5
		4. Failure Mode and Effect Analysis	11-6
		5. Design Reviews	11-7

SECTION VI

STABILITY AND CONTROL AND FLYING QUALITIES

A. STABILITY AND CONTROL

1. Basic Airframe Stability

MIL-F-8785 (ASG) (Class III, up to 25,000 ft, 450-knot speed or limit Mach number of 0.8) is used as a guide for the handling qualities requirements in the conventional wing-borne flight mode. Class III is a fighter airplane.

a. General Definitions

(1) Airplane Loadings

The airplane weight for the center-of-gravity at 30 percent mean aerodynamic chord is a working value and does not correspond to the normal service loading given in Section V.

(2) Altitudes

(a) Low altitude: sea level

(b) High altitude: 25,000 ft or 80 percent of the service ceiling, whichever is higher

(c) Medium altitude: not investigated because the service ceiling is below 40,000 feet

The high altitude conditions are excluded in consideration of configuration L (landing), PA (power approach), WO (wave-off), and TO (take-off).

(3) Operational Flight Envelope

For the altitudes defined in (2) above, Mach-number vs. normal-acceleration envelopes for the airplane are provided in Section VII. Within these boundaries, there is no objectionable buffet, trim or stability changes, or

other irregularities which might detract from the effectiveness of the airplane in executing its intended mission.

(4) Maximum Speed Envelope

The maximum speed envelope is derived from considerations of dives entered at V_H (high speed). This envelope defines, at each altitude, the maximum speed from which a recovery can be made which will result in level flight at an altitude not less than 2000 feet above sea level without encountering intolerable buffet, loss of control, uncontrollable trim changes, or other dangerous airplane behavior during the entire dive or pull-out.

(5) Deceleration Devices

All Class III airplanes are to be capable of deceleration, dive speed limitation, and constant-speed glide-path control to any degree desired by the pilot within limits which shall be stated. Except in flight path control for the research flight pattern, there are no other stated minimum conventional flight performance requirements.

(6) Configurations

The basic airplane configurations are described in Table IV-1. Items of configuration not defined, such as cockpit enclosures, or cooling air flaps are in their normal settings for the particular configuration.

b. Longitudinal Stability and Control

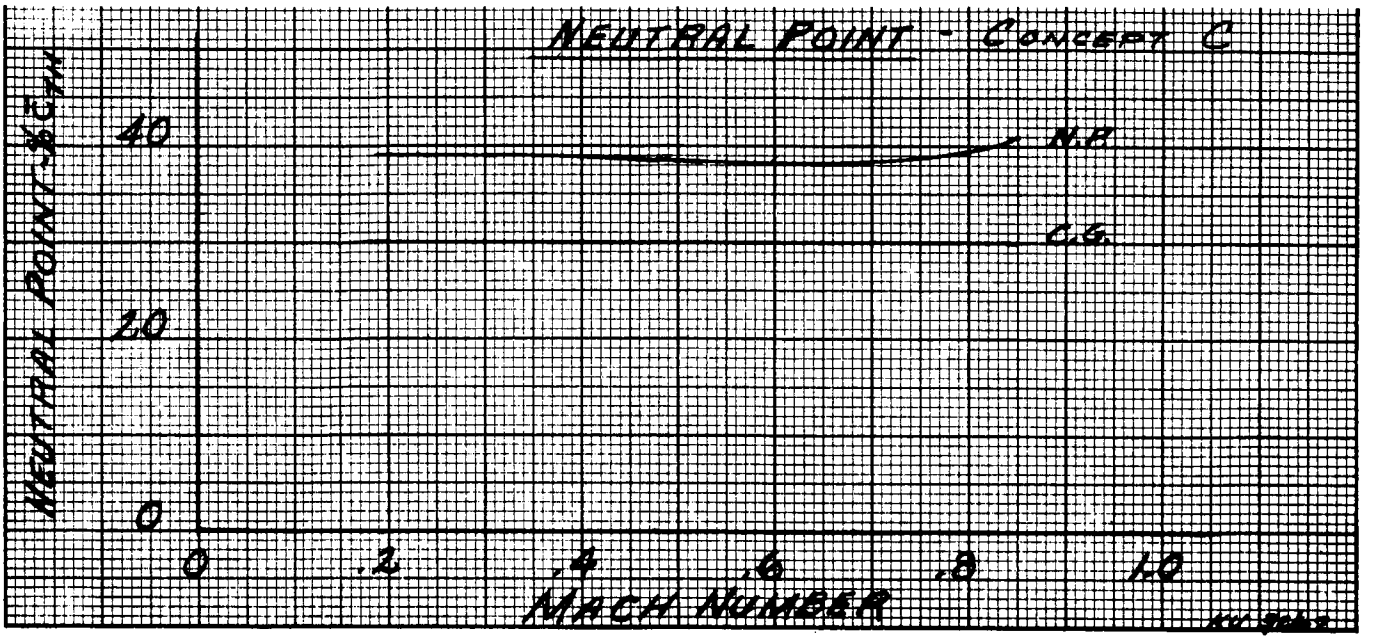
(1) Stabilator Fixed Static Stability

In the flight conditions and throughout the speed ranges in columns 1 and 2 of Table VI-2, the stabilator-fixed neutral points are aft of the center-of-gravity position in the aft critical loading. The neutral point variation with Mach number is presented on Figure VI-1. The center-of-gravity variation is presented on Figure V-1 in Section V. - Weight and Balance.

At the aft critical loading, and in the flight conditions and throughout the speed range listed in columns 1 and 2 of Table VI-2, the stabilator-fixed static longitudinal stability with respect to angle-of-attack at constant speed is positive. This also applies to the wave-off configuration (WO) at 1.15 V_{S_L}.

TABLE VI-1 CONFIGURATIONS

SYMBOL	CR	G	L	P	PA	WO	TO	VL	VTO	VPA	VWO
	CRUISE	GLIDE	LANDING	POWER ON CLEAN	POWER APPROACH	WAVE OFF	TAKE OFF	VERTICAL LANDING	VERTICAL TAKE OFF	VERTICAL POWER APPROACH	VERTICAL WAVE OFF
POWER	LEVEL FLT.	OFF	OFF	NORMAL	LEVEL FLT.	TAKE-OFF	TAKE-OFF	IDLE	TAKE-OFF	NORMAL	TAKE-OFF
POWER AUGMENT	—	—	—	—	—	—	ON	—		—	—
LIFT POWER	—	—	—	—	—	—	—	NORMAL	MAXIMUM	NORMAL	MAXIMUM
SPEED	TRIM	—	—	—	$1.15\ V_{SL}$	—	—				
FLAPS	CRUISE	UP	LANDING	UP	APPROACH	LANDING	TAKE-OFF	LANDING	TAKE-OFF	APPROACH	LANDING
GEAR	UP	UP	DOWN	UP	DOWN	DOWN	DOWN	DOWN	DOWN	DOWN	DOWN
CANOPY	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED
LIFT ENG. DOORS	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED	OPEN	OPEN	OPEN	OPEN



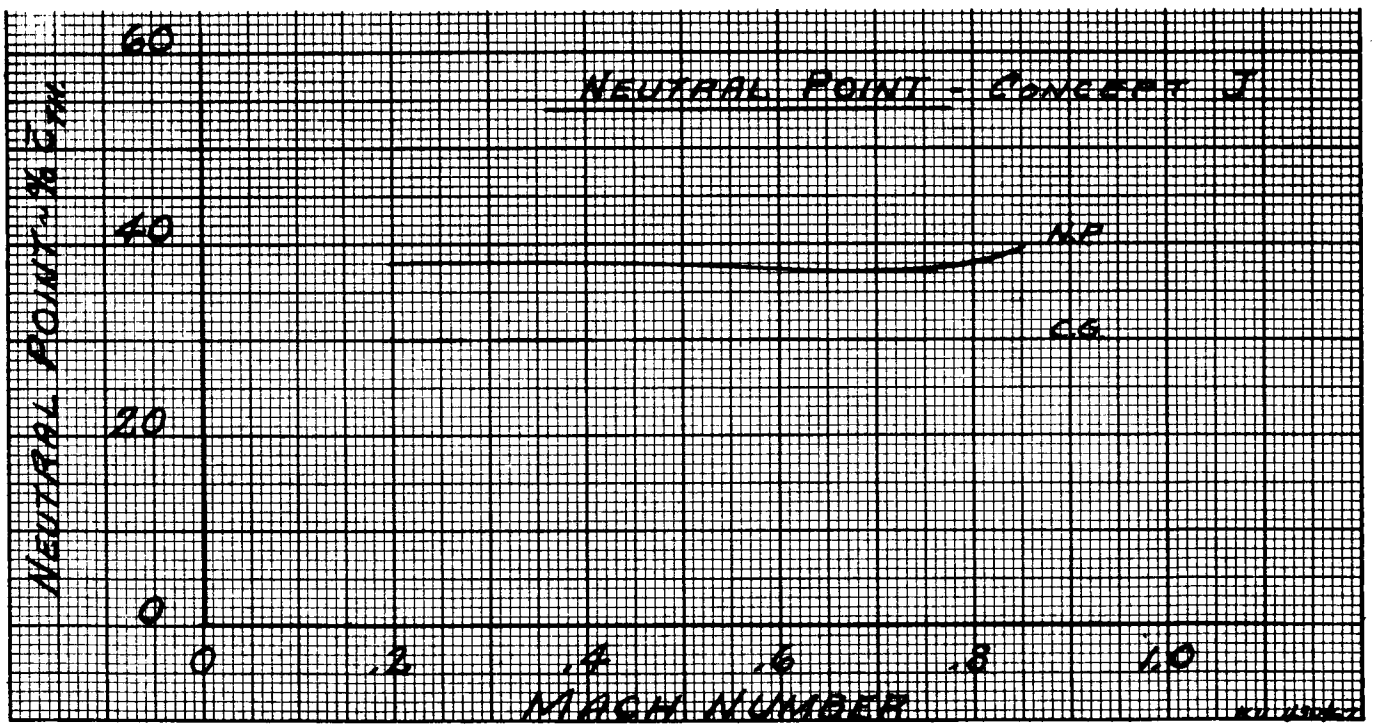


Figure VI-1. Neutral Point Variation with Mach Number Concepts C and J.

TABLE VI-2. CONDITIONS FOR LONGITUDINAL STATIC STABILITY

Configuration	Speed Range	"Trim Speeds"
Cruise (CR)	$1.4\ V_{S_G}$ to V_{NRP}	Speed for max. range $1.4\ V_{S_G}$ V_{NRP}
Clean (P)	$0.75\ V_{NRP}$ to V_H	V_{NRP}, V_H
Clean (P) (Climb)	$0.85\ V_{R/C}$ or $1.15 V_{S_G}$, $1.3\ V_{R/C}$	$V_{R/C}$
Glide (G)	V_{S_G} to V_H	$1.4\ V_{S_G}$, V_H
Landing (L)	V_{S_L} to limit structural speed	$1.4\ V_{S_L}$
Power Approach (PA)	V_{S_L} to limit structural speed	$1.5\ V_{S_L}$

Note: V_{S_G} = stall speed in glide configuration

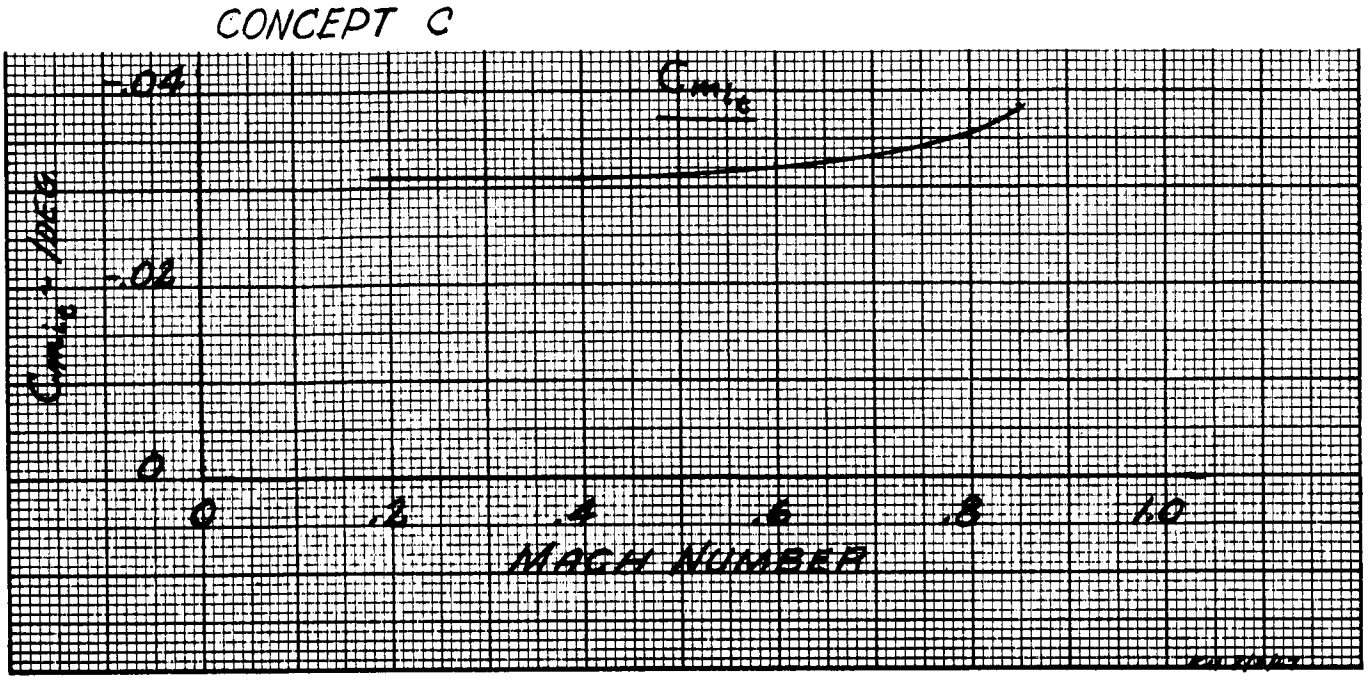
V_{S_L} = stall speed in landing configuration

$V_{R/C}$ = airspeed at optimum rate of climb

V_{NRP} = airspeed at normal rated power

V_H = high speed

Pitching moment and lift effectiveness coefficients of the Concept C and J controls are shown in Figures VI-2 and VI-3.



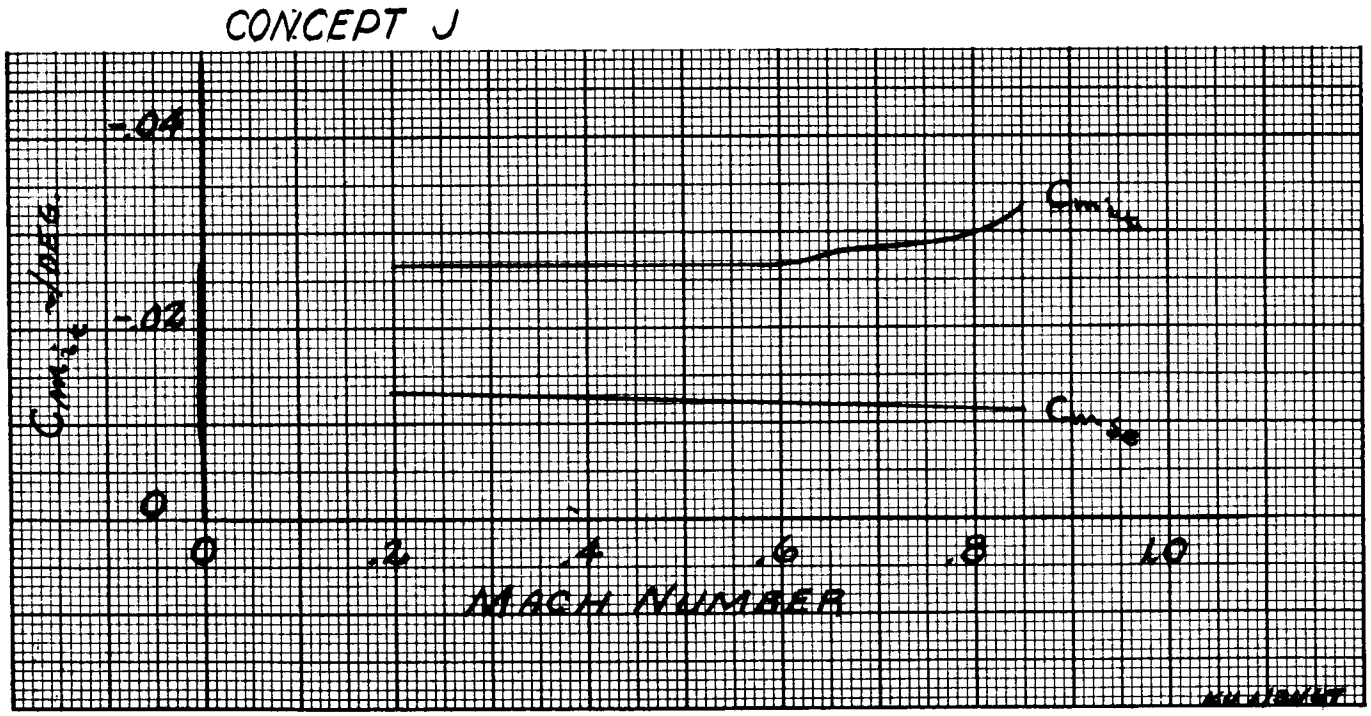


Figure VI-2. Longitudinal Control Power, Pitching Moment Slopes for Concepts C and J.

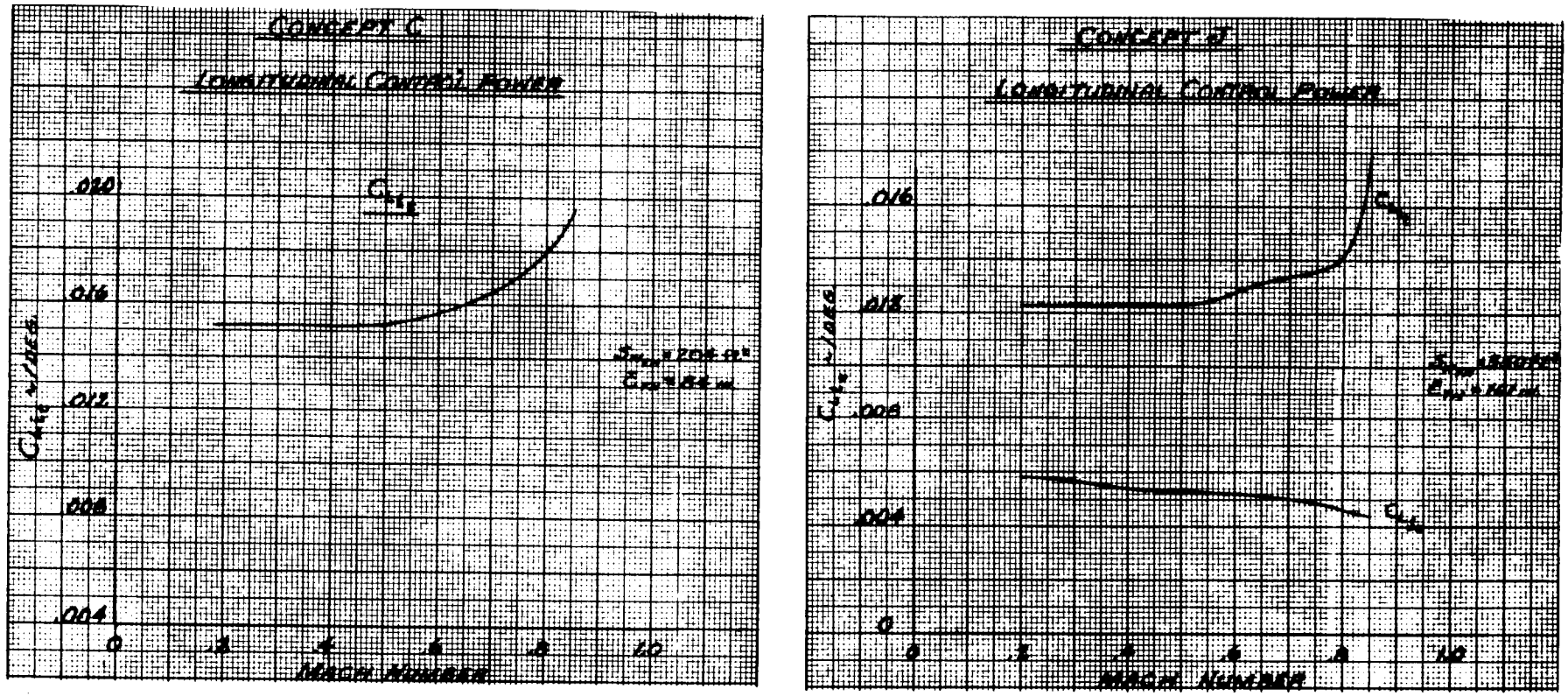


Figure VI-3. Longitudinal Control Power Concept C and J.

(2) Stability in Accelerated Flight

The slope of the curve of stabilator deflection versus normal acceleration (g) at constant speed is stable (increasing negative deflection required for increasing normal acceleration) throughout the range of attainable load factors in all configurations and in all conditions of flight. The stable slopes of n per δ_e are shown in Figure VI-21 for Concept C and in Figure VI-22 for Concept J.

(3) Short-Period Oscillations

The dynamic oscillations of normal acceleration, which occur at approximately constant speed and which may be produced by abruptly deflecting and returning the stabilator control to the trimmed position, is damped to 1/10 amplitude in one cycle, and the magnitude of any residual oscillations do not exceed ± 0.05 g at the pilot location. Residual oscillations in angular attitude shall not be of objectionable magnitude and shall not adversely affect the research utility of the airplane. All longitudinal oscillations with periods less than 6 seconds comply with the conditions.

When the elevator is abruptly deflected and released, the motion of the elevator following the release is essentially deadbeat. There is no tendency for a sustained or uncontrollable oscillation resulting from efforts of the pilot to maintain steady flight.

The unaugmented (airframe alone) dynamic longitudinal stability characteristics are shown in Table VI-3 (Concept C) and Table VI-4 (Concept J) in terms of damping ratio, damped natural frequency, period, and time and cycles to damp to both 1/2 and 1/10 amplitude. Figure VI-4 (Concept C) and Figure VI-5 (Concept J) shows the natural frequency, ω_n and damping ratio, ζ of the airframe alone as a function of altitude and Mach number. The intersection of the altitude and Mach number curves represent computed data points. Also shown are boundaries of desirable and satisfactory regions. Although the indicated boundaries do not represent specification requirements it is generally agreed that the handling qualities degenerate as the natural frequency and damping move away

TABLE VI-3 DYNAMIC LONGITUDINAL STABILITY - CONCEPT C (1/2)

ALT	M	ζ	ω_n	ω_D	P_{sec}	$T_{1/2}$ per	$T_{1/10}$ sec	$C_{1/2}$	$C_{1/10}$		
S.L	.2	.425	.181	.164	6.08	1.43	4.75	.235	.718		
	.3	.433	.274	.247	4.05	.931	3.09	.230	.763		
	.4	.441	.370	.332	3.01	.677	2.25	.224	.746		
	.5	.450	.461	.412	2.43	.531	1.76	.219	.728		
	.6	.457	.559	.497	2.01	.432	1.44	.215	.714		
	.7	.456	.766	.628	1.59	.342	1.14	.215	.715		
	.8	.455	.935	.832	1.20	.259	.861	.216	.717		
	.85	.448	1.06	.950	1.05	.232	.770	.220	.731		
5000	.2	.400	.164	.150	6.67	1.69	5.61	.253	.841		
	.3	.407	.246	.225	4.44	1.10	3.65	.247	.822		
	.4	.415	.332	.302	3.30	.798	2.65	.242	.803		
	.5	.424	.415	.376	2.66	.628	2.08	.235	.784		
	.6	.430	.502	.453	2.20	.510	1.70	.232	.769		
	.7	.430	.635	.573	1.74	.404	1.34	.232	.770		
	.8	.429	.840	.759	1.32	.306	1.02	.232	.772		
	.85	.421	.956	.866	1.15	.274	.91	.237	.787		
10000	.2	.374	.147	.136	7.34	2.01	6.66	.273	.908		
	.3	.381	.221	.204	4.89	1.31	4.34	.267	.888		
	.4	.389	.299	.275	3.63	.949	3.15	.261	.868		
	.5	.397	.373	.342	2.92	.746	2.48	.255	.847		
	.6	.403	.451	.413	2.42	.606	2.01	.256	.821		
	.7	.403	.570	.521	1.92	.486	1.60	.250	.831		
	.8	.402	.754	.690	1.45	.364	1.21	.251	.834		
	.85	.395	.857	.788	1.26	.325	1.08	.256	.851		

TABLE VI-3 DYNAMIC LONGITUDINAL STABILITY - CONCEPT C (2/2)

ALT	M	ζ	ω_n	ω_D	P_{SEC}	$T_{1/2\,sec}$	$T_{1/10\,sec}$	$C_{1/2}$	$C_{1/10}$		
15000	.2	.349	.132	.124	8.10	2.40	7.97	.296	.984		
	.3	.356	.198	.185	5.39	1.56	5.19	.290	.962		
	.4	.363	.268	.249	4.01	1.13	3.77	.283	.940		
	.5	.371	.334	.309	3.23	.891	2.96	.276	.917		
	.6	.377	.403	.374	2.68	.725	2.41	.271	.900		
	.7	.377	.510	.472	2.12	.574	1.91	.271	.901		
	.8	.376	.675	.625	1.60	.435	1.44	.272	.903		
	.85	.369	.768	.713	1.40	.389	1.29	.277	.922		
25000	.2	.300	.105	.100	9.98	3.50	11.63	.351	1.17		
	.3	.306	.158	.156	6.65	2.28	7.58	.343	1.14		
	.4	.313	.213	.202	4.94	1.66	5.56	.335	1.11		
	.5	.320	.265	.251	3.80	1.30	4.32	.327	1.09		
	.6	.325	.321	.303	3.30	1.06	3.51	.321	1.07		
	.7	.325	.405	.383	2.61	.838	2.79	.321	1.07		
	.8	.324	.536	.507	1.97	.635	2.11	.322	1.07		
	.85	.318	.610	.579	1.73	.568	1.88	.329	1.09		

TABLE VI-4. DYNAMIC LONGITUDINAL STABILITY - CONCEPT J (1/2)

Alt.	M	ζ	ω_n	ω_D	$P_{sec.}$	$T_{1/2\,sec.}$	$T_{1/2\,cycles}$	$C_{1/2}$	$C_{1/10}$			
S.L.	.2	.514	.223	.191	5.22	.962	3.19	.184	.612			
	.3	.521	.338	.289	3.46	.625	2.08	.180	.600			
	.4	.532	.456	.386	2.59	.454	1.51	.175	.583			
	.5	.538	.574	.484	2.07	.356	1.19	.172	.574			
	.6	.546	.697	.584	1.71	.290	.962	.169	.562			
	.7	.571	.840	.690	1.45	.230	.763	.158	.526			
	.8	.587	1.09	.881	1.14	.173	.574	.152	.506			
	.85	.562	1.26	1.04	.958	.155	.516	.162	.539			
5K	.2	.486	.200	.175	5.72	1.14	3.77	.198	.659			
	.3	.493	.302	.263	3.80	.739	2.45	.195	.646			
	.4	.504	.407	.352	2.84	.536	1.78	.189	.628			
	.5	.510	.513	.441	2.26	.421	1.40	.186	.618			
	.6	.518	.623	.532	1.87	.342	1.14	.182	.605			
	.7	.542	.749	.629	1.56	.271	.901	.170	.567			
	.8	.558	.968	.803	1.24	.204	.673	.164	.545			
	.85	.534	1.13	.952	1.05	.184	.616	.175	.581			
10K	.3	.458	.179	.159	6.30	1.35	4.48	.214	.712			
	.3	.465	.271	.240	4.17	.877	2.91	.210	.698			
	.4	.476	.364	.320	3.12	.637	2.12	.204	.678			
	.5	.481	.458	.401	2.49	.500	1.66	.201	.667			
	.6	.489	.555	.484	2.06	.406	1.35	.197	.654			
	.7	.513	.667	.570	1.75	.322	1.07	.184	.612			
	.8	.528	.861	.731	1.37	.242	.805	.177	.589			
	.85	.504	1.003	.866	1.15	.218	.725	.189	.627			

TABLE VI-4 DYNAMIC LONGITUDINAL STABILITY - CONCEPT J (2/2)

ALT	M	ζ	ω_n	ω_D	P_{sec}	$T_{1/2}$ sec	$T_{1/10}$ sec	$C_{1/2}$	$C_{1/10}$			
15K	.2	.429	.159	.144	6.95	1.61	5.36	.232	.771			
	.3	.436	.241	.217	4.61	1.05	3.48	.227	.756			
	.4	.447	.324	.290	3.45	.761	2.53	.221	.734			
	.5	.452	.407	.364	2.75	.598	1.99	.218	.723			
	.6	.460	.494	.439	2.28	.485	1.61	.213	.708			
	.7	.484	.592	.518	1.93	.385	1.28	.200	.663			
	.8	.498	.763	.662	1.51	.290	.964	.192	.638			
	.85	.475	.891	.785	1.27	.261	.866	.205	.680			
5K	.2	.372	.125	.117	8.57	2.35	7.83	.274	.913			
	.3	.378	.190	.176	5.68	1.53	5.09	.270	.896			
	.4	.388	.255	.235	4.25	1.11	3.69	.262	.870			
	.5	.393	.321	.295	3.96	.874	2.90	.258	.856			
	.6	.400	.388	.356	2.81	.710	2.36	.253	.839			
	.7	.423	.464	.426	2.38	.563	1.87	.237	.786			
	.8	.436	.597	.537	1.86	.424	1.41	.227	.756			

CONCEPT C DYNAMIC LONGITUDINAL STABILITY

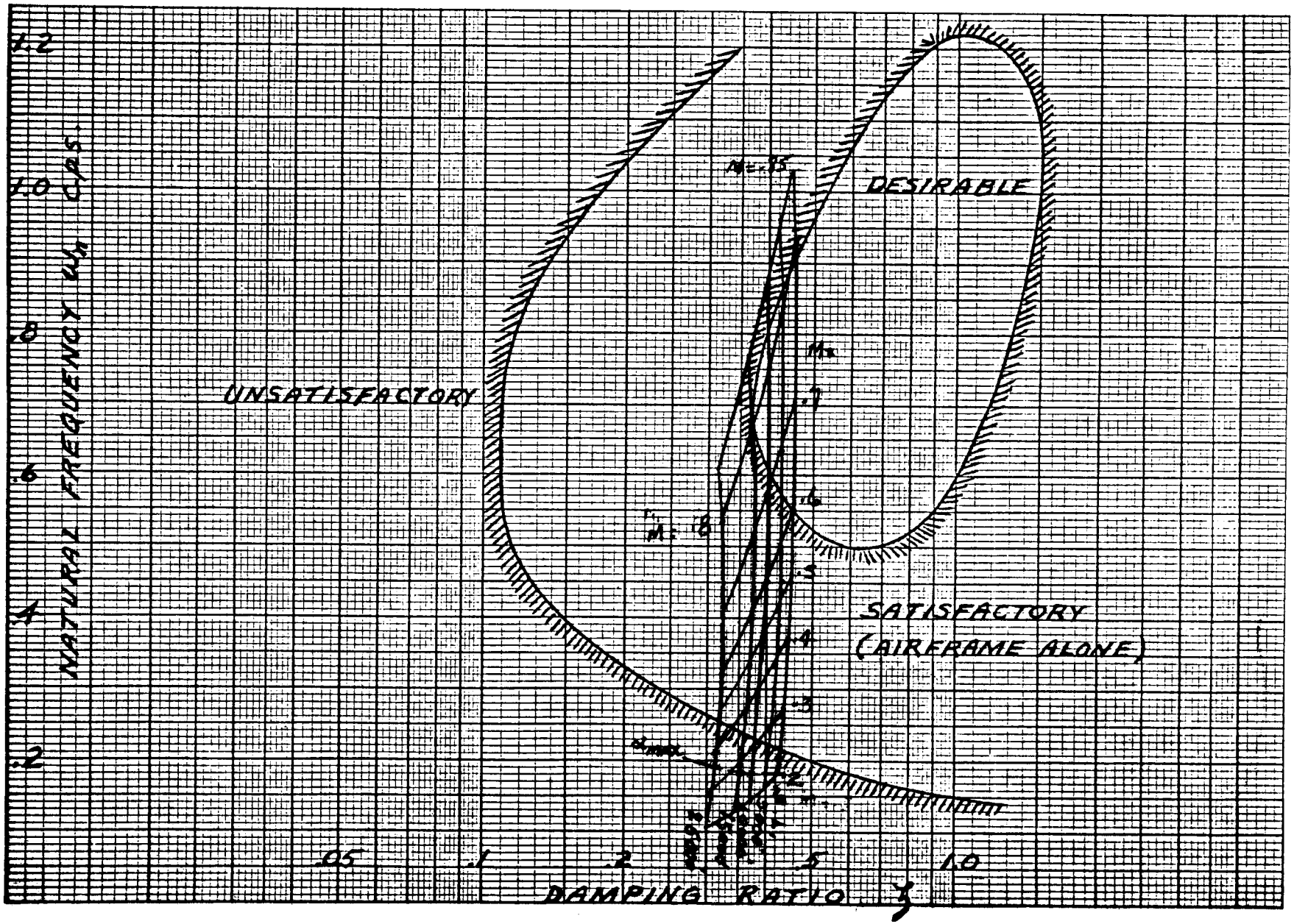


Figure VI-4.

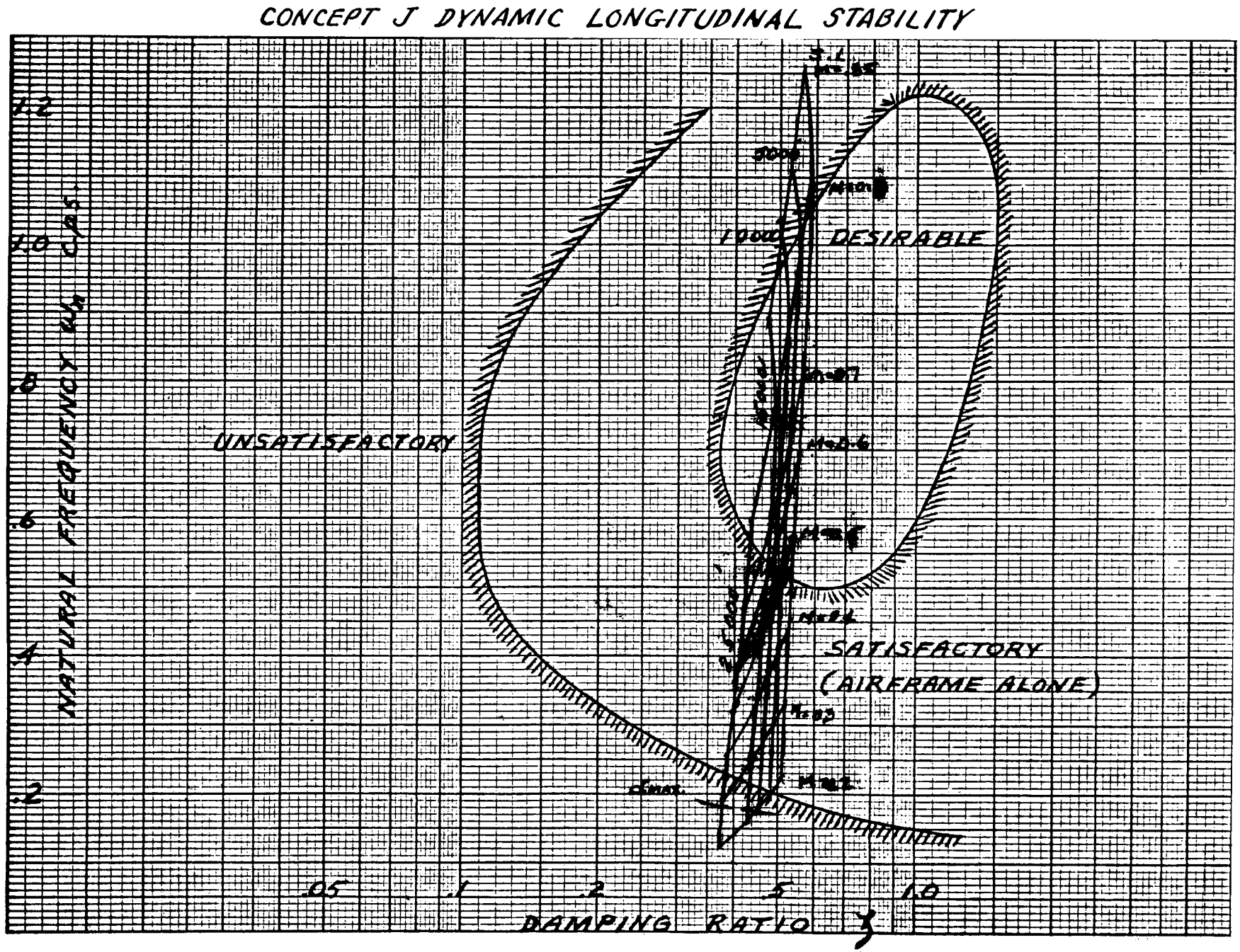

CONCEPT J DYNAMIC LONGITUDINAL STABILITY
NATURAL FREQUENCY ω_n CPS
1.2
1.0
.8
.6
.4
.2
.05
.1
.2
.5
1.0
DAMPING RATIO ζ
DESIRABLE
UNSATISFACTORY
SATISFACTORY
(AIRFRAME ALONE)
ζ_{max}

Figure VI-5

from the indicated desirable region. The basic airframe characteristics vary considerably over the flight range, however, for most flight conditions the characteristics are satisfactory or desirable.

c. Lateral-Directional Stability and Control

(1) Damping of the Lateral-Directional Oscillation

In the configuration and over the corresponding speed ranges specified in Table VI-5, the damping of the lateral-directional oscillations, with controls fixed and with controls free when excited by rudder pulses; is such that the damping parameter $1/C_{1/2}$ has a value not less than that required by curve A of Figure VI-6. The conditions listed in Table VI-5 were used for determination of compliance with these requirements.

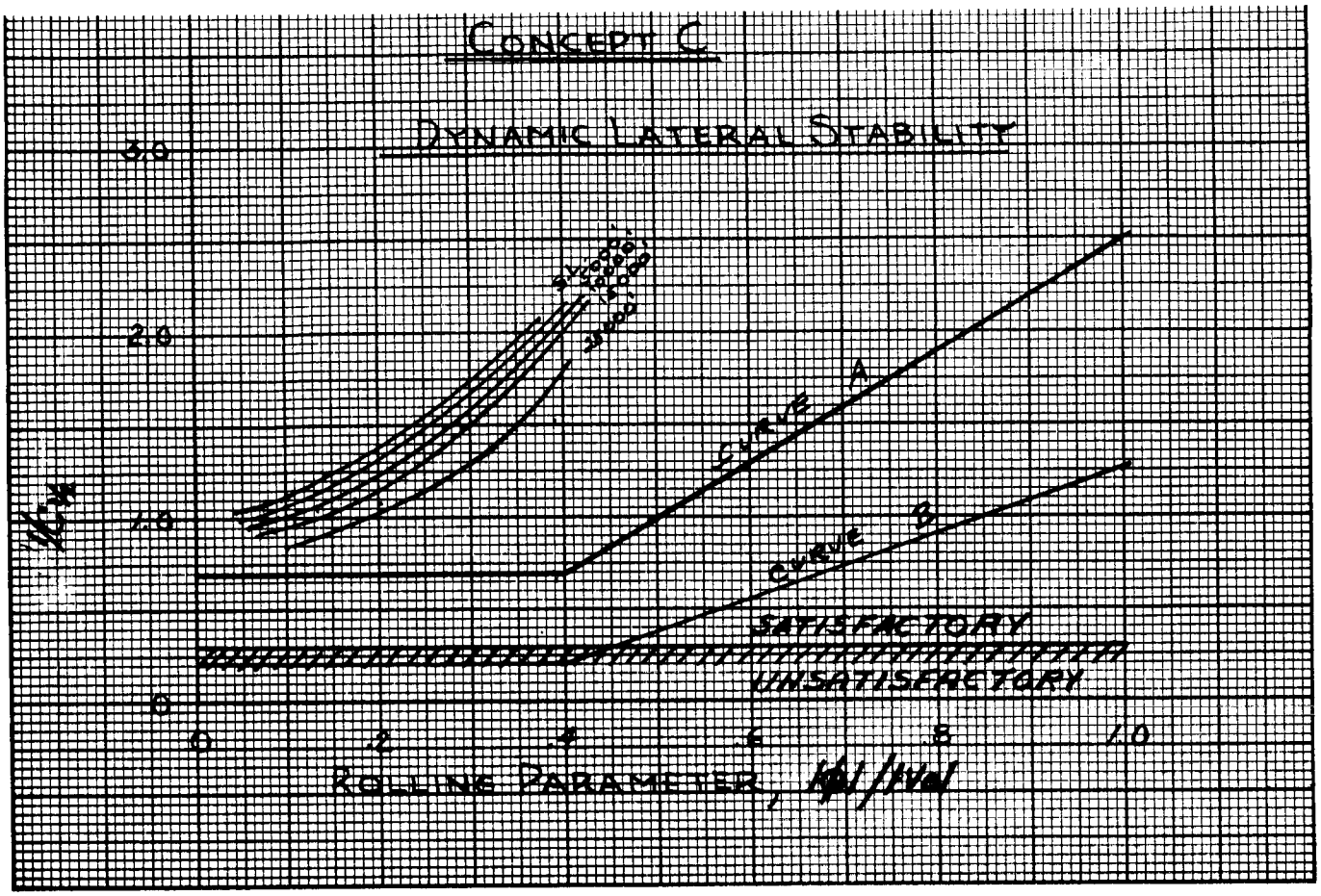
TABLE VI-5. FLIGHT CONDITIONS FOR INVESTIGATION OF LATERAL-DIRECTIONAL DAMPING

Configuration	Altitude	Speed
Cruise (CR)	25,000 ft	V for max. range
Clean (P)	S. L.	V for level flight
Clean (P)	25000 ft.	V for level flight
Power Approach (PA)	S. L.	1.15 V_{S_L}
Landing (L)	S. L.	1.4 V_{S_L}

If an artificial stabilization device is employed, the damping parameter $1/C_{1/2}$ with the artificial device inoperative is above the minimum of 0.24 in all configurations. In the power approach configuration (PA), this parameter is greater than the minimum values presented by curve B of Figure VI-6.

(2) Spiral Stability

Spiral stability is not required. Where the spiral motion is divergent, the rate of divergence following a small disturbance in bank with controls fixed is such that the bank angle is not doubled in less than 20 seconds in the power approach (PA) and cruise (CR) conditions of Table V. The bank angle is not doubled in less than 4 seconds in any other flight conditions of Table VI-2.



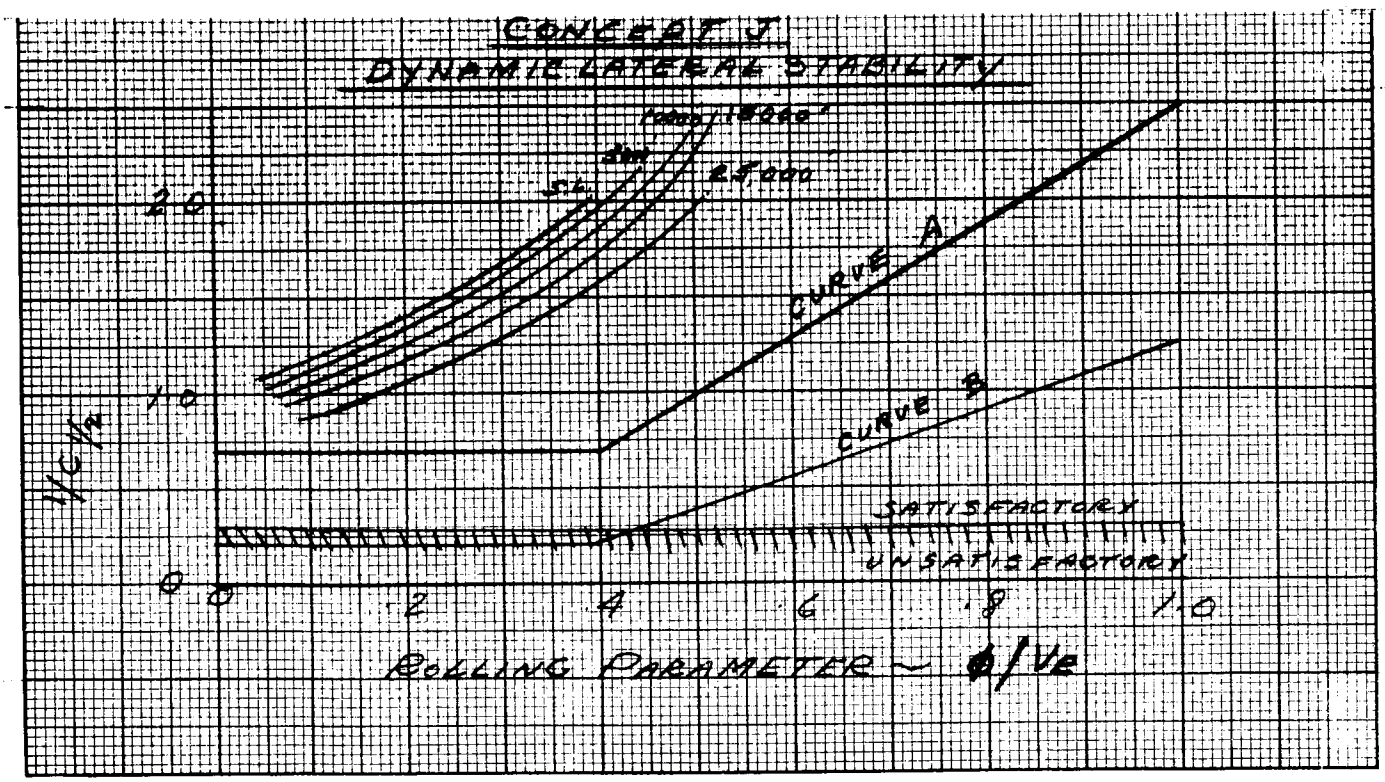


Figure VI-6. Damping of the Lateral-Directional Oscillation, Concepts C and J

(3) Steady Sideslip Conditions

The MIL-F-8785 (ASG) requirements were applied in straight path (zero trim rate) sideslips up to the sideslip angles produced by full rudder deflection or 250 pounds of rudder force or full aileron deflection, whichever is reached first. The requirement is met at the lightest normal service loading, in the configurations and speed ranges specified in Table VI-2, with the airplane trimmed for wings-level straight flight.

In addition, the requirements are met in the wave off configuration (WO) at all permissible speeds above $V_{S_{PA}}$ and in the power approach configuration (PA) with the airplane trimmed for wings-level straight flight at 1.15 V_{S_L}.

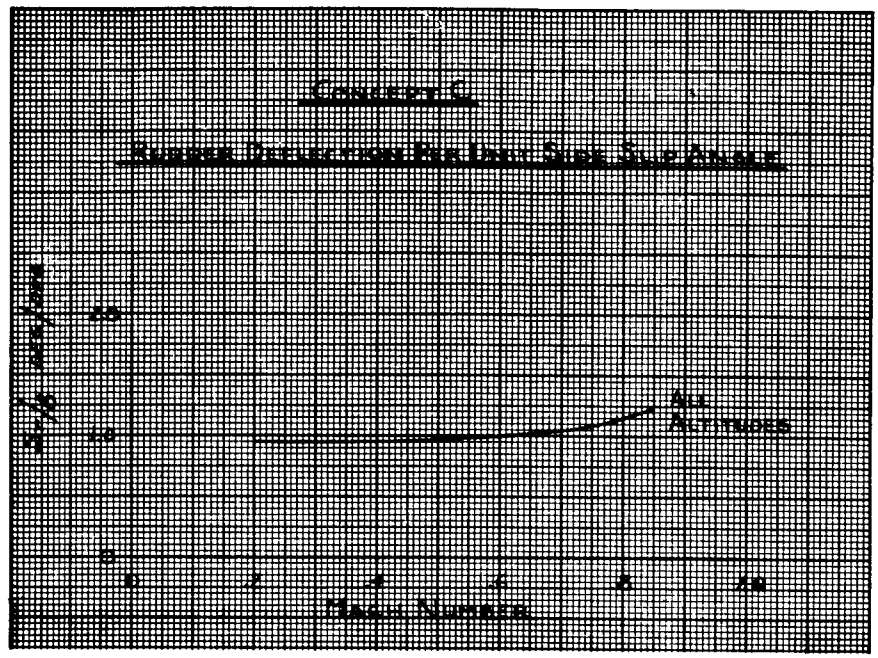
(a) Static Directional Stability (Rudder Position)

The airplanes possess rudder-fixed directional stability such that right rudder pedal deflection from the wings-level position is required in left sideslips, and left rudder pedal deflection is required in right sideslips. For angles of sideslip between ± 15 degree from the wings-level condition, the variation of sideslip angle with rudder pedal deflection is essentially linear. Throughout the remainder of the range of required pedal deflections, an increase in pedal deflection will always be required for an increase in sideslip.

Concept C has positive stability at all speeds as shown in Figure VI-7. Concept J exhibits control reversal at speeds below Mach 0.27 to 0.43 (the tendency increases with angle of attack). A vertical tail redesign is dictated for Concept J on this basis.

(b) Dihedral Effect (Aileron Position and Spoiler Position)

The airplane exhibits positive control-fixed dihedral effect as indicated by the variation of aileron cockpit control deflection with sideslip. Left aileron deflection is required for left sideslip, and right aileron deflection is required for right sideslip. The positive effective dihedral is such that less than 75 percent of full aileron cockpit control deflection is required in any of the sideslip angles which might be required in normal operation.



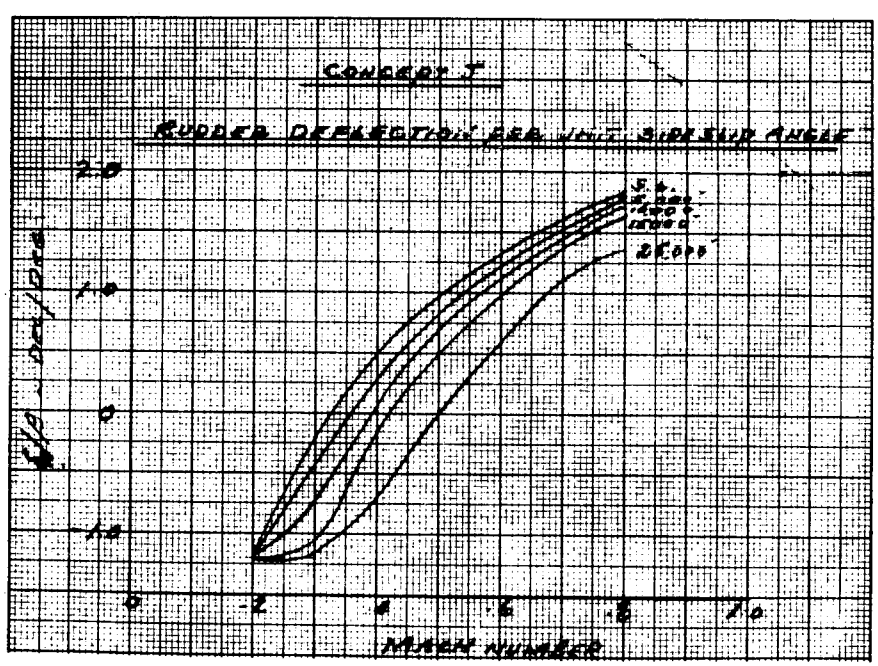


Figure VI-7. Static Directional Stability Concepts C and J

Analysis of the aircraft for compliance with this requirement is generally an analog process. The components in this analysis are presented.

1 Yawing Moment

The yawing moment due to sideslip is positive as shown on Figures VI-8 (Concept C)and VI-9 (Concept J). Concept C has -4 degrees dihedral and Concept J has +3 degrees dihedral.

The yawing moment due to lateral control deflection is constant with Mach number. Concept C has ailerons and spoilers, $C_{n_{\delta_a}} = -0.00197/$ deg and $C_{n_{\delta_s}} = +0.00033/\text{deg}$. Concept J has only ailerons, $C_{n_{\delta_a}} = -0.00187/\text{deg}$.

2 Rolling Moment

The rolling moment due to sideslip is negative and increases negatively with angle of attack. Concept C is presented in Figure VI-8 and Concept J is presented on Figure VI-9.

The rolling moment due to lateral controls is positive. Concept C is presented on Figure VI-10 and Concept J is presented on Figure VI-11.

Rolls initiated by lateral controls but performed with rudder free will have the rolling velocity in the correct direction. This maneuver requires a dynamic justification. It is reviewed in Section VI-B.3.

(c) Side Force in Sideslips

The side force characteristics are such that an increase in right bank angle accompanies a right sideslip, and an increase in left bank angle accompanies an increase in left sideslip. This is shown on Figure VI-8 (Concept C) and Figure VI-9 (Concept J) as C_{Y_β}.

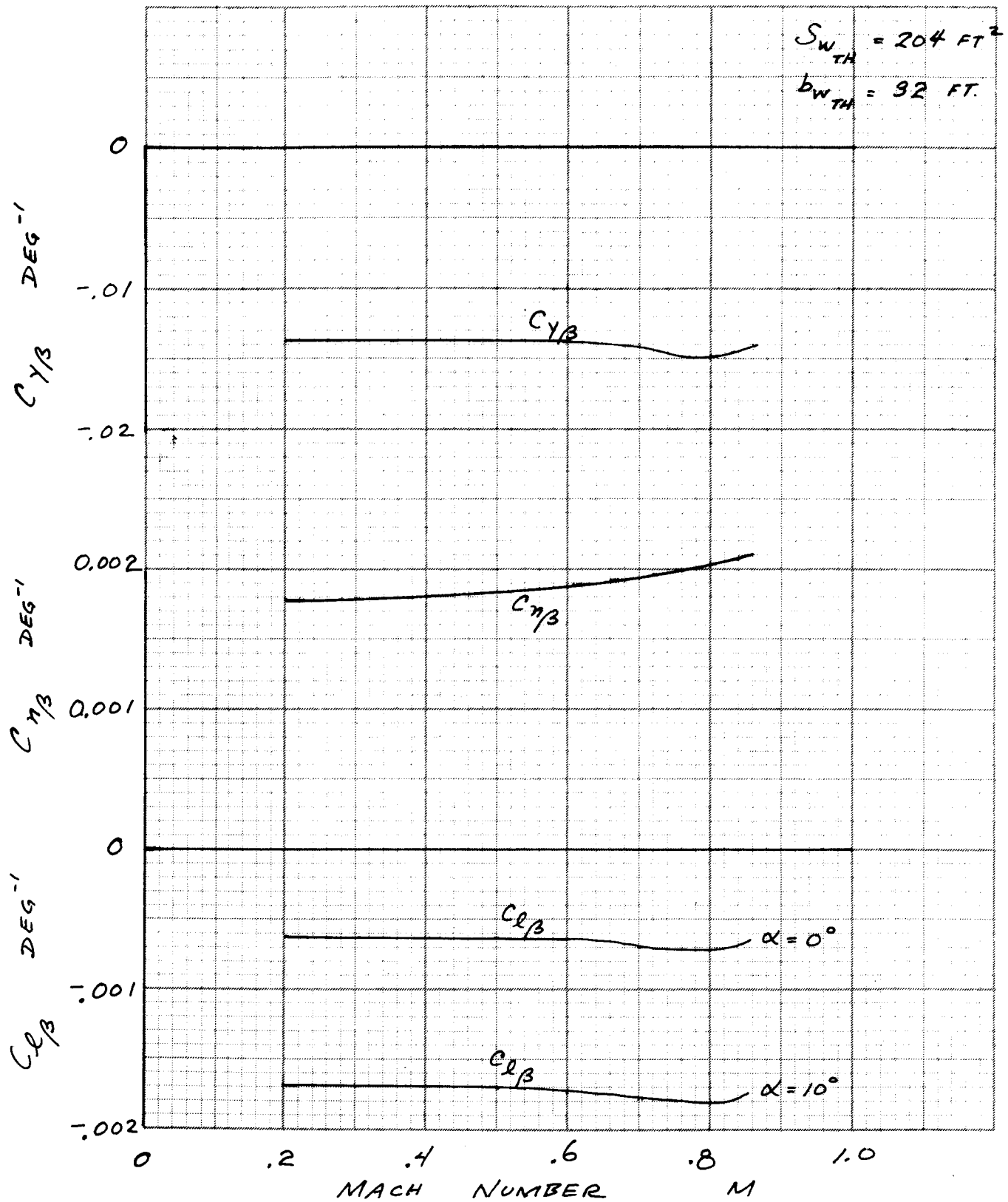


Figure VI-8. Static Lateral-Directional Stability Concept C

CONCEPT J
STATIC LATERAL-DIRECTIONAL STABILITY

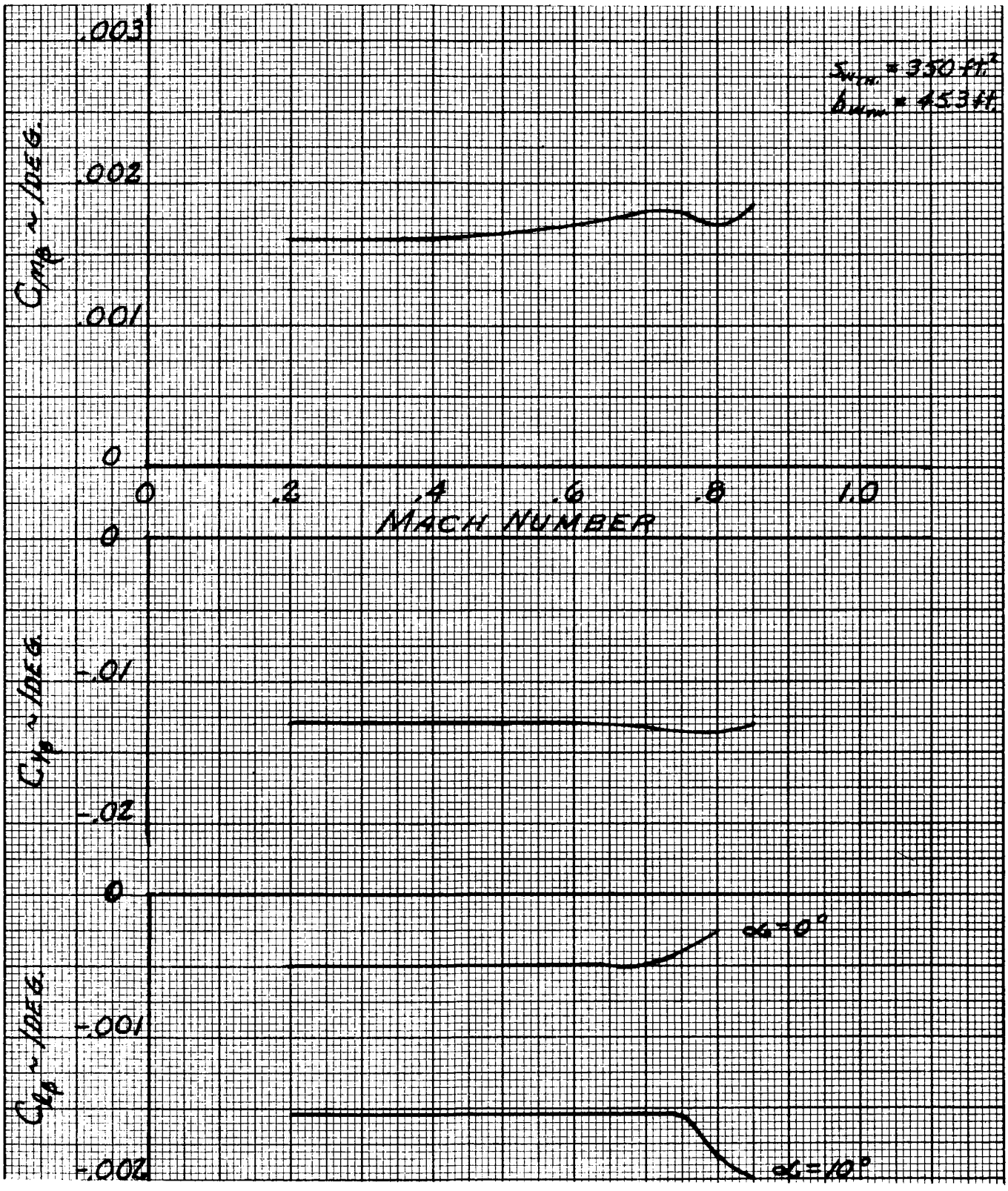


Figure VI-9

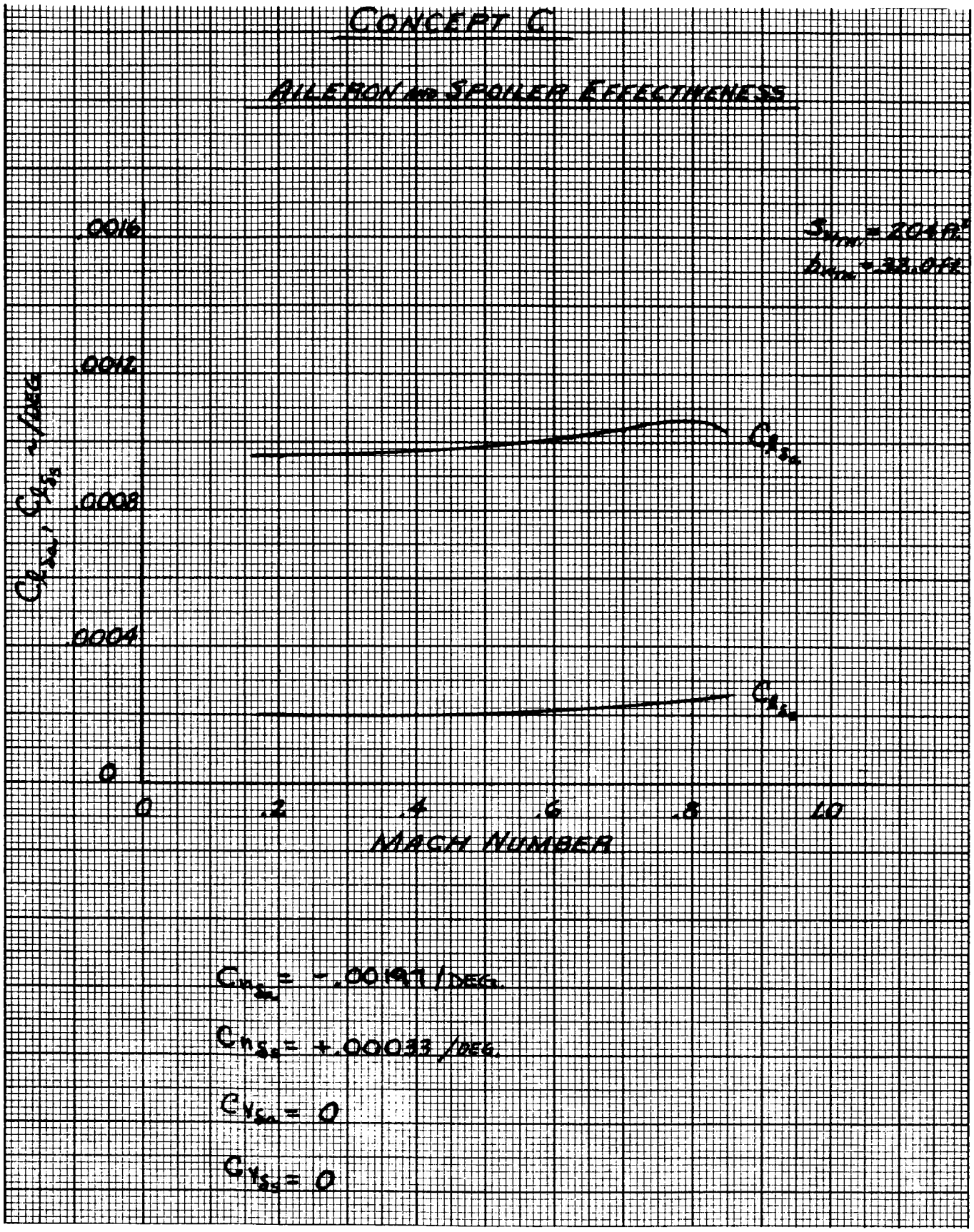

CONCEPT C
AILERON AND SPOILER EFFECTIVENESS
.0016
.0012
.0008
.0004
0
$C_{l_{\delta_a}}, C_{Y_{\delta_s}}$ /DEG
$C_{l_{\delta_a}}$
$C_{Y_{\delta_s}}$
0
.2
.4
.6
.8
1.0
MACH NUMBER
$C_{n_{\delta_a}} = -.00191$ /DEG.
$C_{n_{\delta_s}} = +.00033$ /DEG.
$C_{Y_{\delta_a}} = 0$
$C_{Y_{\delta_s}} = 0$

Figure VI-10

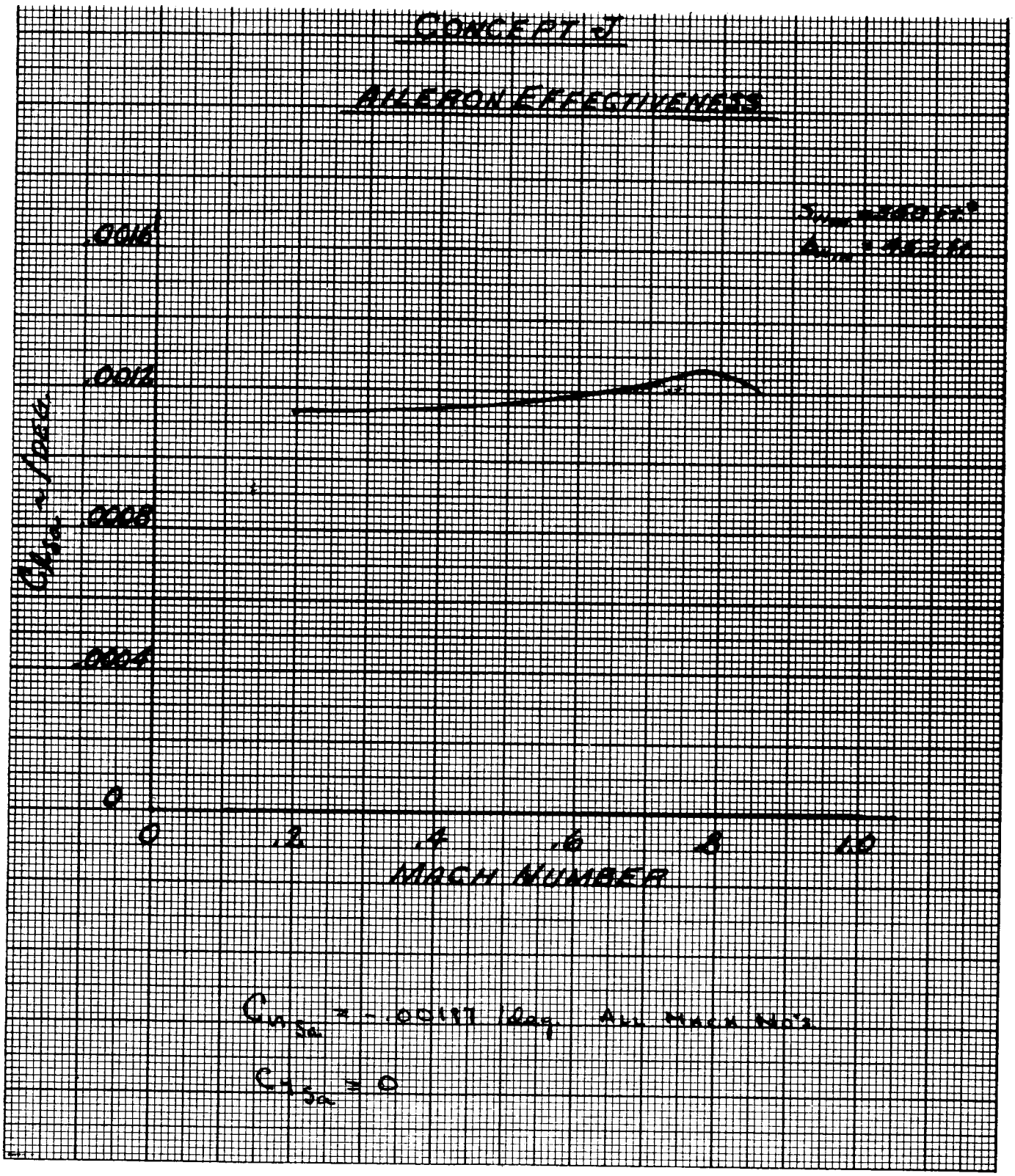

CONCEPT J
AILERON EFFECTIVENESS
.0016
.0012
.0008
.0004
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
$C_{l_{\delta a}}$ ~ /DEG.
$C_{n_{\delta a}}$ = -.00197 /deg. ALL MACH NO'S
$C_{Y_{\delta a}}$ = 0

Figure VI-11

(d) Adverse Yaw

The angle of sideslip developed during a rudder-pedal-fixed abrupt roll out of a trimmed, level, steady 45 degree banked turn at 1.4 $V_{S_{CR}}$ in the cruise configuration (CR) and at 1.4 $V_{S_{PA}}$ in the power approach configuration (PA) does not exceed 15 degrees.

(e) Asymmetric Cruise Engine Power (Rudder Free)

The airplane motions following rudder failure of one cruise engine is such that dangerous flight conditions can be avoided by normal pilot correction control action. In the clean configuration, power on (P), with the left cruise engine inoperative (with RPM simulating the static condition after an engine has failed in flight) and with the right cruise engine developing normal rated power, it is possible to maintain steady straight flight at all speeds above 1.4 V_{S_G} with rudder free by sideslipping and banking. The weight corresponds to the lightest normal service loading, and trim is as required for wings-level straight flight with symmetric power.

d. Stall Characteristics

Stall characteristics were studied at all permissible center-of-gravity positions, for configurations glide (G), cruise (CR), landing (L), and power approach (PA) in straight unaccelerated flight and with normal accelerations up to the limits of the operational flight envelope.

The approach to the complete stall is accompanied by an easily perceptible stall warning which occurs between 1.05 and 1.15 times stalling speed in configurations glide (G), landing (L), and cruise (CR), and between 1.05 and 1.10 times the stalling speed in the power approach configuration (PA). The stall warning consists of a buffeting or shaking of the airplane.

The airplane does not exhibit uncontrollable rolling or downward pitching at the stall in excess of 30 degrees.

It is possible to prevent the complete stall by normal use of the controls at the onset of the stall warning. In the event of a complete stall, it is possible to recover by normal use of the controls with reasonable control forces and without excessive loss of altitude or build up of speed.

2. Damping

a. Pitch Damping

Coefficients of damping in pitch due to pitching velocity (q) and vertical translation ($\dot{\alpha}$) are presented in Figure VI-12 (Concept C) and Figure VI-13 (Concept J). The pitch damping derivatives C_{mq} and $(C_{m_q} + C_{m_{\dot{\alpha}}})$ are primarily a function of the horizontal tail power, the damping in pitch is quite adequate.

b. Yaw Damping

Coefficients of damping in yaw due to the yawing velocity (r) are presented in Figure VI-14 for Concept C and Figure VI-15 for Concept J.

c. Roll Damping

Damping in roll due to the rolling velocity (p) is presented in Figures VI-16 and -17 for Concept C and Figures VI-18 and -19 for Concept J.

3. Trim Characteristics

a. Longitudinal Control Effectiveness

In erect unaccelerated flight at any altitude, the attainment of any permissible speed above the stalling speed is not limited by the effectiveness of the longitudinal control at all airplane configurations and permissible loadings.

In the forward critical loading, when trimmed at any permissible speed and altitude in the configurations listed in Table VI-2, it is possible to develop at the trim speed, by the use of the stabilator control alone, the limit load factor,

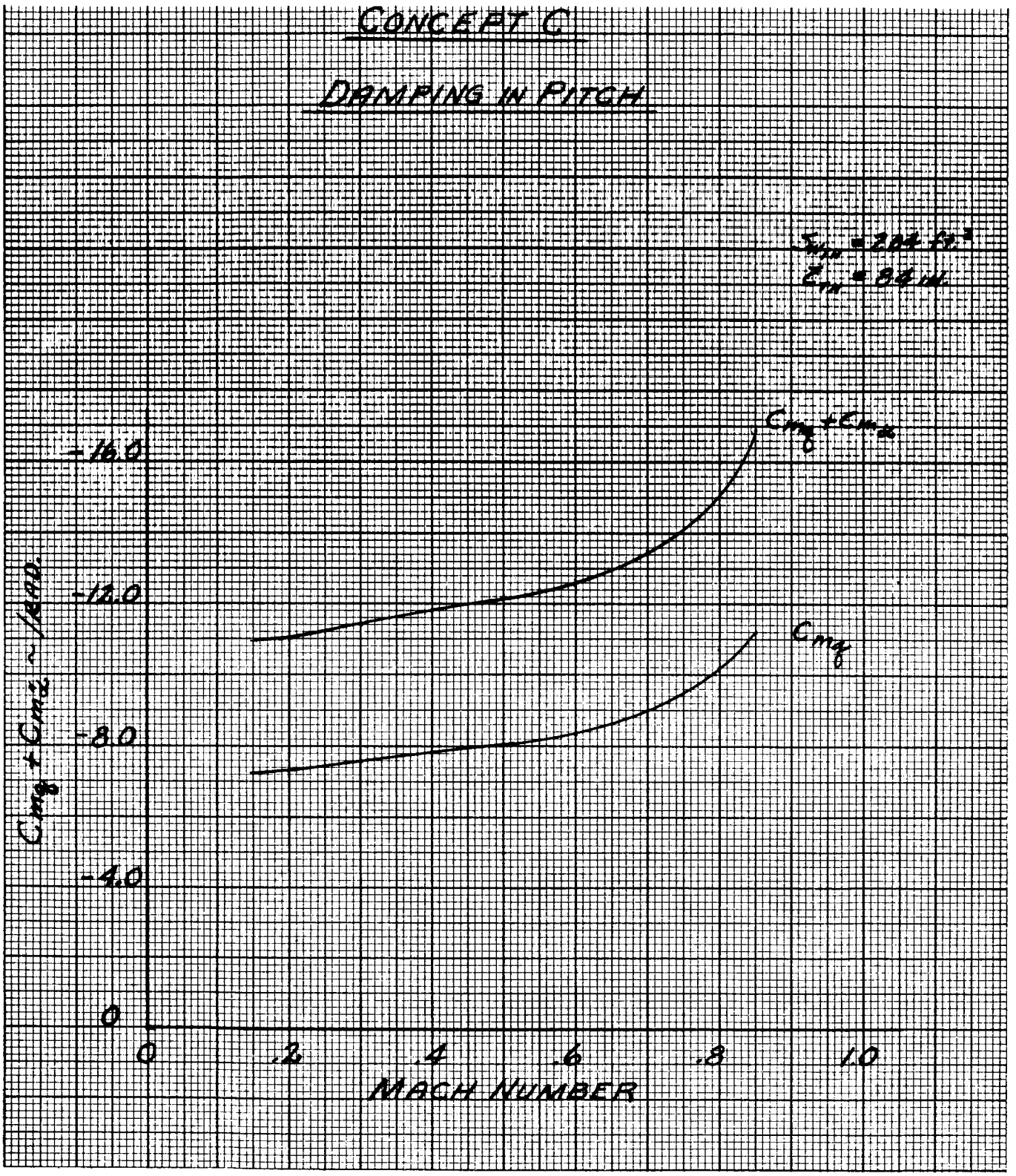

CONCEPT C
DAMPING IN PITCH
$S_{ref} = 204$ ft.^{2}
$\bar{c}_{ref} = 84$ in.
$C_{m_q} + C_{m_{\dot{\alpha}}}$
C_{m_q}
$C_{m_q} + C_{m_{\dot{\alpha}}} \sim$ /RAD.
-16.0
-12.0
-8.0
-4.0
0
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-12

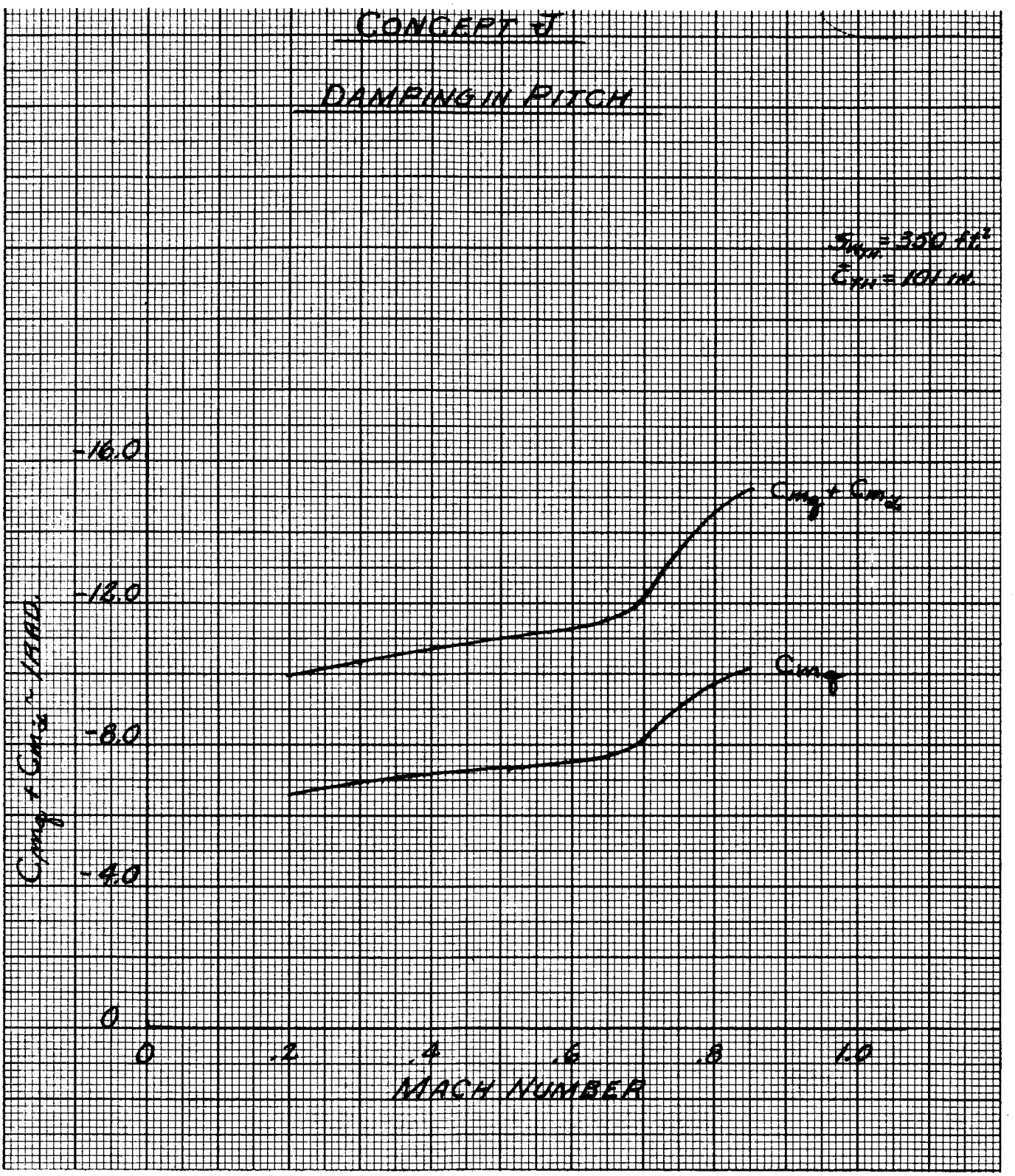

CONCEPT J
DAMPING IN PITCH
$S_{ref} = 350\ ft^2$
$\bar{c}_{ref} = 101\ in.$
-16.0
-12.0
-8.0
-4.0
0
$C_{m_q} + C_{m_{\dot{\alpha}}} \sim 1/RAD.$
$C_{m_q} + C_{m_{\dot{\alpha}}}$
C_{m_q}
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-13

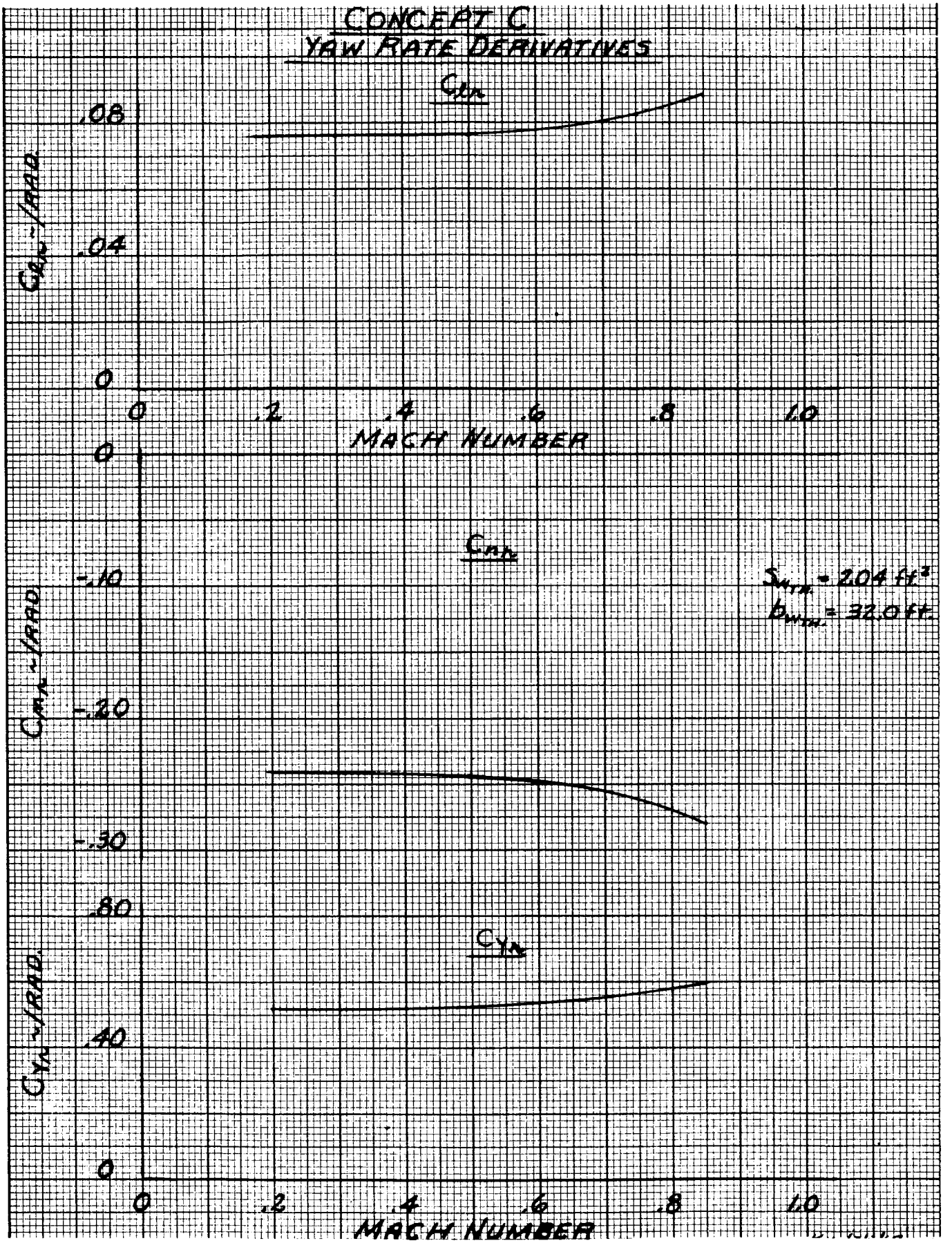

CONCEPT C
YAW RATE DERIVATIVES
C_{l_r}
.08
.04
0
C_{l_r} ~/RAD.
0
.2
.4
.6
.8
1.0
MACH NUMBER
0
C_{n_r}
-.10
-.20
-.30
C_{n_r} ~/RAD.
$S_{WTR} = 204\ ft^2$
$b_{WTR} = 32.0\ ft.$
.80
C_{Y_r}
.40
0
C_{Y_r} ~/RAD.
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-14

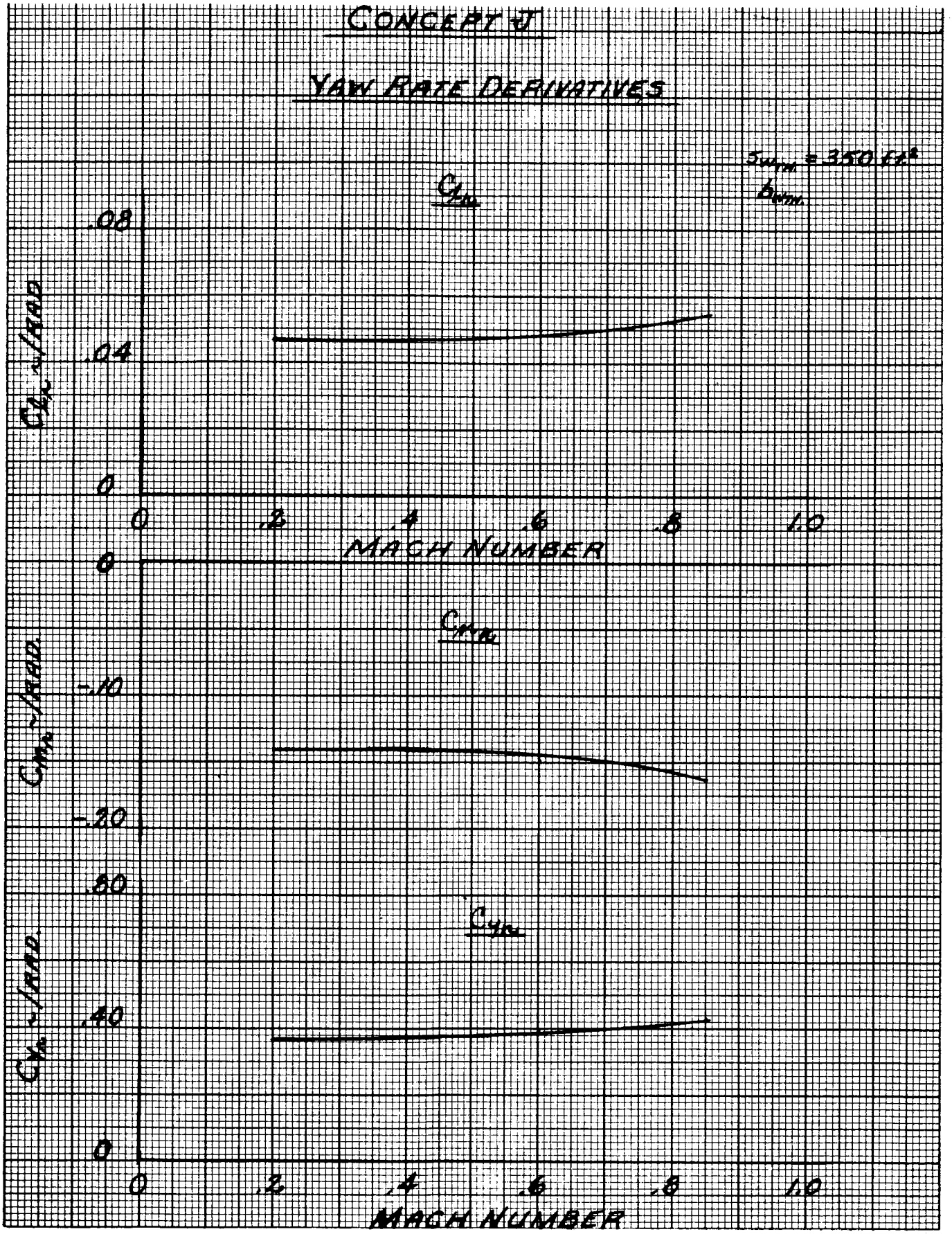

CONCEPT J
YAW RATE DERIVATIVES
$S_{WTM} = 350$ FT^2
b_{WTM}
C_{L_r}
C_{L_r} ~/RAD
.08
.04
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
C_{m_r}
C_{m_r} ~/RAD
0
-.10
-.20
C_{Y_r}
C_{Y_r} ~/RAD
.80
.40
0
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-15

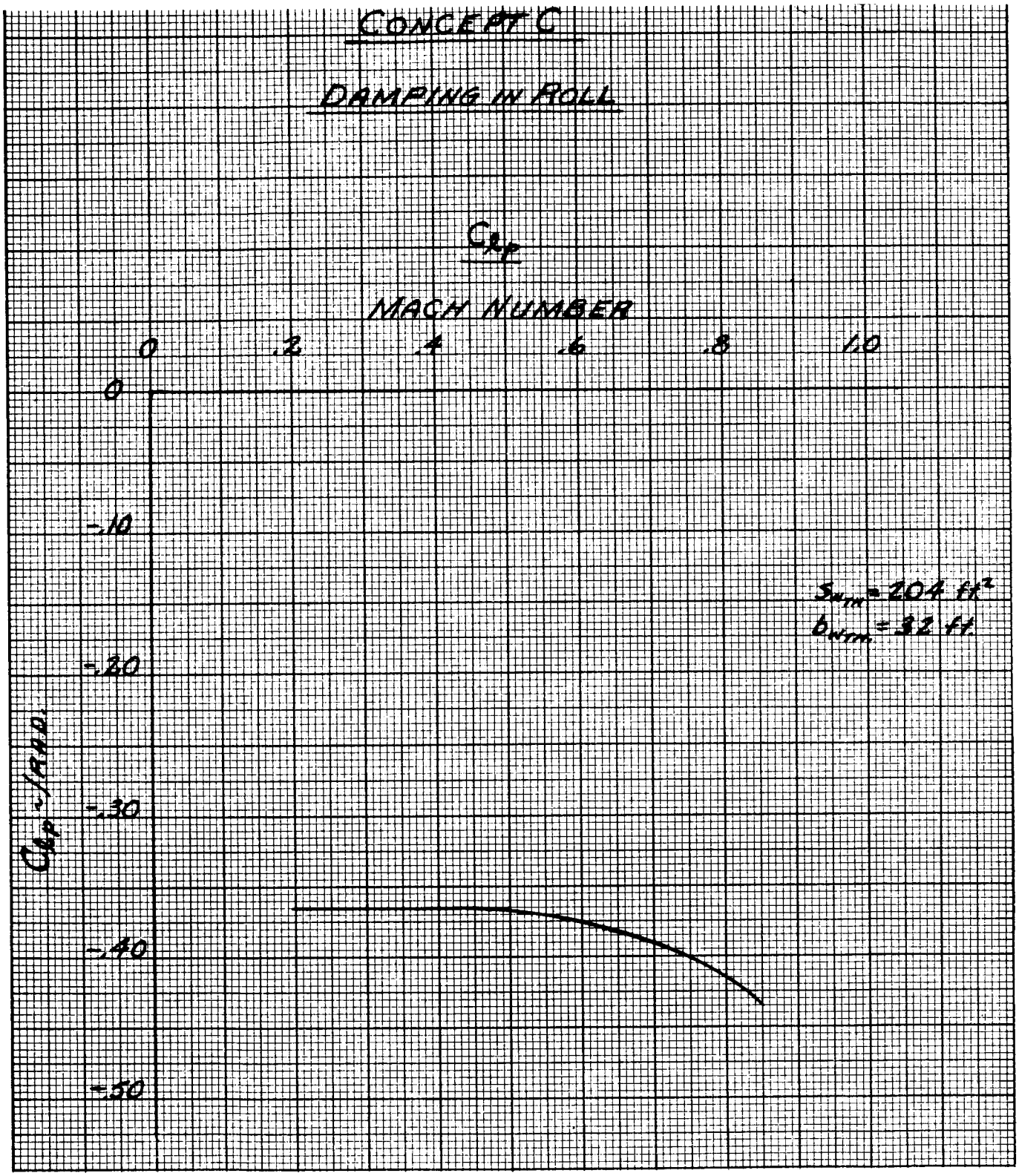

CONCEPT C
DAMPING IN ROLL
C_{l_p}
MACH NUMBER
0
.2
.4
.6
.8
1.0
0
-.10
-.20
-.30
-.40
-.50
C_{l_p} /RAD.
S_{w_m} = 204 ft²
b_{w_m} = 32 ft.

Figure VI-16

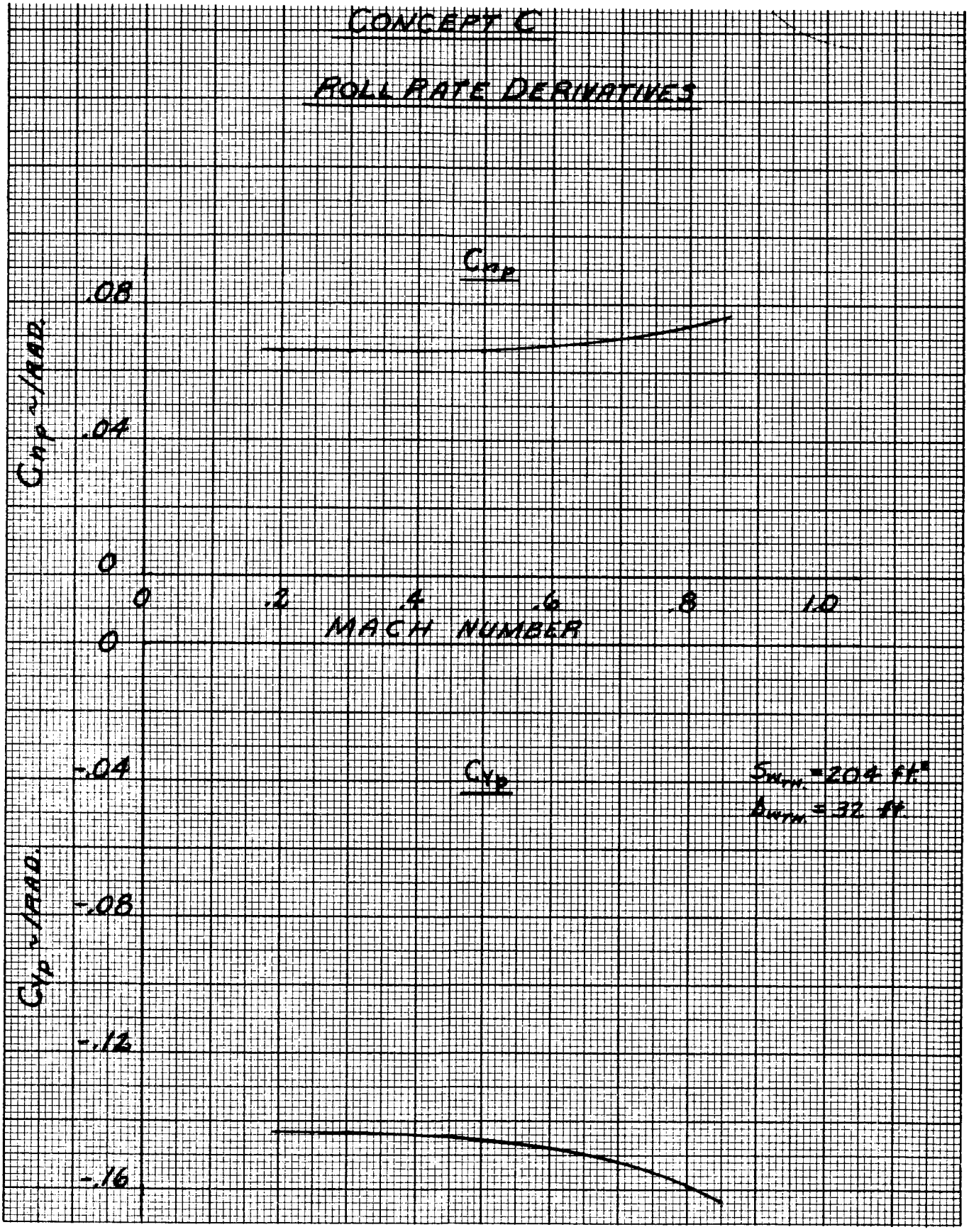

CONCEPT C
ROLL RATE DERIVATIVES
C_{n_p}
C_{n_p} ~/RAD.
.08
.04
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
0
-.04
-.08
-.12
-.16
C_{l_p}
C_{l_p} ~/RAD.
$S_{W_{ref.}}$ = 204 ft.²
$b_{W_{ref.}}$ = 32 ft.

Figure VI-17

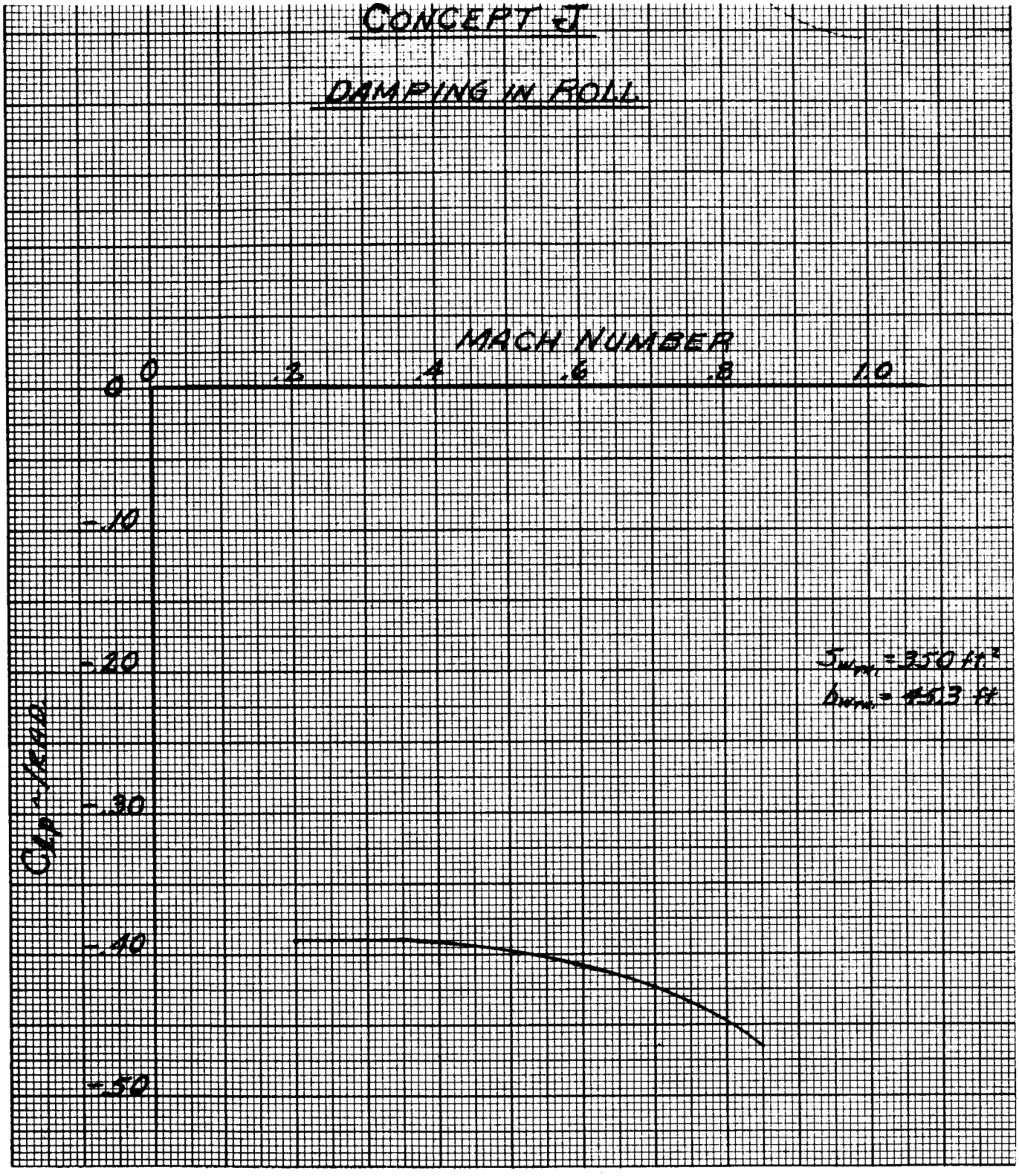

CONCEPT J
DAMPING IN ROLL
MACH NUMBER
0
.2
.4
.6
.8
1.0
0
-.10
-.20
-.30
-.40
-.50
C_{l_p} ~ 1/RAD
$S_{w_{TH}}$ = 350 ft.²
$b_{w_{TH}}$ = 45.3 ft

Figure VI-18

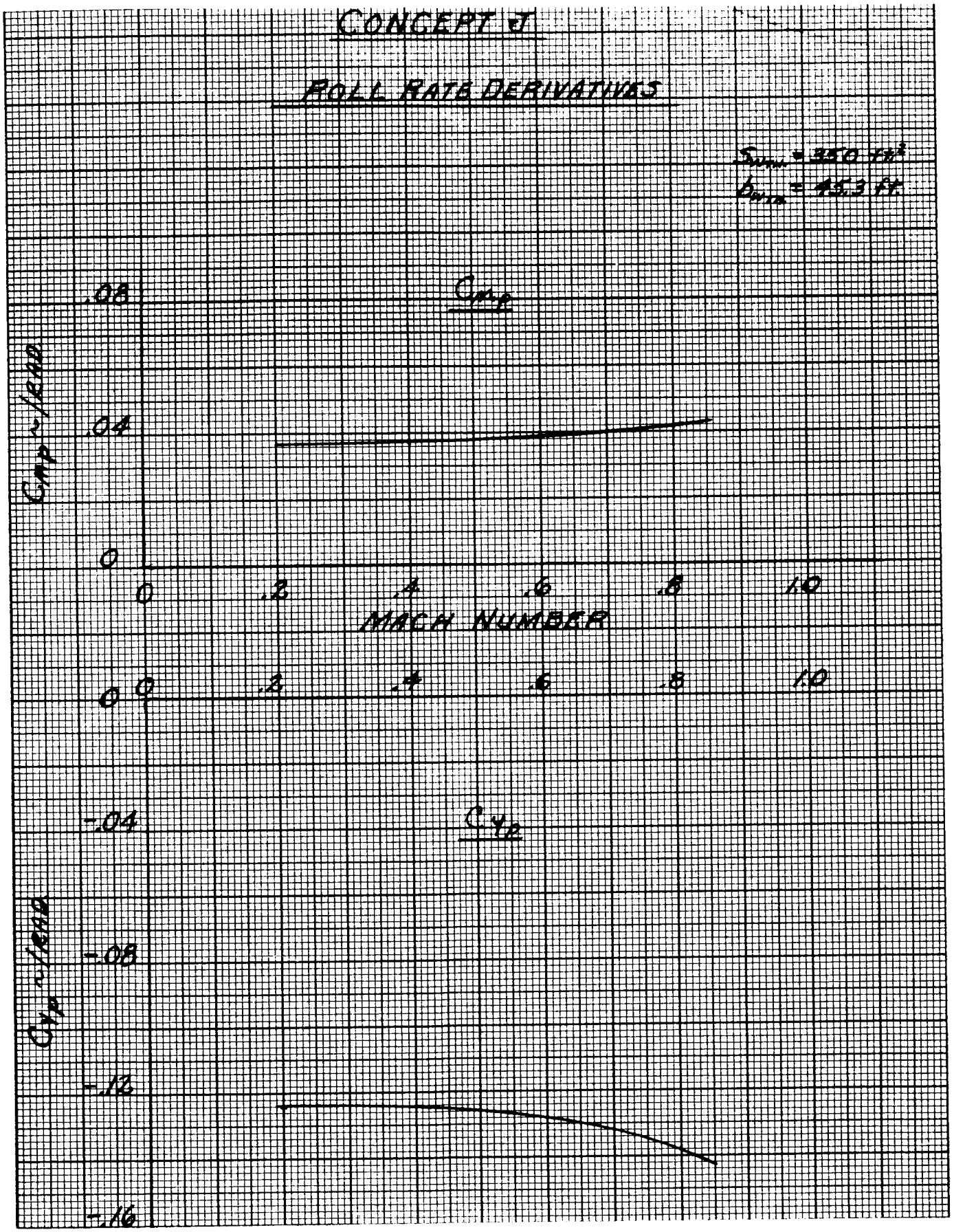

CONCEPT J
ROLL RATE DERIVATIVES
S_{wing} = 350 ft²
b_{wing} = 45.3 ft.
C_{m_p}
C_{m_p}/RAD
.08
.04
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
0
0
.2
.4
.6
.8
1.0
C_{y_p}
C_{y_p}/RAD
-.04
-.08
-.12
-.16

Figure VI-19

the lift coefficient corresponding to stall speed, or a load factor consistent with the operational flight envelope.

In steady turning flight and in pull-outs, increase in pull force will be required to produce increases in positive normal acceleration throughout the range of attainable acceleration.

The ability to obtain maximum load factor on Concept J is shown on Figure VI-20. Concept C can easily perform to this limit. The variation of load factor with longitudinal control deflection is shown in Figures VI-21 (Concept C) and VI-22 (Concept J).

Stabilator effectiveness will not restrict the take-off performance of the airplane. It is possible, on a hard-surface runway at a minimum speed no greater than stall speed, to obtain take-off attitude. These requirements are met with the airplane loading which produces the most critical nose-heavy moment. The loadings considered include all full and partial loads which might normally be employed during training, as well as operational takeoffs.

At the forward critical loading, with the airplane trimmed for 1.2 V_{S_L} in the power approach configuration (PA), longitudinal control is sufficiently effective that, in the landing configuration (L), the landing speed or landing stall speed can be obtained in close proximity to the ground.

The angle-of-attack and longitudinal control deflection required for level flight trim throughout the flight spectrum is presented on Figure VI-23 (Concept C) and Figure VI-24 (Concept J). Trim angle of attack is less than 10 degrees above Mach 0.4 and longitudinal control will not restrict effectiveness in attaining the maximum lift. Lower trim angle-of-attack can be achieved by increased horizontal tail area.

b. Longitudinal Trim Changes

The longitudinal trim changes caused by changes in power, flap setting, gear operation, lift engine operation, etc. will not be so large that peak longitudinal control forces in excess of 10 pounds are required when such configuration

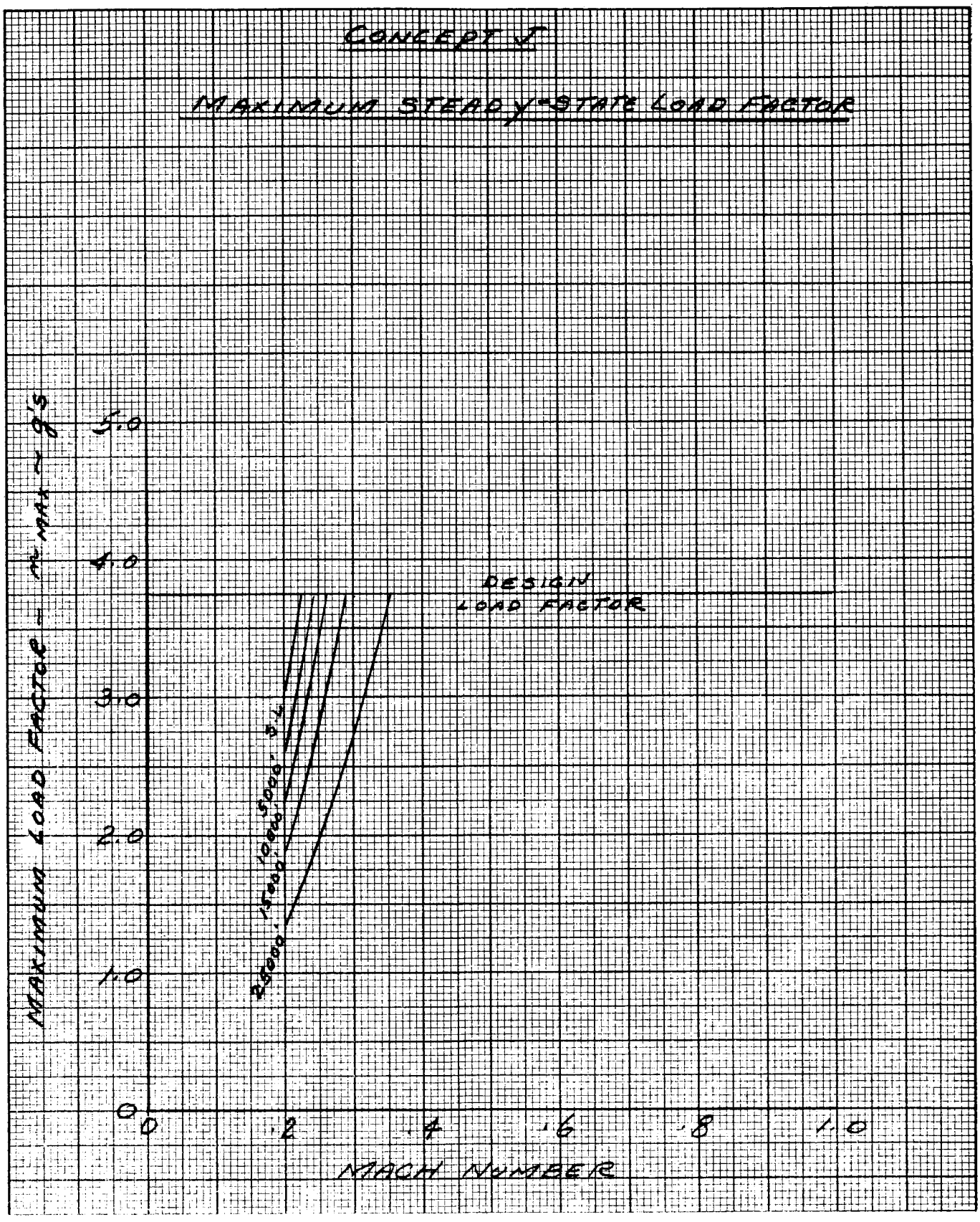

CONCEPT V
MAXIMUM STEADY-STATE LOAD FACTOR
MAXIMUM LOAD FACTOR = n_{MAX} ~ g's
5.0
4.0
3.0
2.0
1.0
0
DESIGN LOAD FACTOR
S.L.
5000'
10000'
15000'
25000'
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-20

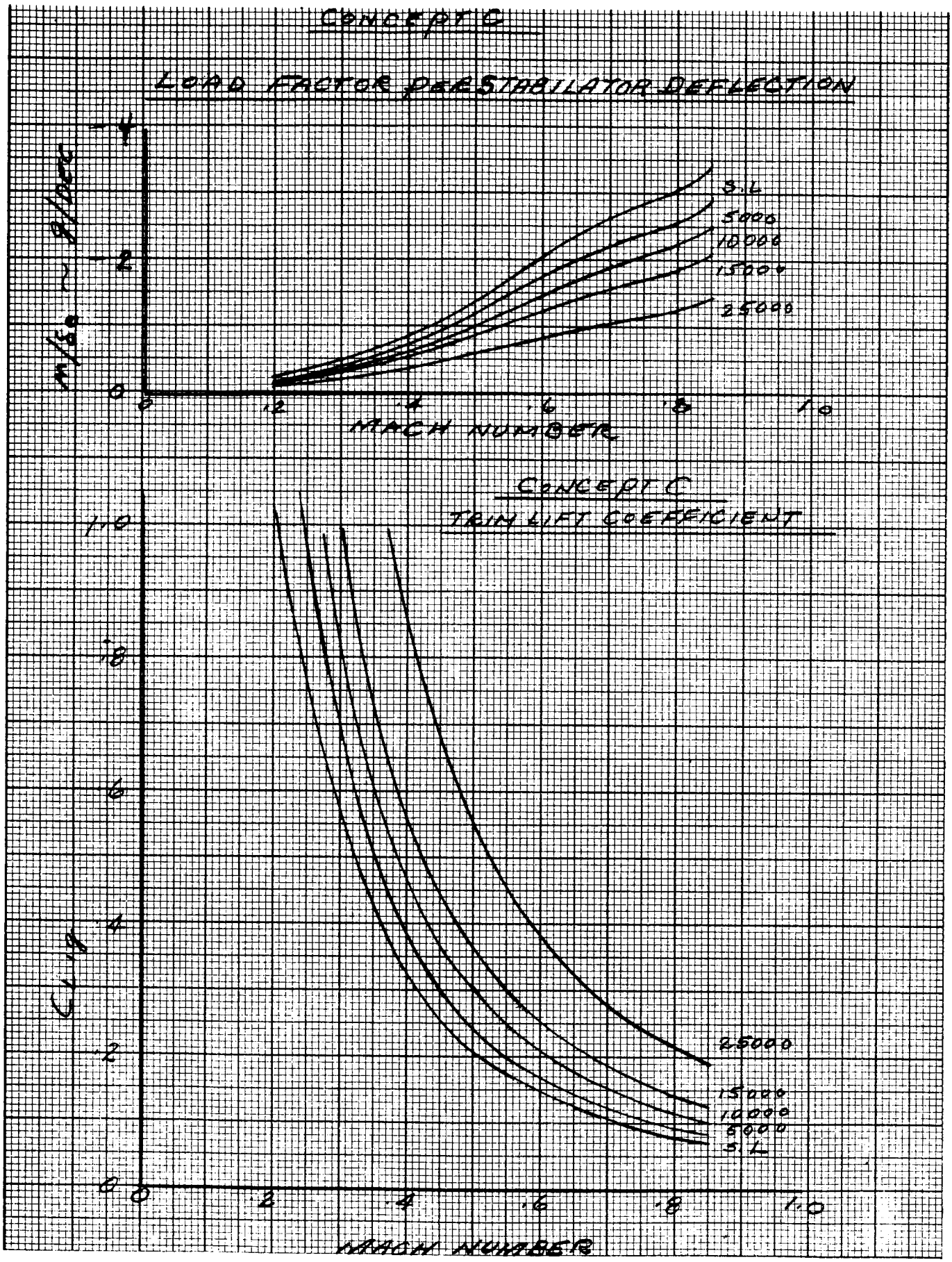

CONCEPT C
LOAD FACTOR PER STABILATOR DEFLECTION
m/δe ~ g/DEG
-4
-2
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
S.L
5000
10000
15000
25000
CONCEPT C
TRIM LIFT COEFFICIENT
CLIG
1.0
.8
.6
.4
.2
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
25000
15000
10000
5000
S.L

Figure VI-21

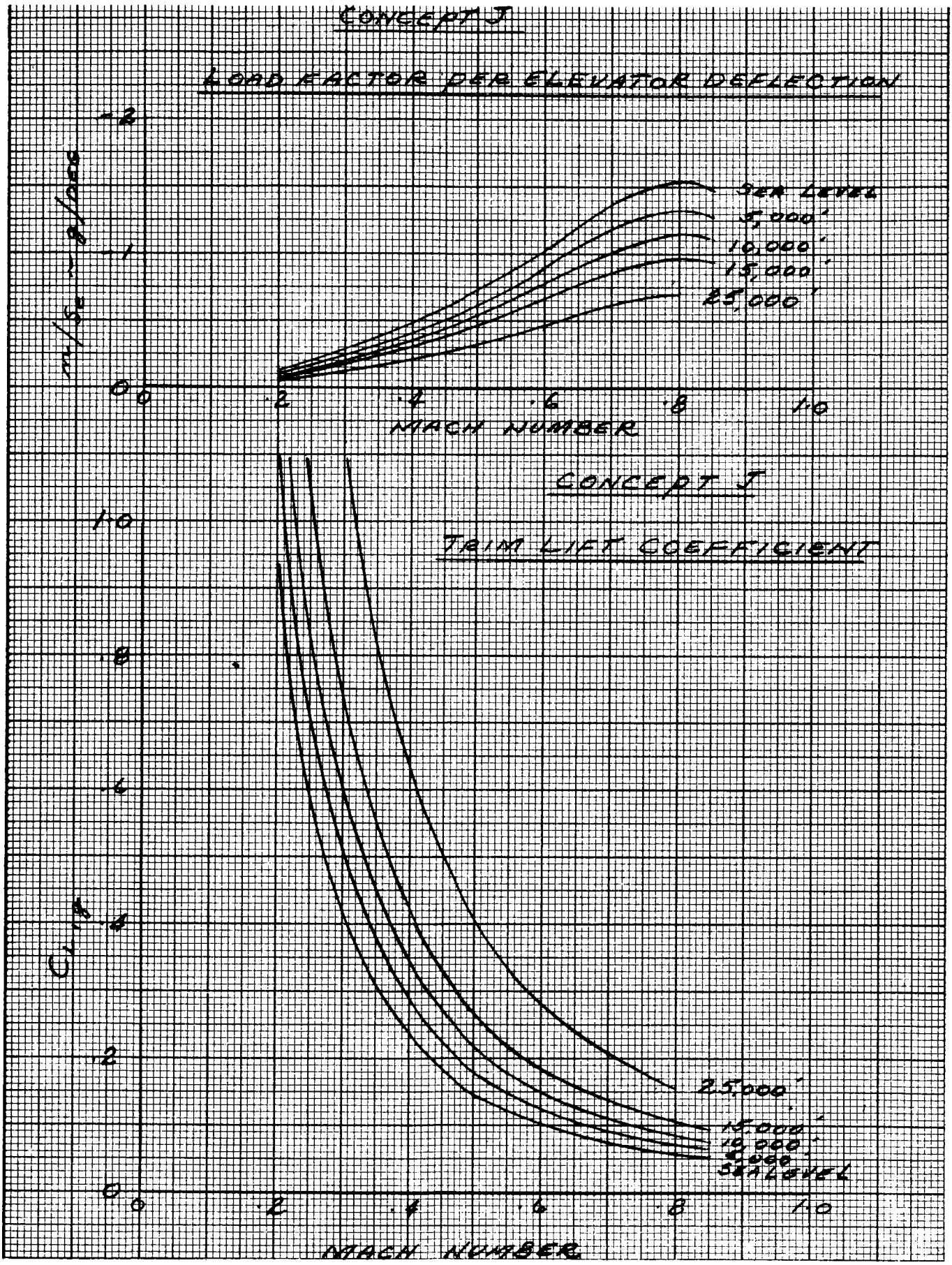

CONCEPT J
LOAD FACTOR PER ELEVATOR DEFLECTION
n/δe
-2
-1
0
SEA LEVEL
5,000'
10,000'
15,000'
25,000'
0
.2
.4
.6
.8
1.0
MACH NUMBER
CONCEPT J
TRIM LIFT COEFFICIENT
C_{L_T}
1.0
.8
.6
.4
.2
0
25,000'
15,000'
10,000'
5,000'
SEA LEVEL
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-22

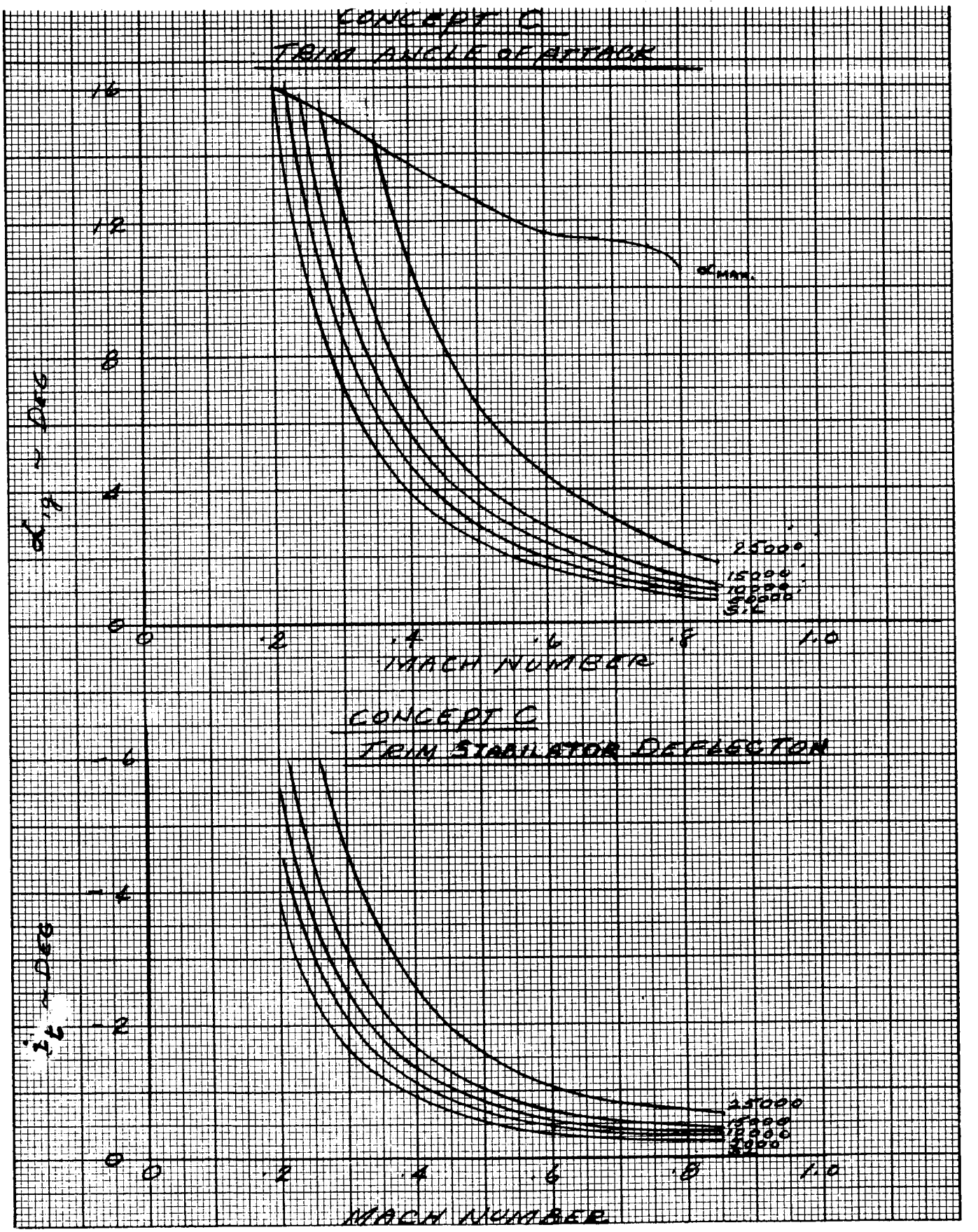

CONCEPT C
TRIM ANGLE OF ATTACK
16
12
8
4
0
$\alpha_{MAX.}$
25000
15000
10000
5000
S.L.
0
.2
.4
.6
.8
1.0
MACH NUMBER
CONCEPT C
TRIM STABILATOR DEFLECTON
-6
-4
-2
0
25000
15000
10000
5000
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-23

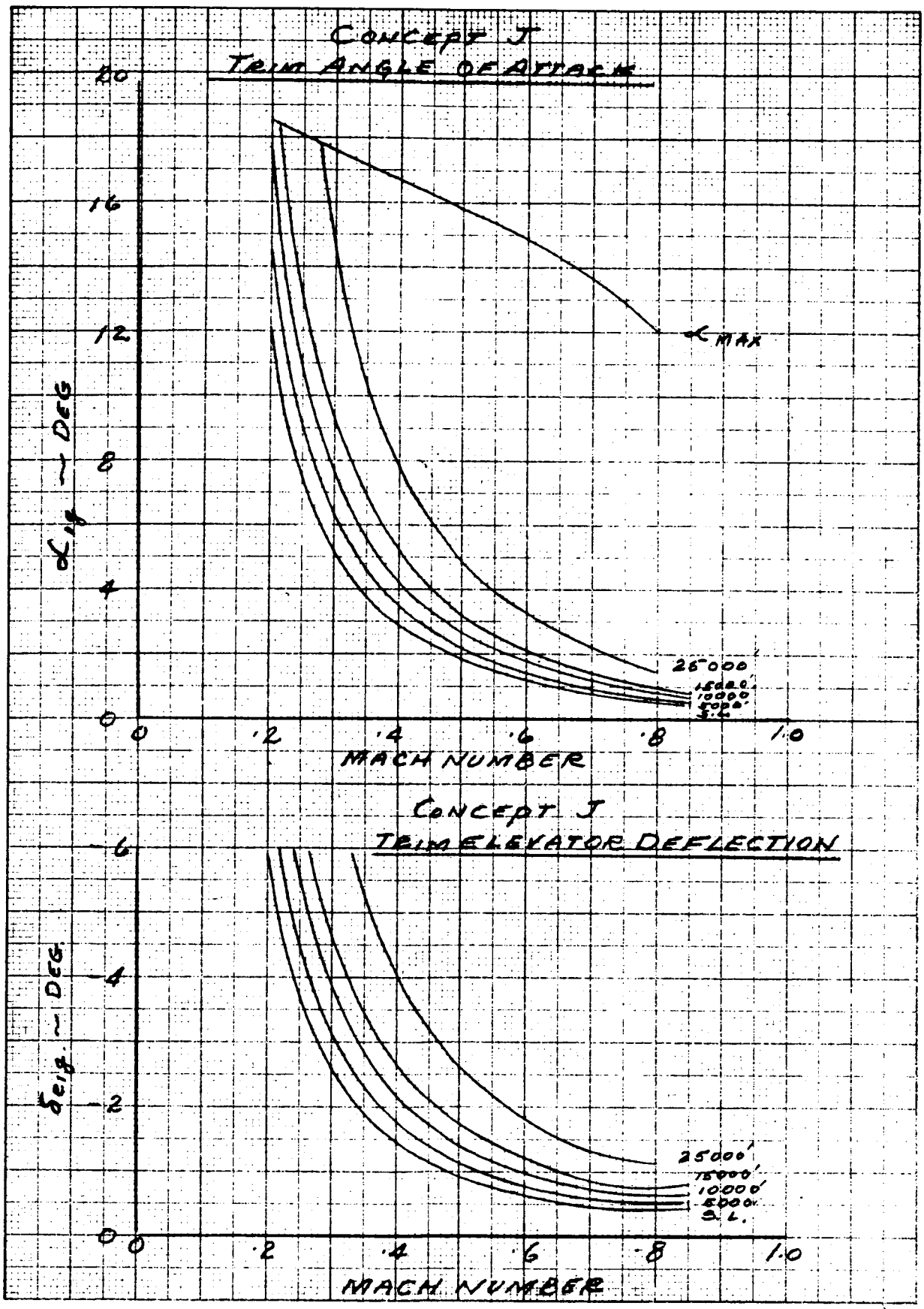

CONCEPT J
TRIM ANGLE OF ATTACK
20
16
12
8
4
0
α_{ig} ~ DEG
α_{MAX}
25000'
15000'
10000'
5000'
S.L.
0
.2
.4
.6
.8
1.0
MACH NUMBER
CONCEPT J
TRIM ELEVATOR DEFLECTION
-6
-4
-2
0
δ_{eig} ~ DEG.
25000'
15000'
10000'
5000'
S.L.
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-24

changes are made in flight under conditions representative of operational procedure. The conditions listed in Table VI-6 will suffice for determination of compliance with this requirement. With the airplane trimmed for each specified initial condition, the peak force required to maintain the specified constant parameter following the specified constant parameter following the specified configuration change will not exceed 10 pounds push or pull. This requirement applies to a time interval of at least five seconds following the completion of the pilot action initiating the configuration change. The magnitude and rate of trim change subsequent to this period is such that the forces are easily trimmable by the use of the normal trimming device.

TABLE VI-6. LONGITUDINAL TRIM CHANGE CONDITION

Case	Altitude	Initial Trim Condition			Power		Configuration Change	Parameter to be Held Constant
		Speed	Gear	Flaps	Cruise	Life		
1.	Low	$1.4\ V_{S_G}$	Up	Up	Normal	Off	Gear Down	Altitude
2.	Low	$1.4\ V_{S_G}$	Down	Up	Normal	Off	Flaps Down	Altitude
3.	Low	$1.4\ V_{S_L}$	Down	Down	Normal	Off	Idle Power	Speed
4.	Low	$1.5\ V_{S_L}$	Down	Down	Normal	Off	Takeoff Power	Altitude
5.	Low	$1.3\ V_{S_{TO}}$	Down	Take-off	Takeoff	Off	Gear Up	Rate of Climb
6.	Low	$1.5\ V_{S_{TO}}$	Up	Take-off	Takeoff	Off	Flaps Up	Rate of Climb
7.	Low	$1.5\ V_{con}$	Down	Down	Normal	Off	Open Lift Engine Doors	Altitude
8.	Low	V_{con}	Down	Down	Normal	Off	Lift Engine at Idle	Altitude
9.	Low	V_{con}	Down	Down	Normal	Idle	Divert Cruise Thrust to Lift	Altitude

To maintain constant speed in sideslips with the airplane trimmed for straight flight in each of the configurations specified in Table VI-2, the longitudinal control force required will not exceed numerically the lowest force which in the same configuration will produce a normal acceleration change of 1.0 g in the maneuvers in steady accelerated flight. In no event, however, will the force exceed 10 pounds of pull or 3 pounds of push. The sideslip considered will include angles up to the largest obtainable with 50 pounds of rudder pedal force applied in either direction from wings-level trimmed flight.

c. Directional Control Effectiveness

(1) Directional Control (Symmetric Power)

Directional control is sufficiently effective to maintain wings-level straight flight in the configurations and speed ranges specified in Table VI-2 with rudder control forces not greater than 180 pounds when the airplane is trimmed directionally at the trim speeds specified in Table VI-2. In the wave-off configuration (WO) at the lightest normal loading, directional control is sufficiently effective to maintain wings-level flight at all speeds down to $V_{S_{PA}}$. In the power approach configuration (PA) at $1.15V_{S_L}$, the rudder control force does not exceed 100 pounds when trimmed.

The rudder control derivatives are presented in Figure VI-25 (Concept C) and Figure VI-26 (Concept J).

(2) Directional Control (Asymmetric Power)

In the take-off configuration (TO) with the most critical outboard engine (left cruise engine) inoperative (with RPM simulating failure in flight), it is possible at the lightest normal take-off loading and with take-off power on the remaining cruise engine to achieve and maintain straight flight with a bank angle not greater than 5 degrees, at all speeds above $1.2\ V_{S_{TO}}$. With trim settings normally employed in a symmetric power takeoff, the rudder pedal force required to maintain straight flight with asymmetric power does not exceed 180 pounds.

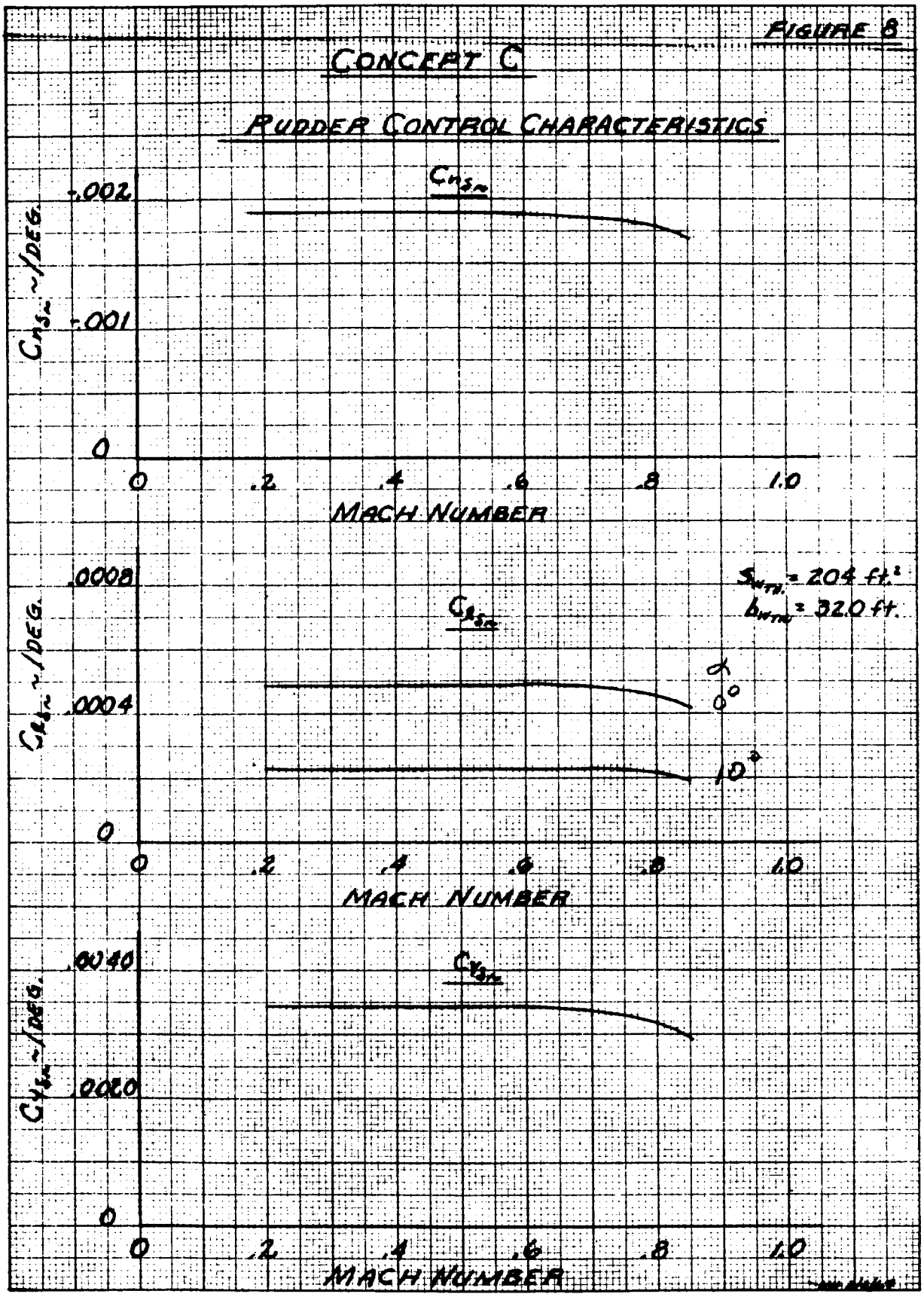


FIGURE 8
CONCEPT C
RUDDER CONTROL CHARACTERISTICS
$C_{n_{\delta_r}}$
$C_{n_{\delta_r}}$ ~/DEG.
.002
.001
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
$S_{W_{TH}}$ = 204 ft.²
$b_{W_{TH}}$ = 32.0 ft.
$C_{l_{\delta_r}}$
$C_{l_{\delta_r}}$ ~/DEG.
.0008
.0004
0
α
0°
10°
MACH NUMBER
$C_{Y_{\delta_r}}$
$C_{Y_{\delta_r}}$ ~/DEG.
.0040
.0020
0
MACH NUMBER

Figure VI-25

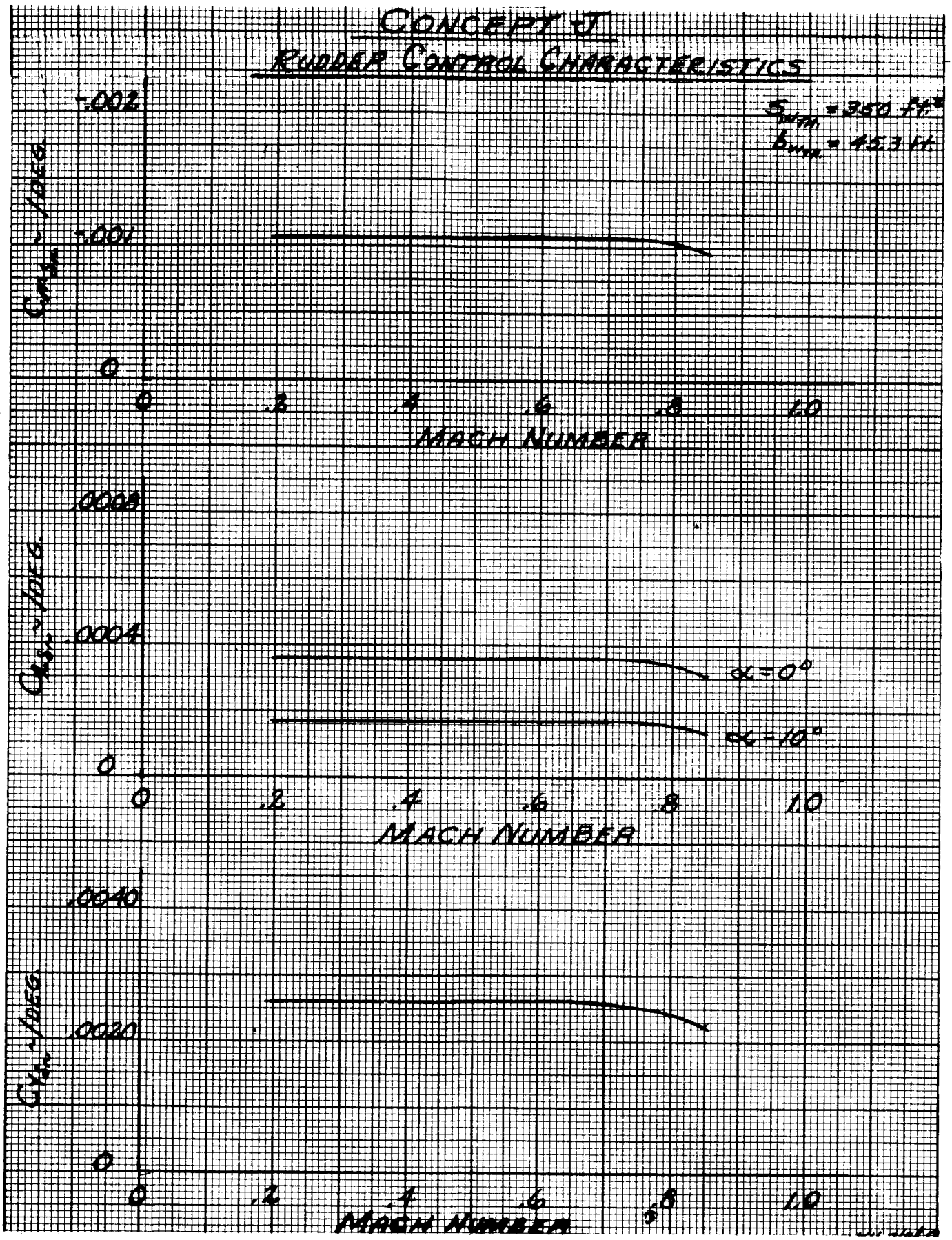

CONCEPT J
RUDDER CONTROL CHARACTERISTICS
$S_{wing} = 300$ ft²
$b_{wing} = 45.3$ ft
$C_{N_{\delta_r}}$ /DEG
-.002
-.001
0
0
.2
.4
.6
.8
1.0
MACH NUMBER
$C_{m_{\delta_r}}$ /DEG
.0008
.0004
0
$\alpha = 0°$
$\alpha = 10°$
0
.2
.4
.6
.8
1.0
MACH NUMBER
$C_{Y_{\delta_r}}$ /DEG
.0040
.0020
0
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-26

(3) Directional Control During Takeoff and Landing

The rudder control, in conjunction with the stabilator, aileron, and spoiler controls, is adequate to maintain straight paths on the ground during normal takeoffs and landings in calm air and in 90-degree cross winds of at least 30 percent of V_{S_L} or 40 knots, whichever is less. This requirement is met with not more than 180 pounds of pedal force.

d. Lateral Control Effectiveness

Lateral control is adequate for compliance with the rolling performance specified in Table VI-7. The steady state rolling performance is presented in Figure VI-27 for Concept C and in Figure VI-28 for Concept J. In cases where the flight conditions coincide, the highest rolling requirements for pb/2V of 0.07 is met by the aileron-spoiler performance of Concept C as shown by Table VI-7 and Figure VI-27. Concept J does not meet this requirement as shown by Figure VI-28 because the outboard ailerons are inadequate. The elapsed time listed in Table VI-7 is measured from the instant of initiation of pilot control action.

TABLE VI-7. MINIMUM ROLLING PERFORMANCE REQUIREMENTS

Configuration	Speed and Altitude	Rolling Performance
Clean (P)	S. L. and 25,000 ft.	$\frac{pb}{2V} = 0.07$ above $1.1V_{S_G}$
	7,500 ft.	Bank angle = 90 degrees in 1 second at V_H
	20,000 ft to 25,000 ft.	Bank angle = 90 degrees in 1 second from $V_{R/C\ max}$ to V_H
Landing (L) Power Approval (PA)	S.L. at 1.1 V_{S_L}	Average $\frac{pb}{2V} = 0.05$ for first 30 degrees of bank based on elapse time from initiation.

For specifications in terms of peak pb/2V, the rate of roll is limited to 220 deg/sec (3.84 rad/sec). Concept C approaches this value at sea

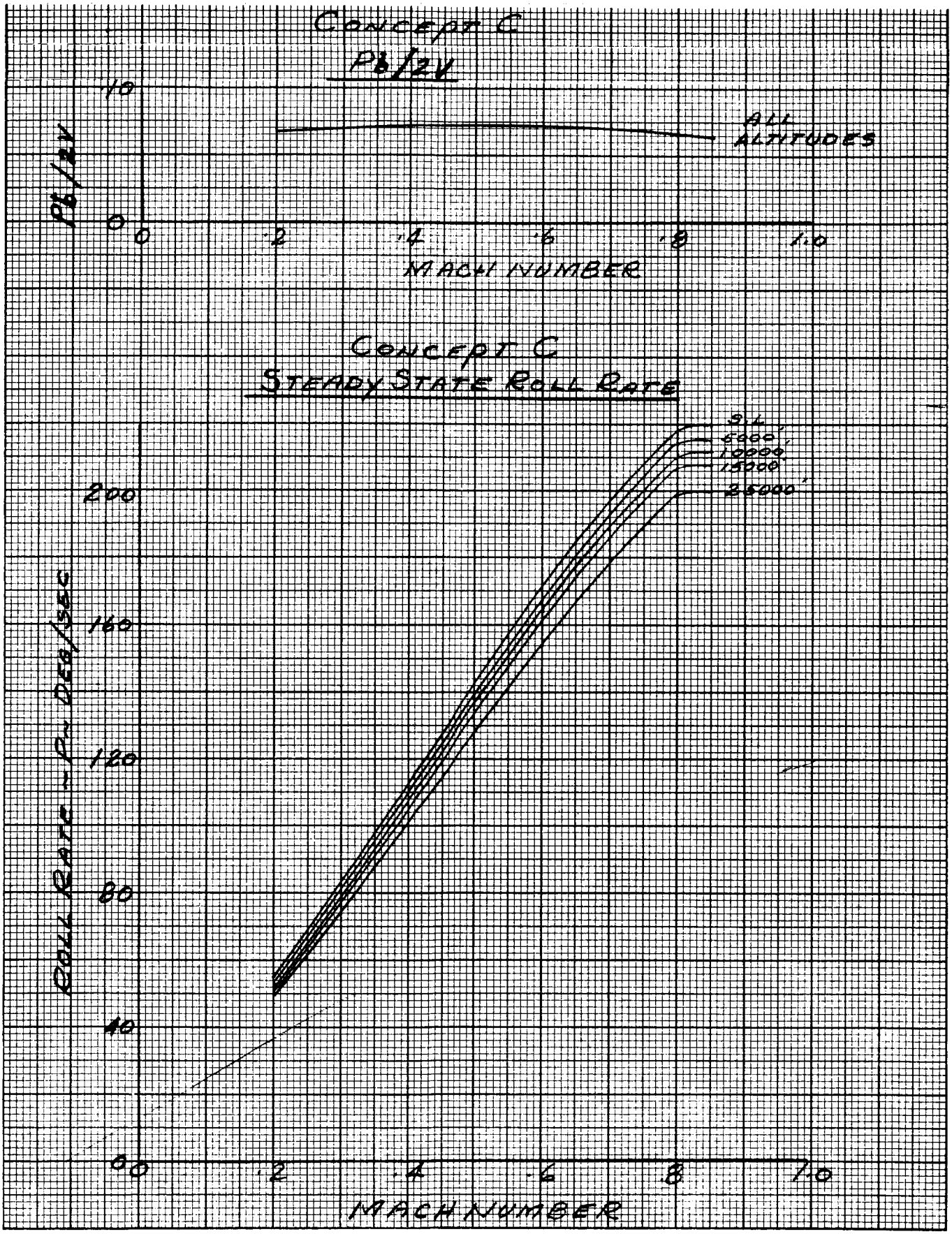

CONCEPT C
Pb/2V
ALL ALTITUDES
Pb/2V
MACH NUMBER
CONCEPT C
STEADY STATE ROLL RATE
SL
5000'
10000'
15000'
25000'
ROLL RATE - P~ DEG/SEC
MACH NUMBER

Figure VI-27

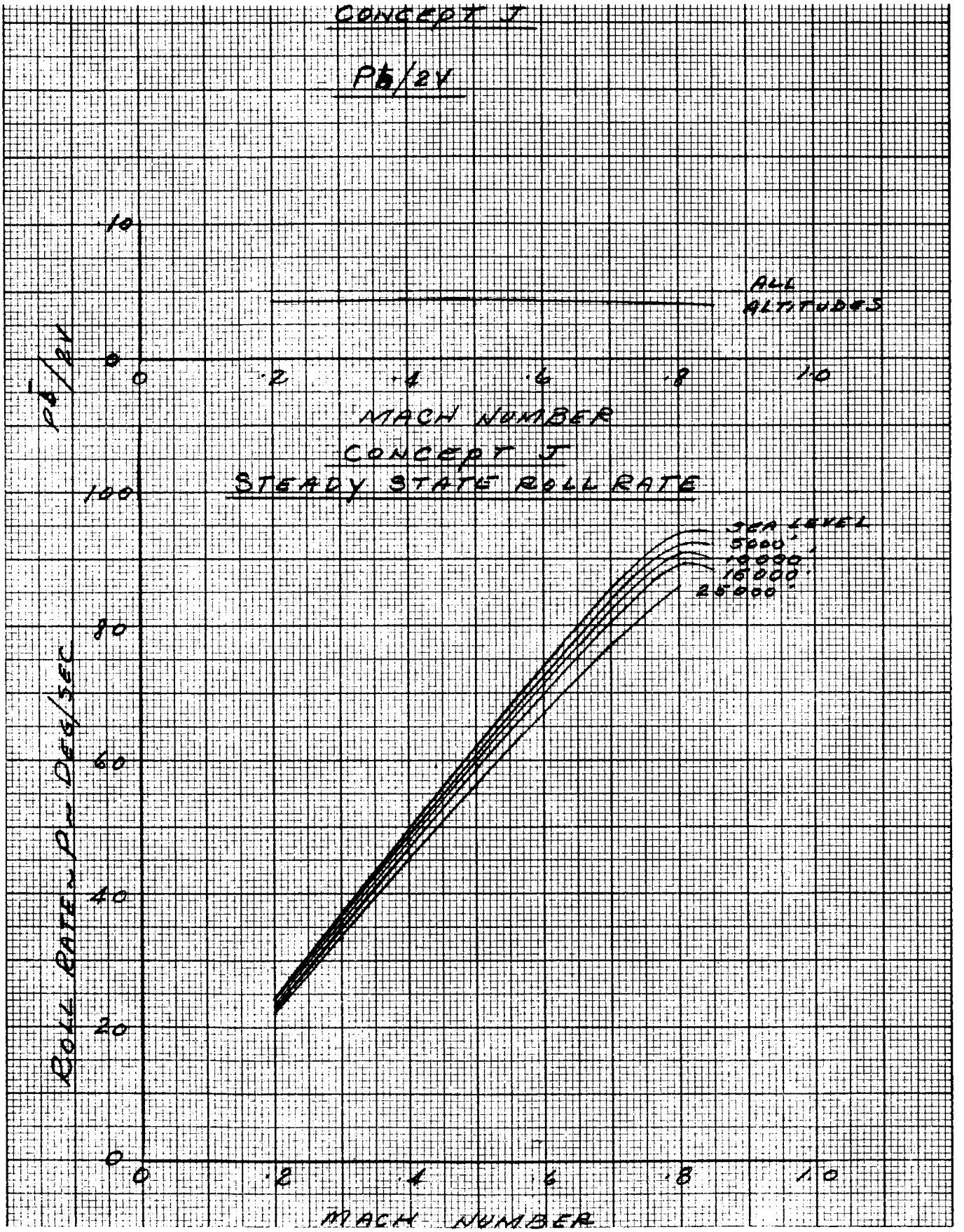

CONCEPT J
Pb/2V
.10
0
ALL ALTITUDES
0
.2
.4
.6
.8
1.0
MACH NUMBER
CONCEPT J
STEADY STATE ROLL RATE
100
SEA LEVEL
5000
10000
15000
25000
80
60
40
20
0
ROLL RATE ~ P ~ DEG/SEC
0
.2
.4
.6
.8
1.0
MACH NUMBER

Figure VI-28

level, Mach 0.80 as shown by Figure VI-29 for the unaugmented aircraft in conventional flight. The flying qualities requirements for Navy and Air Force aircraft are shown in the form of pilot opinion boundaries and were obtained from NATO AGARD report R-336. The roll characteristics for Concept C lie in the good to acceptable areas for all conditions in the useable flight range. The rudder pedals remain fixed in the position required for steady flight prior to the roll.

The maximum steady-state roll rate for Concept J is similarly shown on Figure VI-30. The roll rates are unacceptably low in the low-speed range and become acceptable in the high-speed range. Higher aerodynamic roll power is required for this concept.

The peak lateral control force required to obtain the rolling performance specified in Table VI-7 does not exceed 20 pounds of stick force. At 0.8 V_H, the peak lateral control force required to obtain the rolling performance specified in Table VI-7 is not less than 10 pounds of stick force.

Lateral control power is sufficiently effective to balance the airplane laterally under asymmetric power (rudder free) and directional control (symmetric and asymmetric power).

There are no objectionable nonlinearities in the variation of rolling response with lateral deflection or force causing sensitivity or sluggishness in response to small cockpit deflections or force.

e. General Control and Trimmability

(1) Control for Spin Recovery

In configurations glide (G) and landing (L), the normal control will be adequate to provide consistant prompt recoveries from fully developed erect and inverted spins. Recovery characteristics will be adequate to permit demonstration and will be shown to present no abnormal effort on the part of the pilot, and recovery control forces will be shown not to exceed 250 pounds (rudder), 75 pounds (stabilator), or 35 pounds (aileron and spoilers).

CONCEPT C

ISO OPINION BOUNDARIES OF ROLL CHARACTERISTICS, AIRFRAME ALONE

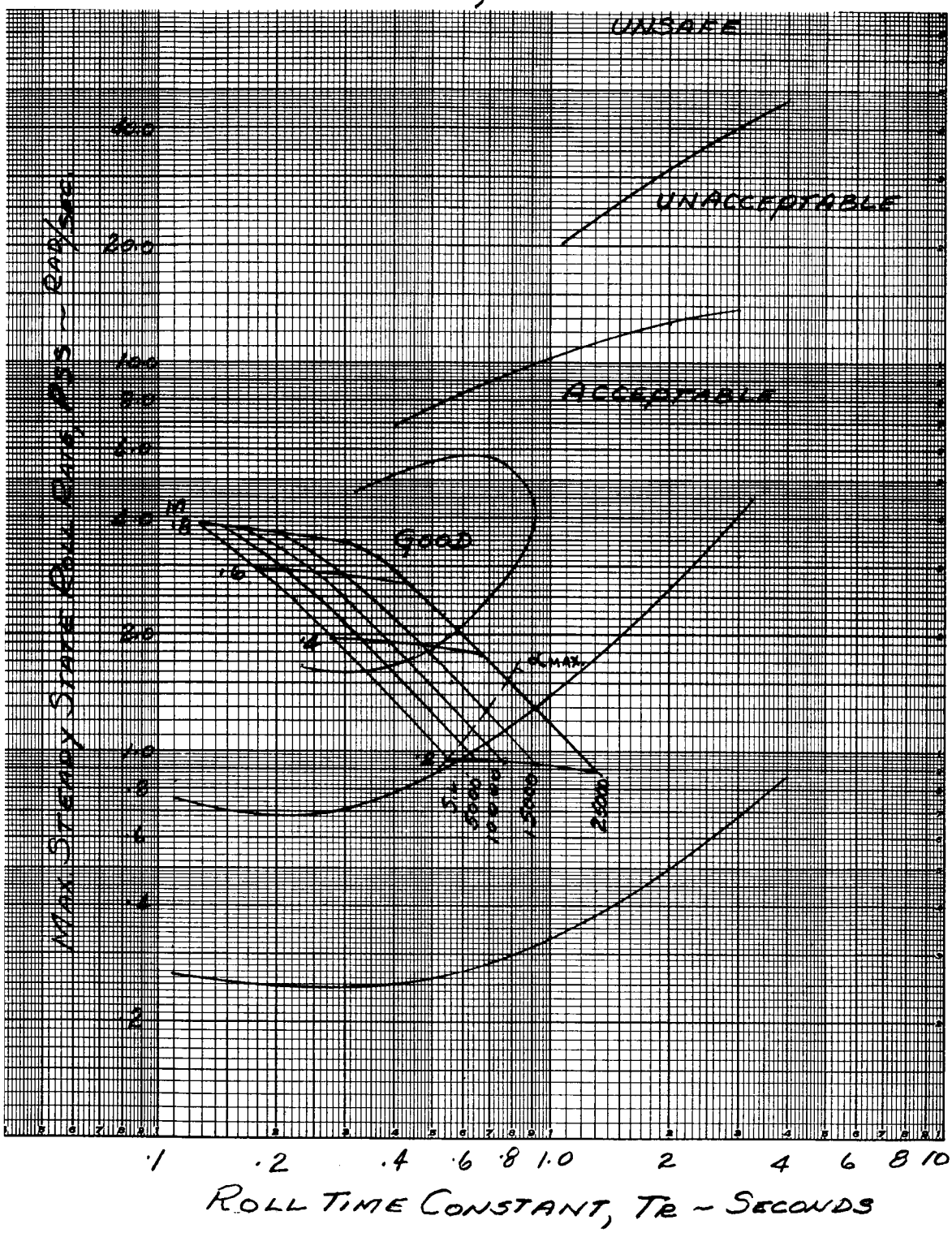


Figure VI-29

CONCEPT J

ISO - OPINION BOUNDARIES OF ROLL CHARACTERISTICS, AIRFRAME ALONE

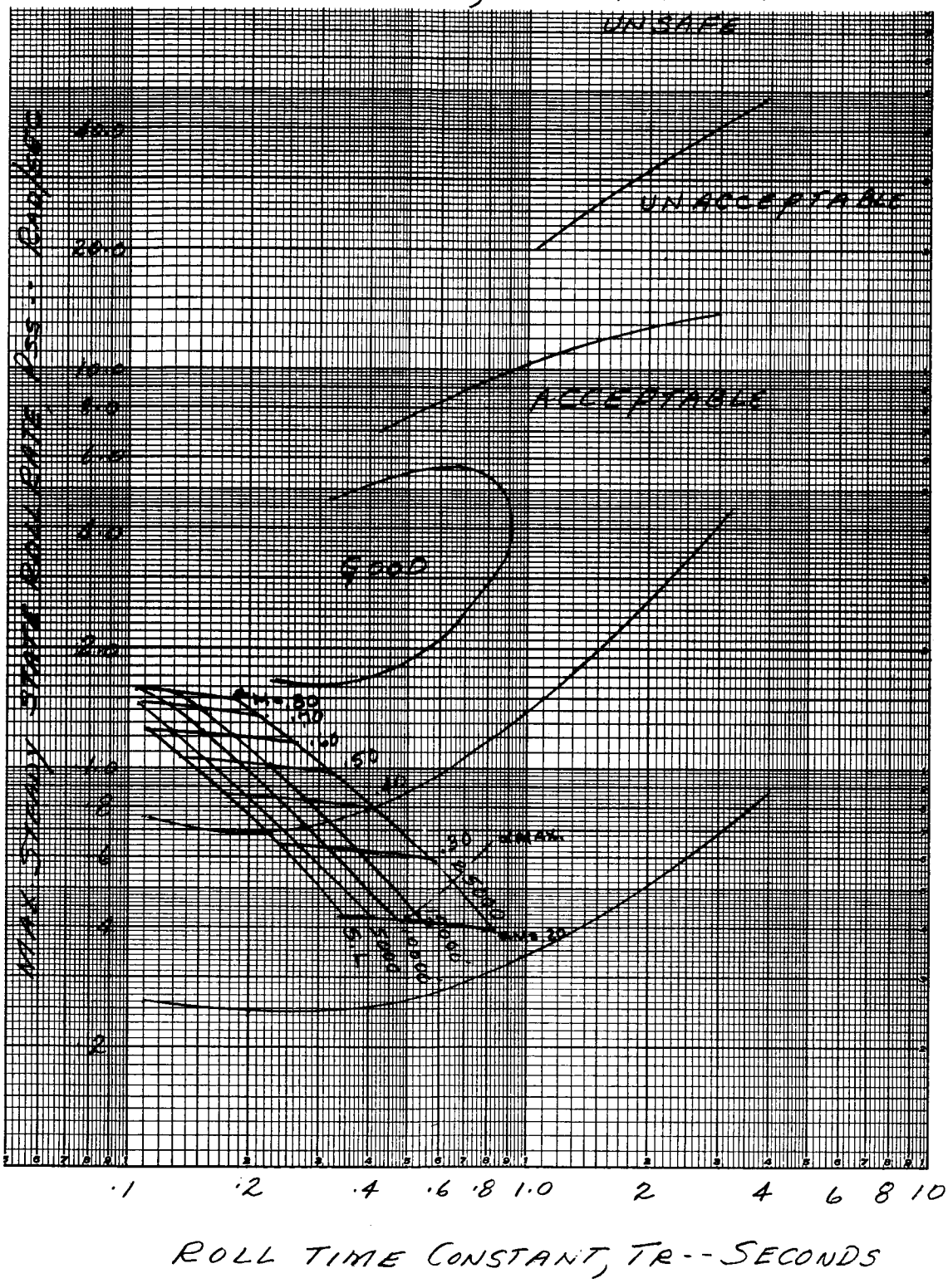


Figure VI-30

(2) Control for Taxiing

All normal taxiing operations can be performed without undue pilot effort or inconvenience.

(3) Primary Flight Control Trimmability

The trimming devices are capable of reducing the stabilator, rudder, and aileron control forces to zero at all speeds between the minimum trim speeds specified in Table VI-8 and the upper limits of the speed ranges specified in Table VI-2.

TABLE VI-8. CONDITION FOR TRIMMING TO ZERO CONTROL FORCES AT FORWARD AND AFT CRITICAL LOADING

Configuration	Maximum Trim Speed
Clean (P)	$1.2\ V_{S_{CR}}$
Landing (L)	$1.4\ V_{S_L}$
Power Approach (PA)	$1.4\ V_{S_L}$
Cruise (CR) with left cruise engine inoperative, wings level	Speed for Max. Range

(4) Trim System Failure

Failure of the power-actuated trim system (including sticking or runaway in either direction) will not result in an unsafe flight condition. Following such failure, it is possible to cruise for extended periods and to make a safe landing.

(5) Roll-Pitch-Yaw Coupling

In rudder-and-stabilator-cockpit-control-fixed rolls through 360 degrees at all altitudes and permissible speeds, entered from straight flight or from turns, push-overs, or pull-ups ranging from zero g to 2/3 N_L (limit load factor), the resulting yaw motion, sideslip angle, and normal acceleration will not exceed

structural limits nor cause dangerous flight conditions such as uncontrollable oscillations. This condition is covered in Paragraph VI. C. 3.

4. Methods of Obtaining Control

a. Attitude Control Devices

The aircraft control devices are the conventional control surfaces, the jet reaction devices, the engine throttle control, and the engine thrust vectoring control. Independent dual power-operated controls are provided on all axes. Positive stops are provided in the control system to prevent movement of the controls beyond safe aerodynamic or engine compressor bleed limits. The power control system is capable of continuous operation under all positive and negative loading conditions within the design flight envelope of the aircraft.

(1) Attitude Control: V/STOL Mode

(a) Normal Control Response Characteristics

For the V/STOL flight conditions, the handling qualities recommendations of AGARD Report 408 are considered as requirements except that the response characteristics for full control input will be increased.

TABLE VI-9. NORMAL CONTROL RESPONSE CHARACTERISTICS

Item	Control Axis		
	Pitch	Yaw	Roll
NASA Response Factor	1.5	1.5	2.0
AGARD Response (degrees in first second)	$\frac{300.0}{(W+1000.0)^{1/3}}$	$\frac{180.0}{(W+1000.0)^{1/3}}$	$\frac{300.0}{(W+1000.0)^{1/3}}$
NASA Response (degrees in first second)	$\frac{450.0}{(W+1000.0)^{1/3}}$	$\frac{270.0}{(W+1000.0)^{1/3}}$	$\frac{600.0}{(W+1000.0)^{1/3}}$

The NASA level of damping augmentation is that required to provide the minimum characteristics recommended in AGARD Report 408 for normal operating conditions. These are presented in Section VI-B. 1. The control-power-to-response ratio is the same for AGARD and NASA.

$$\frac{\ddot{\theta}}{\theta} = \frac{1}{\frac{\zeta}{C}\left\{t - \frac{\zeta}{C}\left(1 - e^{-t\,C/\zeta}\right)\right\}}$$

where $\ddot{\theta}$ = control power - rad/sec^2

θ = response - rad.

ζ = moment of inertia - slug ft^2

C = damping - (ft. lb.)/(rad/sec)

(therefore ζ/C has the units of seconds)

t = elapsed time minus time lag - seconds.

The critical axis is that axis requiring the maximum bleed air supply to satisfy the single-axis normal control response characteristics. The higher inertia in pitch results in pitch control power establishing the critical axis. This is the control thrust required shown by point C in Figure III-59 for Concept C and in Figure III-60 for Concept J.

(b) Simultaneous Control Displacement

With all controls displaced fully about all axes, the control power about any axis will not be less than 60 percent of the maximum available about that axis. This is shown as point D in Figure III-59 for Concept C and Figure III-60 for Concept J.

(c) Auxiliary Control Power Source

The attitude control power for operation in the V/STOL mode (where conventional aerodynamic controls are ineffective) is derived from compression bleed air from the normal YJ85-GE-19 engine operation which is ducted to reaction control nozzles. The control power required for the AGARD report 408 response (except 1.5 minimum in roll) is supplied by the normal propulsion units. The control power required above these minimum may be supplied by an auxiliary control power source.

As the total control power for controls fully displaced on all axes is similar to the requirements for simultaneous control displacement and the total minimum control power available is greater than the control power required

on any single axis, a special or auxiliary control power source need not be considered.

(2) Attitude Control: Conventional Flight Mode

(a) Pitch Control

Attitude control in pitch is provided by an all-movable stabilator on Concept C and a trimmable horizontal tail and an elevator on Concept J.

(b) Yaw Control

Attitude control in yaw is provided by a vertical tail with an adjustable fin and a rudder.

(c) Roll Control

Attitude control in roll is provided by ailerons and spoilers on Concept C and ailerons above on Concept J.

b. Lift Control Devices

(1) Lift Control: V/STOL Mode

The research aircraft utilizes the mixed propulsion system (lift-only plus lift-cruise engines). With this system it is also possible to study the pure lift mode arrangement with separate lift and cruise engines where the cruise engine is not deflected downward for low speed flight. The aircraft is designed for the composite (mixed) propulsion system mode of operation with design provisions for later field conversion to the pure lift mode. Structure and service provisions are provided for the pure lift mode.

In line with the general philosophy of using fully qualified hardware to the maximum extent feasible for minimum cost and delivery time, engine selection and ratings are restricted to those flight rated for at least 150 hours for the lift-cruise engines and 25 hours for the lift-engines, or to those fully-funded to achieve these ratings within the time period required. Two engines were selected by NASA for Part I and II; these were the Rolls Royce RB 162-81 and the General Electric YJ85-GE-19. The final designs use only the YJ85-GE-19 engine.

(a) Exhaust Flow Vectoring

The lift engines include vectoring for directing the exhaust from 20 degrees forward to 30 degrees rearward of the vertical. The vectoring requirements assume thrust recovery factors similar to those expected from a spherical vectoring nozzle. Important considerations in this thrust vectoring are jet-induced lift losses during transition and longitudinal trim and stability throughout the flight range.

(b) Jet-Induced Lift Losses in Hover

The lift losses in hover both in and out of ground effect are presented in Figure VI-31 for Concept C and Concept J.* Concept C is the 8 (J-85) engine version and Concept J is the 10 (J-85) engine version. The losses in ground effect at the landing gear extend height and out of ground effect are tabulated.

TABLE VI-10. BASE LIFT LOSS

Concept	Base Lift Loss/Total Lift Thrust	
	In Ground	Out of Ground
C	0.095	0.015
J	0.150	0.027

(c) Jet-Induced Lift Losses and Moments in Transition

During acceleration from hover, the jets cause a pitching moment and an additional lift loss.* Figure VI-32 presents these data for Concept C and Figure VI-33 presents these data for Concept J. These interference effects are small for dynamic pressure square root ratios of less than 0.12, when the square root ratio is:

$$Q_{RT} = \sqrt{\frac{\rho_\infty V_\infty^2}{\rho_{jet} V_{jet}^2}}$$

The effects of flaps and the effects of angle-of-attack below 10 degrees are small.

* Data presented here are based on the results correlation by Fairchild Hiller, Republic Aviation Division, in a limited distribution report: Part II - Technical Data Substantiation Report - V/STOL Jet Operations Research Airplane Design Study, Report No. FHR 3391-3I, 7 April 1967.

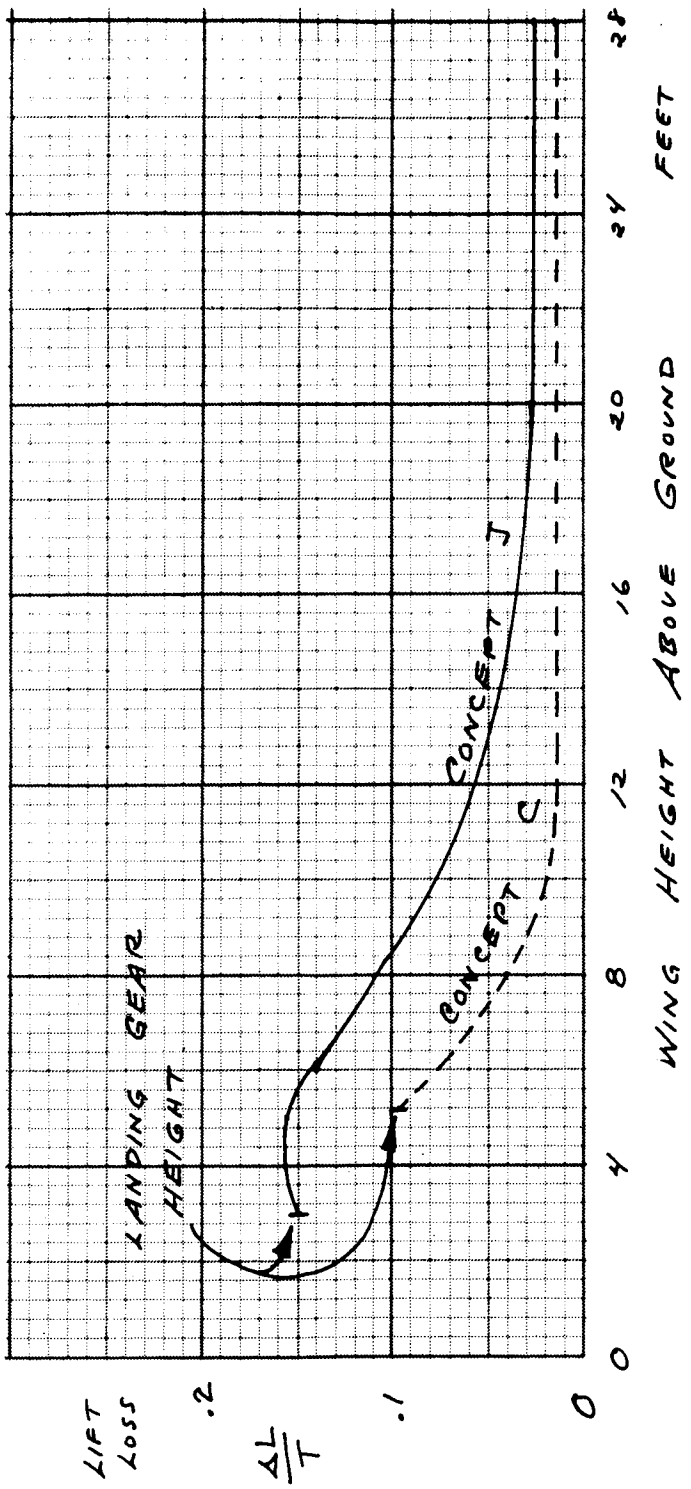


Figure VI-31. Lift Loss Due to Ground Effects

JET INTERFERENCE EFFECTS

CONCEPT C

CONFIGURATION

LIFT	6	J-85
CRUISE	2	J-85

$S_\gamma = 90°$

HOVER BASE LOSSES NEGLECTED

HORIZONTAL TAIL OFF

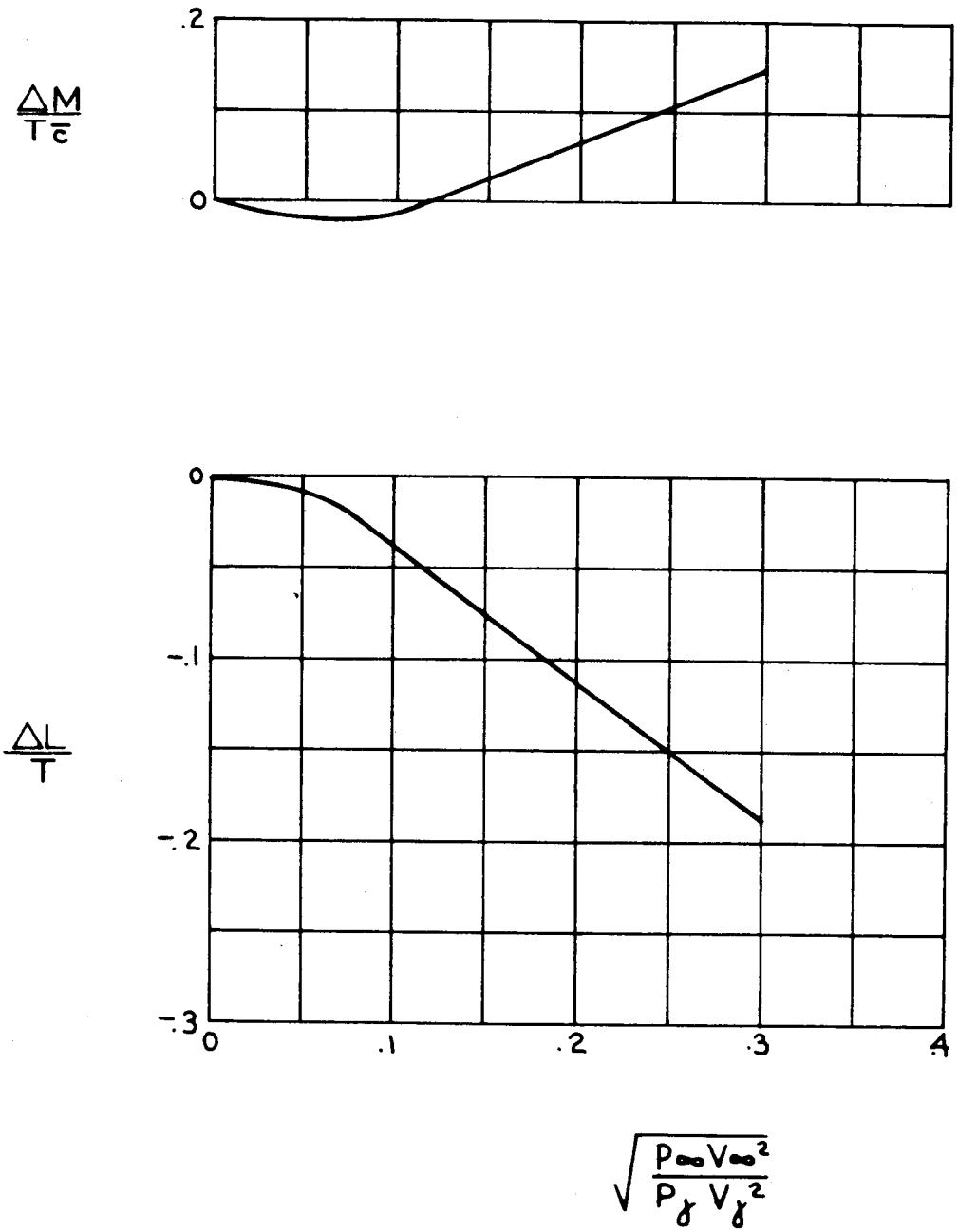


Figure VI-32

JET INTERFERENCE EFFECTS

CONCEPT J

CONFIGURATION

LIFT	8	J-85
CRUISE	2	J-85

$S_\gamma = 90°$

HOVER BASE LOSSES NEGLECTED

HORIZONTAL TAIL OFF

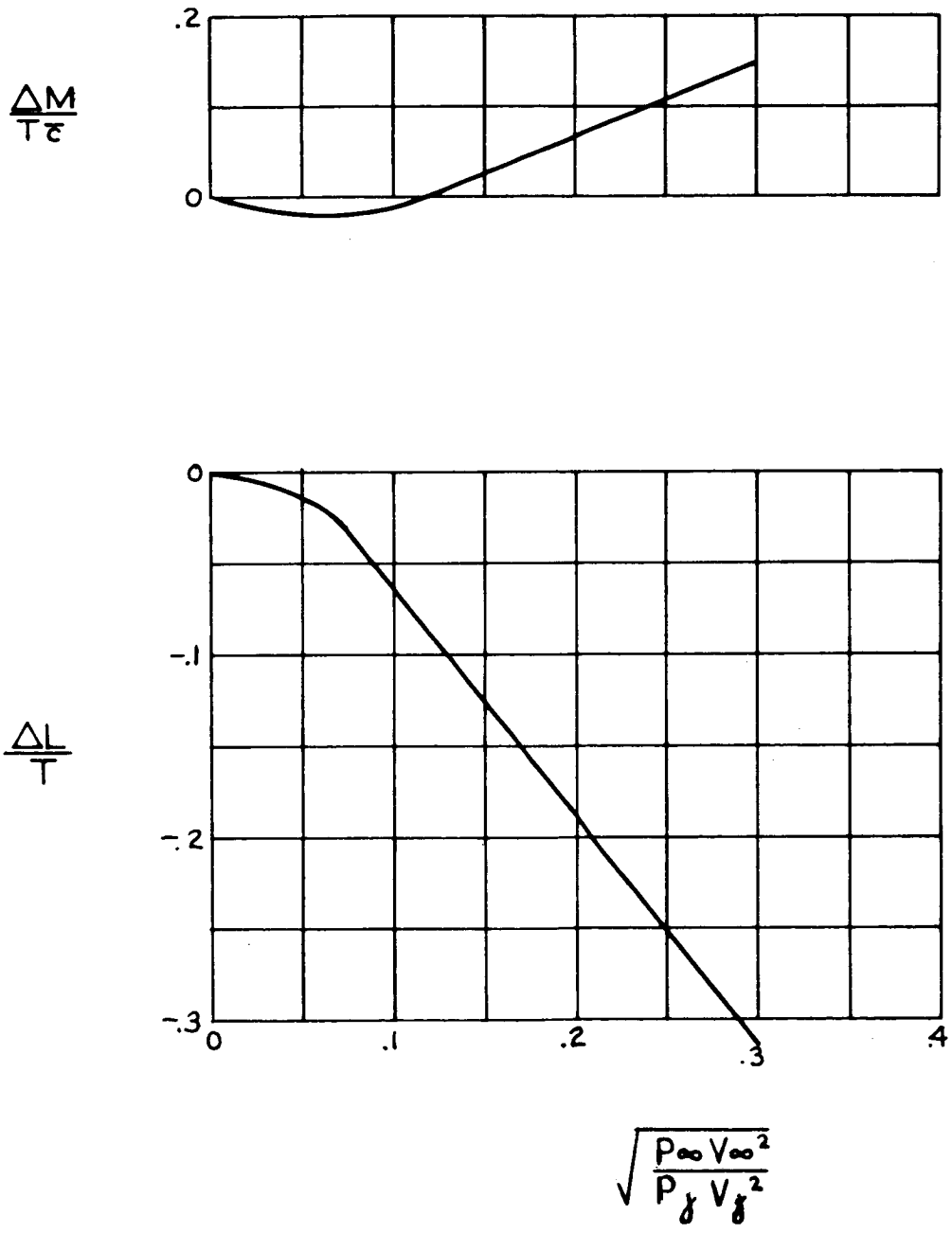


Figure VI-33

(d) Jet-Induced Rolling Moments in Sideslip

During acceleration from hover, the increment in the effective dihedral parameter C_{ℓ_β} has been estimated and presented in Figure VI-34.* The values for Concept C are presented by the solid lines. The effect of low wings increases the induced rolling moments as shown by the dotted lines for Concept J.

(2) Lift-Control: Conventional Flight Mode

The effects of lift control by Trailing Edge Flaps, Leading Edge Flaps, and Spoilers are detailed in Section III-C.2.

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* Data presented here are based on the results correlation by Fairchild Hiller, Republic Aviation Division, in a limited distribution report: Part II - Technical Data Substantiation Report - V/STOL Jet Operations Research Airplane Design Study, Report No. FHR 3391-3I, 7 April 1967.

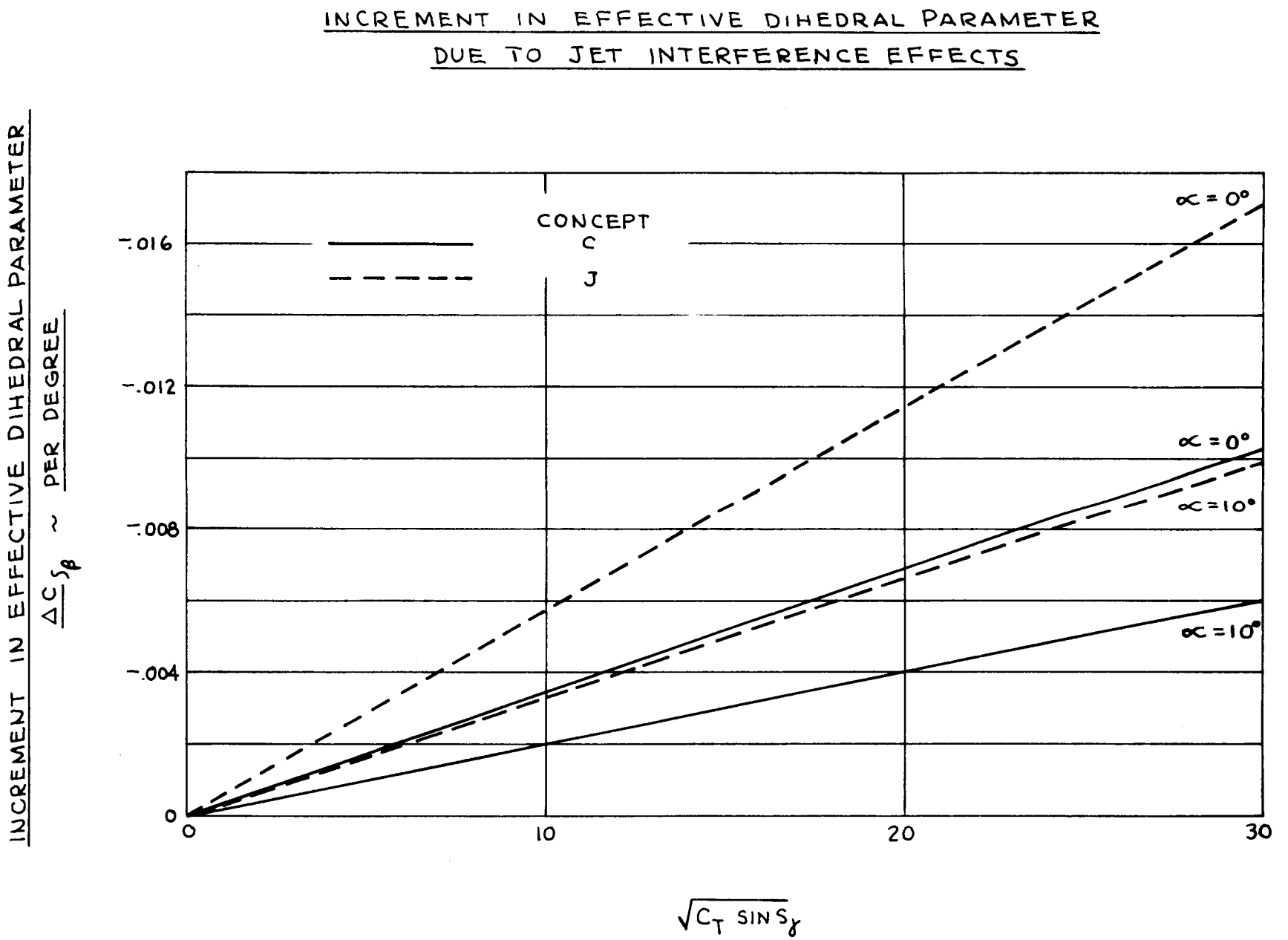

INCREMENT IN EFFECTIVE DIHEDRAL PARAMETER
DUE TO JET INTERFERENCE EFFECTS
INCREMENT IN EFFECTIVE DIHEDRAL PARAMETER
ΔC_{ζ_β} ~ PER DEGREE
CONCEPT
C
J
$\alpha = 0°$
$\alpha = 0°$
$\alpha = 10°$
$\alpha = 10°$
-.016
-.012
-.008
-.004
0
0
10
20
30
$\sqrt{C_T \sin S_\gamma}$

Figure VI-34

B. FLYING QUALITIES

1. Stability Augmentation System (SAS)

A dual stability augmentation system is installed in the aircraft to provide additional rate damping about all three axes. The level of damping augmentation is at least that required to provide the minimum characteristics as specified in AGARD Report 408.

TABLE VI-11. DAMPING CHARACTERISTICS

	Damping (lb/ft (rad/sec))		
Conditions	Pitch	Yaw	Roll
Normal Operation	$15\ (I_y)^{0.7}$	$27\ (I_z)^{0.7}$	$25\ (I_x)^{0.7}$
Emergency Operation	$8\ (I_y)^{0.7}$	$14\ (I_z)^{0.7}$	$18\ (I_x)^{0.7}$

The pilots are able to disengage the SAS. In normal operation, the stability augmentation system will be operating any time the evaluation pilot's variable stability equipment is disengaged.

a. System Description

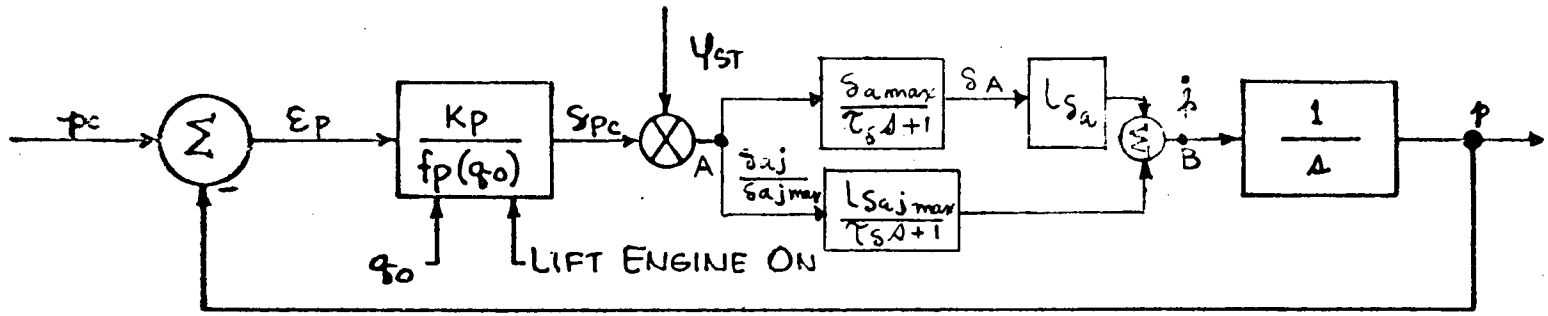
The SAS for the aircraft will provide artificial damping for each of the angular control axes. Rate gyros will measure angular rates about the body axes. The gyro outputs will go through an amplifier whose gain varies as a function of dynamic pressure, q_o. The amplifier outputs are fed to series actuators which superimpose additional surface and jet nozzle motions on top of those commanded by the parallel actuator, if the evaluation pilot is controlling, or by the direct mechanical linkage, if the safety pilot is flying. These series actuators will have limited authority and inputs to them will not be mechanically reflected to the pilot's stick.

In VTOL, the SAS must provide all of the damping while, in conventional flight, it must supplement the natural damping characteristics of the aircraft to ensure satisfactory handling qualities over the flight envelope. By

making the rate damping gain variable as a function of dynamic pressure, the loop dynamics remain constant for most flight conditions. Figure VI-35 shows a block diagram of the roll control loop. For the VSS mode, the model roll rate command, p_c, activates the loop and the stick command input, φ_{ST}, is open. In the SAS mode, φ_{ST} drives the loop and p_c is open. The $\dot{p}/\varphi_{ST}$ transfer function of Figure VI-35 contains the jet and surface roll acceleration contribution as well as the aileron actuator dynamics which is represented by a first order lag. The numerator of this transfer function is a function of dynamic pressure, q_o.

The $f_p(q_o)$ term in the loop is provided to cancel this variable gain and provide a constant loop gain, K_p, for most flight conditions. This variable gain feature is discussed in detail under the VSS section of this report. A lower value of loop gain, K_p, is necessary for the SAS than in the VSS mode since for this mode, K_p is in the feedback path and a high value of K_p therefore tends to slow down the response to a stick input.

For the proposed control loop, K_p is equal to the damping term defined in AGARD 408 divided by the roll axis moment of inertia. The minimum value of K_p is therefore specified by AGARD 408. By using this value of K_p, therefore, the minimum one second response requirement of 23.6 degrees will be met. By similarly choosing the values K_q and K_r in pitch and yaw, one second responses of 17.7 degrees in pitch and 10.6 degrees in yaw will result. For an aircraft weight of 15300 pounds these responses represent 1.5 times the AGARD requirements for pitch and yaw and 2 times the roll requirement as specified by NASA.



K_p = Vss ROLL RATE LOOP GAIN (SEC.)

$f_p(q_o) = \delta_{a_m} L_{\delta a} + L_{\delta aj}$

$L_{\delta a} = \frac{q S b}{I_x} C_{l_{\delta a}}$

$L_{\delta aj} = \frac{\text{MAX JET ROLL MOMENT}}{I_x}$

τ_δ = POWER ACTUATOR TIME CONSTANT = .05 SEC

I_x = ROLL MOMENT of INERTIA

K_{SF} = MAXIMUM SURFACE DEFLECTION (rad)

ε_p = ROLL RATE ERROR

$(\delta_p)_c$ = ACTUATION COMMAND

p = ROLL RATE

$$\frac{p}{p_c} \quad \frac{1}{\frac{s^2}{\omega_p^2} + \frac{2\zeta_p}{\omega_p} s + 1}; \quad \zeta_p = .7, \quad K_p = \frac{1}{4\zeta_p^2 \tau_\delta}$$

$$\omega_p = 2 K_p \zeta_p$$

Figure VI-35. Simplified Roll Rate Loop

b. Summary of Analog Simulation Effort

The analog simulation effort of this program comprised three phases as follows:

1) A six degree of freedom simulation of the research aircraft. Initially, Concept F was mechanized since the aerodynamic data was available for that configuration first. Concept C was later substituted for F.

2) A six degree of freedom simulation of a model aircraft. The characteristics of a supersonic V/STOL fighter bomber were used as being representative of the type of aircraft which would be of interest to the NASA research program.

3) Finally, the five model following loops were closed between the model simulation outputs and the research aircraft control loops. The angular rate following loops were closed and checked out individually and finally the longitudinal and normal load factor following loops were closed. At this point, it was possible to perform complete flights from lift-off through transition and back to touch down with all five model following loops operating. The result of these efforts are discussed under the VSS section.

The equations used to represent the research aircraft are presented below.

NASA V/STOL 6 D.O.F. SIMULATION EQUATIONS

1) $$\mathring{u} = -g \sin\theta - w q + r v + \frac{X_A}{m} + \frac{K_T T \cos\delta_V}{m}$$

2) $$\mathring{v} = -u r + w p + g \cos\theta \sin\varphi + Y_v v + Y_p p + Y_r r + Y_{\delta_r} \delta_r$$

3) $$\mathring{w} = u q - v p + g \cos\theta \cos\varphi + \frac{Z_A}{m} + Z_{\delta_e} \delta_e - \frac{K_T T \sin\delta_V}{m} + \frac{F_j}{m}$$

4) $$\mathring{p} = \frac{I_{xz}}{I_x} (\mathring{r} + p q) - \frac{(I_z - I_y)}{I_x} r q - \frac{H_z}{I_x} q + L_v v + L_p p + L_r r + L_{cg}$$

5) $$\mathring{q} = \frac{(I_z - I_x)}{I_y} r p - \frac{H_x}{I_y} r + \frac{H_z}{I_y} p + \frac{M_A}{I_y} + M_{cg} + M_q q$$

6) $$\mathring{r} = \frac{I_{xz}}{I_z} (\mathring{p} - r q) - \frac{(I_y - I_x)}{I_z} p q + \frac{H_x}{I_z} q + N_v v + N_r r + N_p p + N_{cg}$$

7) $$\mathring{\theta} = q \cos\varphi - r \sin\varphi$$

8) $$\mathring{\varphi} = p + \mathring{\psi} \sin\varphi$$

9) $$\mathring{\psi} \cos\theta = q \sin\varphi + r \cos\varphi$$

10) $$\mathring{h} = u \sin\theta - v \sin\varphi \cos\theta - w \cos\theta \cos\varphi$$

11) $$X_A = \bar{q} S C_{X_A}$$

12) $$Z_A = \bar{q} S C_{Z_A}$$

13) $C_{X_A} = f_1(\alpha)$

14) $C_{Z_A} = f_2(\alpha)$

15) $M_A = x_a Z_A$

16) $Z_{\delta_e} = \dfrac{-\bar{q}\, S\, C_{L_{\delta_e}}}{M} \cos \alpha$

17) $M_{cg} = M_{\delta_e} \delta_e + M_{\delta_{e_j}} \delta_{e_j} \; f(L.E.)$

18) $M_{\delta_e} = \dfrac{\bar{q}\, S\, \bar{c}\, C_{m_{\delta_e}}}{I_y}$

19) $\delta_e = [\; .349\ \theta_{ST} - .087 + K_q q \;] \dfrac{1}{1 + \tau_a S} \quad -.436 \le \delta_e \le .262 \text{ RAD}$

20) $\theta_{ST} = \dfrac{\text{\% of Stick Travel (Pitch)}}{100}$

21) $M_{\delta_{e_j}} = \dfrac{\text{max jet pitch moment available}}{I_y} = .763 \text{ RAD/sec}^2$

22) $\delta_{e_j} = -\ \theta_{ST} - K_{q_j} q$

23) $L_{cg} = L_{\delta_a} \delta_a + L_{\delta_r} \delta_r + L_{\delta_{a_j}} \delta_{a_j} \; f(L.E.)$

24) $L_{\delta_a} = \dfrac{\bar{q}\, S\, \bar{c}\, C_{\ell_{\delta_a}}}{I_x}$; $L_{\delta_r} = \dfrac{\bar{q}\, S\, \bar{c}\, C_{n_{\delta_r}}}{I_x}$

25) $\delta_{a_j} = \varphi_{ST} - K_{p_j} p$

26) $\varphi_{ST} = \dfrac{\text{\% of Stick Travel (Roll)}}{100}$

27) $N_{cg} = N_{\delta_a} \delta_a + N_{\delta_r} \delta_r + N_{\delta_{r_j}} \delta_{r_j} \; f(L.E.)$

28) $N_{\delta_r} = \dfrac{\bar{q}\, S\, \bar{c}\, C_{n_{\delta_r}}}{I_z}$; $N_{\delta_a} = \dfrac{\bar{q}\, S\, \bar{c}\, C_{n_{\delta_a}}}{I_z}$

29) $\delta_r = [\, .435\, \psi_{ST} + K_r\, r\,] \dfrac{1}{1 + \tau_a\, S}$ $\quad -.435 \leq .435$ RAD

30) $\delta_{r_j} = -\psi_{ST} - K_{r_j}\, r$

31) $\delta_a = [\, .7\, \varphi_{ST} - K_p\, p\,] \dfrac{1}{1 + \tau_a\, S}$ $\quad -.7 \leq \delta_a \leq .7$ RAD

32) $L_{\delta_{a_j}} = \dfrac{\text{max jet roll moment available}}{I_x} = 1.407$ RAD/sec^2

33) $N_{\delta_{r_j}} = \dfrac{\text{max jet yaw moment available}}{I_z} = .527$ RAD/sec^2

34) $\psi_{ST} = \dfrac{\%\text{ of Rudder Pedal Travel}}{100}$

35) $F_j = .11\, T$ at 10% engine bleed

36) T = actual thrust from all engines = $f(\theta_{TH})$

37) $\theta_{TH} = \dfrac{\%\text{ Throttle Command}}{100}$

38) K_T = 1 when lift engines are on
= .25 when lift engines are off

39) $\delta_V = \int \dot{\delta}_V\, dt$ $\qquad 60 \leq \delta_V \leq 105°$
$-30°/\text{sec} \leq \dot{\delta}_V \leq +30°/\text{sec}$

40) f (L.E.) = 1 when L.E. are on
= 0 when L.E. are off

41) $\bar{q} = 1/2\, \rho\, V^2$

42) $V \cong \dfrac{w^2}{2u} + u$

43) $\alpha \cong \dfrac{w}{u}$

As shown by the first six equations, all of the significant stability derivatives, as well as the kinematic and gyroscopic coupling terms are included. The major area where liberties were taken, in order to conserve equipment, is in the engine representation which will be discussed under "Assumptions." The aerodynamic force derivatives C_{X_A} and C_{Z_A} are generated as functions of angle of attack, α, (Figure VI-36). The L_{cg}, M_{cg} and N_{cg} terms include the aerodynamic plus the reaction jet forcing function terms. All pilot command inputs are non-dimensional values representing percentage of full command. The resulting aerodynamic surface positions, δ_e, δ_a and δ_r are functions of the pilot commands and the rate feedback terms with a first order lag to represent actuator dynamics (.05 second time constants have been used for each actuator). The bias term in the δ_e relation, equation (20), is to provide the asymmetric elevator position vs θ_{ST} relation shown in Figure VI-37.

Figure VI-38 shows the static thrust versus throttle relationship. The δ_{a_j}, δ_{r_j}, δ_{e_j} equations represent equivalent aerodynamic surface deflections and are dimensionless quantities. These are used to generate the reaction jet moment terms which are linear out to their limits. The aerodynamic moments reach their limits at the same time as the jet moments.

Rotation of the thrust vector is accomplished with a three way button which feeds either a positive, negative or zero voltage into an integrator whose output is limited to values of vector angle, δ_v, between 60° and 105° (jet efflux varies from 15° forward of vertical to 30° aft of vertical). Diversion of the lift cruise diverter valve is accomplished with the lift engine cut out switch. Actuation of this switch simultaneously reduces total thrust to 25 percent and rotates the thrust vector to the cruise position.

An engine-out switch is provided to simulate engine failure. Manual actuation of this switch simultaneously introduces pitch and roll disturbance moments and reduces the engine thrust by 12.5 percent.

Available test data showing the transient thrust response of the J85 engine to step throttle commands indicates a nominally linear relationship between rate of change of thrust, $\mathring{T}$, and instantaneous thrust, T. The dynamics of the engine were simulated by generating a $\mathring{T}$ proportional to T. If the difference, ϵ, between T and the commanded

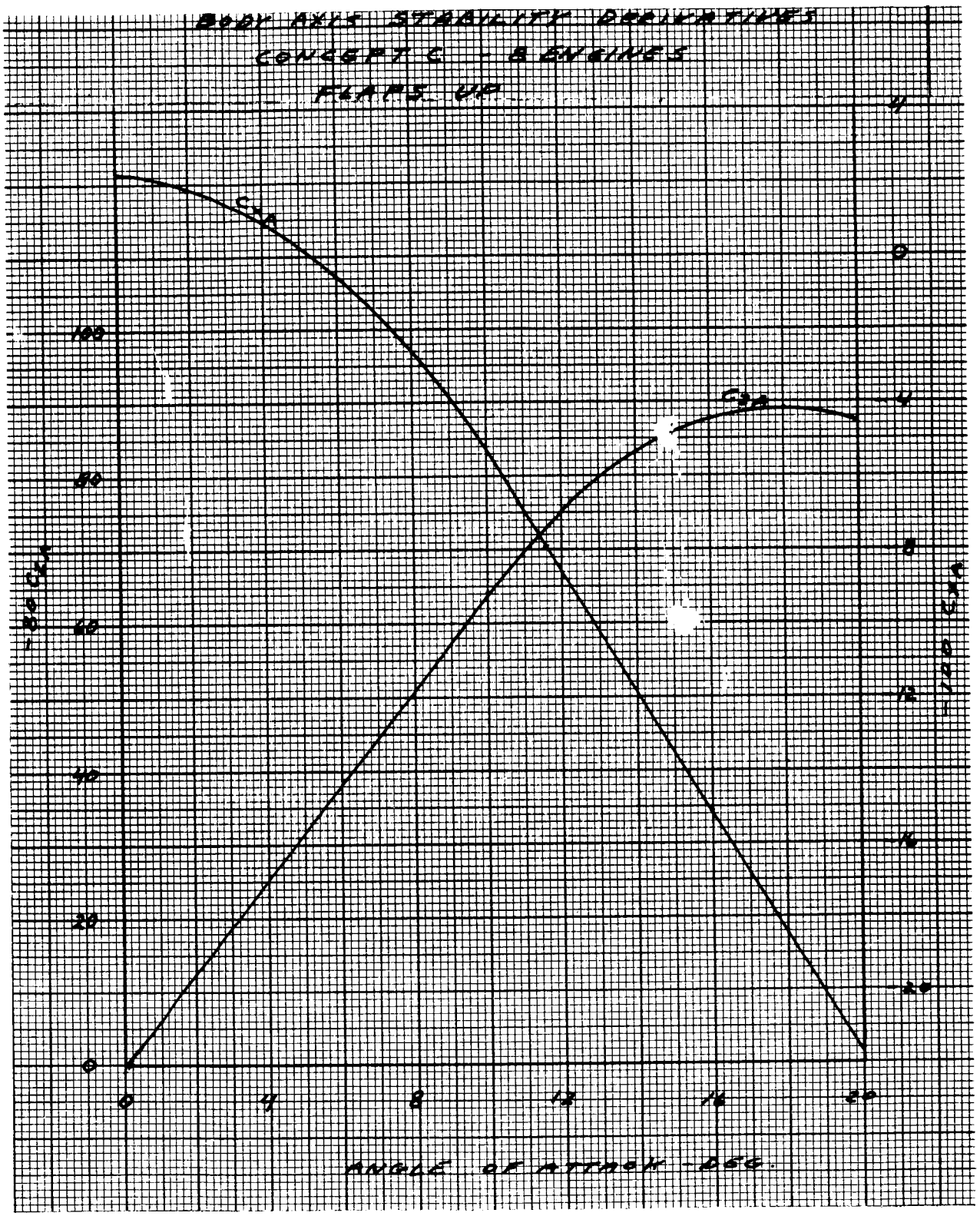

BODY AXIS STABILITY DERIVATIVES
CONCEPT C - 8 ENGINES
FLAPS UP
C_{xA}
C_{zA}
-80 C_{zA}
-100 C_{xA}
100
80
60
40
20
0
4
0
4
8
12
16
20
0
4
8
12
16
20
ANGLE OF ATTACK - DEG.

Figure VI-36.

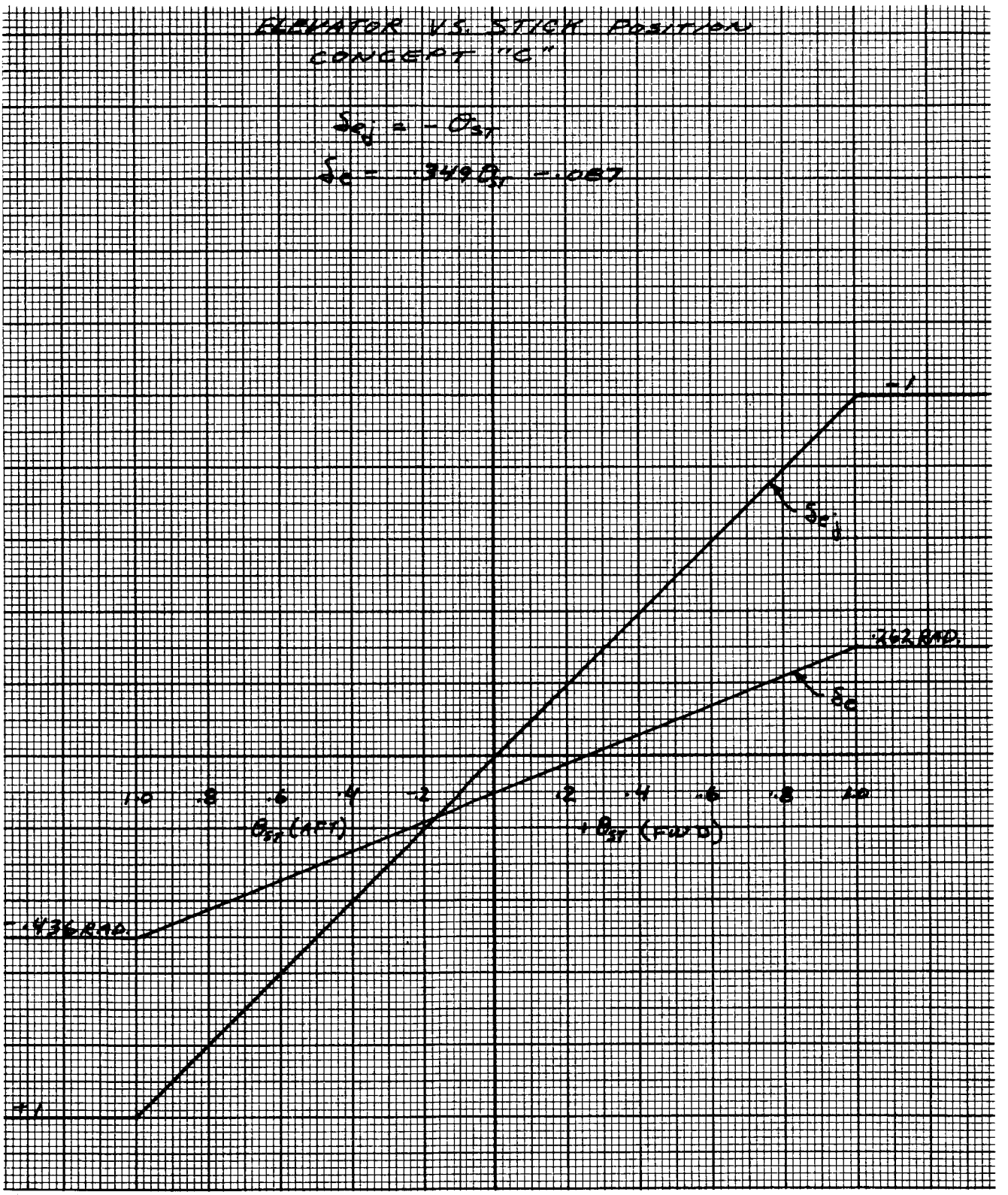

ELEVATOR VS. STICK POSITION
CONCEPT "C"
$\delta_{e_d} = -\theta_{ST}$
$\delta_e = .349\theta_{ST} - .087$
-1
δ_{e_d}
.262 RAD.
δ_e
1.0
.8
.6
.4
.2
.2
.4
.6
.8
1.0
$-\theta_{ST}$ (AFT)
$+\theta_{ST}$ (FWD)
-.436 RAD.
+1

Figure VI-37

J-85 ENGINE THRUST VS THROTTLE CHARACTERISTIC

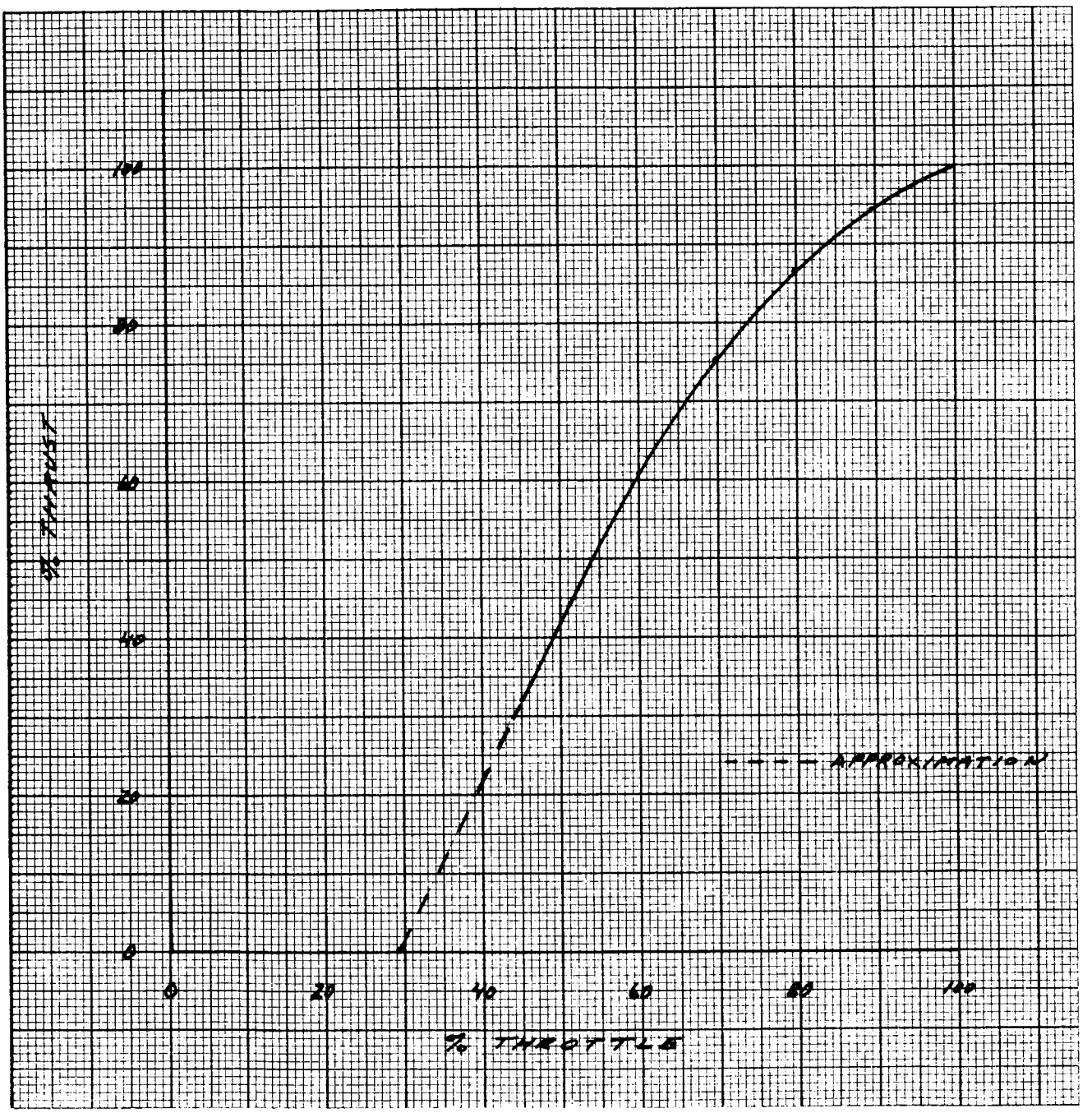


Figure VI-38

thrust exceeds a reference value, ϵ_o, $\mathring{T}$ is integrated to reduce the error. When ϵ reaches ϵ_o, a relay switches out the $\mathring{T}$ signal and substitutes the error signal, after suitable filtering, as the input to the integrator. This provides a second order decay of the error to zero. A higher $\mathring{T}/T$ slope is provided for negative throttle commands than for positive commands.

It should be noted that the simulation equations do not reflect the variable feedback gains discussed under "System Description." They do, in fact, show separate SAS rate feedback gains for the surface and jet controls (K_q and K_{q_j} for pitch, K_p and K_{p_j} for roll and K_r and K_{r_j} for yaw). The change to a single variable feedback gain was made prior to the tie-in of the model outputs with the configuration C simulation for the variable stability studies.

c. Assumptions Used in Simulation

The following assumptions were made in order to minimize simulation complexity:

1) All engines produce the same thrust.

2) The net thrust vector always goes through the C.G. Provision was later made to include the pitch moment which occurs during vectoring. This effect was included in the variable stability studies and is discussed in the VSS section.

3) Control moment commands produce pure couples from the hover controls.

4) All of the bleed thrust acts to produce lift which is a function only of total thrust.

5) Full control power demand in each axis produces the control power specified by NASA.

6) No ram drag, interference or ground effects are included.

7) Constant angular momentum values are assumed for gyroscopic coupling terms.

8) Except for C_{X_A} and C_{Z_A}, constant values for stability derivatives are assumed.

9) Zero flap was assumed.

The following conventions were assumed for the simulation:

1) $+\theta_{ST}$ is stick forward resulting in trailing edge of elevator down and in negative pitching moments (down) from both aerodynamic and jet controls.

2) $+\psi_{ST}$ is left rudder forward resulting in rudder motion to the left and negative yawing moment (nose to left) from both aerodynamic and jet controls.

3) $+\varphi_{ST}$ is stick to the right resulting in left aileron down and right aileron up. Positive roll moments (right wing down) are produced by aerodynamic and jet controls.

4) Vector angle, δ_V, is zero when the reaction thrust is along the aircraft longitudinal axis and 90° when normal to it.

2. Variable Stability System (VSS)

a. System Description

A conceptual block diagram of the VSS is presented in Figure VI-39 This diagram serves as a basis for description of the VSS.

The pilot controls are inputs to the simulation of the model aircraft. In addition, the research aircraft pitch and roll angles and wind velocity are required. The model consists of a set of 6 degree of freedom equations of motion written in the body axis coordinate system. The analog computer used for this simulation will have sufficient capability to allow inclusion of all significant terms in these equations. The outputs of the simulation are model aircraft roll, pitch, and yaw rates (p_c, q_c, r_c) and axial and normal load factors (n_{x_c}, n_{z_c}). These outputs, possibly summed with the outputs of appropriate feed forward filters, form the commands to the VSS following loops. The sensors required for this system are a 3-axis ratio gyro package, x and z body axis mounted accelerometers, and a dynamic pressure (q_c) sensor.

A VSS system of the type described above has been synthesized on a simplified analytical basis. The resulting system was simulated in 6 degrees of freedom on an analog computer. Its performance, when used with a 6 degree of freedom model of a typical, high performance, V/STOL aircraft, was then examined during a variety of dynamic situations covering all critical phases of typical V/STOL flight. On the basis of the results, to be presented subsequently, this system appears to be satisfactory. It is therefore concluded that it is feasible to build a fairly simple, 5 degree of freedom, VSS system for use with the type of research aircraft considered.

In the following sections, brief descriptions of some of the studies conducted to date as part of the VSS synthesis effort are presented.

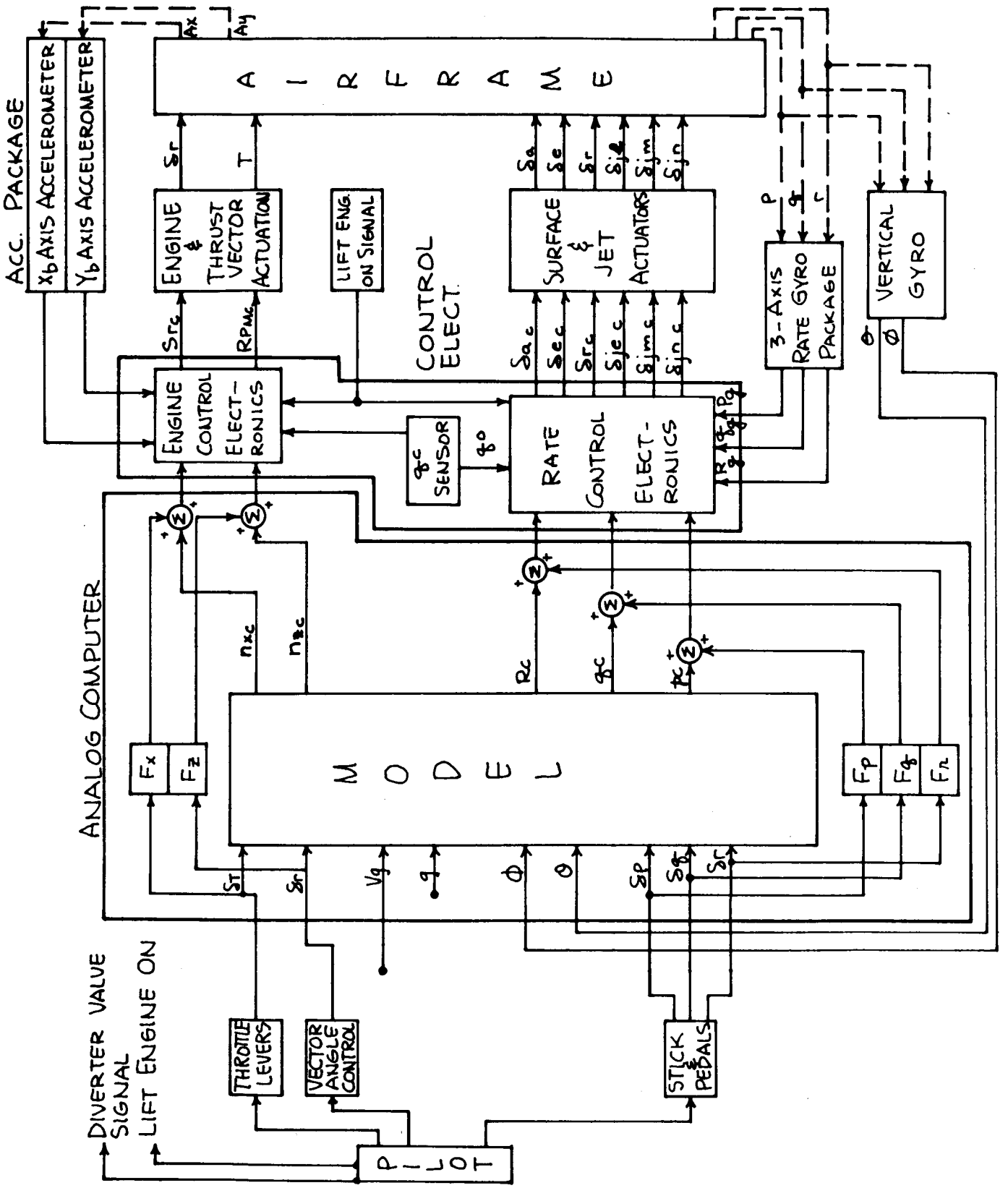


Figure VI-39. VSS Conceptual Block Diagram

b. Rate Control System

(1) General Description

The three rate control loops used in the VSS are all essentially equivalent. A typical loop is shown in Figure VI-40. A qualitative description of the operation of this loop is presented below.

The evaluation pilot's control motions serve as inputs to a model aircraft simulation. The model angular rate output is the basic command supplied to the research aircraft rate following control loop. In addition, a feed forward filter, whose input is derived from control stick motion and, possibly, from variables available in the model, may be used to provide a supplementary input which serves to enhance the rate following loop performance. The total input command is compared with the research aircraft angular rate as obtained from a rate gyro. The resulting error signal is multiplied by a gain factor which is made a simple function of q_o. This variable gain factor is motivated by the desire to maintain high performance over a wide range of operating conditions. The output of the gain element is then shaped to obtain favorable dynamic characteristics. This actuation signal is fed to a power actuator which produces corresponding control surface deflections and control jet valve motion. These motions are coordinated in such a way that both control effectors saturate simultaneously. Due to such factors as power actuator velocity limiting, linkage friction and backlash, etc; it is deemed advisable to make tentative provision for a dynamically tight, series actuation path for the crucial, low dynamic pressure, phase of the mission. To this end, the series, limited authority, actuator channel is provided.

It is emphasized that the system described above is, quite possibly, more complex than will be required to meet performance goals. Extensive analog computer studies will be made to determine whether or not this is the case. For example, if the indicated signal shaping can be eliminated then the corresponding modulators and demodulators can be eliminated, i.e., an "all-AC" system will be possible.

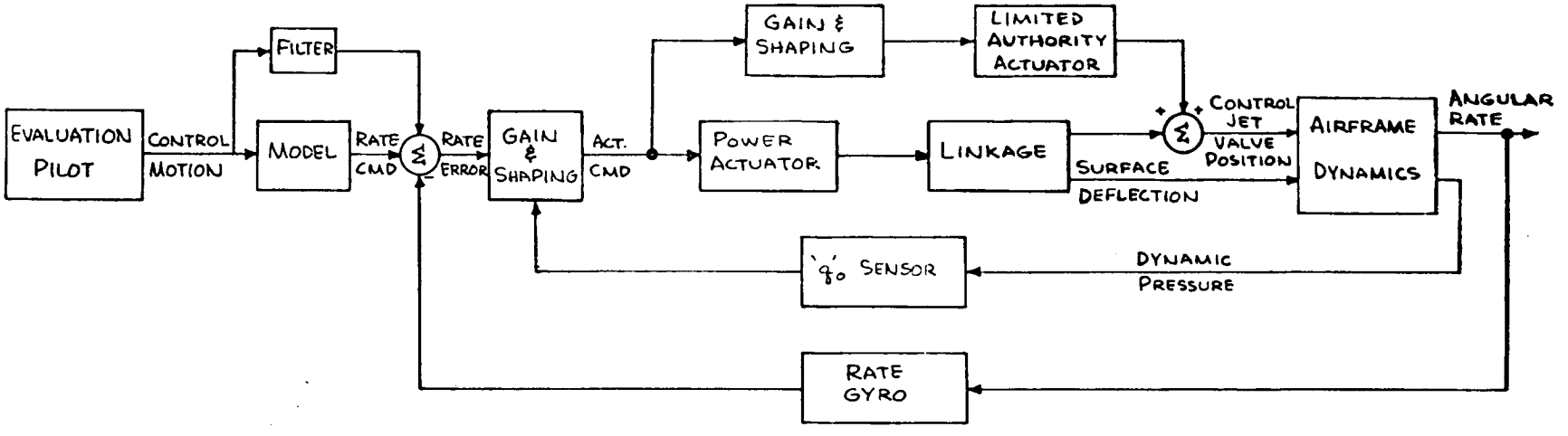


Figure VI-40. Typical VSS Control Loop Block Diagram - Single Axis

(2) Linear Rate Loop Design Considerations

For the purposes of establishing preliminary rate loop gain values, a very simple, linear, characterization of the system was utilized. The corresponding block diagram for roll rate loop is presented in Figure VI-35. The simplified pitch and yaw rate loops are similar.

If a fixed rate gain is used, then the system performance characteristics vary considerably over the operating envelope. In order to preserve high performance over a large portion of the operating envelope, the simple function of dynamic pressure presented is utilized. In addition, since control jet operation is slaved to the lift engines, a lift engine ON signal is provided. The damping ratio and natural frequency of the roll and yaw rate loops were chosen as $\zeta = 0.5$ and $\omega_n = 20$ rad/sec, respectively. These choices lead to gains of $K_p = K_r = 20 \ \text{sec}^{-1}$. These gain factors have been utilized in subsequent computer studies.

The pitch rate control must provide adequate dynamic compensation for the effect of the pitching moment due to lift engine vectoring. This moment is given by:

$$M_{\delta_v} = 2.3 \, T_M \cos \delta_v \quad \text{(ft-lb)}$$

where: $60° \leq \delta_v \leq 110°$ (thrust vector angle)

T_M = thrust magnitude $\leq$ 20,000 lb

The results of a computer study indicate that a value of $K_q = 200 \ \text{sec}^{-1}$ is high enough to provide good dynamic suppression of the effects of this moment while preserving adequate system stability.

The required variable gain factors are presented in Figure VI-41. In order to ease mechanization problems, gain variation was restricted to 5:1. This permits realizing the desired gain values over most of the velocity range.

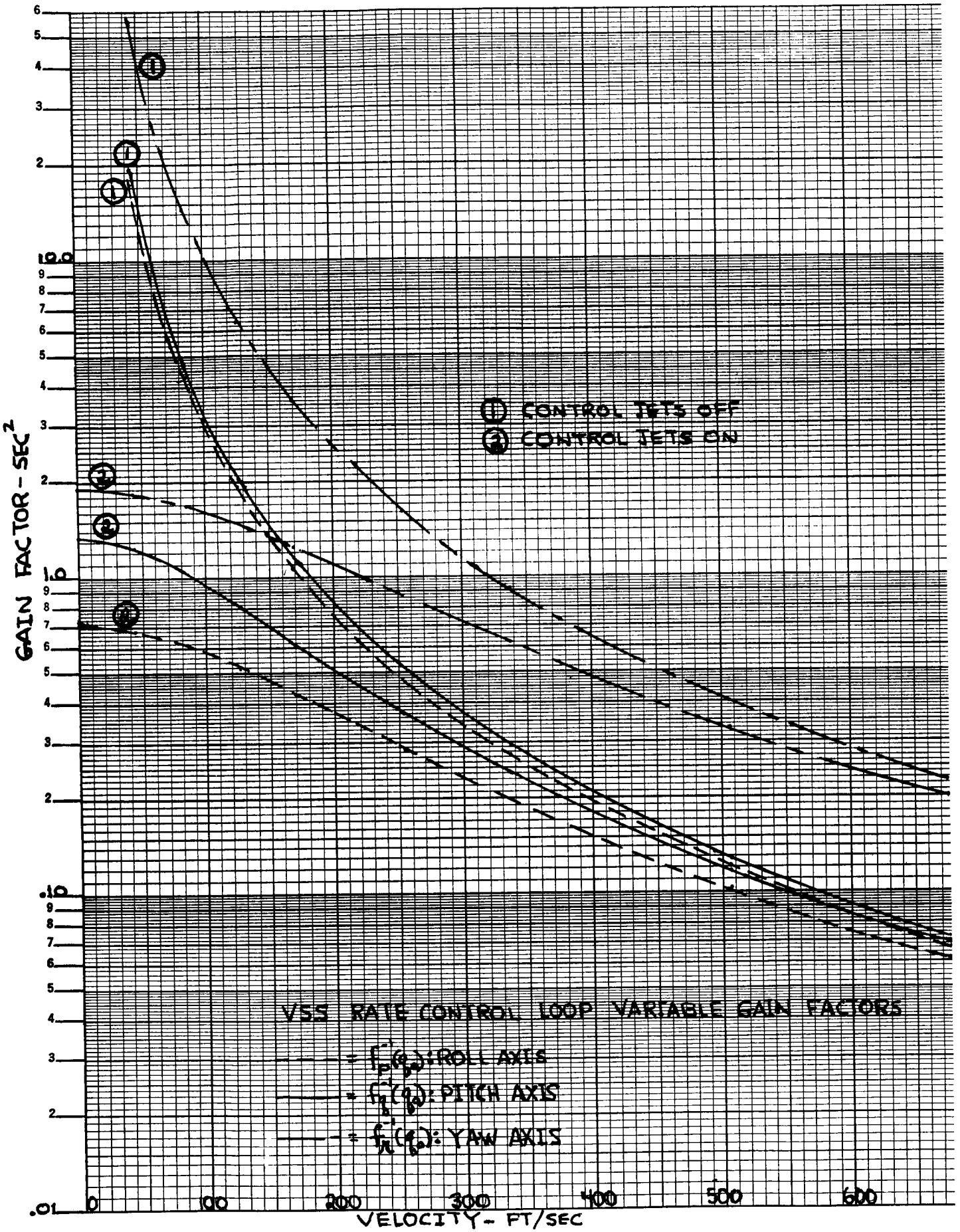

GAIN FACTOR - SEC²
VELOCITY - FT/SEC
0
100
200
300
400
500
600
.01
.10
1.0
10.0
① CONTROL JETS OFF
② CONTROL JETS ON
VSS RATE CONTROL LOOP VARIABLE GAIN FACTORS
= $f_p^{-1}(q_p)$: ROLL AXIS
= $f_q^{-1}(q_q)$: PITCH AXIS
= $f_r^{-1}(q_r)$: YAW AXIS

Figure VI-41.

(3) Nonlinear Rate Effects

(a) Actuator Velocity Limiting

A primary consideration in the design of a high quality rate control system is determination of the effects of non-linearites which are bound to be present in the mechanized system. This information is required for two reasons:

1) It must be verified that state-of-the-art hardware is more than adequate

2) Insight into the effects of nonlinearities may allow for simple compensation, if it is found to be required.

Two general types of non-linearities may be delineated, ie, threshold and saturation types. It is felt that, for purposes of preliminary design, the saturation type should be initially considered since they effect the gross or large signal system performance. Based on initial estimates and previous experience, velocity limiting in the power actuator is felt to be the effect which should be most critical.

In order to make some qualitative statements about the nature velocity limiting effects a typical actuator system was chosen for brief investigation. The system dynamics were:

$$\frac{\delta}{\delta_c} = \frac{1}{\frac{s^2}{\omega_a^2} + \frac{2\zeta_a}{\omega_a} s + 1}$$

where:

δ = output

δ_c = input = $\delta_0 \sin \omega_c t$

ω_a = 20cps = 126 rad/sec

ζ_a = .7

$\dot{\delta}_b$ = 1 rad/sec

$\ddot{\delta}_b$ = 500 rad/sec^2

If the acceleration limit is ignored, the region of nonlinear operation for sinusoidal inputs is obtained from the expression

$$\omega^4 + \omega^2\left[1 - 2K\tau - \left(\delta_o/\dot{\delta}_L\right)^2 K^2\right]\frac{1}{\tau^2} + \frac{K^2}{\tau^2} = 0$$

where:

$$K = 89.5 \text{ sec}^{-1}$$

$$\tau = .572 \times 10^{-2} \text{ sec}$$

The results are plotted in Figure VI-42. While quantitative statements are difficult to make, it maybe seen that, in the upper region of the frequency range of interest ($\omega > 15$ rad/sec), nonlinear operation occurs for relatively modest input levels, ie, $\delta_o/\dot{\delta}_L \sim .1$. This result tends to confirm the intuitive notion that velocity limiting level must be carefully set so that required performance levels may be achieved.

Closed loop frequency response plots for such an actuator, obtained by computer simulation are presented in Figure VI-43. It can be seen that, as the input level increases, large departures from linear operation may be expected. It is recognized, however, that the command inputs from the model will be rather slowly varying which should diminish the severity of this problem.

The effect of these departures from linear operation on stability is most easily discussed from a describing function point of view. Figure VI-44 presents a gain-phase plot for a typical rate loop. The major point to be made is that while the linear equivalent system may appear quite stable, a large amplitude, low frequency limit cycle may well be possible unless care is taken during the design phase to preclude this possibility.

(b) Feedforward Filtering

The basic motivation for the use of a feedforward filter is to compensate for non-ideal following loop dynamics, ie, finite bandwidth. The reasons for non-ideal following may be broadly categorized as unavoidable non-linearities and linear stability considerations. The design of feedforward filters for compensation of following loop nonlinearities is best accomplished by use of a computer. The

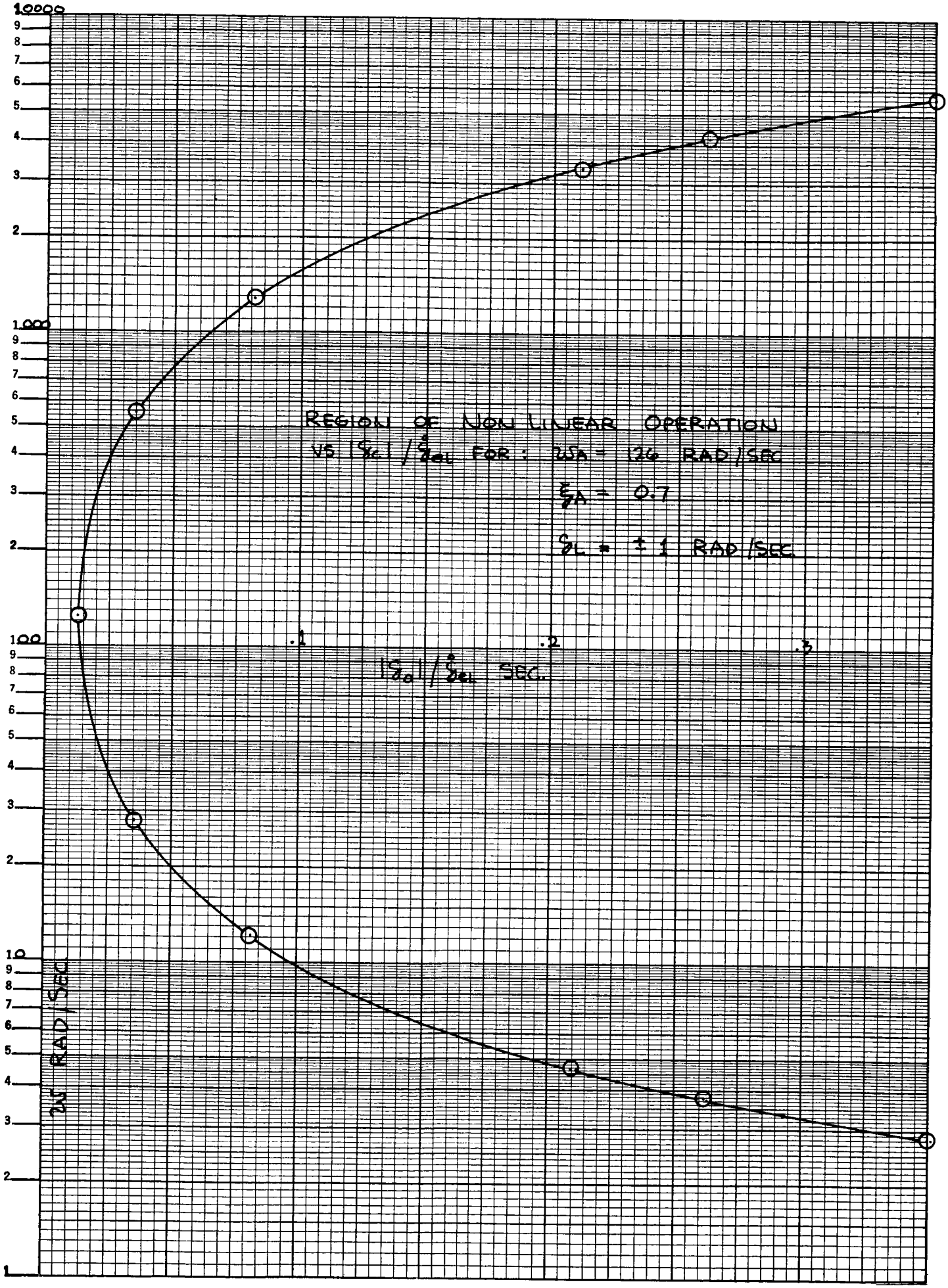

REGION OF NON LINEAR OPERATION
VS $|\delta_c| / \dot{\delta}_{eL}$ FOR : ω_A = 126 RAD/SEC
ξ_A = 0.7
δ_L = ± 1 RAD/SEC
$|\delta_o| / \dot{\delta}_{eL}$ SEC.
ω RAD/SEC
10000
1000
100
10
1
.1
.2
.3

Figure VI-42

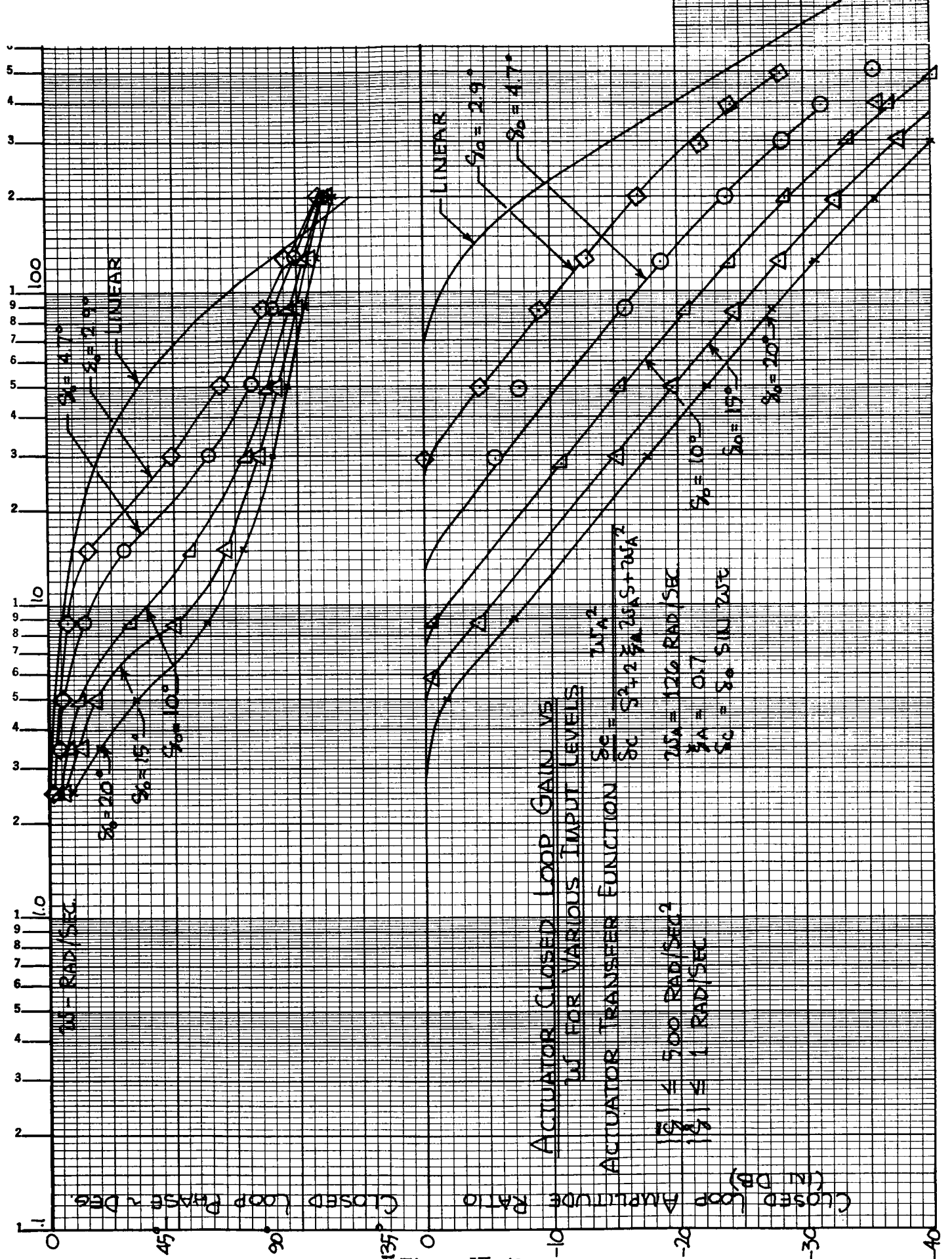

ACTUATOR CLOSED LOOP GAIN VS
ω FOR VARIOUS INPUT LEVELS
ACTUATOR TRANSFER FUNCTION
$\frac{\delta_c}{\delta_c} = \frac{\omega_A^2}{S^2 + 2\zeta_A \omega_A S + \omega_A^2}$
$\omega_A = 120$ RAD/SEC
$\zeta_A = 0.7$
$\delta_c = \delta_0 \sin \omega t$
$|\frac{\delta}{\delta_c}| \leq 500$ RAD/SEC^2
$|\frac{\dot{\delta}}{\delta_c}| \leq 1$ RAD/SEC
$\delta_0 = 10°$
$\delta_0 = 15°$
$\delta_0 = 20°$
$\delta_0 = 2.9°$
$\delta_0 = 4.7°$
LINEAR
$\delta_0 = 4.7°$
$\delta_0 = 2.9°$
LINEAR
$\delta_0 = 20°$
$\delta_0 = 15°$
$\delta_0 = 10°$
$\delta_0 = 5°$
ω - RAD/SEC
Closed Loop Amplitude Ratio (in DB)
Closed Loop Phase ∠ Deg.
0
-10
-20
-30
-40
0
45°
90°
135°

Figure VI-43

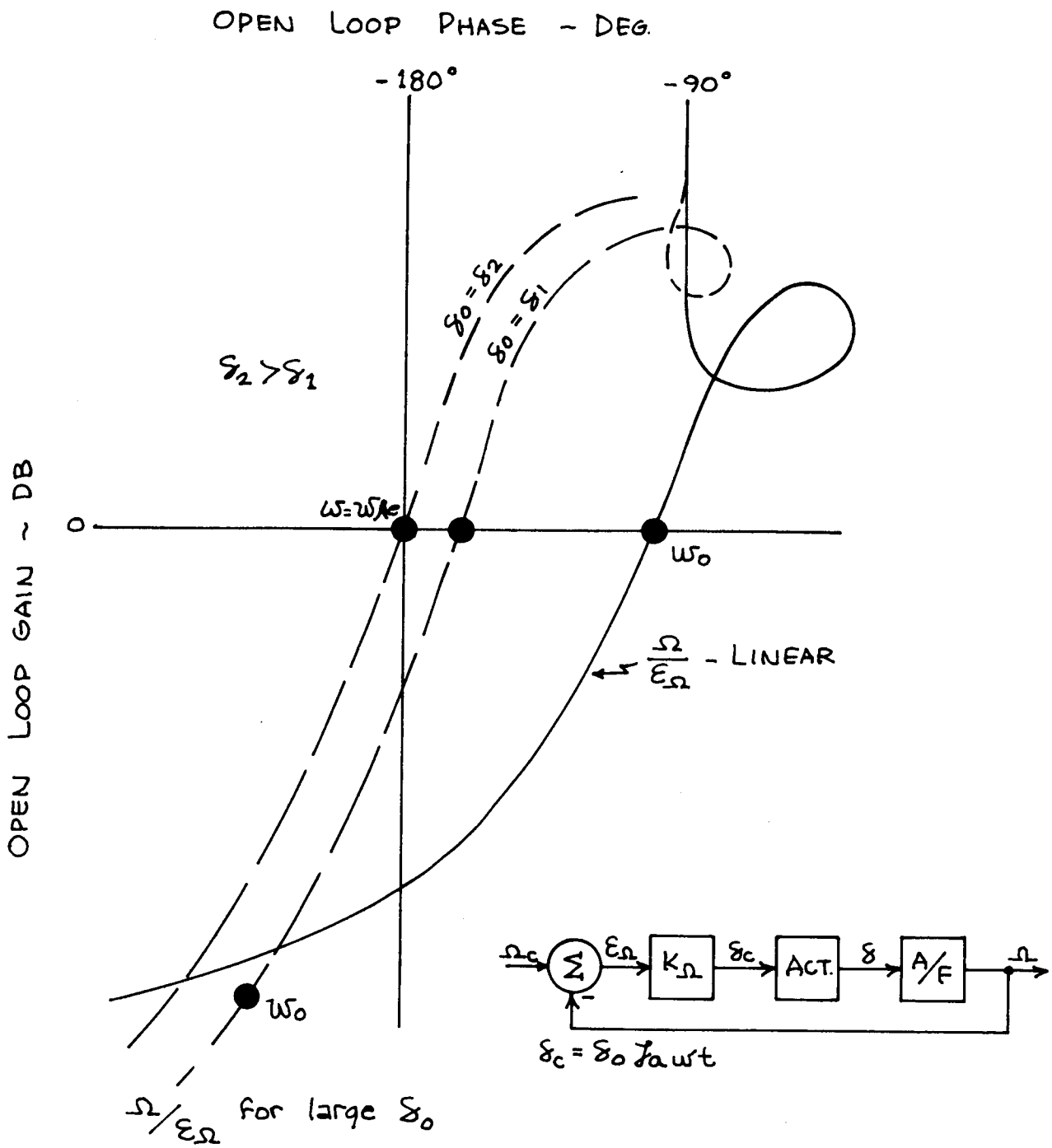


Figure VI-44. Gain Phase Plot for a Typical Aircraft Rate Control Loop

design of filters for compensation of undesirable linear performance characteristics is very straightforward.

A linear structure which is suitable for study of VSS feedforward filtering is presented in Figure VI-45. It is seen that the calculation of the feedforward filter is simple once the required system functions have been obtained. The type of filter used can, however, significantly affect the complexity of mechanization. A simple example of the types of filters obtained when the filters shown in Figure VI-45 are calculated is presented below:

$$M(s) = \frac{K_M s}{\alpha_{1M} s^2 + \alpha_{2M} s + 1} \quad , \quad F_v(s) = \frac{K_v}{s(\tau_v s + 1)}$$

$$G_2(s) = \frac{K_M'}{s(\tau_M' s + 1)}$$

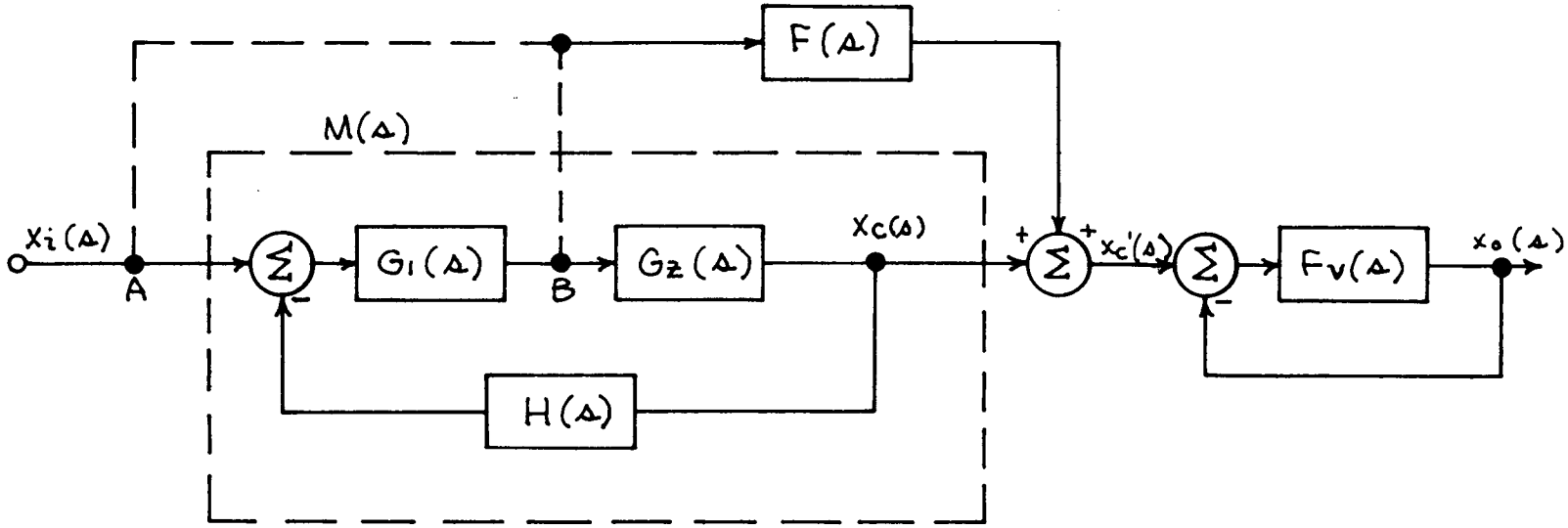
$$F_A(s) = \frac{K_M}{K_v} s^2 \left[\frac{\tau_v s + 1}{\alpha_{1M} s^2 + \alpha_{2M} s + 1} \right]$$

$$F_B(s) = \frac{K_M'}{K_v} \left(\frac{\tau_v s + 1}{\tau_M s + 1} \right)$$

It can be seen that mechanization of $F_A(s)$ is extremely difficult while mechanization of $F_B(s)$ is quite easy. The point to be made is that careful consideration of the types of models is one of the requirements in feedforward filter synthesis.

Due to the approximations required in a linearized model used for filter synthesis and due to the fact that coefficients are never known exactly, there is a likelihood that feedforward filters may produce hang-off errors. In order to eliminate this possibility, it is helpful to include a term of the form

$$\frac{\lambda s}{\lambda s + 1}$$



FEED FORWARD FILTER FOR $\frac{X_o}{X_c}(s) = 1$:

FILTER INPUT FROM PT. A : $F_A(s) = \frac{M(s)}{F_v(s)}$

FILTER INPUT FROM PT. B : $F_B(s) = \frac{G_2(s)}{F_v(s)}$

ALTERNATE LOOP MECHANIZATION

Figure VI-45. Linear Feedforward Configuration

in the filter function or to, in fact, use the exact form of the filter to derive a first approximation for λ using the above expression as the complete filter.

The computer studies conducted have not shown a need for this type of filtering. It is possible, however, that the more detailed studies required in a final design effort might illuminate problems which would make this sort of filter desirable.

c. VSS Load Factor Control System

This section outlines the design philosophy being used in order to produce a high performance control system for axial and normal load factor control.

The analysis presented below is for the hover case with lift engines on and cruise engine thrust diverted. The analysis also assumes that all engines are coordinated. Figure VI-46 defines the body ($\underline{x}_b$, $\underline{z}_b$) and engine ($\underline{x}_e$, $\underline{z}_e$) coordinate systems used.

The notation defined by this figure was chosen for analytical convenience. The figures which present analog computer time histories have been changed, however, to conform to standard notation, i.e., lift position for the engines corresponds to zero vector angle.

If, at some instant, the engine thrust vector angle, δ_N, has some value, and it is required to produce body axis forces F_{x_b} and F_{z_b}, the problem can be solved in the following way. First resolve the forces into the engine axis system so that the required forces become F_{x_e} and F_{z_e}. It can be seen that the required thrust is $T = F_{x_e}$ and the vector angle is, approximately $\Delta\delta_N = \delta_N - \delta_{N_c} = -F_{z_e}/T$ This simple concept is the basis of the system described below.

The axial and normal acceleration equations are:

$$m(\dot{u} - vr + wq) = T\cos\delta_N + X_{aero} - W\sin\theta$$

$$m(\dot{w} - uq + vp) = -T\sin\delta_N + Z_{aero} + W\cos\theta\cos\phi$$

where standard notation is used. Setting $\phi = 0$ and neglecting aerodynamic forces, the corresponding load factor equations are:

$$n_x = T/W \cos\delta_N - \sin\theta$$

$$n_z = -T/W \sin\delta_N + \cos\theta$$

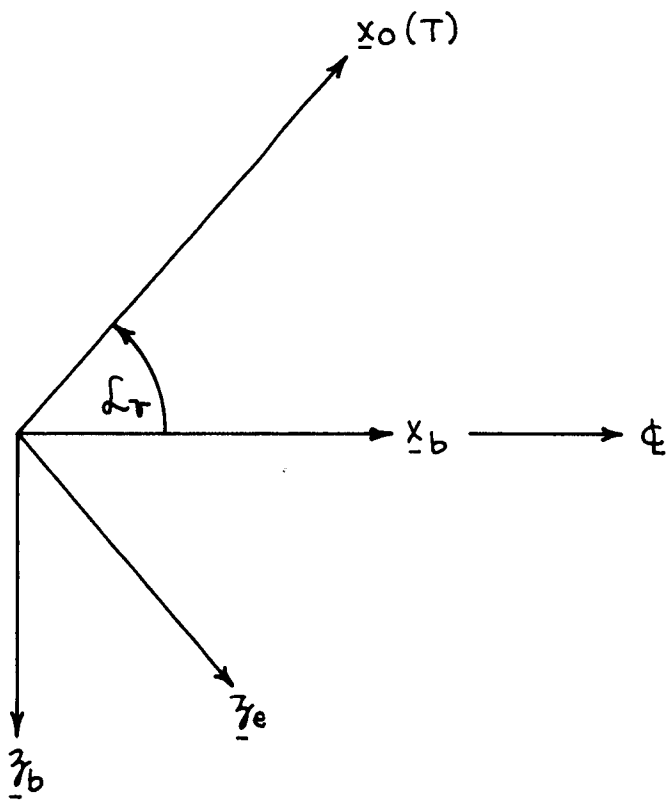


Figure VI-46. Body and Engine Coordinate Systems

setting $T' = T/W$ and denoting the system operating point as $(\bar{T}', \bar{\delta}_N, \bar{\theta}; \bar{n}_x, \bar{n}_z)$
The system perturbation equations become:

$$\begin{bmatrix} \Delta n_x \\ \Delta n_z \end{bmatrix} = \begin{bmatrix} -\bar{T}' \sin\bar{\delta}_N & \cos\bar{\delta}_N \\ -\bar{T}' \cos\bar{\delta}_N & -\sin\bar{\delta}_N \end{bmatrix} \begin{bmatrix} \Delta\delta_N \\ \Delta T' \end{bmatrix} \quad \text{or} \quad \underline{\Delta n}(s) = \underline{G}_P \, \underline{\Delta u}(s) \qquad (1)$$

The basic error signals generated are the difference between the load factor commands generated by the model and the load factors of the research aircraft as derived from accelerometer outputs, ie,

$$\begin{bmatrix} \Delta\varepsilon_x(s) \\ \Delta\varepsilon_z(s) \end{bmatrix} = \begin{bmatrix} \Delta n_{c_x}(s) \\ \Delta n_{c_z}(s) \end{bmatrix} - \begin{bmatrix} n_x(s) \\ n_z(s) \end{bmatrix} \quad \text{or} \quad \underline{\Delta\varepsilon} = \underline{\Delta n}_c - \underline{\Delta n} \qquad (2)$$

these errors resolved into engine axes are

$$\begin{bmatrix} \Delta\varepsilon_{x_e}(s) \\ \Delta\varepsilon_{z_e}(s) \end{bmatrix} = \begin{bmatrix} m_1 & m_4 \\ m_2 & m_3 \end{bmatrix} \begin{bmatrix} \Delta\varepsilon_x(s) \\ \Delta\varepsilon_z(s) \end{bmatrix} \quad \text{or} \quad \underline{\Delta\varepsilon}_e = \underline{M}(\bar{\delta}_N)\, \underline{\Delta\varepsilon} \qquad (3)$$

where from geometry:

$$m_1 = \cos\bar{\delta}_N \quad , \; m_3 = -\cos\bar{\delta}_N$$

$$m_2 = -\sin\bar{\delta}_N \quad , \; m_4 = -\sin\bar{\delta}_N$$

Finally, the thrust and vector angles are assumed to be related by linear dynamics to the resolved error signals, as follows:

$$\begin{bmatrix} \Delta\delta_N(s) \\ \Delta T'(s) \end{bmatrix} = \begin{bmatrix} 0 & G(s) \\ F(s) & 0 \end{bmatrix} \begin{bmatrix} \Delta\varepsilon_{x_e}(s) \\ \Delta\varepsilon_{z_e}(s) \end{bmatrix} \quad \text{or} \quad \underline{\Delta u} = \underline{G}_C \, \underline{\Delta\varepsilon}_e \qquad (4)$$

The system equation is easily obtained from equations (1) - (4) as:

$$\underline{\Delta n} = \frac{1}{\mathrm{Det}\,\underline{\Lambda}} \, \underline{\Lambda}_0 \, \underline{G}_P \, \underline{G}_C \, \underline{M} \, \underline{\Delta n}_c \qquad (5)$$

where: $\underline{\Lambda} = \underline{G}_P \underline{G}_C \underline{M} + \underline{I}$

$$\underline{\Lambda}^{-1} = (\mathrm{Det}\,\underline{\Lambda})^{-1} \underline{\Lambda}_0$$

for simplicity set $\underline{\Delta n} = \dfrac{1}{\mathrm{Det}\,\Lambda} \, \underline{\Gamma} \, \underline{\Delta n}_c \qquad (6)$

where

$$\underline{\Gamma} = \begin{bmatrix} \gamma_{11} & \gamma_{12} \\ \gamma_{21} & \gamma_{22} \end{bmatrix}$$

Ignoring the cross coupling coefficients γ_{12} and γ_{21} or, alternatively, assuming that both input commands are not applied simultaneously, the system transfer functions are

$$\frac{\Delta n_x}{\Delta n_{xc}} = \frac{\gamma_{11}}{\mathrm{Det}\,\Delta} \quad ; \quad \frac{\Delta n_z}{\Delta n_{zc}} = \frac{\gamma_{22}}{\mathrm{Det}\,\Delta} \tag{8}$$

where, when the indicated operations are performed, it is found that:

$$\begin{aligned} \mathrm{Det}\,\Delta &= \bar{T}'FG + F + \bar{T}'G + 1 \\ \gamma_{11} &= \bar{T}'FG + (\cos^2\bar{\delta}_N)F + (\bar{T}'\sin^2\bar{\delta}_N)G \\ \gamma_{22} &= \bar{T}'FG + (\sin^2\bar{\delta}_N)F + (\bar{T}'\cos^2\bar{\delta}_N)G \end{aligned} \tag{9}$$

It is found that suitable choices for F and G are:

$$\begin{aligned} F(s) &= \frac{K_T}{s(\tau_e s+1)} \qquad \text{where } \tau_e = \text{equivalent engine time constant} \\ G(s) &= \frac{K_\delta}{s} \end{aligned} \tag{10}$$

when equations (10) are substituted into equations (9) and the hover values of $\bar{\delta}_N = \pi/2$ and $\bar{T}' = 1$ are used, equations (8) become:

$$\begin{aligned} \frac{\Delta n_x}{\Delta n_{xc}} &= \frac{1}{1/K_\delta s + 1} \\ \frac{\Delta n_z}{\Delta n_{zc}} &= \frac{1}{\frac{\tau_e}{K_T}s^2 + \frac{1}{K_T}s + 1} = \frac{1}{\frac{s^2}{\omega_n^2} + \frac{2\zeta}{\omega_n}s + 1}, \quad \omega_n = \sqrt{\frac{K_T}{\tau_e}}, \quad \zeta = \frac{1}{2\sqrt{K_T\tau_e}} \end{aligned} \tag{11}$$

The preliminary design criteria used for determining the parameter values for this system were that an axial acceleration of 3 ft/sec^2 should produce a thrust vector angle rate of 20°/sec and that the normal acceleration loop should have $\zeta = .7$ for an equivalent engine time constant of $\tau_e = .1$ sec. These criteria yield values of $K_T = 5.1$ sec^{-1} and $K_\delta = 3.2$ sec^{-1}. These values were used as the preliminary values for an analog computer simulation. During the computer study, it was found that doubling K_δ so that $K_\delta = 6.4$ sec^{-1} significantly improved system response. This value was used for the computer runs to be presented subsequently.

The concept implemented is to vary the m_i of equation (3) so that they are always equal to the required values. A block diagram of the system is shown in Figure VI-47. A major advantage of this method of acceleration control is that its implementation is very simple, i.e., the required coordinate system transformation may be accomplished by use of potentiometers slaved to vector angle.

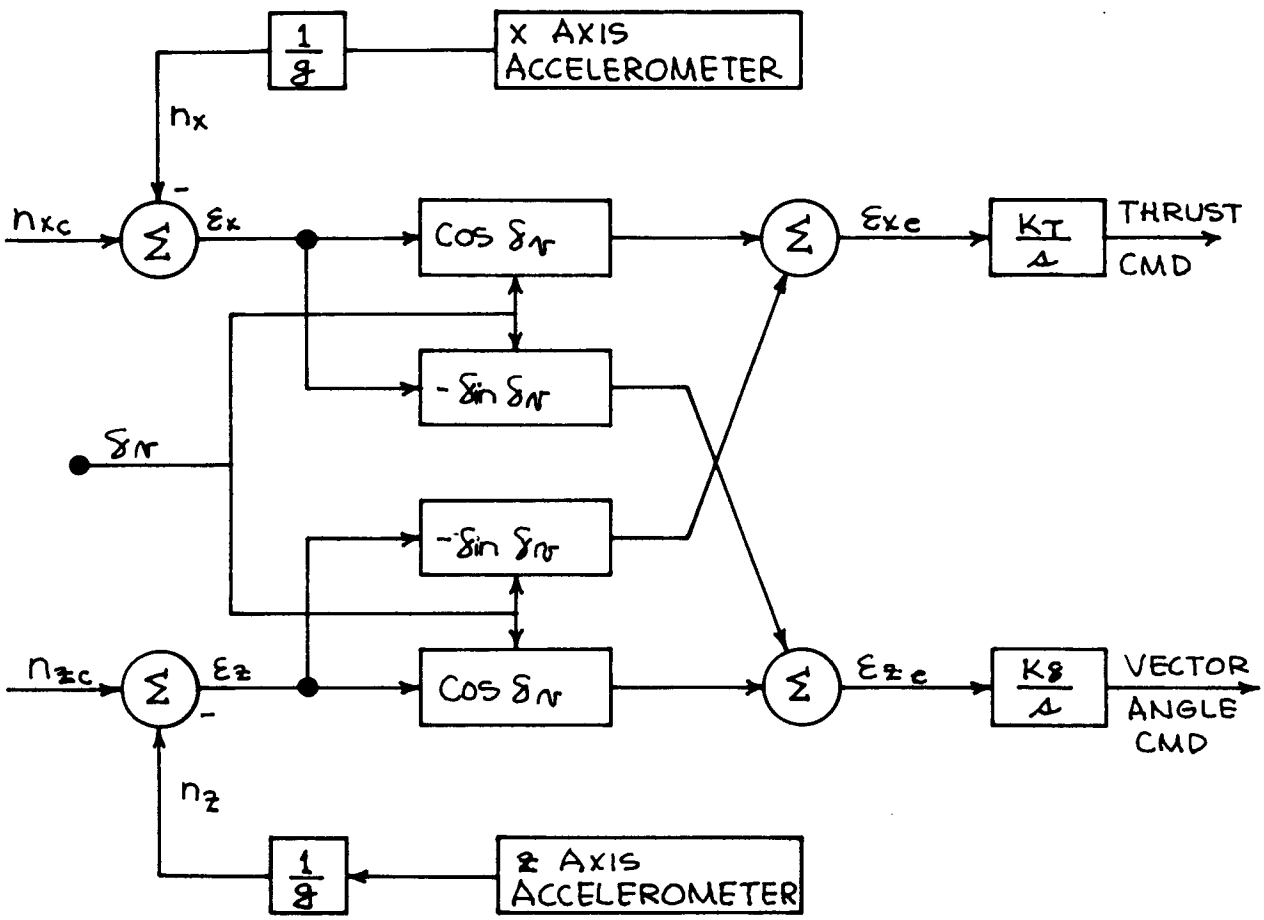


Figure VI-47 . Acceleration Control Loop Conceptual Block Diagram

The results of a computer simulation, to be presented in Flight Characteristics Simulation subsequently, show that this type of system performs quire well. On the basis of these results other possible, more complex, load factor control schemes were not studied.

3. Flight Characteristics Simulation

a. SAS Performance Simulation

The following tabulations present pertinent parameter values used for the configuration C simulation.

CONFIGURATION C PARAMETERS

Longitudinal Stability Derivatives

$C_{L_{\delta_e}}$ = .87 per rad

$C_{m_{\delta_e}}$ = -1.79 per rad

C_{m_q} = -7.3 per rad

C_{X_A} = see Figure VA-6

C_{Z_A} = see Figure VA-6

Lateral-Directional Stability Derivatives

C_{y_B} = -.78 per radian

C_{y_r} = .515

C_{y_p} = -.143

C_{ℓ_B} = -.092

C_{ℓ_r} = .076

C_{ℓ_p} = -.365

C_{n_B} = .104

C_{n_r} = -.242

C_{n_p} = .066

$C_{\ell_{\delta_a}}$ = .072 per radian

$C_{\ell_{\delta_r}}$ = .0206

$C_{n_{\delta_a}}$ = -.0845

$C_{n_{\delta_r}}$ = -.109

$C_{y_{\delta_a}}$ = 0

$C_{y_{\delta_r}}$ = .196

Miscellaneous Parameters

x_a = .63 ft

$\bar{c}$ = 7 ft

b = 32 ft

S = 204 ft^2

ρ = 2.308×10^{-3} $slugs/ft^3$

I_x = 5781 slug ft^2

I_y = 33505 slug ft^2

I_z = 36,676 slug ft^2

m = 475 slugs

H_x = 1635 lb ft sec (cruise engines angular momentum)

H_z = -4900 lb ft sec (lift engines angular momentum)

K_p = .58 per rad/sec

K_q = .622 per rad/sec

K_r = 2.44 per rad/sec

K_{p_j} = 1.32 per rad/sec

K_{q_j} = .862 per rad/sec

K_{r_j} = 2.19 per rad/sec

$M_{\delta_{e_j}}$ = .673 rad/sec^2

$L_{\delta_{a_j}}$ = 1.407 rad/sec^2

$N_{\delta_{r_j}}$ = .527 rad/sec^2

The results of three analog simulation studies performed with configuration C are presented below:

(1) Typical Transition Time History

Figure VI-48 shows a typical transition time history for Concept C. This record provides an indication of the amount of pilot effort required in performing the task as shown by the pitch and roll stick as well as throttle motions.

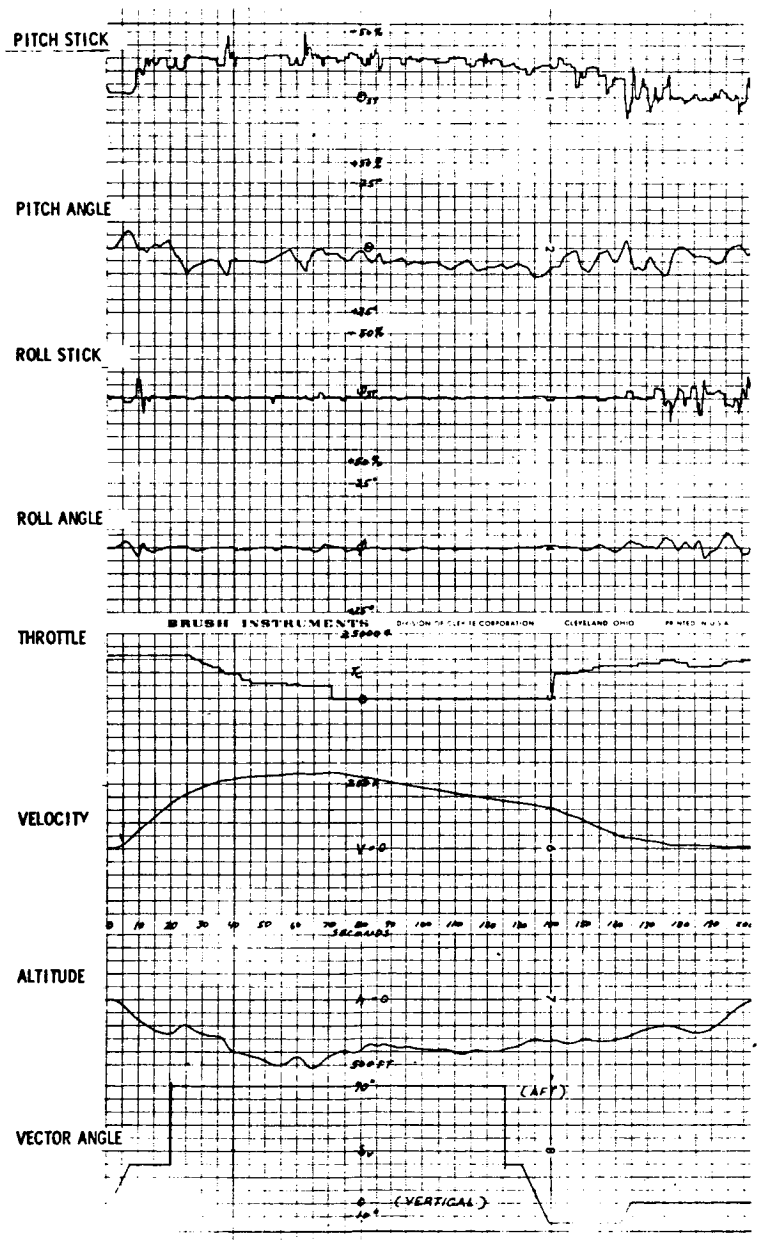


Figure VI-48. Transition Time History - Concept C

The task consisted of the following sequences:

(a) Vertical Lift Off to Aerodynamic Flight

- Lift off and vector the reaction thrust from the lift position to 30° forward (the thrust deflection limit) in approximately 5 seconds.
- At approximately 160 knots, shut down the lift engines and simultaneously divert the lift-cruise diverter valve to the cruise position.
- Reduce throttle and adjust the pitch stick to achieve level flight at approximately 250 knots and 400 feet altitude.

(b) Aerodynamic Flight to Vertical Let Down

- Reduce the throttle to idle power and allow the speed to bleed off while maintaining altitude with the pitch stick.
- At approximately 160 knots, re-start the lift engines and divert the lift-cruise engines to the 30° vector position.
- Rotate vector angle to -15° (aft component of reaction thrust) using the throttle and pitch stick to achieve a gradual descent and deceleration.
- At approximately 40 knots, restore the thrust vector to the lift position, 0°, adjusting the sink rate with the throttle and forward speed with the pitch stick to bring both to zero simultaneously.

The final descent phase of the task was particularly touchy because the pilot was trying for zero sink rate at touchdown rather than 15 feet per second. Also the simulation had an altitude display but no sink rate display. The addition of a sink rate display plus a fine control on vector angle would substantially help in this phase.

(2) Single Engine Failure in Hover

Figure VI-49 shows the response of the aircraft and the pilot to a lift engine failure during hover. The increased pilot action after engine failure is evident in all of the control traces. The pilot's task was to maintain level attitude and constant altitude.

(3) Engine Gyroscopic Effects

The Concept C analog simulation was employed to determine the effects of the gyroscopics on control power demand in each axis and on total control thrust demand during hover. Although gyroscopics due to engine angular momentum provide the dominant coupling moments, the results of this study include the effects of all sources of coupling moments.

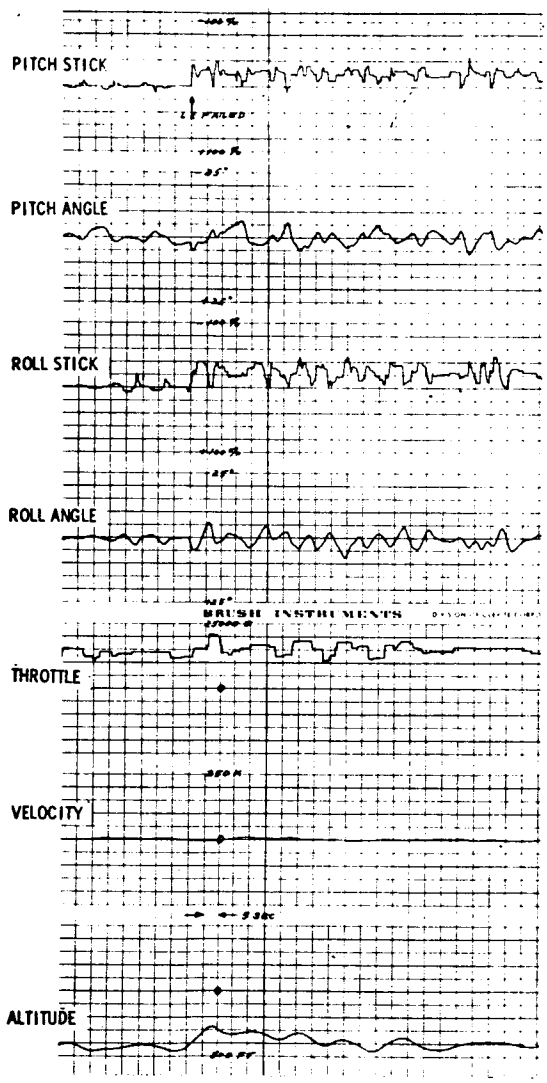


Figure VI-49. Engine Failure in Hover - Concept C

Maximum pitch, roll and yaw stick commands were used individually as well as simultaneous inputs of 80% pitch, 50% roll and yaw. In the case of combined inputs, all possible combinations of plus and minus values were used. Maximum available control powers of .763, 1.407 and .527 rad/sec^2 in pitch, roll and yaw, respectively, were assumed with a maximum total thrust of 2350 pounds.

For the single axis input runs, the coupling moments cause motions about the other axes. The rate feedbacks then cause moments to be generated by the jets to counteract the disturbances. In the axis which is commanded, the rate feedback causes the control power demand in that axis to decay as the body rate builds up to its steady state value. In all cases, the rate of decay of demand in the commanded axis is much higher than the rate of buildup of demand in the coupled axes so that the total demand is always maximum at the time of the command input. Figure VI-50 shows the responses to three single axis inputs. For the pitch command case, pitch rate saturation occurs in the simulation at about 40 degrees per second invalidating the data after the tick marks on the pitch command traces. It is not likely that the 40 degree per second rate would ever be reached in practice without resulting in excessive angular attitudes. The coupling values at the time of pitch rate saturation are therefore good estimates of maximum values that would actually occur. Table VI-12 summarizes the peak coupling effects as percentages of available power at ten percent engine bleed. The maximum total bleed demand for single axis inputs corresponds to 5 percent engine bleed. This is recommended as the nominal bleed requirement for the engines.

Where two values are shown, they represent the initial and the minimum values respectively. This table indicates that a full pitch command makes the largest demands on the control system in terms of initial total demand and amount of coupling effects.

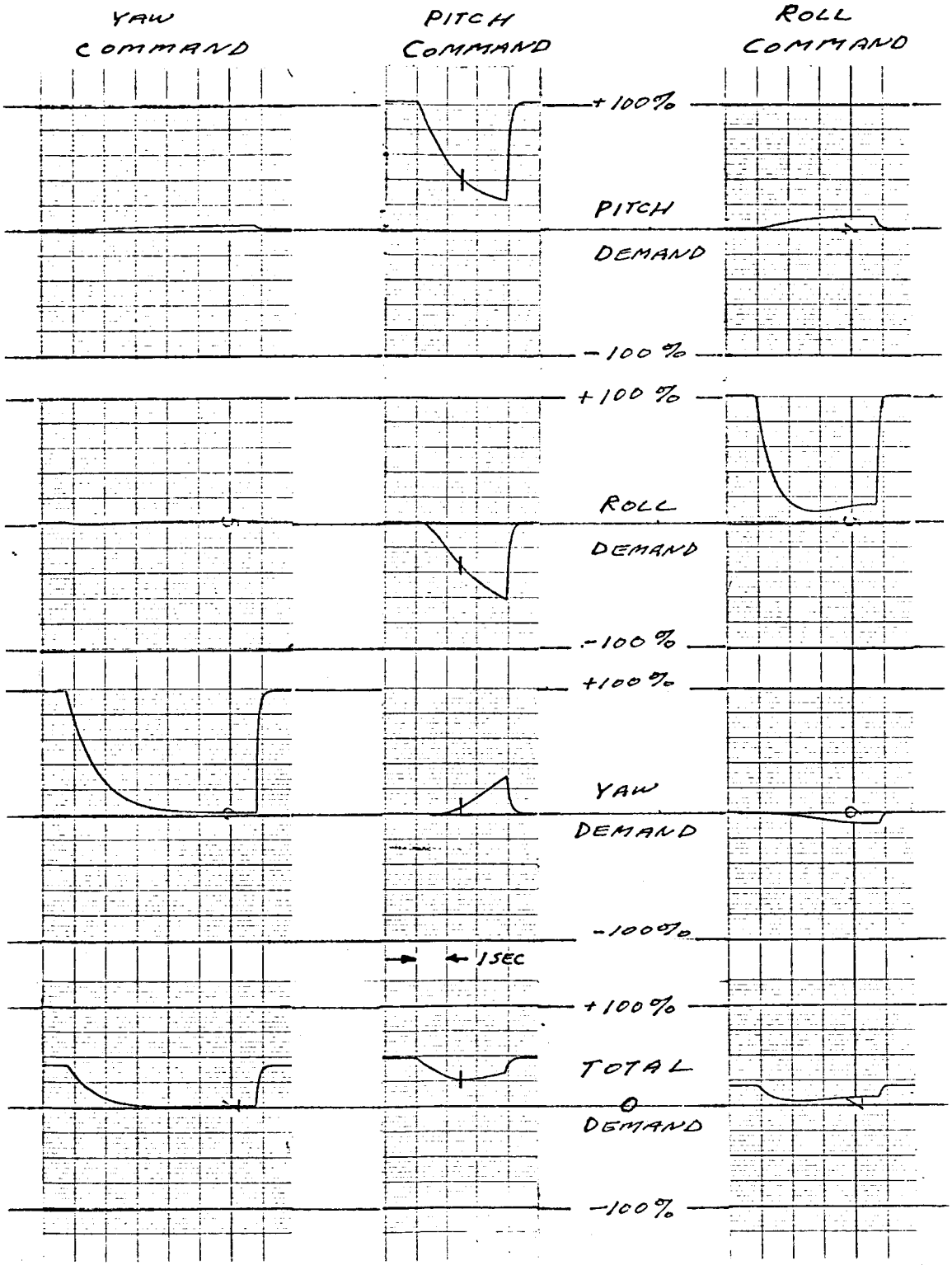


Figure VI-50. Control Power Demands - Single Axis Inputs

TABLE VI-12. PEAK INTER-AXIS COUPLING DATA FOR SINGLE AXIS INPUTS

Input Axis	Pitch Axis Demand - %	Roll Axis Demand - %	Yaw Axis Demand - %	Total Thrust Demand - %
+ Pitch	100 - 38	36	6	50 - 27
- Pitch	100 - 32	44	24	50 - 30
+ Roll	11	100 - 8	8	20 - 5
- Roll	8	100 - 8	8	20 - 5
+ Yaw	4	2	100 - 2	42 - 1
- Yaw	2	2	100 - 1	42 - 2

Figures VI-51 and VI-52 present the control power demand traces for combined inputs. As in the cases for single axis inputs, the total demand is always maximum at the start of the transient. In cases (2) and (5) the yaw axis demand decreases initially but then builds up to a value higher than the initial value after approximately 2-3 seconds. The roll axis demand behaves similarly in cases (3) and (8). It should be noted that this combined input would result in pitch and roll attitudes in excess of 30° in 2 seconds. This is not realistic for a hovering vehicle. It is therefore not likely that the control demands of the individual axes will ever exceed the initial values for any combined maneuver in hover. A third observation from Figures VI-51 and VI-52 is that in some cases the roll and yaw demands reverse sign. This is because the coupling effects from the pitch command are in the proper direction to cause the reversal. From the single axis results it was seen that the major coupling effects were due to pitch commands and in these combined runs, the largest step command was in the pitch axis.

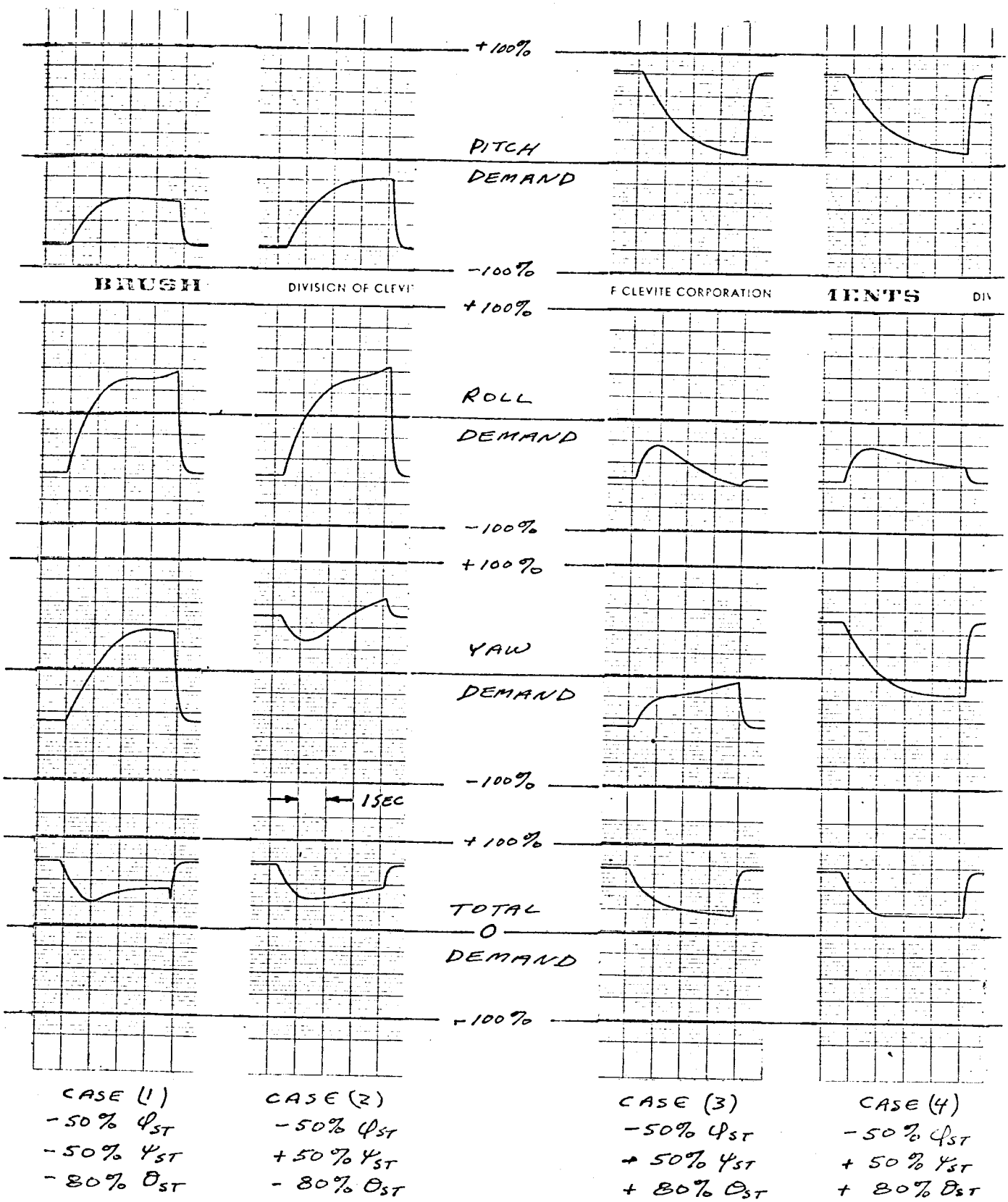


Figure VI-51. Control Power Demands - Simultaneous Commands

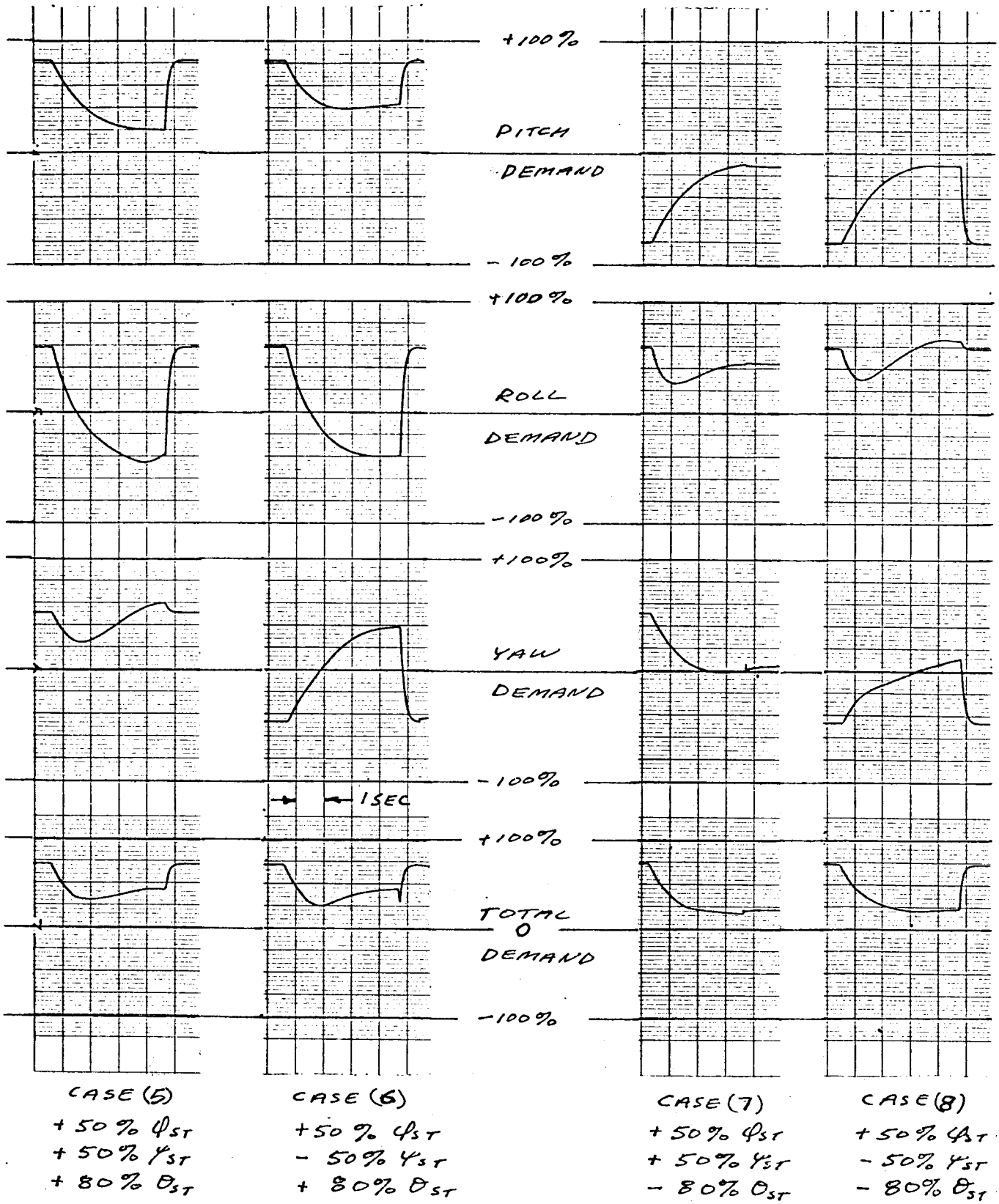


Figure VI-52. Control Power Demands - Simultaneous Commands

b. VSS System Performance

(1) Introduction

This section presents the results of an analog computer simulation conducted to verify the system design. System parameter values were obtained from the simplified analysis previously presented. These values were modified where parameter variation studies indicated that such changes would result in improved performance.

After a final set of system parameters had been evolved, various time histories were obtained in order to demonstrate the sort of performance which might be expected from the simple VSS system considered.

It is to be expected that higher performance could be obtained if a more complex system were utilized. At this preliminary stage, however, such complexity is not required since the goal of this study is to delineate the performance levels which are reasonable for a fairly simple, 5 degree of freedom, VSS.

The simulation consisted of a simplified, 6 degree of freedom, model of a typical high performance V/STOL aircraft together with a 6 degree of freedom simulation of the research aircraft.

The equations implemented for the model aircraft are virtually the same as for configuration with the following exceptions:

1) Some of the less significant kinematic and gyroscopic coupling terms were deleted.

2) The Euler angle and altitude rate computations were eliminated. Pitch and roll attitude angles as well as air speed of configuration C were used in the model.

3) Attitude feedback was included in the pitch and roll hover control loops.

4) Engine dynamics were simulated by a first order lag with a .3 second time constant.

(2) VSS Rate Control System Performance

The simple rate control loops considered in this study performed in a satisfactory manner over the entire flight envelope. Some time histories which present the loop responses at various flight conditions are presented and their characteristics discussed.

Figure VI-53 presents a time history of the pitch rate loop response at hover. The pitch stick was excited with a stop to stop motion by the pilot. Model pitching rates of as much as 32 deg/sec resulted from this input. The aircraft (VSS) pitch rate loop reproduced this motion very well. There was little discernible attenuation or phase shift of the aircraft pitch rate with respect to the model pitch rate.

Figure VI-54 presents the same sort of time history for a speed of 300 ft/sec. Once again, the model following is excellent.

These traces illustrate that the model following concept produces good response. While the simulation did not include all of the system non-linearities which tend to degrade following performance, it is felt that techniques such as feed-forward filters, etc. can be used to reduce undesirable effects to within tolerable limits.

Figure VI-55 presents a time history of the response of the VSS roll and yaw rate control loops to roll stick and rudder pedal inputs at hover while Figure VI-56 presents the same sort of time history for a velocity of 300 ft/sec. Once again the following is found to be excellent with following errors generally less than .1 - .2 sec for all flight conditions examined, i.e., V = 0, 300, 500 ft/sec.

In summary, it is felt that these results indicate that the design of an adequate VSS rate control system is relatively straightforward.

(3) VSS Load Factor Control System Performance

In order to illustrate the capabilities of the VSS load factor control system the throttle and vector angle controls were exercised at hover. The resulting time history is presented in Figure VI-57.

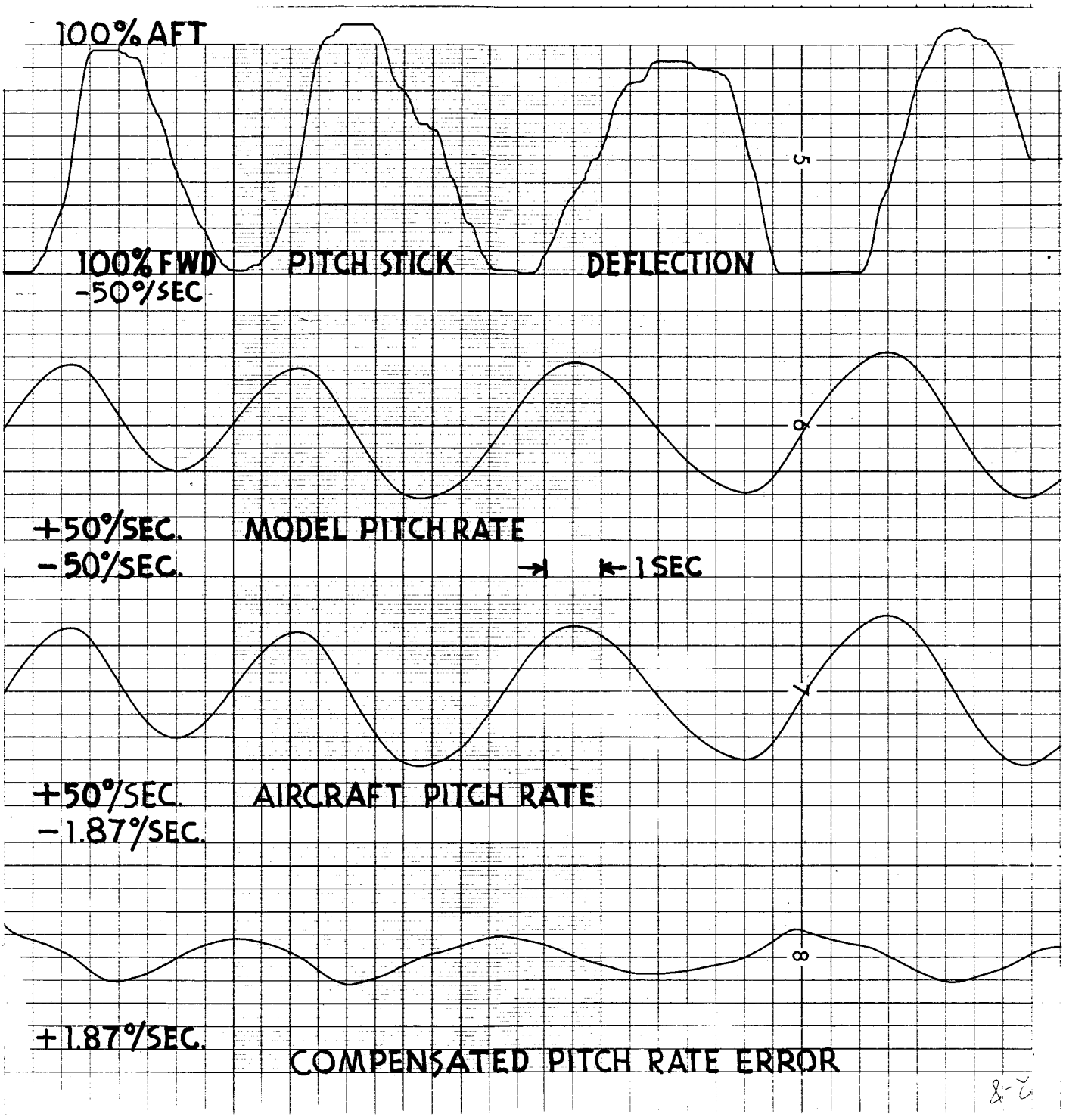


Figure VI-53. VSS Pitch Rate Loop Performance at Hover

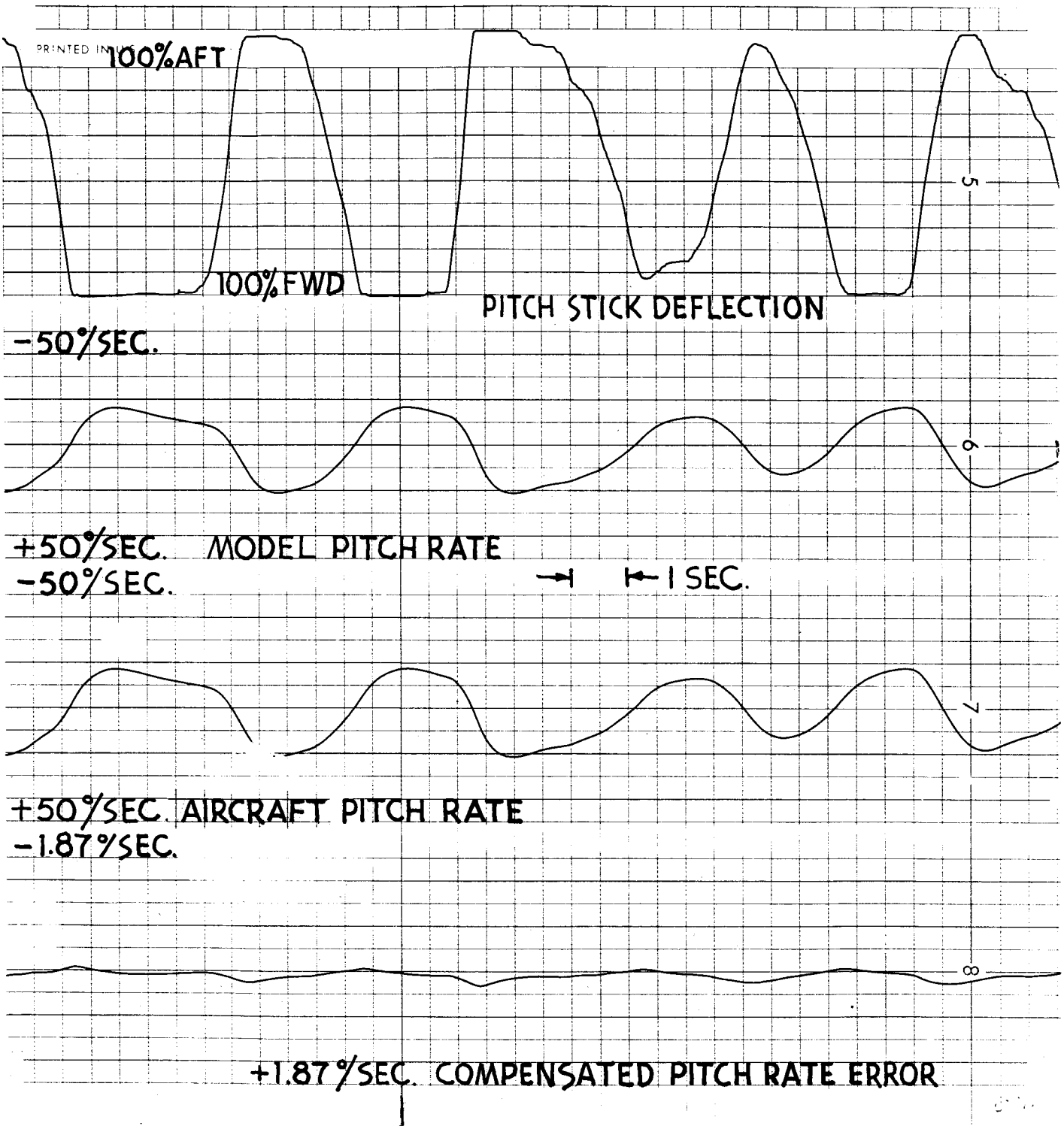


Figure VI-54. VSS Pitch Rate Loop Response at V = 300 ft/sec

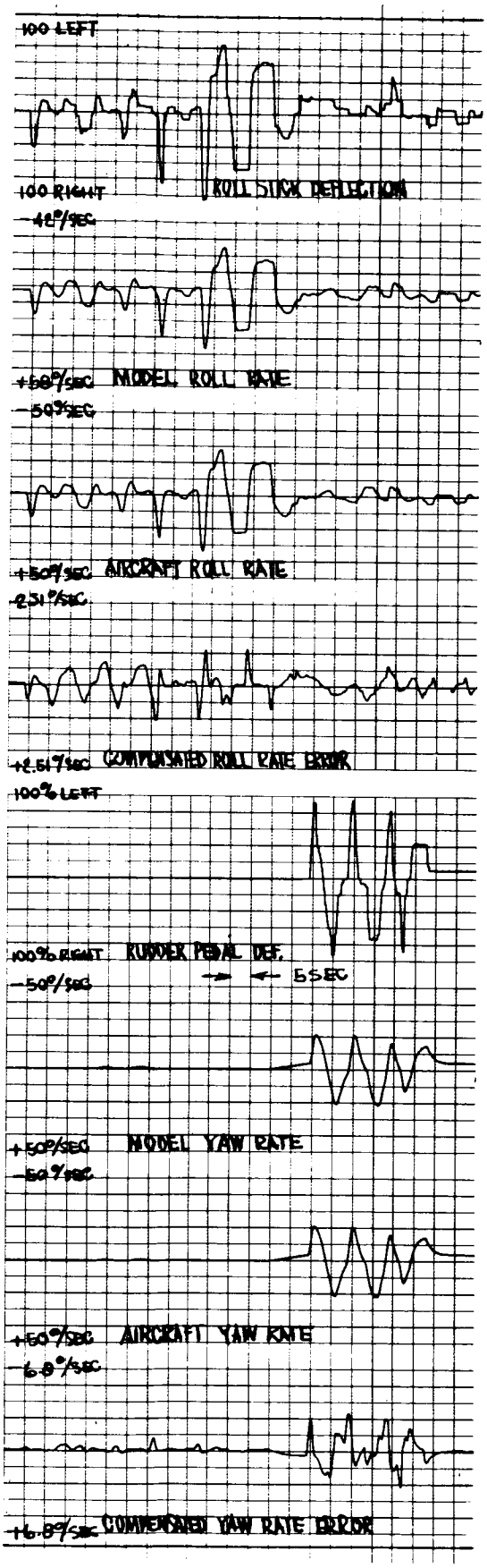


Figure VI-55. Roll-Yaw Rate Response at Hover

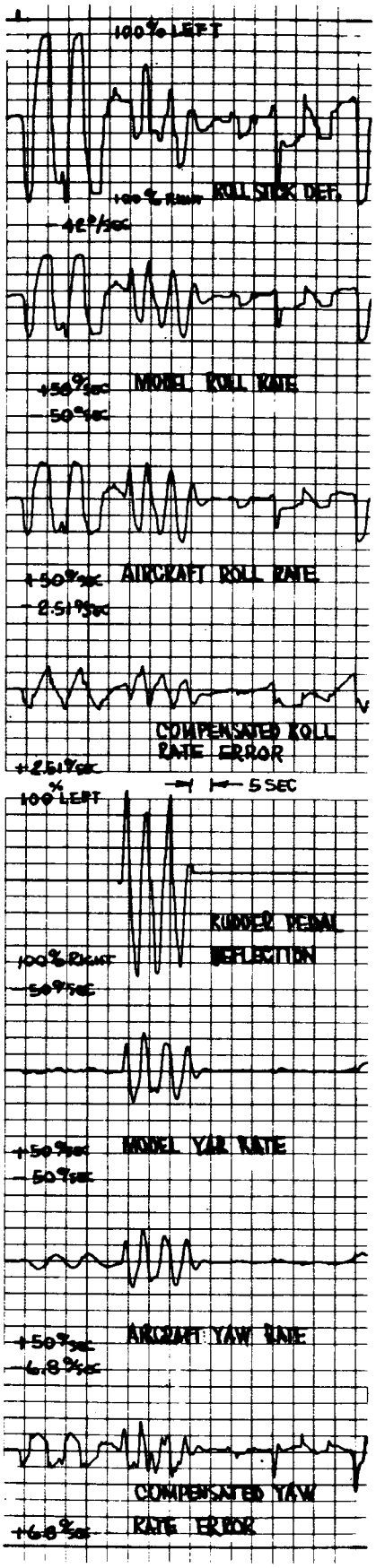


Figure VI-56. Roll-Yaw Response at 300 Ft/Sec

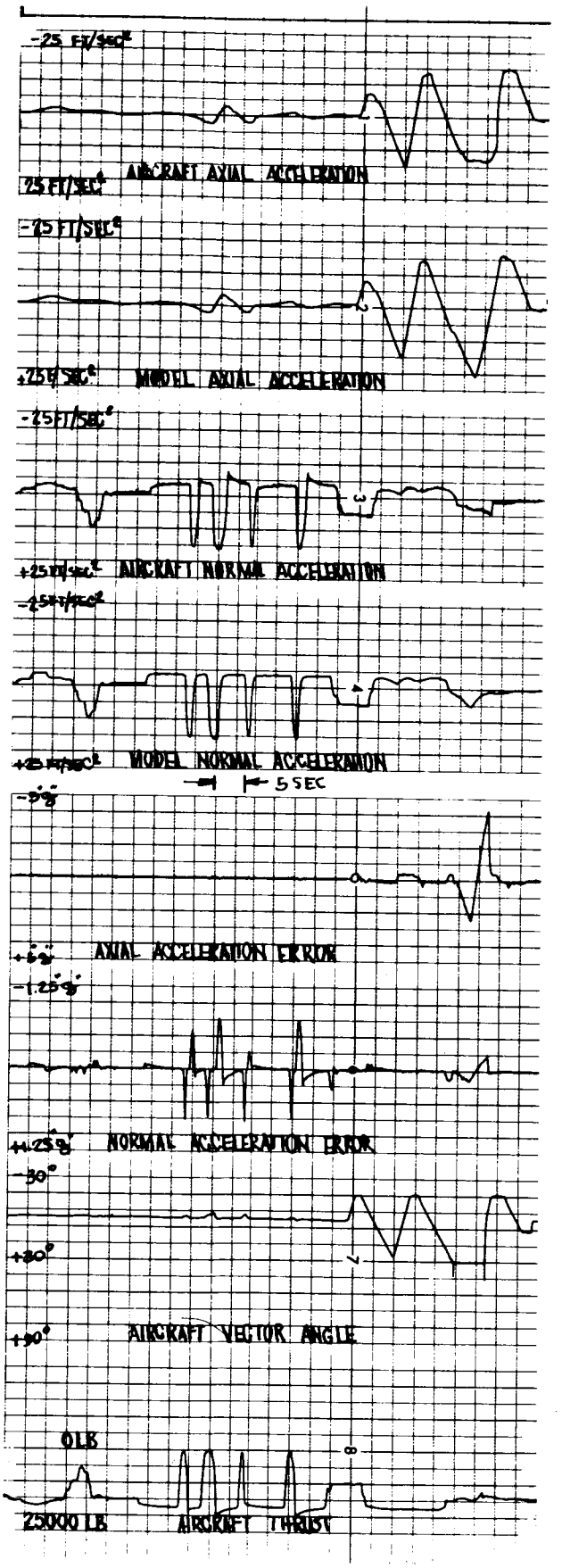


Figure VI-57. Load Factor Control System Performance at Hover

In the first part of the run, the model aircraft thrust was caused to vary between maximum and idle thrust by a periodic throttle lever motion input from the pilot. The resulting model normal acceleration and the VSS controlled, aircraft normal acceleration are also presented. It is seen that, over the acceleration range of -3 to $+16$ ft/sec^2, the model output is followed very well. The research aircraft thrust trace indicates the sort of thrust modulation required by the VSS to follow this type of model normal acceleration input. It should be noted that the engine dynamics for this simulation were carefully modeled for the simulation in order that the response obtained for this type of maneuver be representative of the actual, real world, response.

Axial acceleration outputs were generated by causing the model thrust vector to oscillate about the vertical position by pilot thrust vector rate inputs. The resulting model axial acceleration and the aircraft axial acceleration are seen to track very well during that portion of time when the thrust vector of the research aircraft is not being limited. When limiting occurs there is, of course, some acceleration error build-up.

(4) VSS Transition Time Histories

This section presents complete transition time histories for the VSS mode of operation. The first figure (Figure VI-58) presents a time history for the VSS where a high pitch rate control loop gain is used to suppress the effects of the pitch moment disturbance due to thrust vectoring. In the second time history (Figure VI-59) a relatively low pitch rate gain is used but a bias is computed as a function of thrust level and vector angle and fed into the pitch rate loop in order to compensate for the disturbance moment.

(a) High Pitch Rate Control Loop Gain

In Figure VI-58, the first portion of the run (hover phase) consists of gaining altitude while maintaining relatively low translational velocities. During this initial hover phase, a high thrust level is maintained until an altitude of about 300 ft is achieved. During this phase it can be seen that the model and aircraft axial and normal accelerations track quite well.

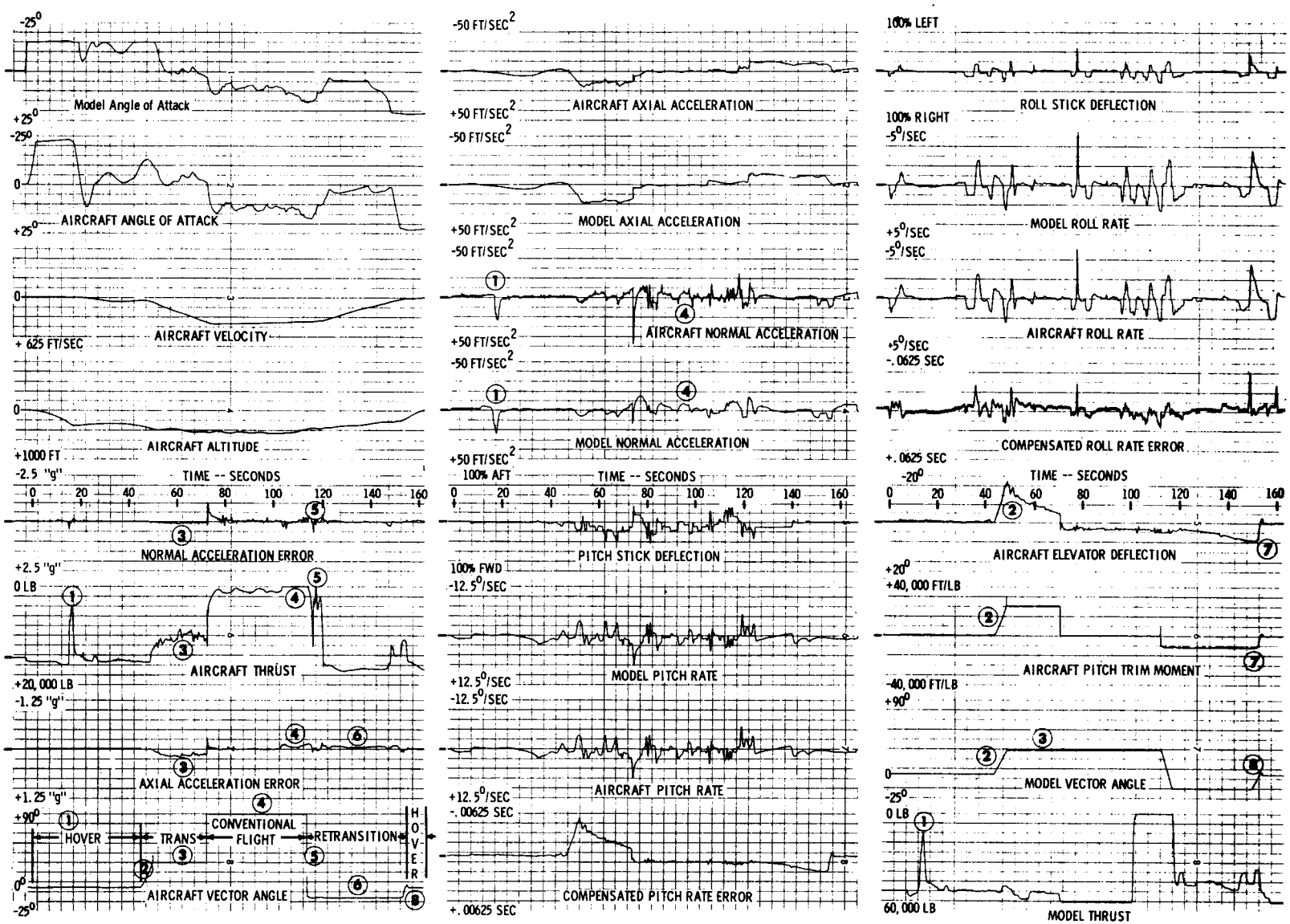


Figure VI-58. Transition Time History-High Pitch Rate Control Loop Gain

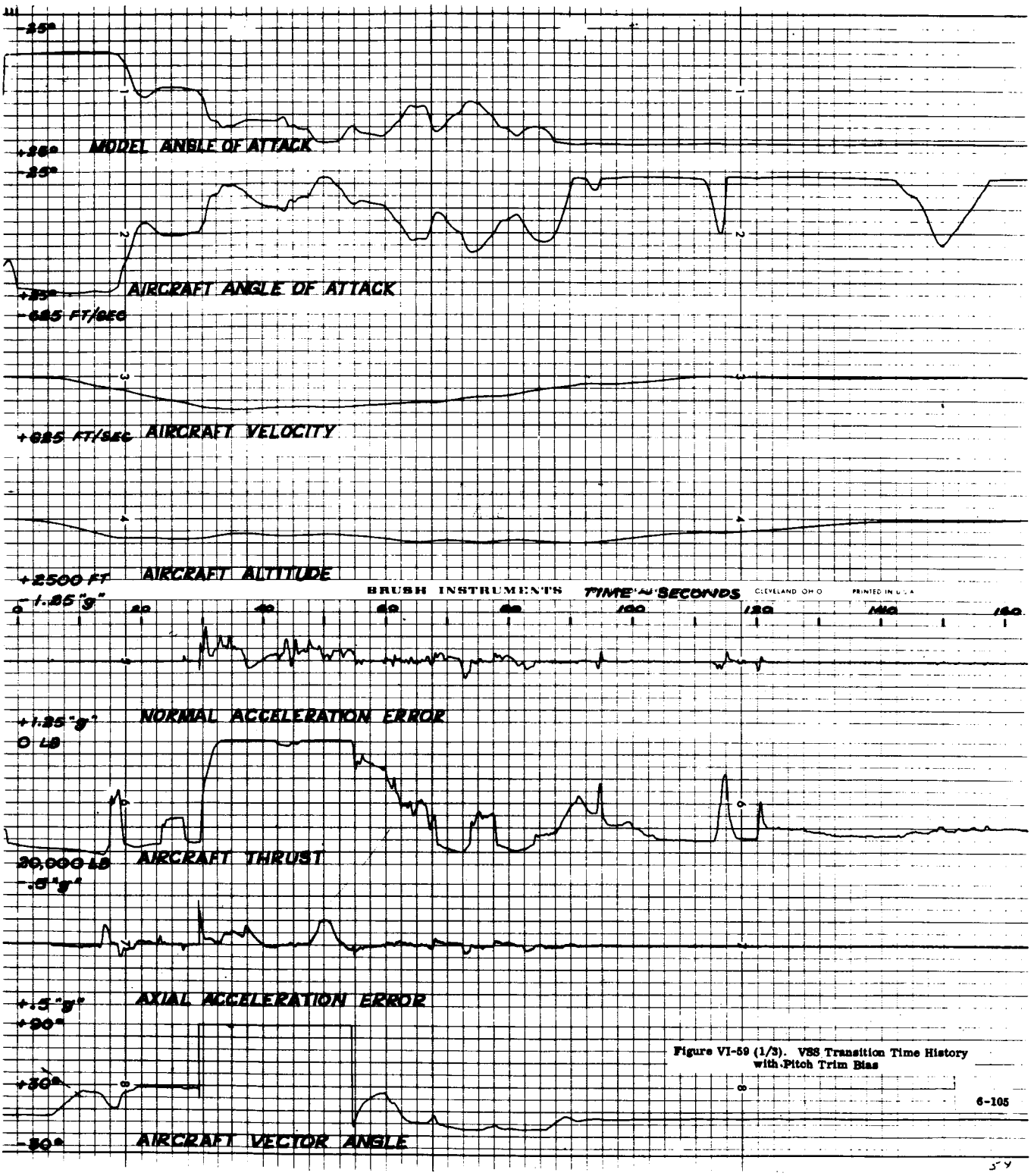


Figure VI-59(1/3). VSS Transition Time History with Pitch Trim Bias

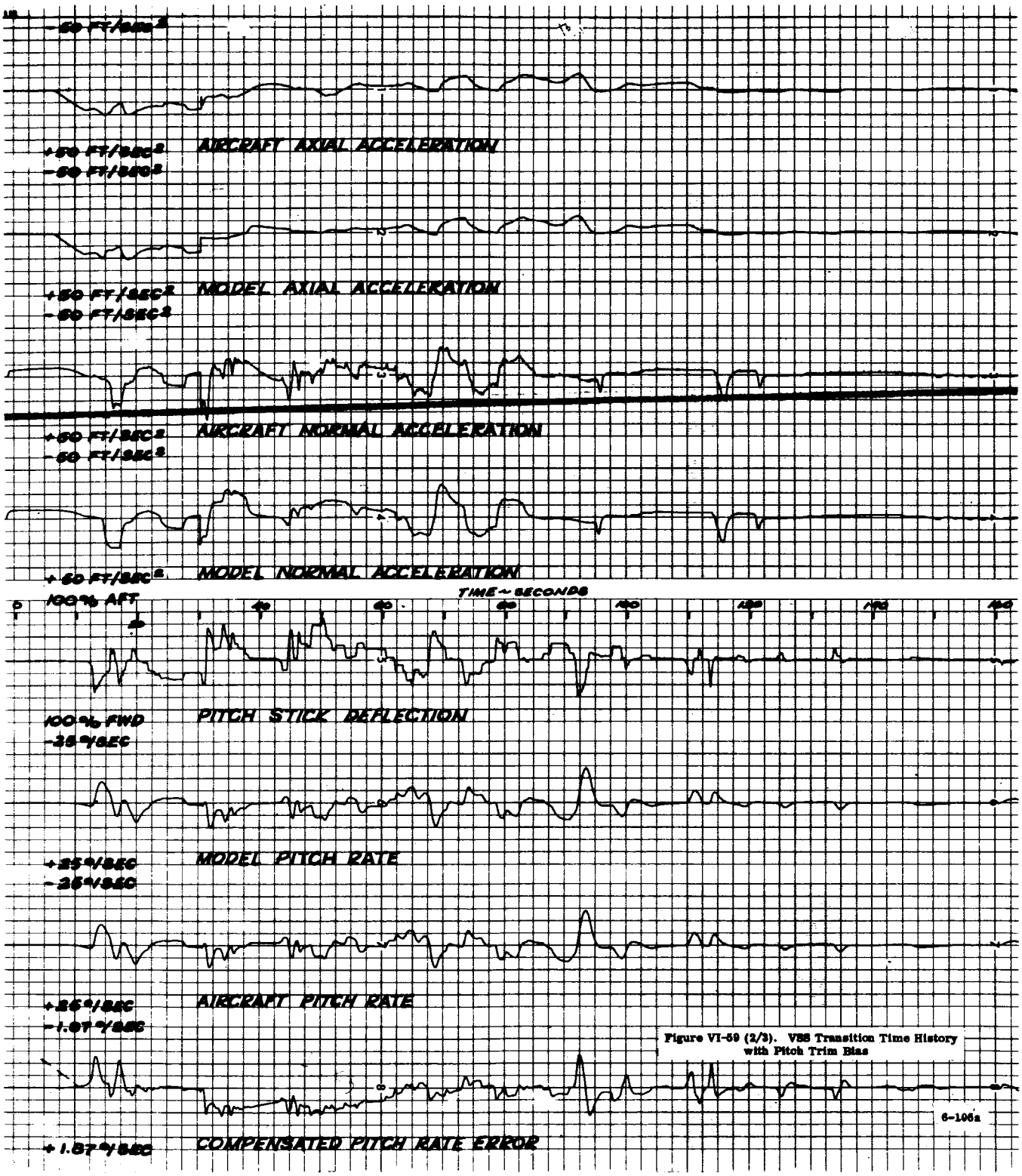


Figure VI-59 (2/3). VSS Transition Time History with Pitch Trim Bias

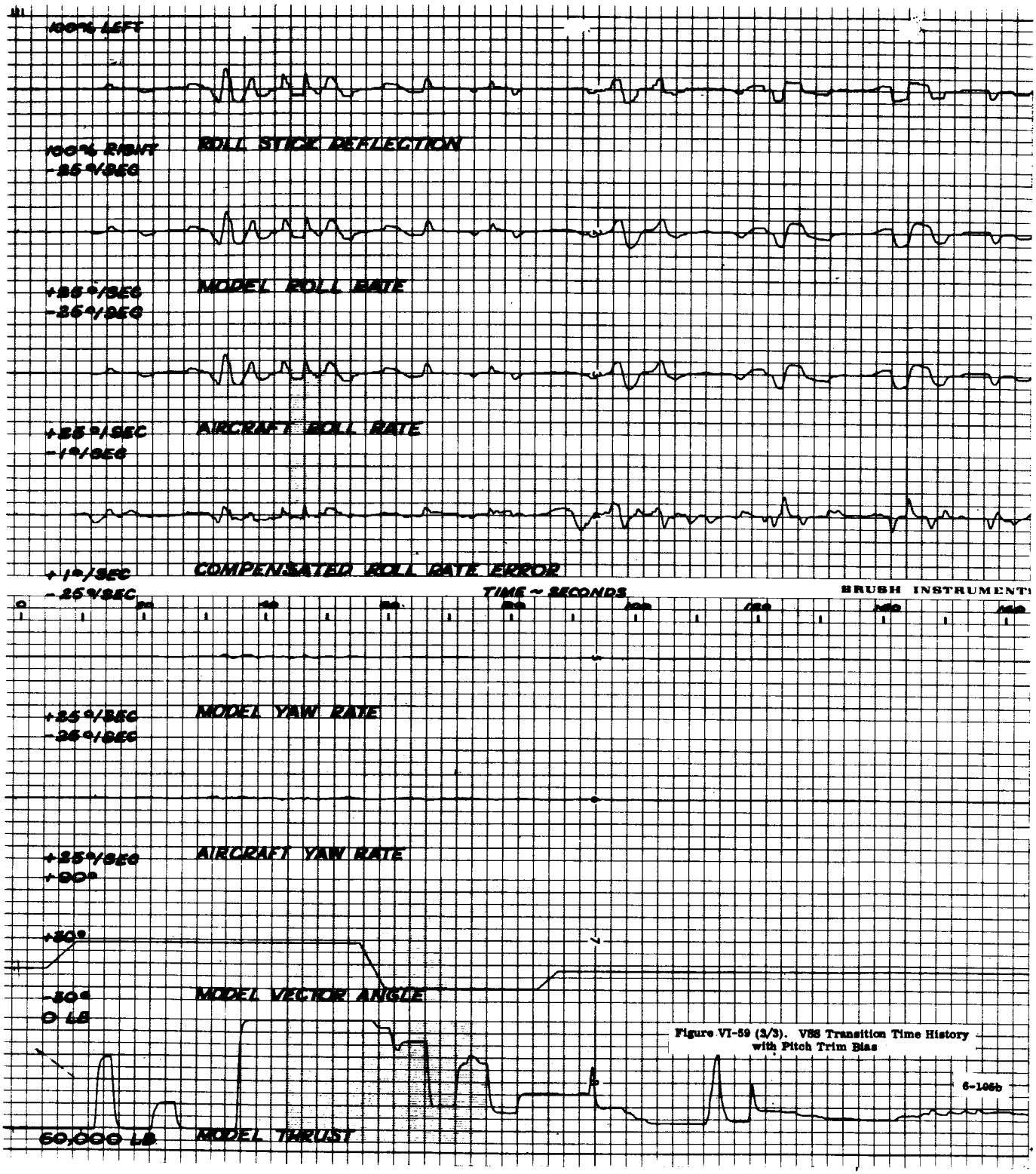


Figure VI-59 (3/3). VSS Transition Time History with Pitch Trim Bias

The model throttle (and thrust) is then decreased (1) in order to diminish altitude rate. It can be seen that the VSS causes a corresponding decrease in aircraft thrust. The tracking of the resulting model normal acceleration is good. It can also be seen that, during this period, the model angular rates are followed accurately.

During the subsequent transition phase the model thrust is vectored (2) in order to increase velocity. The VSS causes the aircraft thrust to be vectored in a similar manner until a limit is reached. In this period the control power demand increases until a peak value of approximately 70% of maximum available control power is reached. While axial acceleration tracking is good, some degradation is experienced because of thrust vector angle limiting (3). In this same time period the aircraft thrust level decreases in order to maintain the normal acceleration at a low level. Again rate tracking is good.

At the start of the conventional flight phase (4) the lift engines are cut. The VSS causes all controlled axes to track well. It is also noted that normal acceleration tracking is good even though this variable is not controlled during conventional flight. Some buildup of axial acceleration error is seen toward the end of the conventional flight phase. This error is due to the fact that, even though the VSS causes the aircraft engines to come to idle, sufficient negative axial acceleration is not produced, i.e., more drag force is required.

At the initiation of the retransition phase the lift engines are started (5). The VSS causes the aircraft thrust level to change in such a way as to null the initial normal acceleration which exists when control of this variable is again exerted. Note that the VSS logic is such that the thrust vector starts at the lift position.

The vector angle is now used to decrease aircraft velocity (6). Again, all controlled variables are tracking well. Once again, however, some axial acceleration error is caused by vector angle limiting.

During this retransition period the control demand builds (7) due to diminished control surface effectiveness as velocity decreases. The peak demand during this period is 35% of available control power and hence, still well within allowable limits.

During the final hover phase the thrust vector angle returns to the hover position (8). Note that the VSS causes the aircraft vector angle to follow the model angle. As with the initial hover phase all controlled variables track well.

(b) Low Pitch Rate Control Loop Gain

It can be seen that the time history presented in Figure VI-59 indicates that VSS performance is not drastically affected by using a lower pitch rate loop gain and a computed pitch trim bias.

It is currently felt that the differences in performance between these two alternatives do not warrant the introduction of the trim moment bias unless later, more detailed, study reveals that a relatively high pitch rate gain cannot be used.

4. Flexibility of Operation

a. Description of VSS and SAS Controls

The evaluation pilot has a VSS power-on and a VSS engage switch. The safety pilot has only a VSS disengage switch. If either pilot disengages VSS, the system automatically reverts to the SAS mode and control of the three rate loops reverts from the rate commands of the VSS computer to the pilot's stick and rudder. When VSS is disengaged, the SAS control of any loop can be removed as well by means of SAS engage switches which are provided for pitch, roll and yaw in each cockpit. For safety reasons, it is necessary that both pilots disengage their switches to remove SAS. For the proposed system, means must be provided to prevent engaging of the VSS mode if any of the SAS loops is disengaged. Referring to Figure VI-35 the SAS switch would break the loop between the summing junctions when disengaged. In the VSS mode, the stick input would also be removed from the loop, leaving no control. This dependence of the VSS operation on the SAS and switches can be avoided if separate control amplifiers are provided for each mode instead of a single amplifier with a gain switch as presently proposed. A final decision would require a more detailed study of both approaches.

b. Switching from SAS to VSS

Normally, switching between modes is controlled by the evaluation pilot and except for the restriction mentioned above, he is free to switch at will.

The possibility of excessive transients when switching from SAS to VSS exists if switching is attempted when the aircraft is not close to trim condition. This is because: the loop gains in VSS are increased thereby increasing the natural frequencies and reducing damping; the loop sees a step error command. This can be prevented either by pilot instruction or by providing logic which will prevent switching if any of the aircraft rates exceed some reference value.

c. Switching from VSS to SAS

In the case of switching from VSS to SAS, it is similarly advisable to switch when in steady flight. The possibility of having to switch during unsteady conditions in an emergency must be considered in this case, however, so provision of means for preventing switching back to SAS during unsteady flight is not practical. Fortunately, in switching to SAS, the loop gains are reduced and damping increased thereby tending to reduce the switching transients.

d. Wave-Off and Aborted Take-Off

No problems are anticipated with either of the above conditions by virtue of the large L/W ratio designed into the aircraft. During a normal let-down the rate of descent in the vicinity of the ground will normally be low enough to permit reversal before touch down. In the event of an engine failure during lift-off, Figure VI-49 indicates there is no problem in maintaining attitude control. At maximum power, with seven engines and bleed at 10%, the L/W is still in excess of one so there is no danger of losing altitude control. A safe abort can therefore be made. A potentially dangerous condition is if an engine failure occurs during let down at high sink rates. The combination of loss of lift, pilot response time and engine response time to full power command could be too much to overcome. Maximum permissible sink rates during let-down must therefore be established which take into account the possibility of engine failure.

SECTION VII
STRUCTURES
STRUCTURES - CONCEPT C

A. STRUCTURAL DESIGN

The structural diagram for the complete aircraft is shown in Figure VII-1. A conventional stressed skin design is provided which considers structure reliability, aircraft maintenance, accessibility to research payload, aircraft transportation,and ease of construction in a prototype shop environment. The principal details of the structure are described below.

Fuselage - The forward fuselage bending section consists of typical cockpit longerons, two side members and a keel member. The side members form the boundary for the cockpit floor and the keel member transitions into the lower caps of the nose wheel beams. The nose gear trunnion is located at the bulkhead aft of the safety pilot and between the nose wheel well beams. Bulkheads separate the engines and the fuel tanks and provide supporting structure for the front and rear wing spars and the stabilizing mount on the lift engines. A vertical beam located at the center of the airplane and extending from the aft bulkhead of the forward fuel tank to the forward bulkhead of the aft fuel tank is used to support the engine mounts and to separate the engines. The side skin and stringer panels extend from the lower longeron to the upper longeron. The bottom of the mid-fuselage section has access doors for fuel cell and engine access and removal. The upper section has the lift engine air inlet ducts, which are removable, and removable access panels for engine adjustments. Ram air start doors cover the air inlet ducts when in the cruise mode. The nacelles which house the lift cruise engines are integral with the fuselage and contribute to the primary structure of the aircraft. The aft fuselage has skin and stringer panels, upper and lower longerons and several primary frames which provide support for the fin and stabilator. The electronic equipment compartment is accessible through a large removable structural access panel on the bottom of the aft fuselage section.

Wing - The wing structure consists of two primary spars forming a box with stiffened skins, a root rib located at the fuselage side skin intersection, numerous primary and secondary ribs, and a special splice rib, for convenience of shipping, located outboard of the main landing gear support rib. The two main

spars pick up two fuselage bulkheads and become integral with that structure. An auxiliary spar provides a means for attachment of spoiler and aileron hinges and for the tracks for the Fowler flap, as well as an aft support point for the main landing gear trunnion. The forward support for the trunnion is in the main box. The flap and aileron is of spar, rib and light skin type construction. Structural access panels on the bottom side of the wing skin will provide access to attitude control ducts, (see Figure VII-2) and the hydraulic lines and actuators for the spoilers, aileron, and flap.

Stabilator - The stabilator structure is a multi-celled box beam consisting of a main structural box with leading and trailing edge boxes. The main box which provides most of the bending material is made of tapered 2024-T3 aluminum alloy skins stabilized by full depth webs. The leading and trailing edge boxes have thin skins supported by ribs. The right and left sides of the stabilator are interconnected by means of a steel torque tube which is supported by two hinges in the fuselage. The stabilator is hydraulically actuated through the steel torque tube. This design permits the manufacture of a stabilator half that can be used on both sides and is readily removable without disconnecting the actuators or controls.

Vertical Tail - The fin primary bending structure consists of front and rear spars with interspar tapered covers stabilized by full depth webs. Bending loads are transferred to the fuselage bulkheads through the front and rear spars only. Three main ribs transfer the rudder hinge loads into the fin.

B. LANDING GEAR DESIGN

The landing gear diagram is shown in Figure VII-3. This shows a tricycle configuration with a single 24 x 5.5 Type VII tire, wheel, and brake on each air-oil oleo type main gear. The nose gear (air-oil) has dual 16 x 4.4 Type VII tires and wheels.

The main gear (Figure VII-4) is mounted with the trunnion chord-wise in the wing and retracts inboard. A folding sidebrace with a latch locks the gear in the down position. A hydraulic actuator is utilized to release the down-lock latch

and retract the gear assembly which shrinks (shortens) during retraction into the wing wheel well. Individual hydraulically powered uplocks will be utilized to lock the gear assembly in the up (retracted) position. Doors and fairings attached to the fuselage, strut and wing enclose the gear assembly after retraction. Separate locks will be used for locking the doors in the up (retracted) position. Emergency provisions will be provided to release and extend the gears in the event of hydraulic system failure. Hydraulically actuated brakes are provided in each main wheel.

The nose gear (also shown in Figure VII-4) is mounted in the forward fuselage and retracts rearward under the aft cockpit floor. A folding drag brace incorporating a latch locks the gear in the down position. A hydraulic actuator releases the down lock, thereby permitting a retracting actuator to fold the drag brace, the motion of which moves the gear to the retracted position achieving shrinkage during retraction. A separate lock will be used to lock the gear in the up position. Doors attached to the fuselage enclose the gear assembly after retraction. Separate locks will be used for locking the doors in the up (retracted) position. Emergency provisions will be provided to release and extend the gear in the event of hydraulic system failure. The nose strut incorporates a hydraulically powered steering unit which will also provide damping. The design meets the landing sink speed requirements at a landing weight of 15,300 pounds.

C. MATERIALS

The materials used in the basic structure are principally aluminum alloys 2024 and 7075. Titanium 6AL-4V annealed is required in the engine bay for primary structure because of the high temperatures and the need for firewalls. The exhaust nozzle engine doors which are bathed in the lift engine exhaust gas are made from Inconel X steel. High load intensities and stiffness requirements to be met within a given structural envelope dictated the use of 4340 steel with 160-180 ksi heat treat for the horizontal tail carry through torque tube and with 260-280 ksi heat treat for the main gear shock strut. A summary list of components and the form and type of materials required are given in Table VII-1.

TABLE VII-1. MATERIALS LIST

Component	Item	Form of Construction	Material	Alloy	Form
Wing	Interspar Covers	Sheet - Stringer Panels	AL	2024 T3	Sheet
	Upper Spar Caps				
	Inboard	Extrusion	AL	2024 T3	Extrusion
	Outboard	Formed Angles	AL	2024 T3	Sheet
	Lower Spar Caps				
	Inboard	Extrusion	AL	2024 T3	Extrusion
	Outboard	Formed Angles			Sheet
	Spar - Fuselage Splice Fitting	Machining	Ti	6AL-4V	Bar
	Former Ribs	Formed	AL	2024 T42	Sheet
	Root Rib	Built-Up - Formed Parts	Ti	6AL-4V	Sheet
	L/G and Flap Support Ribs	Built-Up - Formed Parts	AL	7075 T6	Sheet
	Flaps & Ailerons	Sheet, Stringers Ribs	AL	2024 T3	Sheet
Horizontal Tail	Structural Box Covers	Machined Tapered Sheet	AL	2024 T3	Sheet
	Shear Webs	Stiffened Sheet	AL	2024 T3	Sheet
	L. E. & T. E. Skins	Sheet	AL	2024 T3	Sheet
	Ribs	Formed	AL	2024 T42	Sheet
	Torque Tube Fittings	Machining	AL	2024 T4	Bar
	Torque Tube	Machined Tube	Steel	4340 (160-180 KS1)	Extrusion

TABLE VII-1. MATERIALS LIST (Cont'd)

Component	Item	Form of Construction	Material	Alloy	Form
Vertical Tail	Structural Box Covers	Machined Tapered Sheet	AL	2024 T3	Sheet
	Front & Spars	Built-Up Section	AL	2024 T3 & T42	Sheet
	Shear Webs	Formed	AL	2024 T42	Sheet
	Root Fittings	Machining	AL	2024 T4	Bar
Fuselage	Blk'Hds/Frames (Engine Bay)	Built-Up	Ti	6AL-4V	Sheet
	Tail, Nose Gr. & Other Major Frames	Built-Up	AL	2024 T3 & T42 7075 T6	Sheet
	Former Frames	Formed	AL	2024 T42	Sheet
	Covering-F'wd	Stiffened Sheet	AL	2024 T3 & T42	Sheet
	Covering-Eng. Bay	Stiffened Sheet	Ti	6AL-4V	Sheet
	Covering-Aft	Stiffened Sheet	AL	2024 T3 & T42	Sheet
	Center Beam-Eng. Bay	Built-Up	Ti	6AL-4V	Sheet
	Longerons - Fwd.	Formed	AL	2024 T42	Sheet
	Longerons - Eng. Bay	Formed	Ti	6AL-4V	Sheet
	Longerons - Aft	Formed	AL	2024 T42 7075 T6	Sheet
	Lift Eng. Inlet Doors	Built-Up	AL	2024 T3 & T42	Sheet
	Lift Eng. Exhaust Doors	Built-Up	Steel	Inconel X	Sheet

TABLE VII-1. MATERIALS LIST (Cont'd)

Component	Item	Form of Construction	Material	Alloy	Form
Main Landing Gear	Strut, Side Brace and Shrink Strut	Machined and Welded	Steel	Principally 4340 (260-280 ksi)	Forging
Nose Landing Gear	Strut, Axle And Drag Brace	Machined	AL	Principally 7075 T73	Forging
V.T.O. Controls	Ducting	Welded Tubing	Steel & Titanium	18-8 Type 347 And	Tubing
	Nozzles	Welded Sheet And Machined Valves	Steel & Titanium	Ti 5AL-2.5Sn	Sheet And Plate
Cockpit	Windshield & Canopy	Molded	Stretched Acrylic		

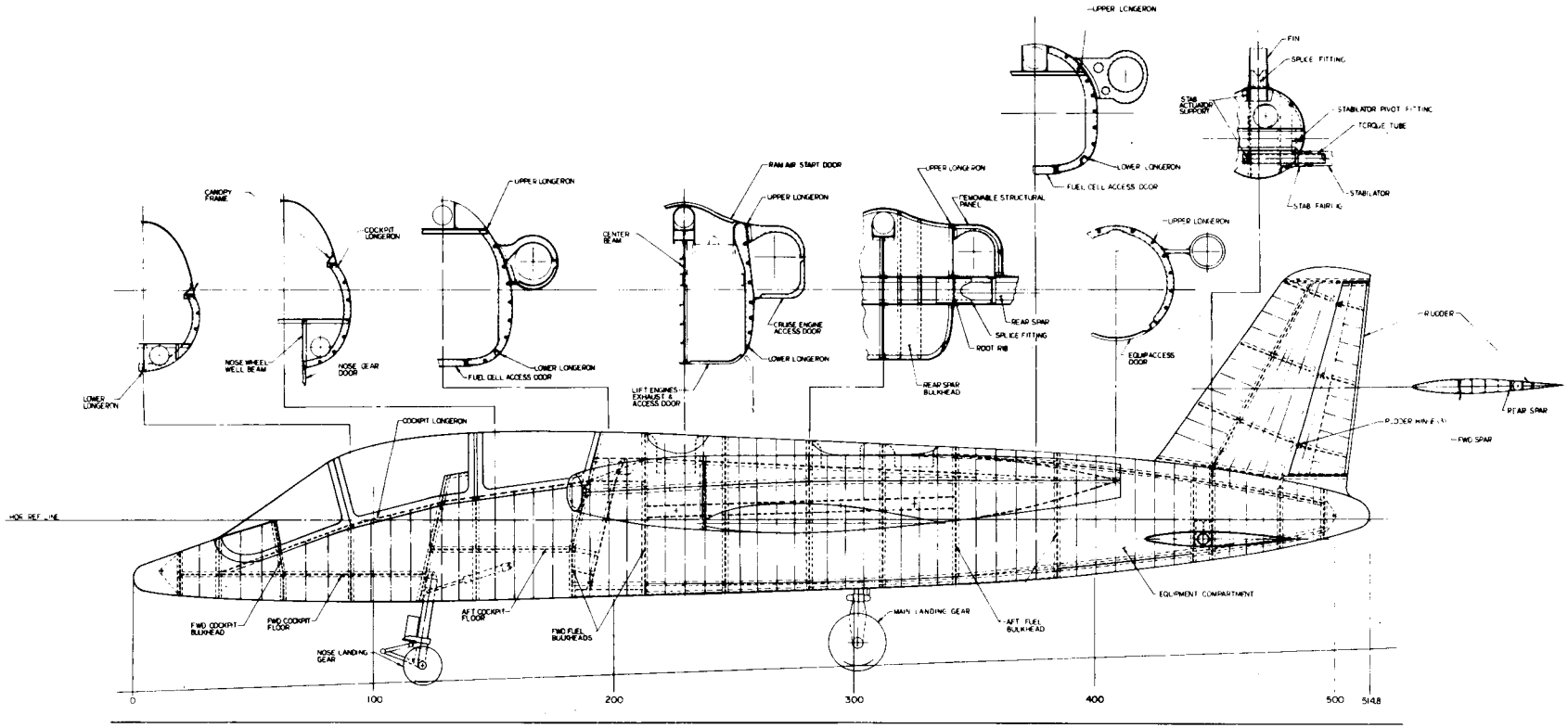


Figure VII-1, Sheet 1, Structural Diagram, Concept C

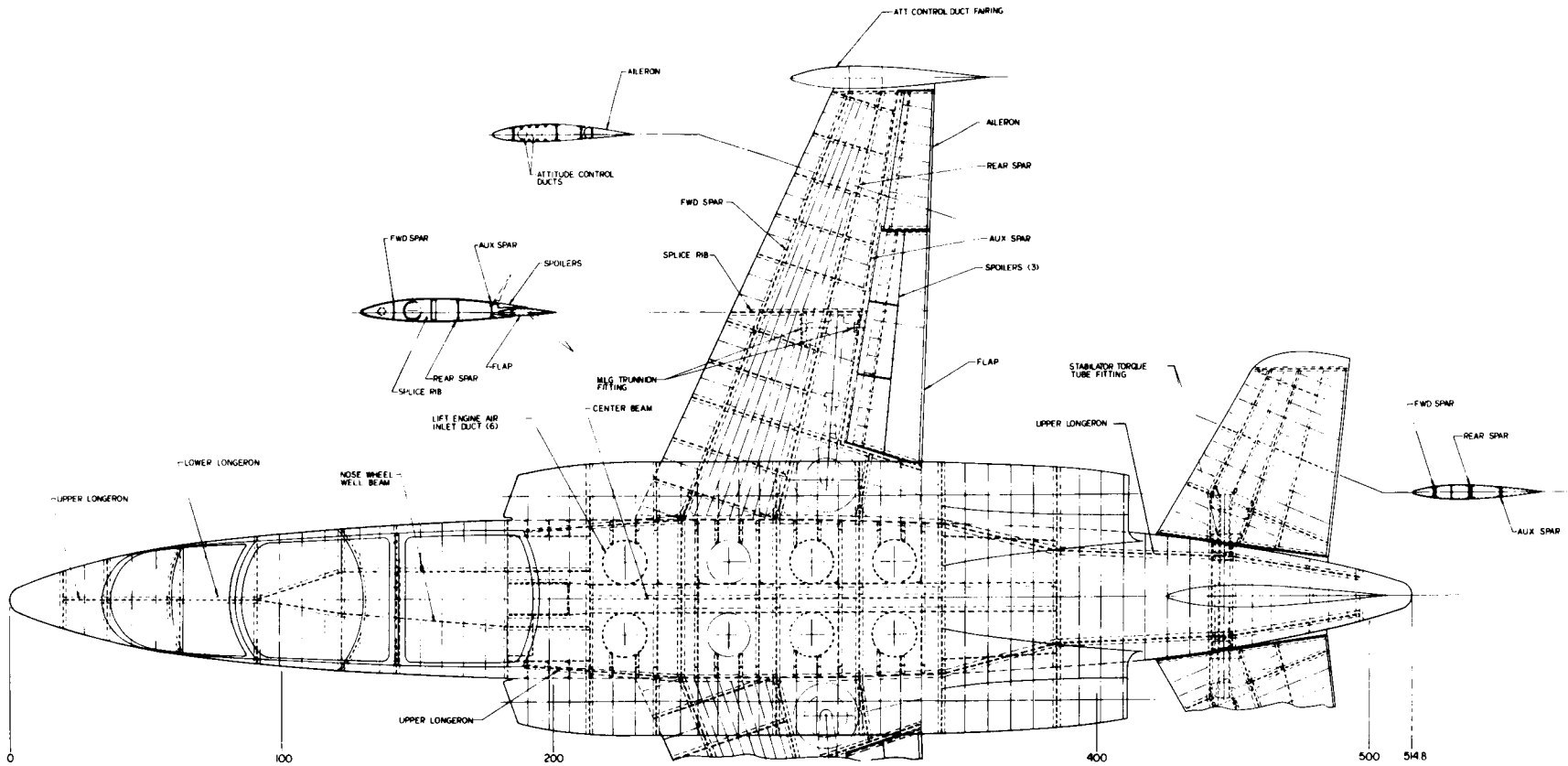
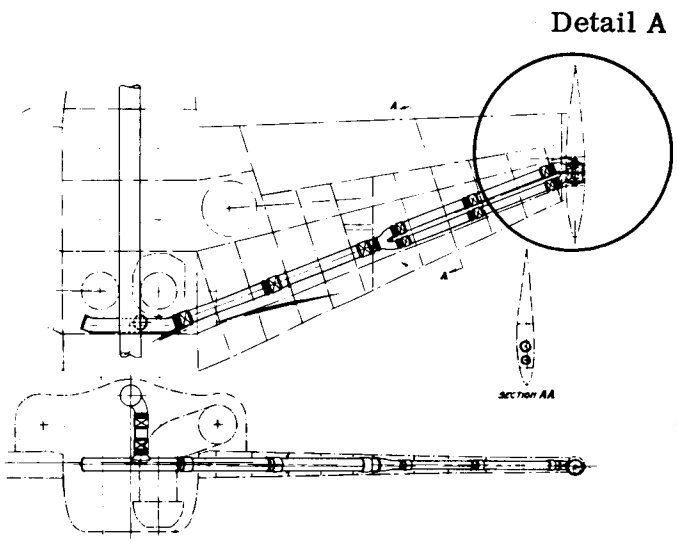


Figure VII-1, Sheet 2. Structural Diagram, Concept C.



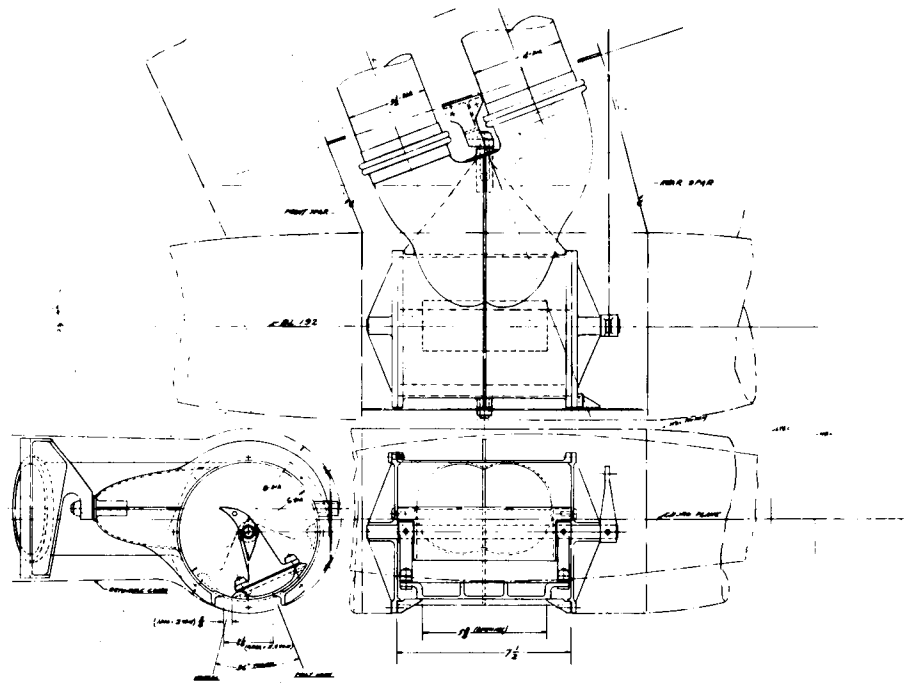


Figure VII-2. Wing V. T. O. Control Ducting and Tip Nozzle Detail

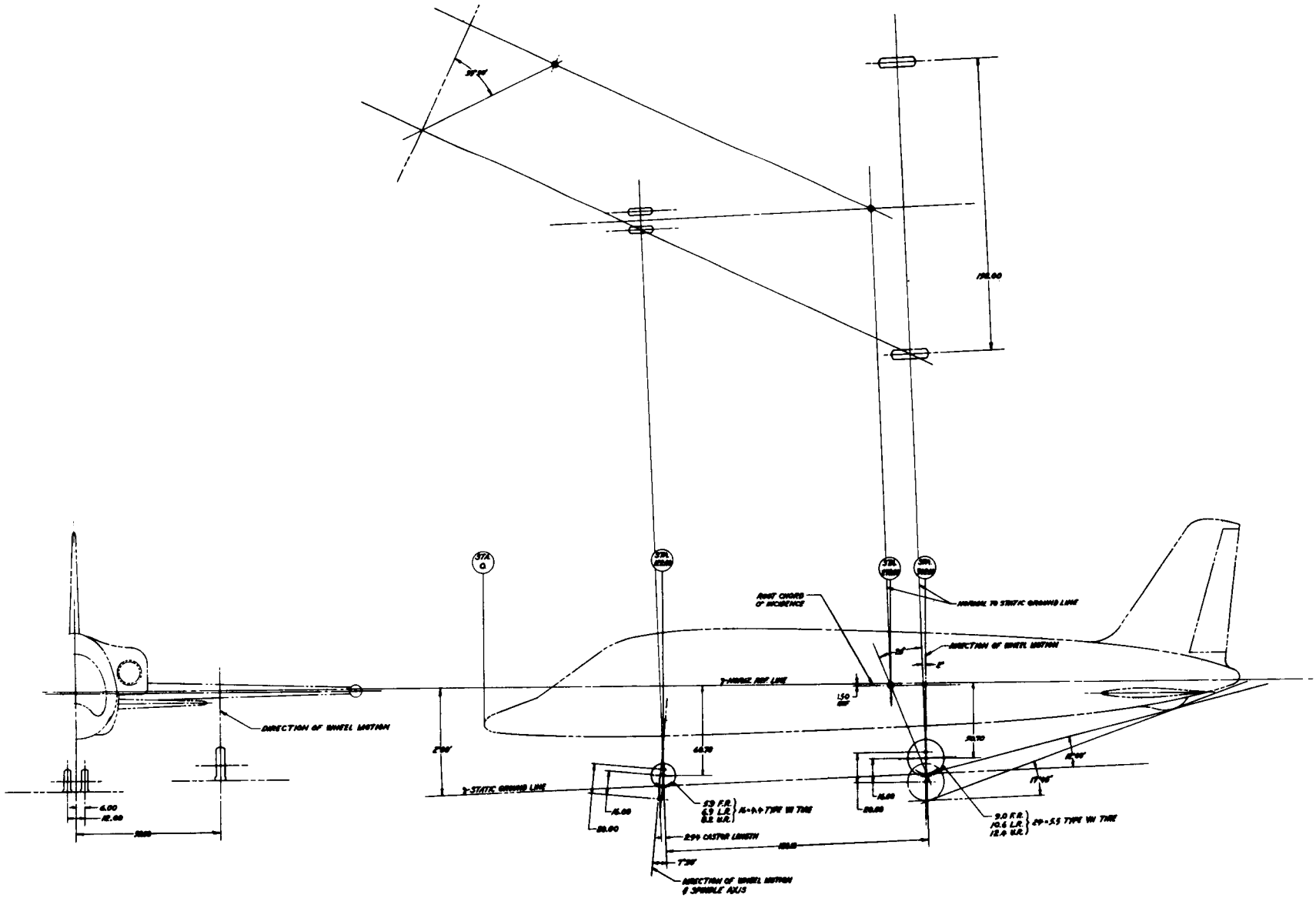


Figure VII-3. Landing Gear Diagram

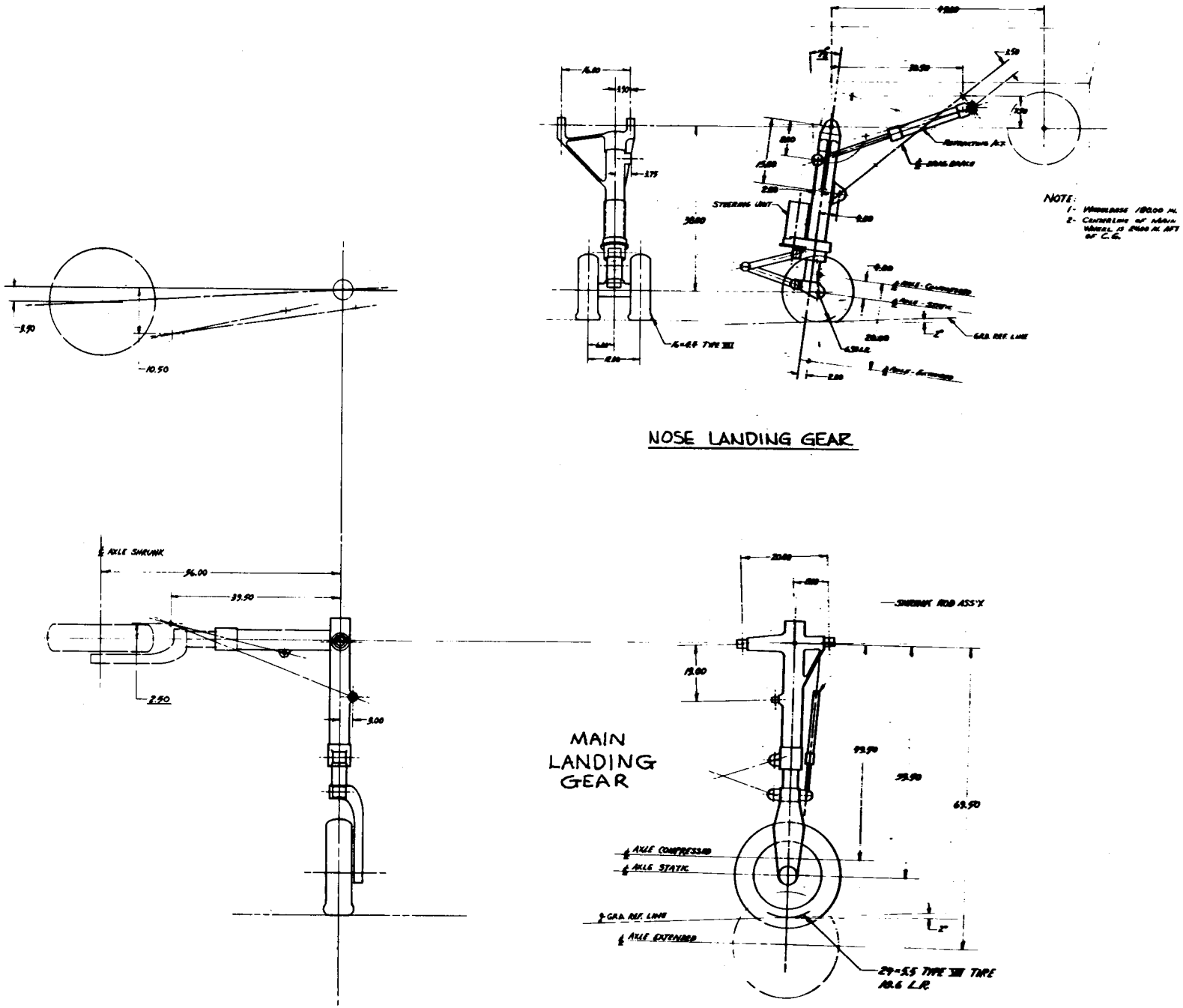


Figure VII-4. Main and Nose Gear Design

D. FLIGHT LOADS

1. General

The flight parameters considered for establishing design loads for the basic flight configuration are:

1) Basic design gross weight of 15,300 pounds
2) Limit positive load factor of 3.75
3) Limit negative load factor of -1.5; and a maximum speed of 0.7 M at S.L. (462 KCAS).

For the high-drag (flaps down) configuration, the flight parameters considered are:

1) Design gross weight of 15,300 pounds
2) Limit positive load factor of 2.0
3) Limit negative load factor of -0.5 and a maximum speed of 0.33M at S.L. (220 KCAS)

The design conditions were selected assuming no violent maneuvers would be initiated by the pilot, but that some might result from a system malfunction. MIL-A-8861 is used as a guide in establishing design loads.

The V-n envelope considered for deriving design loads is presented in Figure VII-5. It is evident from the V-n envelope that vertical gusts are not critical for design. Design flight conditions are selected from the maneuver envelope assuming the only violent maneuvers are a result of a system malfunction. The design airloads are derived assuming a rigid structure, since the effects of flexibility are estimated to be small for the flight regime of the aircraft.

2. Symmetric Conditions

Design loads for symmetrical flight conditions are established to conform to Paragraph 3.2 of MIL-A-8861. A design gross weight of 15,300 pounds is used in conjunction with load factors of +3.75 and -1.5 for the basic configuration of +2.0 and -0.5 for the high drag configuration (flaps down). A pitching acceleration of +5 rad/sec^2, which corresponds to an Air Force Class T of Table I in MIL-A-8861, is used to account for a system malfunction. Table VII-2 presents the design loads for the symmetrical flight conditions. Figures VII-6 and VII-7 present the spanwise airload distributions for the wing and horizontal tail, respectively.

3. Asymmetric Conditions

The vertical tail is a conventional fin-rudder combination. The design criteria were developed in accordance with MIL-A-8861 Specification, specifically, paragraphs 3.3.3 and 3.6 which cover sideslip and yawing maneuvers and horizontal gust, respectively.

The rudder induced maneuvers loads are given in Table VII-3 for the maximum speed design of .7M at sea level. The spanwise airload distribution is presented in Figure VII-8. Design conditions are selected to provide maximum torsion and maximum shear and bending on the overall fin. The conditions supplied are:

1) Maximum sideslip with maximum rudder of 10 degrees. The sideslip angle of this condition reflects a dynamic overswing factor of 1.6 from a steady state sideslip.
2) Steady state sideslip with rudder neutral.

To obtain the maximum gust load, sea level high speed is used. For horizontal gust, paragraph 3.5 of MIL-A-8861 applies. The derived vertical tail load for a 50 foot per second gust is 7600 pounds and is less critical than that derived for the rudder induced maneuvers.

In accordance with paragraph 3.3.3.7 of MIL-A-8861 an analysis was made which indicated sufficient rudder power in the event of power loss of one engine.

E. GROUND LOADS

Both vertical and conventional type landings are considered for design. The criteria used for establishing design loads for the vertical mode are:

1) Landing gross weight of 15,300 pounds
2) Sinking speed of 15 fps
3) Lift to weight ratio of 2/3
4) Side drift of 5 knots
5) Landing bank angle of 5 degrees MIL-A-8862 is used for conventional landings with a design sinking speed of 12 fps at a landing weight of 15,300 pounds

The strut stroke for both the main and nose gear is 20 inches. In the determination of main landing gear loads, the equation of energy also accounted for the structural deflection of the wing. A lateral drift of 5 knots is also combined with the maximum vertical sink speed condition. Obstruction loads encountered during vertical landing are based on paragraphs 3.4.5.1 and 3.4.5.2 of MIL-A-8698. Lateral loads encountered from a 10 degree bank attitude during vertical landing are also considered.

Landing loads for conventional type landings are derived as per MIL-A-8862 for a gross weight of 15,300 pounds at a sinking speed of 12 fps with a lift to weight ratio of 1.0.

Table VII-4 presents the main and nose gear loads and Table VII-5 summarizes the maximum normal load factors.

TABLE VII-2. SYMMETRIC FLIGHT CONDITIONS

G.W. = 15,300 pounds

Condition Number	1	2	3	4	5	6	7
Description	Positive	Balanced Negative	Negative Flaps Dn.	Flaps Up	Flaps Up	Dynamic	Flaps Dn.
Mach No.	.7	.7	.33	.4	.7	.7	.33
Altitude	S. L.	S. L.	S. L.	S. L.	S. L.	S. L.	S. L.
Load Factor	3.75	-1.5	-.59	3.75	1.88	-1.0	0
Pitching Accel. Rad/Sec^2	-	-	-	-5	5	5	5
Total Exposed Wing Load - lb	44650	-17850	1130	34600	32200	-2050	17300
C.P. Fract. of Theor. Wing Root Chord	.544	.544	4.46	.544	.544	.544	.794
Body Load Due to Wing - lb.	9520	-3810	-5080	7350	6900	-440	-1570
C.P. Fract. of Theor. Wing Root Chord	.012	.012	-.026	-.026	.012	.012	-.026
Exposed Horizon. Tail Load - lb.	2530	-1020	-4000	12120	-8120	-10060	-12340
C.P. Fract. of Theor. Hor. Tail Chord	.564	.564	.564	.564	.564	.564	.564
Body Load Due to Hor. Tail-lb.	680	-270	-1100	3310	-2220	-2750	-3390
C.P. Fract. of Theor. Hor. Tail Chord	.414	.414	.414	.414	.414	.414	.414
Wing Load Due to Angle of Attack	-	-	-23600	-	-	-	-7430
C.P. Fract. of Theor. Wing Root Chord	-	-	.544	-	-	-	.544
Wing Load Due to Flap Deflection	-	-	24730	-	-	-	24730
C.P. Fract. of Theor. Wing Root Chord	-	-	.718	-	-	-	.718

TABLE VII-3. ASYMMETRIC FLIGHT CONDITIONS VERTICAL TAIL LOADS

G.W. = 15,300 pounds

Condition Number	8	9
Description	Steady State Sideslip	Maximum Sideslip With Maximum Rudder Deflection
Mach No.	.7	.7
Altitude	S. L.	S. L.
Load Factor	1.0	1.0
Sideslip Angle (Deg)	9.5	15.2
Rudder Deflection (Deg)	0	10
Fin Load Due to Sideslip - lb.	-11300	-18000
C.P. Fraction of Exposed Fin Root Chord	.48	.48
Fin Load Due to Rudder Deflection - lb.	0	5000
C.P. Fraction of Exposed Fin Root Chord	-	.705
Total Fin Load - lb.	-11300	-13000
C.P. Fraction of Exposed Fin Root Chord	.48	.395

TABLE VII-4. LANDING LOADS

G.W. = 15,300 pounds

Condition Number	Mode	Description	Sink Rate (ft/sec)	Lift/ Weight	Main Gear Loads V (lb.)	Main Gear Loads D (lb.)	Main Gear Loads S (lb.)	Nose Gear Loads V (lb.)	Nose Gear Loads D (lb.)	Nose Gear Loads S (lb.)
11	Vertical	2-Point	15	2/3	18000	-	-	-	-	-
12	Vertical	3-Point	15	2/3	15150	-	-	5700	-	-
13	Conventional	Spin-up*	12	1	10300	-7350	-	3350	-1830	-
14		Spring-back*	12	1	10100	7410	-	-	2350	-
15	Vertical	5 Kn Drift	15	2/3	18000	-	+12600 -9000	5700	-	±3990
16	Conventional	Drift	-	1	9000	-	-5400 +7200	-	-	-
17	Vertical	Obstruction	15	2/3	18000	±9000	-	5700	±3990	-
					18000	-	±9000	5700	-	±3990
18	Vertical	10 Degree Bank	10	2/3	18000	-	+3540	-	-	-

* Loads perpendicular and parallel to strut

Sign Convention

V - Positive up, perpendicular to ground

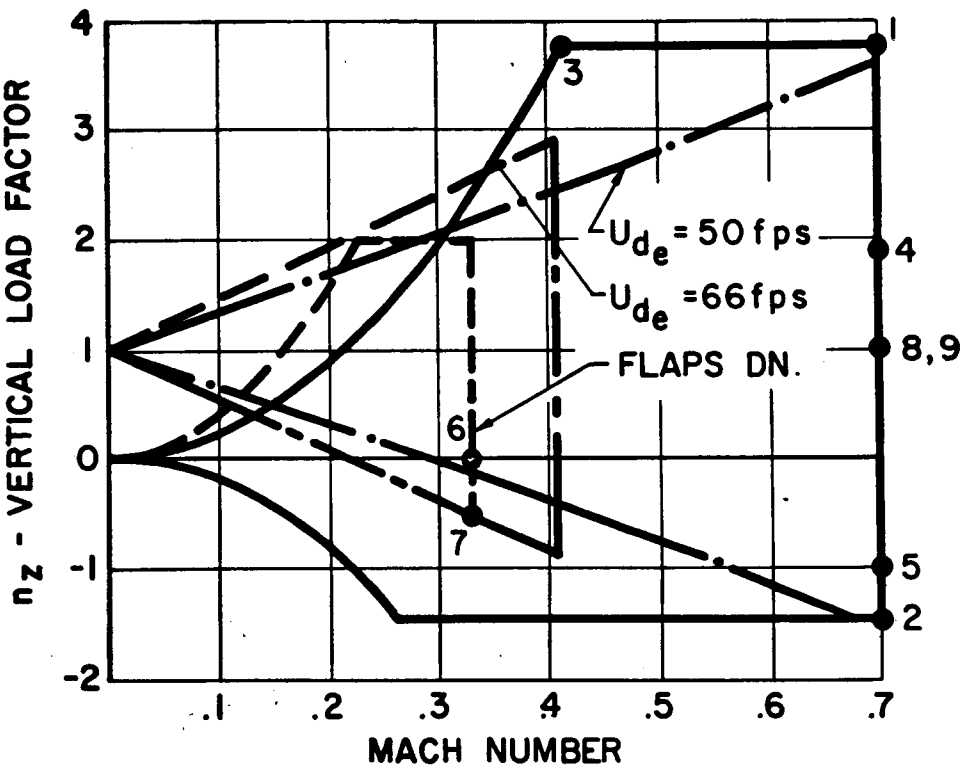
D - Positive forward, parallel to ground

S - Positive inboard, parallel to ground

TABLE VII-5. LANDING LOAD FACTORS

G.W = 15,300 Lbs					
	Type of Landing	Sink Speed (fps)	Lift Load Factor	Ground Reaction Load Factor	Total Load Factor
Main Gear	Vertical 2 Pt.	15	.67	2.36	3.03
Stroke = 20 in.	Conventional 2 Pt.	12	1.0	1.33	2.33

V-n DIAGRAM-CONCEPT C



CONDITION	DESCRIPTION
1.	POSITIVE BALANCED MANEUVER
2.	NEGATIVE BALANCED MANEUVER
3.	PULLUP MANEUVER RECOVERY
4.	PULLUP MANEUVER
5.	PUSHDOWN MANEUVER RECOVERY
6.	RECOVERY FROM MINIMUM FLAPS-DOWN MANEUVER
7.	NEGATIVE GUSTS WITH FLAPS-DOWN
8.	STEADY SIDESLIP MANEUVER
9.	MAXIMUM SIDESLIP MANEUVER

Figure VII-5

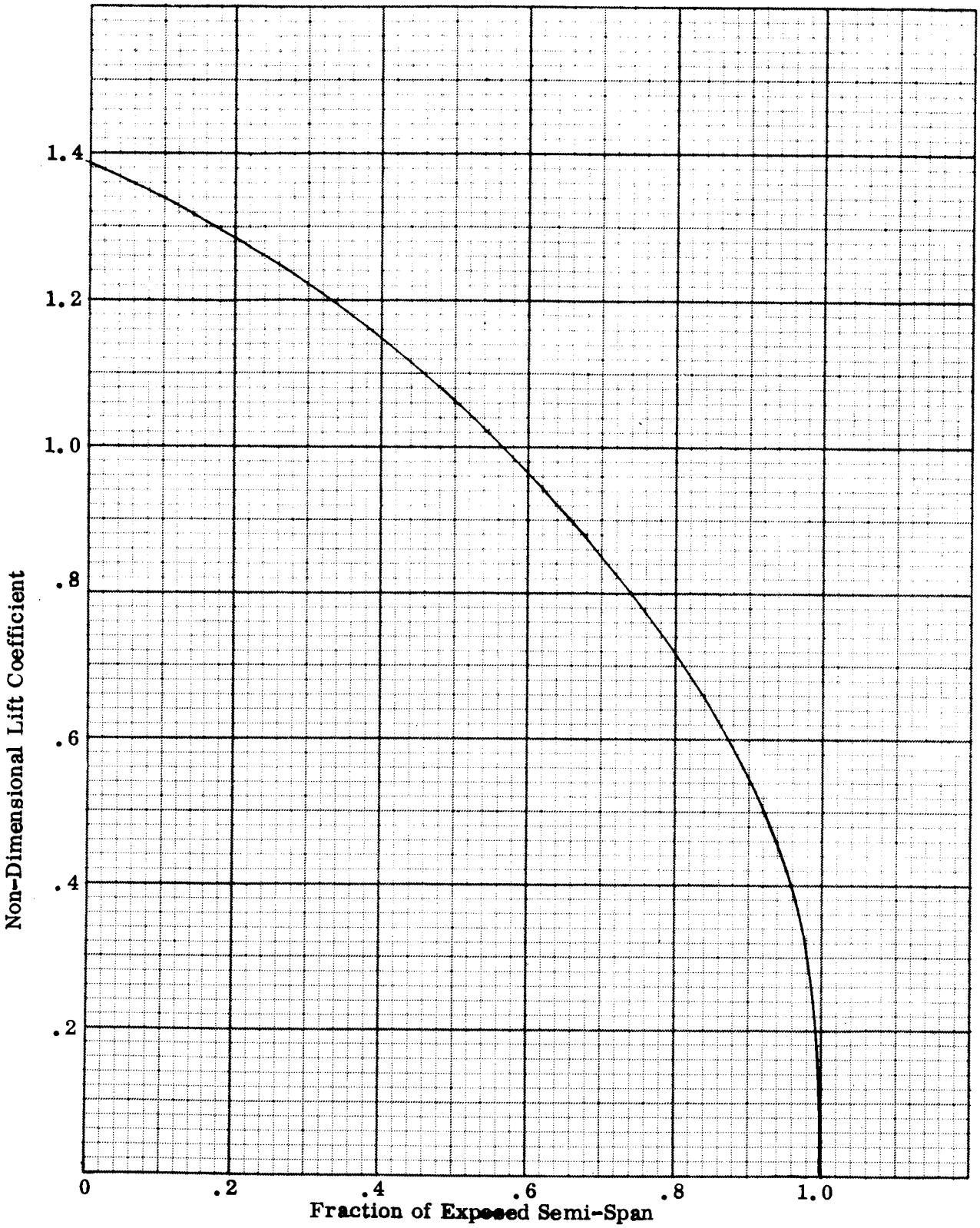


Figure VII-6. NASA V/STOL Concept C Wing-Nondimensional Span Loading

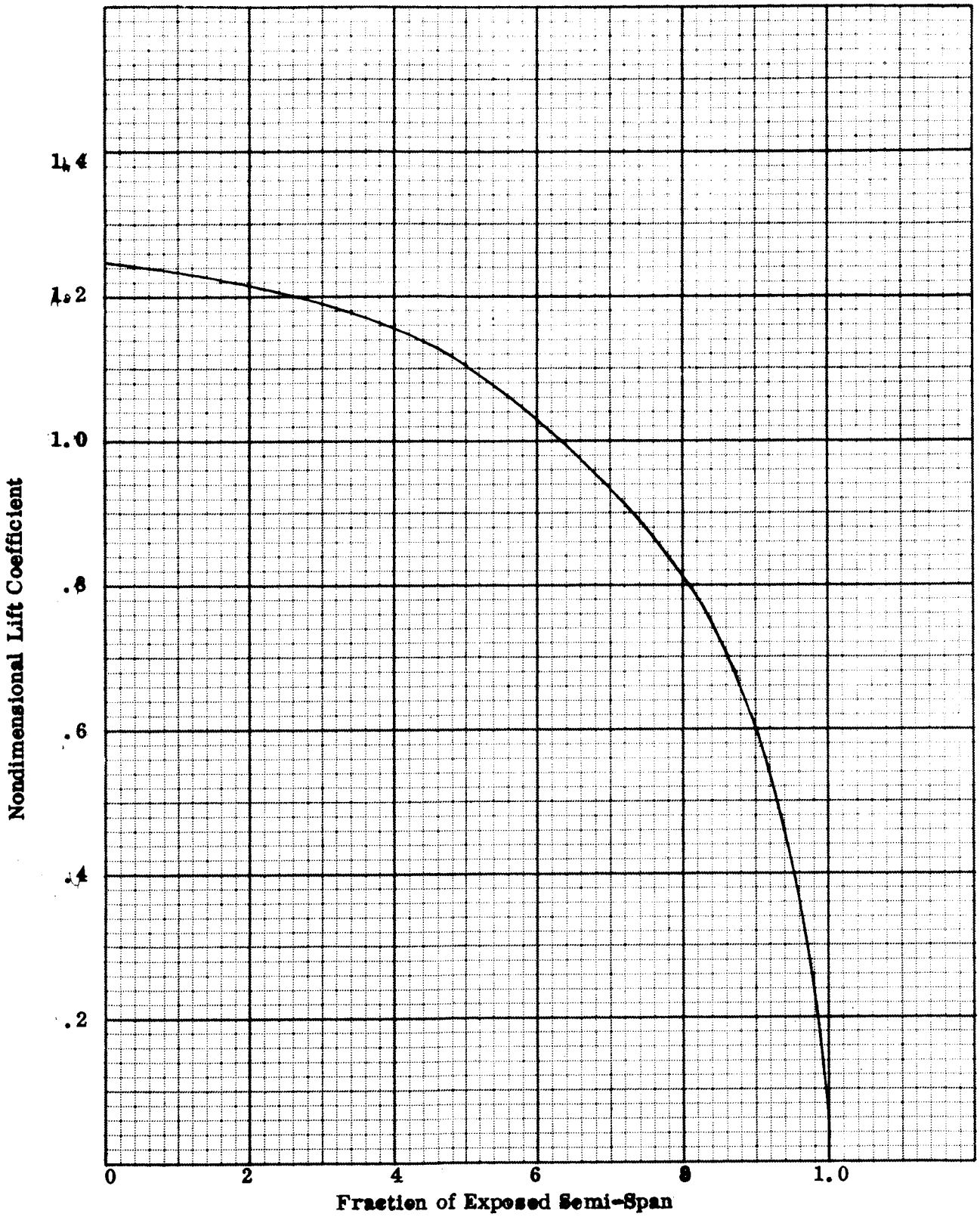


Figure VII-7. NASA V/STOL Concept C Horizontal Tail - Span Loading

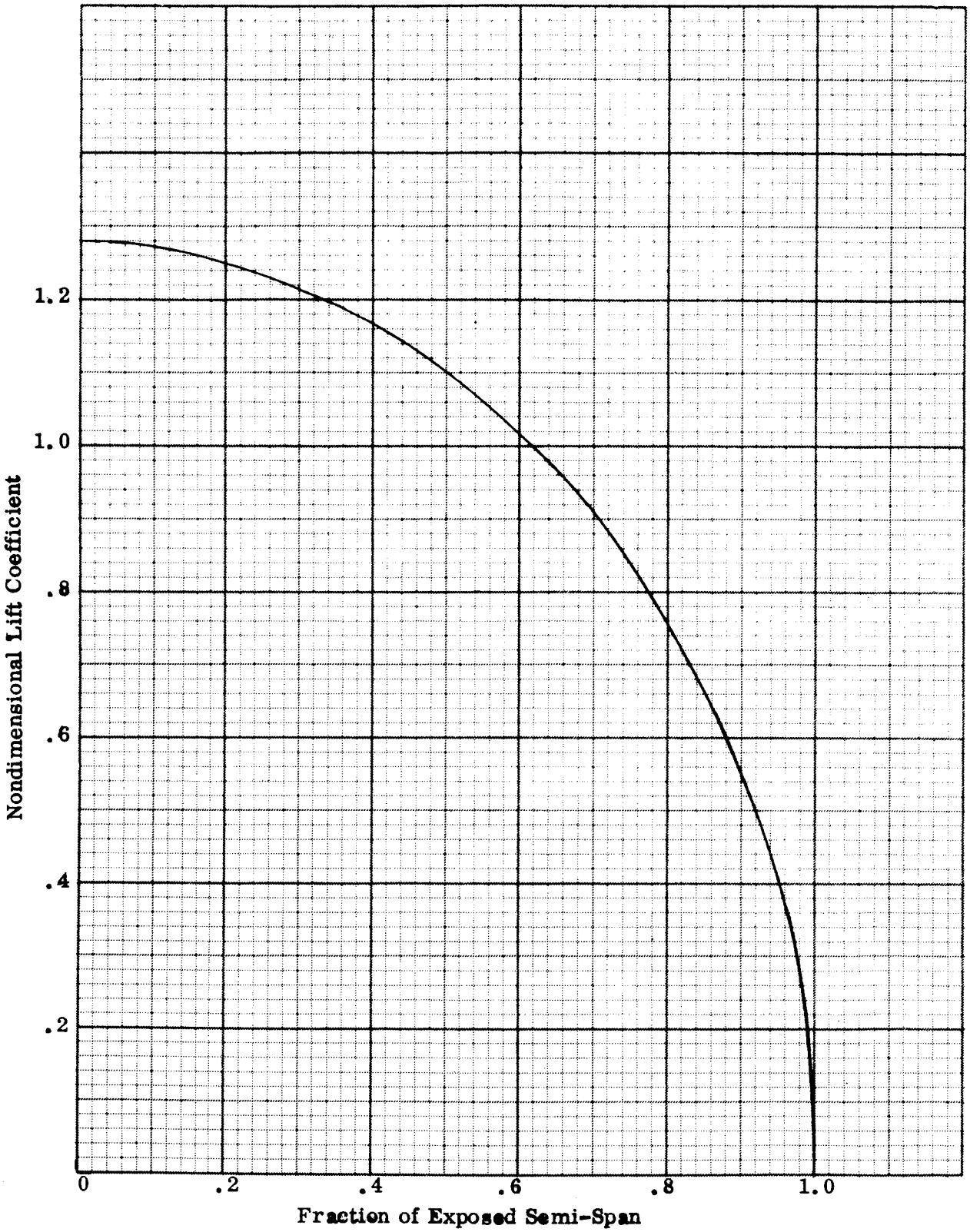


Figure VII-8. NASA V/STOL Concept C Vertical Tail Nondimensional Span Loading

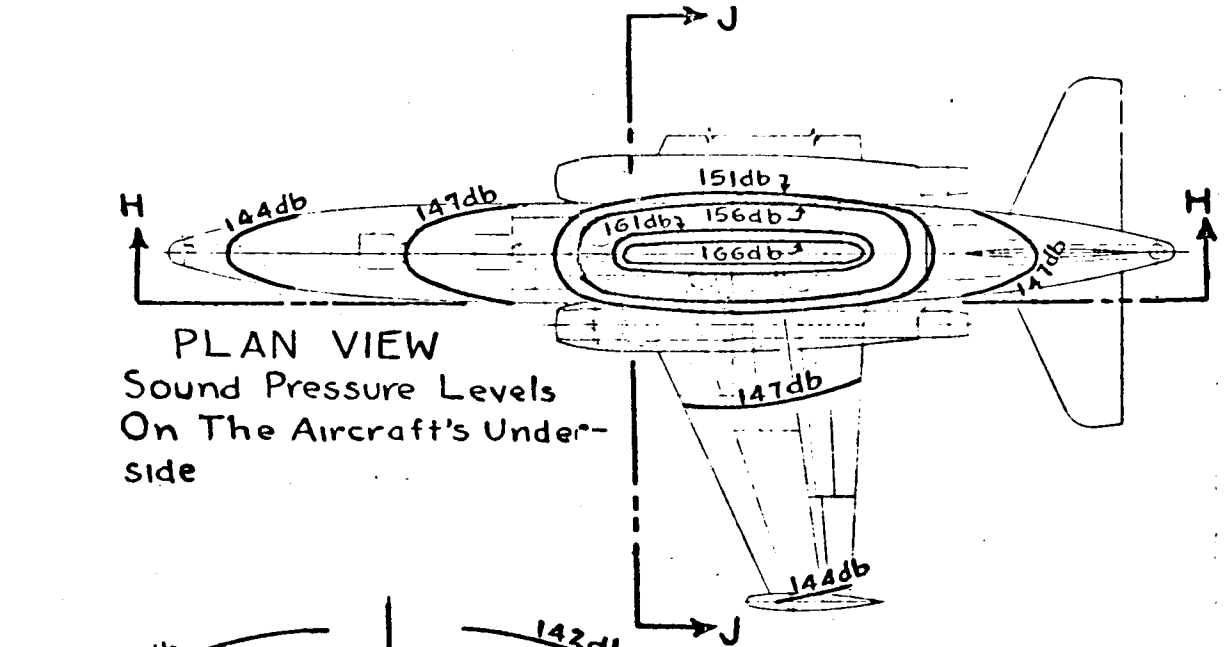
F. ACOUSTIC ENVIRONMENT

The most severe acoustic noise environment for many regions of the airplane will occur for a few seconds during lift-off and landing in the VTOL mode. The noise generated by the turbulent mixing of the jet exhaust gases with the surrounding air was predicted for a single J-85 jet based on the methods given in Reference 1. Estimates for the sound pressure levels for eight J-85 engines exhausting vertically downward were increased by 5 db to account for ground reflection effects. The predicted near field sound pressure level contours for vertical lift-off are shown in Figure VII-9.

When two of the J-85 engines are exhausted to the rear during STOL operation or cruise, the acoustic noise environment in the aft section of the airplane will be more severe than during VTOL operation. The predicted near field sound pressure level contours for the cruise condition are shown in Figure VII-10.

The frequency spectrum of the external noise field varies with location. For the region near the cockpit, the frequency spectrum in octave bands corresponding to an overall level of 147 db is shown in Figure VII-11. The estimated internal cockpit noise when the cockpit is enclosed by a canopy of 5/16 inch thick Plexiglas and a fuselage wall of 0.05 inch thick dural is shown in Figure VII-11 to exceed the requirements of MIL-A-8806. To meet the requirement of this specification for the low frequency noise range requires the use of 0.60 inch thick Plexiglas with 0.25 inch thick dural fuselage. The high frequency noise level can be alleviated by the addition of sound absorption material to reduce reverberation effects. Figure VII-12 shows that a vertical takeoff with 80% engine thrust gives similar external and internal noise levels but reduced by 5 db.

Figure VII-13 shows that the hover condition using 80% engine thrust gives similar noise levels but reduced 10 db from the 100% thrust lift-off condition. 5 db reduction in noise level is due to the lower (80%) engine thrust, while the remaining 5 db reduction is due to the absence of ground reflection effects when the airplane is more than 20 feet off the ground. For this condition the cockpit enclosure noise approximately meets the requirements of MIL-A-8806 for a conventional designed cockpit and canopy.



PLAN VIEW
Sound Pressure Levels
On The Aircraft's Underside

NOTES:

1- SPL Contours Shown Pertain To 100% Engine Thrust.

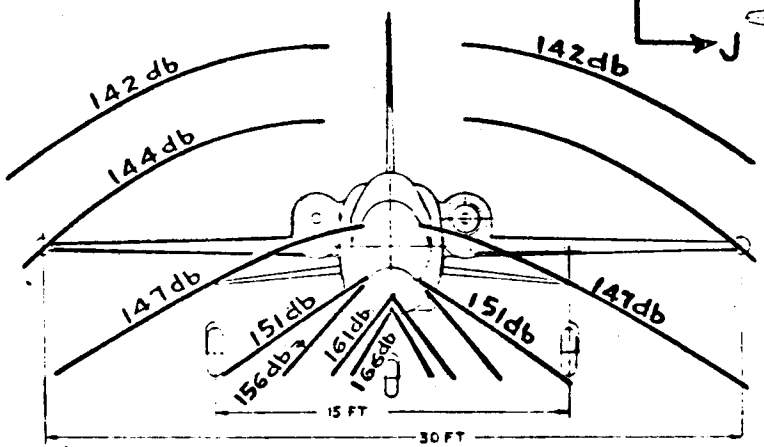
2- SPL Numerical Values Include 5db For Reflection From The Ground.

3- For Heights Where Ground Reflection Effect Is Negligible, SPL's Will Be 5db Less Than The Values Shown.

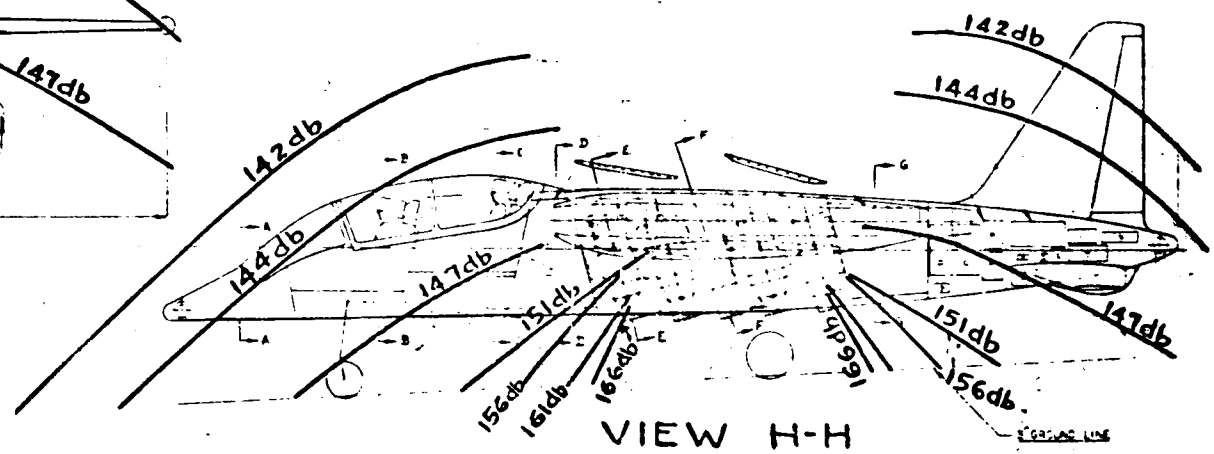
4- SPL Numerical Values Shown Do Not Include Pressure Augmentation Due To Surface Reflection. 4db Should Be Added For Essentially Flat Surfaces And 2db For Curved Surfaces.

5- At 80% Engine Thrust, SPL Numerical Values Are 5db Less Than For 100% Thrust.

6 - Drg shows ten engines values based on eight engines



VIEW J-J
Rotated 90°



VIEW H-H

Figure VII-9 Vertical Lift Off Condition - Near Field Sound Pressure Levels

VERTICAL LIFT-OFF CONDITION
NEAR FIELD SOUND PRESSURE LEVELS

Notes:

1. SPL contours shown pertain to 100% engine thrust.
2. SPL numerical values do not include pressure increase due to surface reflection. 4 db should be added for essentially flat surfaces and 2 db for curved surfaces.

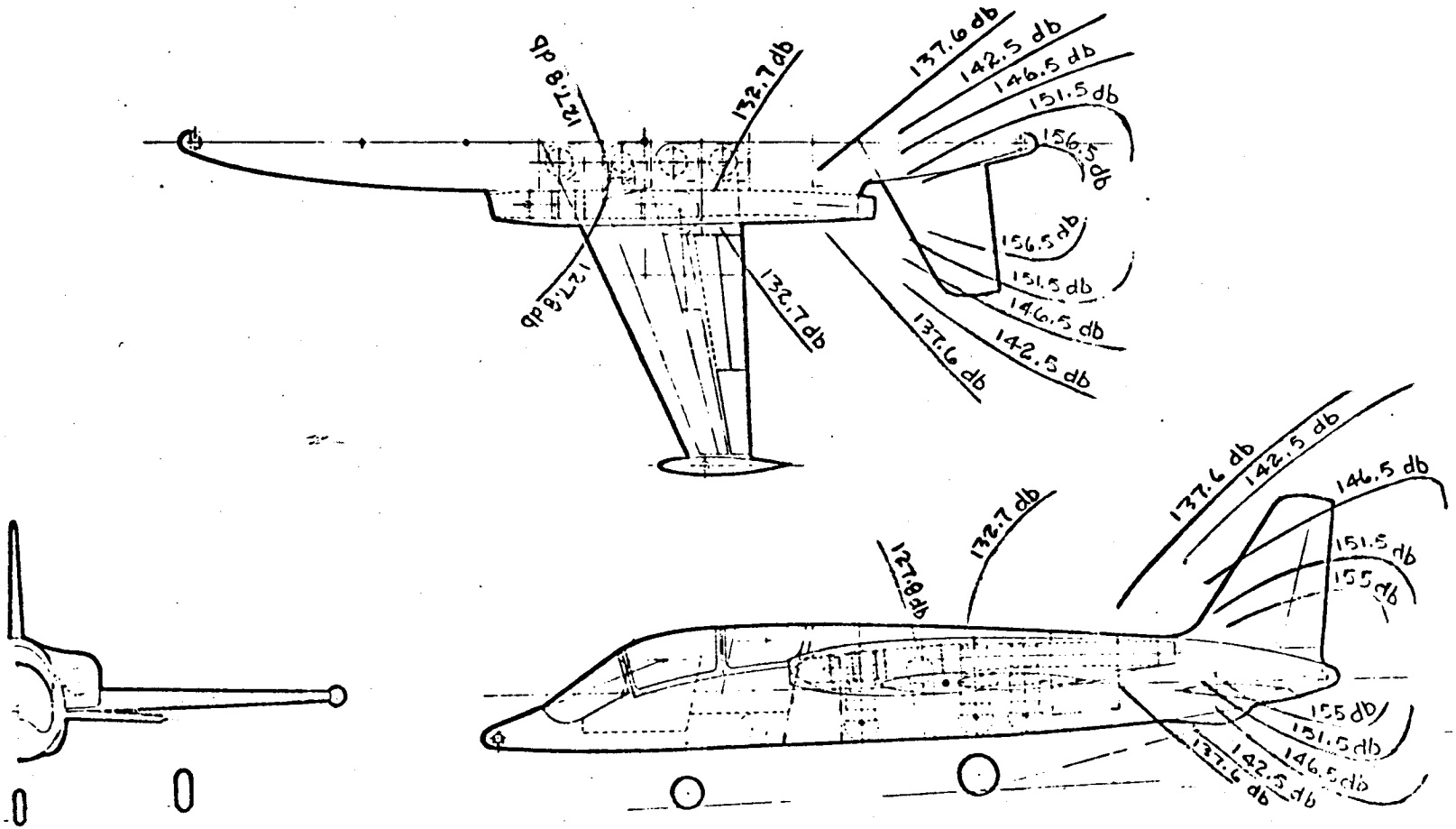


Figure VII-10
Cruise Condition Near Field Sound Pressure Levels

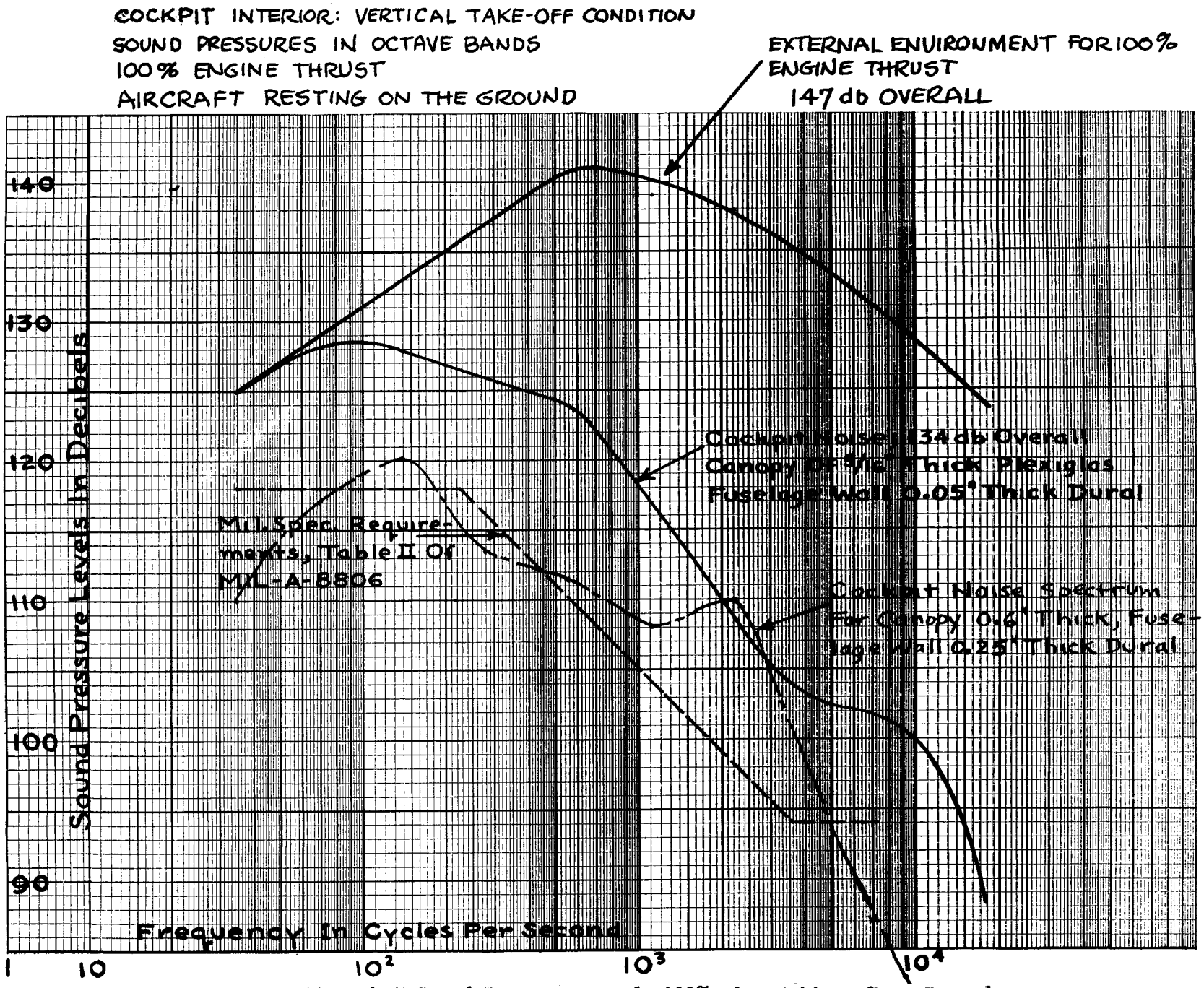


Figure VII-11 Cockpit Sound Pressure Levels 100% Thrust Aircraft on Ground

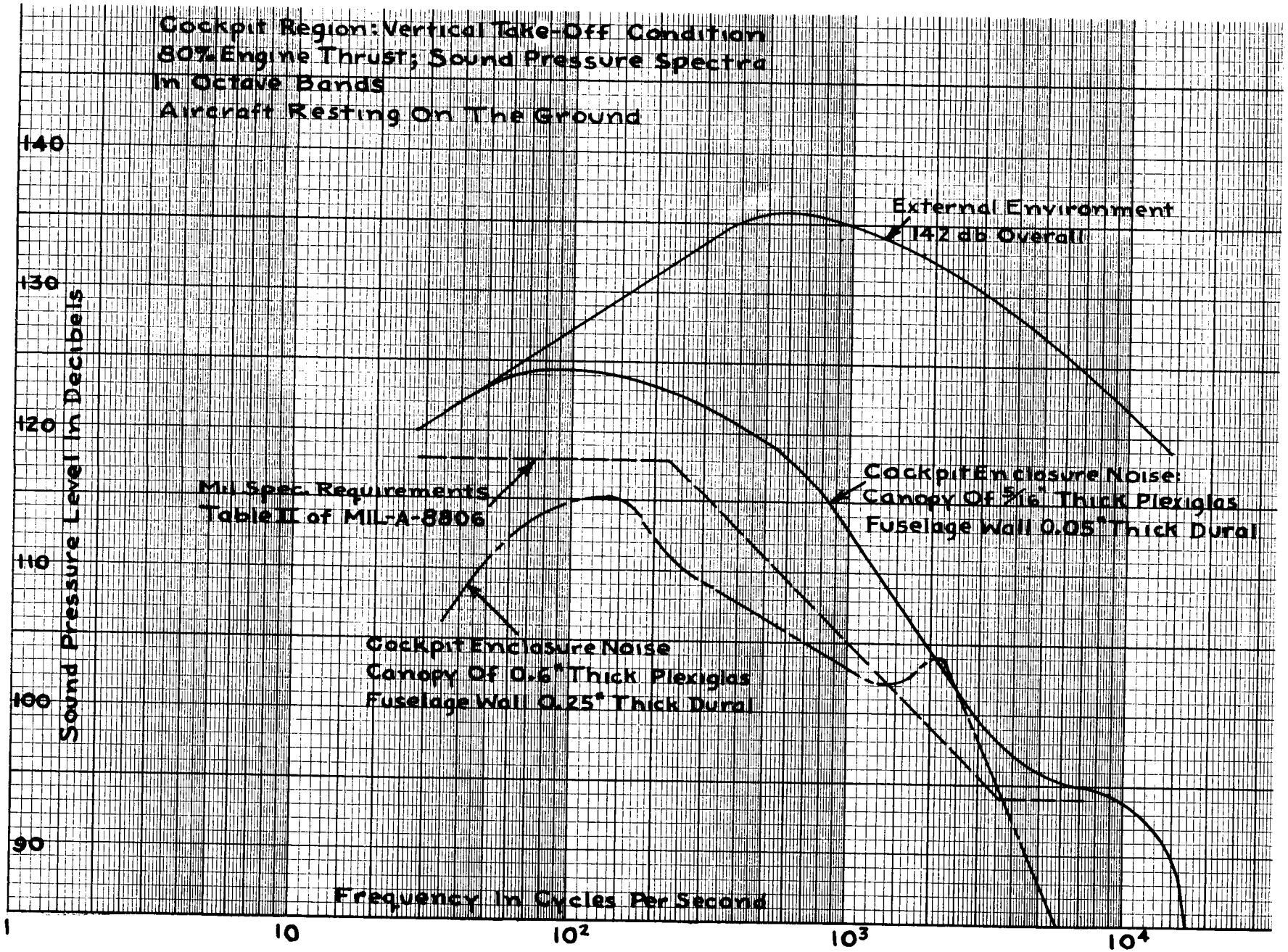


Figure VII-12. Cockpit Sound Pressure Levels 80% Thrust - Aircraft on Ground

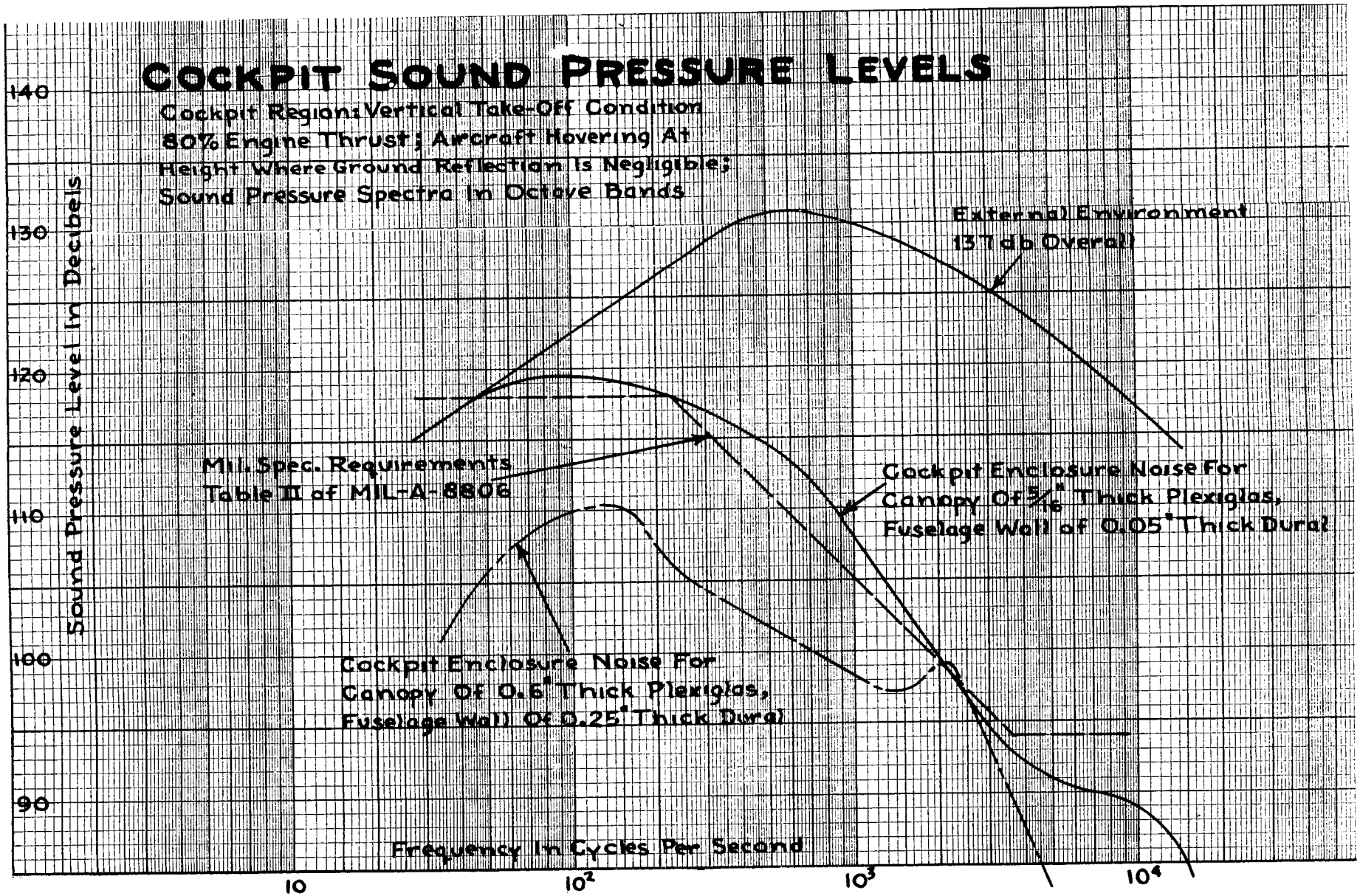


Figure VII-13. Cockpit Sound Pressure Levels
80% Thrust-Aircraft Hovering Above Ground Influence

G. STRUCTURAL TEMPERATURES

Structural temperatures arise internally from hot engine and tailpipe surfaces and externally from direct and recirculated engine exhaust gases. The derivation of the structural temperatures arising from these conditions was given in Section IV.

For design purposes the temperature distribution shown in Figure VII-14 was determined for the structural analysis of the engine bay section. These are the average steady state temperatures occurring in out-of-ground effect hover. A steel stand-off heat shield limits the structural temperature of the bottom fuselage skin between the engine exhausts to 500°F. For simplicity and expediency at this stage of design, these temperatures were assumed to be constant throughout the length of the engine bay.

The external skin temperatures were less than the equilibrium temperatures for out-of-ground hover for an assumed dwell period of 15 seconds on the ground at maximum throttle and exhaust directed vertically.

Engine nozzle exhaust doors are bathed directly in the engine exhaust gas and experience temperatures up to 1200°F.

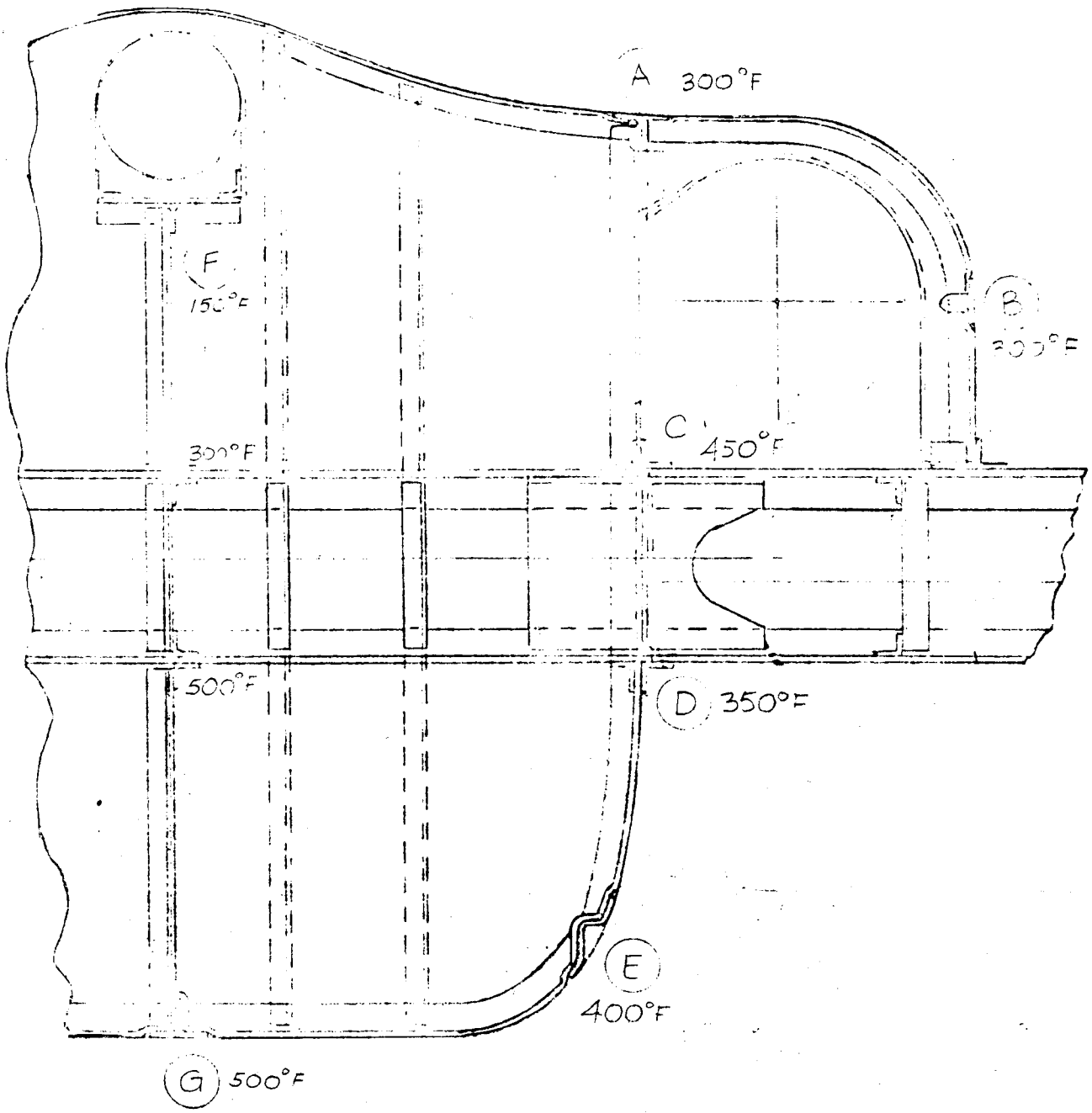


Figure VII-14. Structural Design Temperatures - Engine Bay

H. WING ANALYSIS

The primary structural load path consists of two spars, the upper and lower covers and the leading edge skin. The front and rear spars are located at 16.1 and 49.4 percent of chord respectively. A third, auxiliary spar located at 65 percent of chord, is used to introduce aileron and flap loads into the primary structure. The auxiliary spar is supported at B.L. 104.75 by a full depth chordwise rib, incorporated integrally with the primary structure. Additional support is provided inboard of B.L. 104.75 at B.L. 56.3. At this location a diagonal rib transfers auxiliary spar shear to the primary wing structure and the fuselage bulkhead support at F.S. 312.00. The auxiliary spar is also supported between these ribs by partial depth ribs which are attached between this spar and the rear spar. These ribs also provided support for the wheel well upper cover.

The upper and lower covers of the primary wing structure consist of hat stiffened skin, reinforced transversely by ribs spaced at approximately 17 inches. The rib angularity is made normal to a mean line drawn between the front and rear spars.

A wing splice is made at the full depth chordwise rib at B.L. 104.75.

The primary wing structure outboard of the wing splice is a three cell, three spar box. The upper and lower covers are hat stiffened between the front and rear spars, with ribs transversely reinforcing the covers. In addition, these ribs provide a shear connection at the auxiliary spar for distributing aileron load to the structural box.

The main landing gear forward trunnion support is accomplished at B.L. 99.00 by means of a full depth intercostal supported between the chordwise rib at B.L. 104.75 and a partial span rib existing in the primary box. The aft trunnion attachment is supported primarily by the rib at B.L. 104.75, with some support afforded by the auxiliary spar.

The wing structure was analyzed for bending and torsion as a single cell box beam. The upper and lower covers consisting of the spar caps and the hat

stiffened skins were designed for compression to fail in the primary column mode. The spar webs were analyzed as supporting shear due to torsion and full vertical shear; diagonal tension being utilized.

All of the symmetric flight and landing conditions given in Paragraphs D and E, respectively, were investigated in the design of the wing. Flight Conditions 1, 4 and 5 yield up-loads on the wing' Condition 1 exhibits the larger upload. The wing center of pressure location is the same for Conditions 1, 4 and 5.

Conditions 2 and 6 produce down-loads on the wing, with Condition 2 exhibiting the larger down-load. The wing center of pressure location is the same for Conditions 2 and 6.

Conditions 7 and 3 are flap conditions. The flap load is the same for both conditions. Condition 3 contains the higher wing load, hence develops higher wing torque than Condition 7. Condition 7 and 3 wing loads are down. The wing load of Condition 3 is larger than condition 2 and designs the lower wing cover for compression from the tip to 70 inches inboard. The lower wing cover is designed for compression for Condition 2 from the above point to the fuselage attachment.

The wing bending moments, shear and torques are given for the critical flight conditions in Figures VII-15, VII-16, and VII-17, respectively.

The critical landing conditions are Condition 15 (vertical + 5 kn drift) and Condition 17 (vertical + obstruction). Bending moments, shears and torques are shown for these two conditions in Figures VII-18 and VII-19.

Flight Condition 1 is used to design the upper cover for compression from the wing tip to 106 inches inboard. The landing gear Conditions 15 and 17 design the upper cover between the above point and the fuselage attachment. This is shown graphically in Figure VII-20 where compression cover load (P_i lb/in) is plotted against wing station.

Condition 1 is used to design the spar webs for vertical and torsion shear from the tip to 86 inches inboard. The landing gear obstruction condition is used to design the spar webs from this point to the fuselage attachment. Figures VII-21 and VII-22 depict graphically the maximum shear flow distribution in the plane of the spar webs, plotted against wing station.

The wing support bulkheads at Fuselage stations 246.50 and 281.0 are designed by Condition 1 and the landing gear obstruction condition. The landing gear obstruction condition produces higher vertical reactions at the bulkheads and Condition 1 produces slightly higher spar cap loads in the plane of the fuselage bulkheads. Figure VII-23 contains a summary of the fuselage bulkhead reactions, closure rib shear flows and spar cap loads.

The structural area requirements for the primary structural items of the wing were determined from the above design loadings. These results are plotted in Figure VII-24. A typical wing cross section of the inboard portion of the wing is shown in Figure VII-25. Wing bending and torsional stiffinesses determined for flutter and wing deflection analyses are shown in Figure VII-26. The wing deflection for Flight Condition 1 is shown in Figure VII-27, also shown in this figure is the wing tip deflection when the aerodynamic lift is zero for a 1000 lb concentrated load at the wing tip.

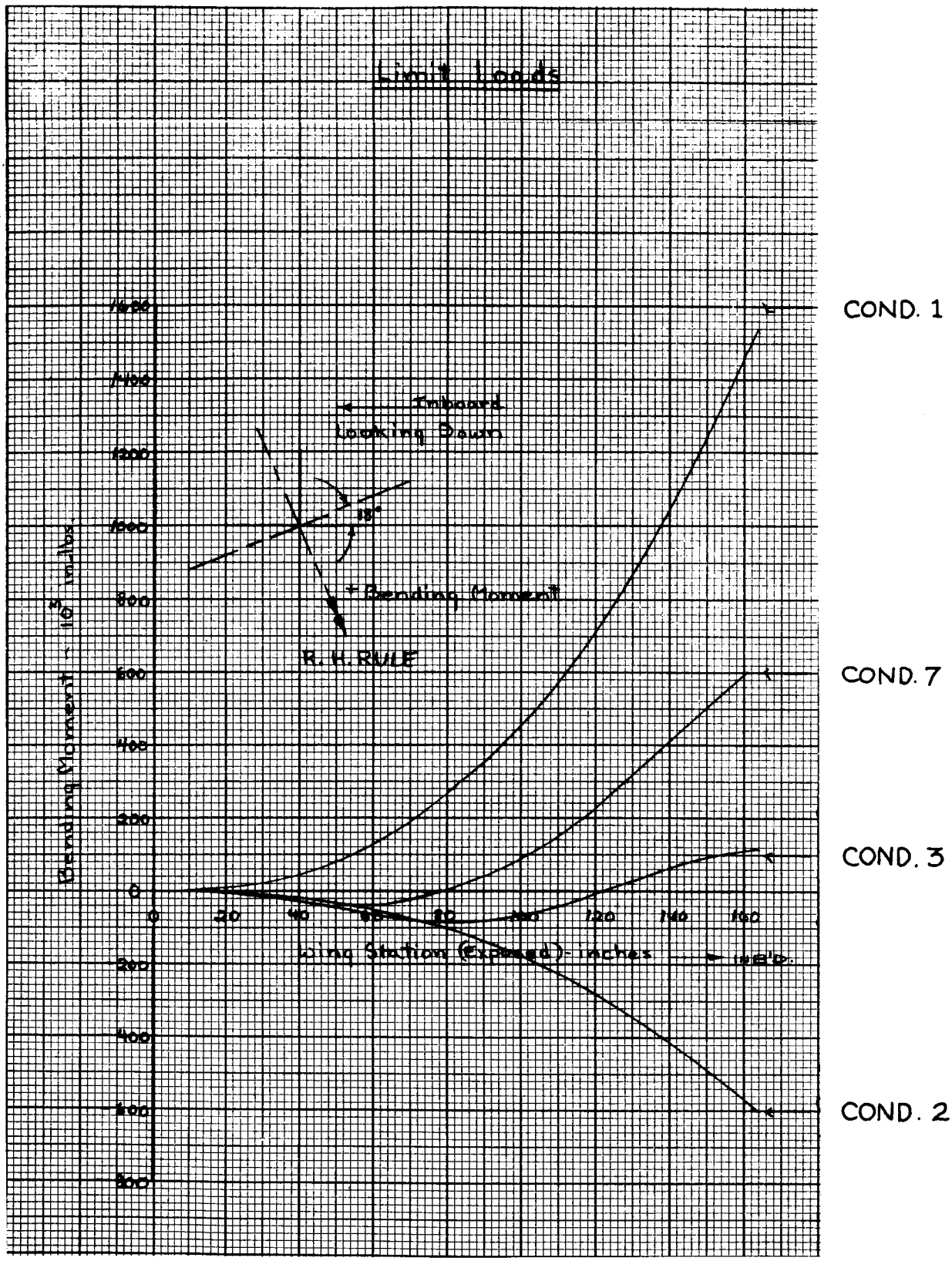


Figure VII-15. Wing Design Bending Moments - Flight Conditions

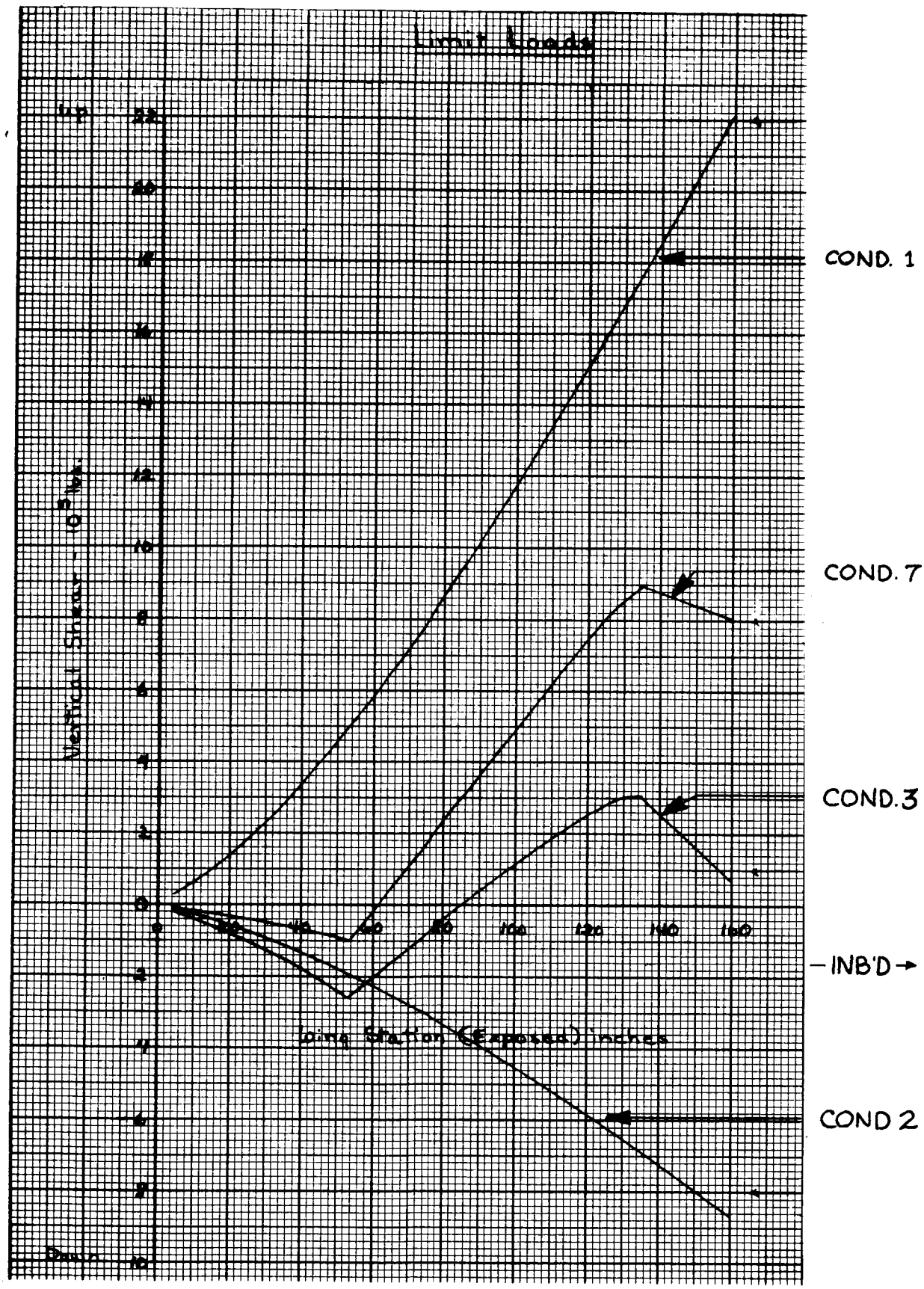


Figure VII-16. Wing Design Shears - Flight Conditions

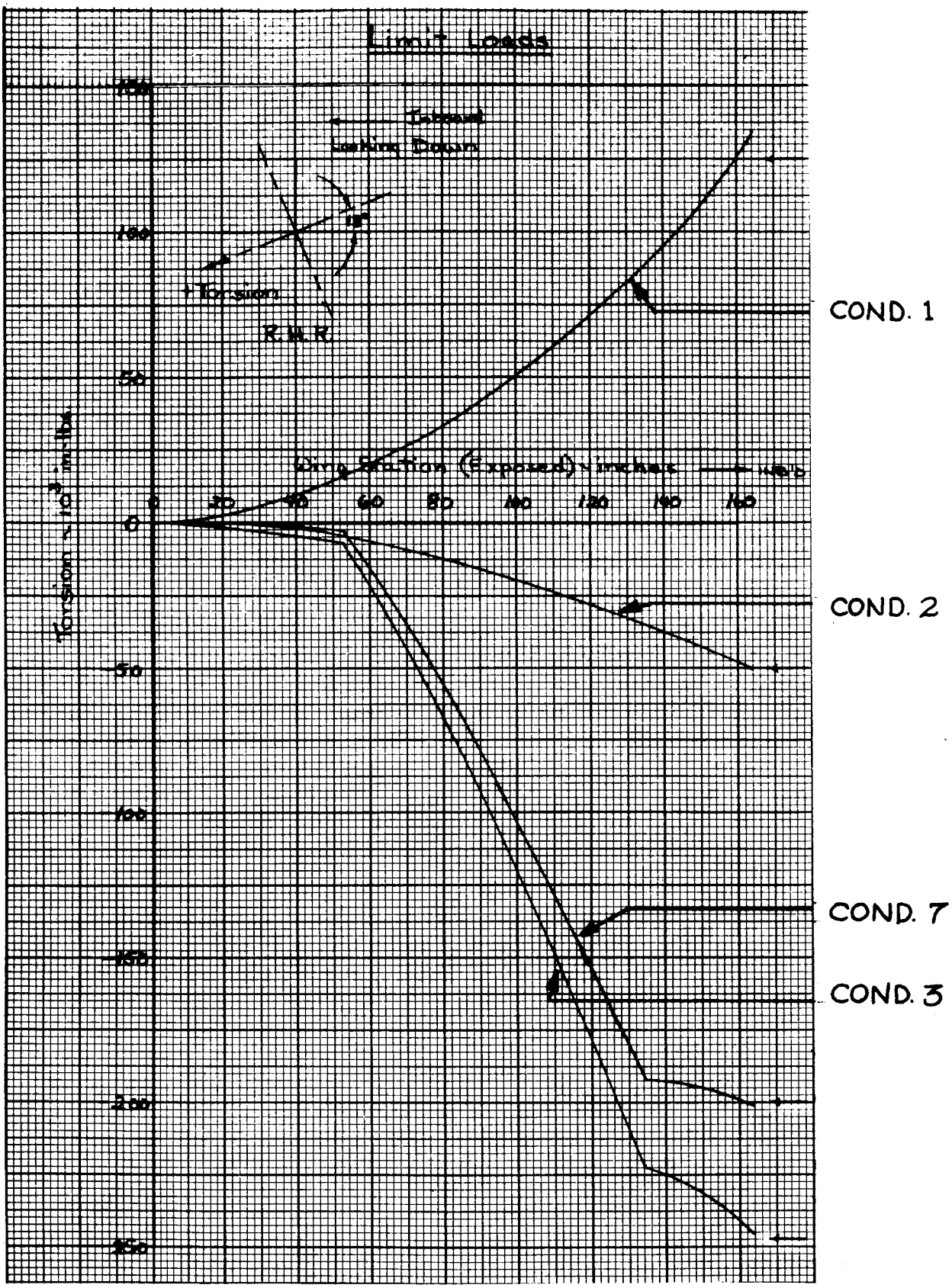


Figure VII-17. Wing Design Torques - Flight Conditions

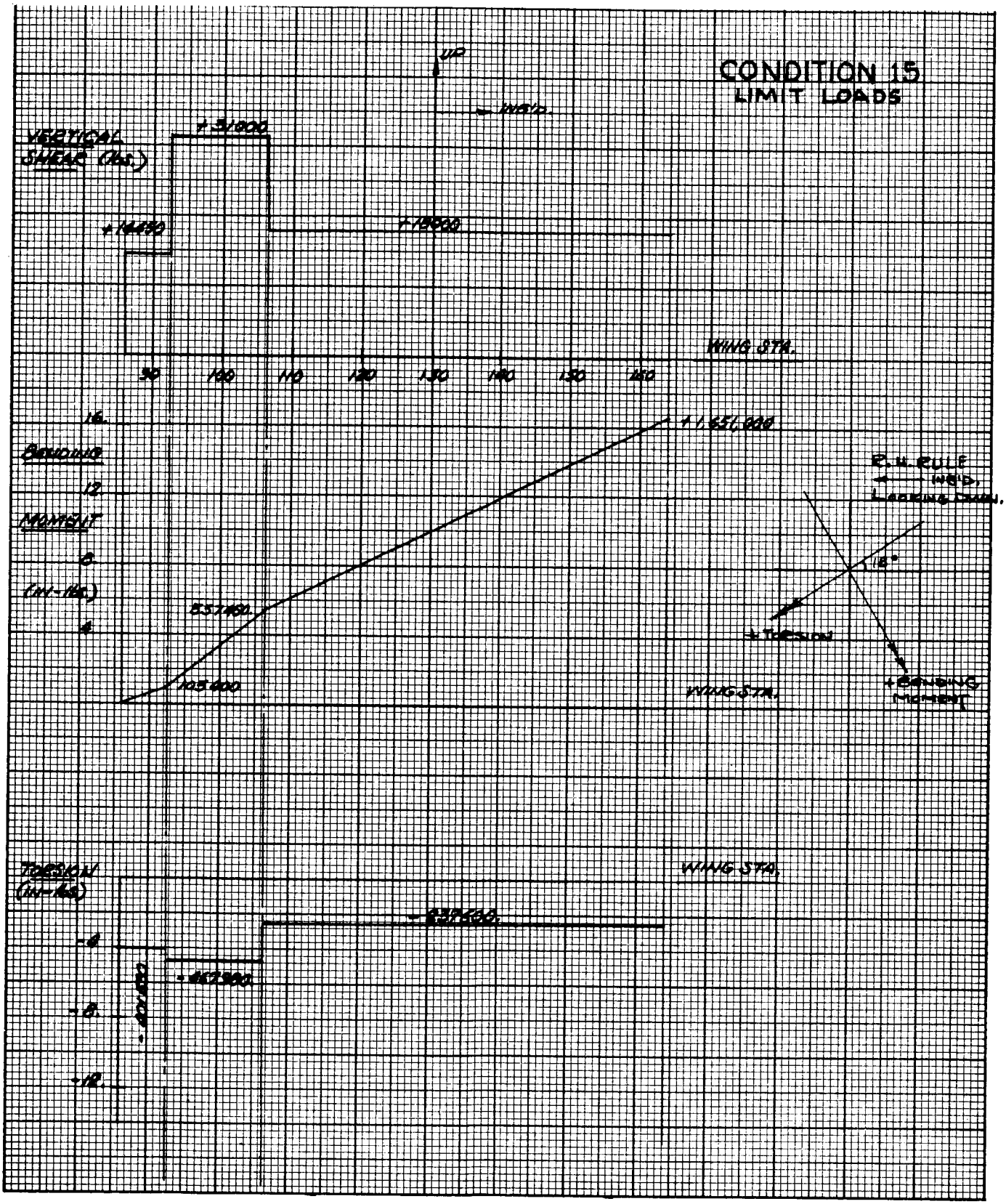


Figure VII-18. Wing Design Loadings - 5 kn. Drift

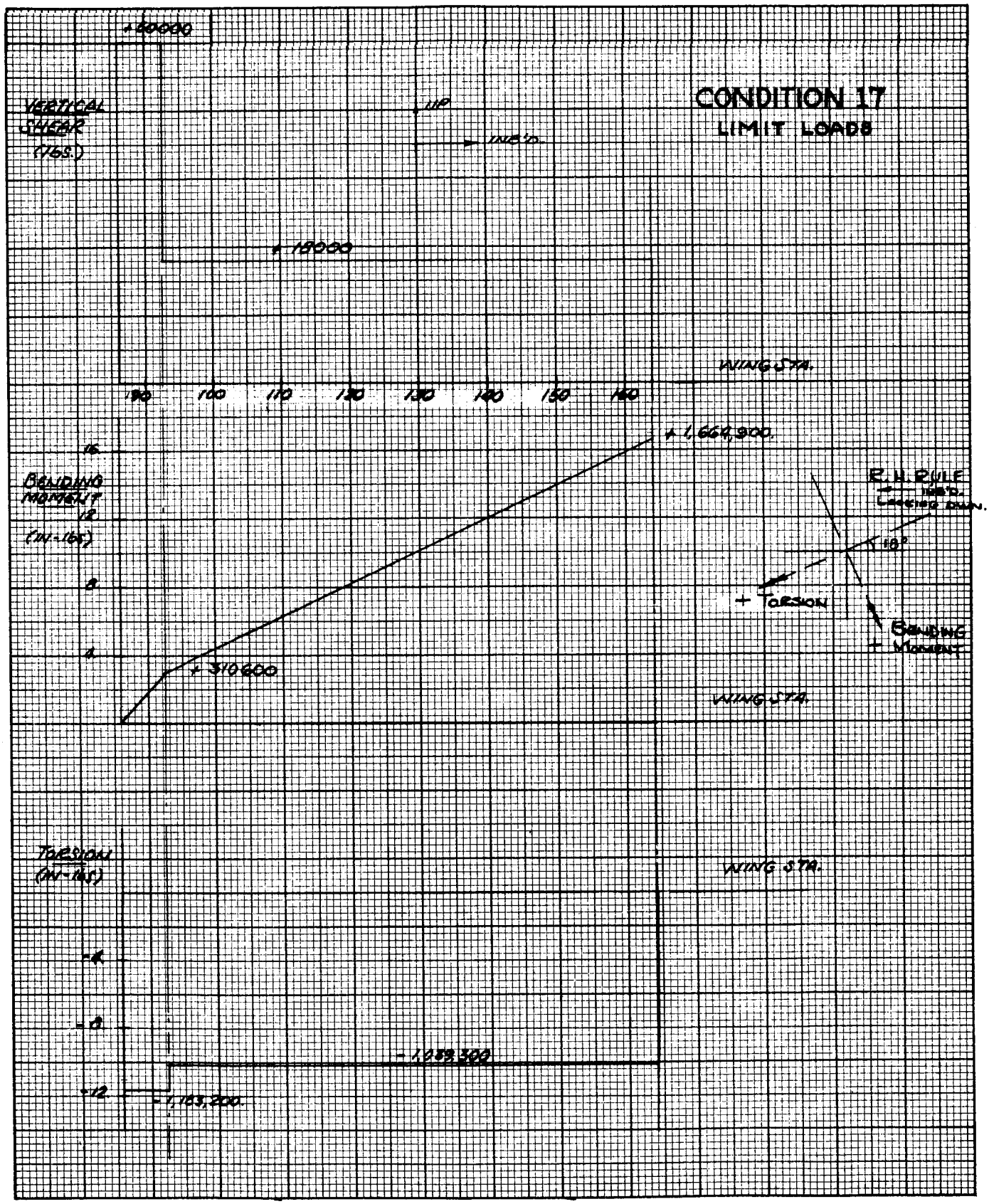


Figure VII-19. Wing Design Loadings - Obstruction Vertical Landing

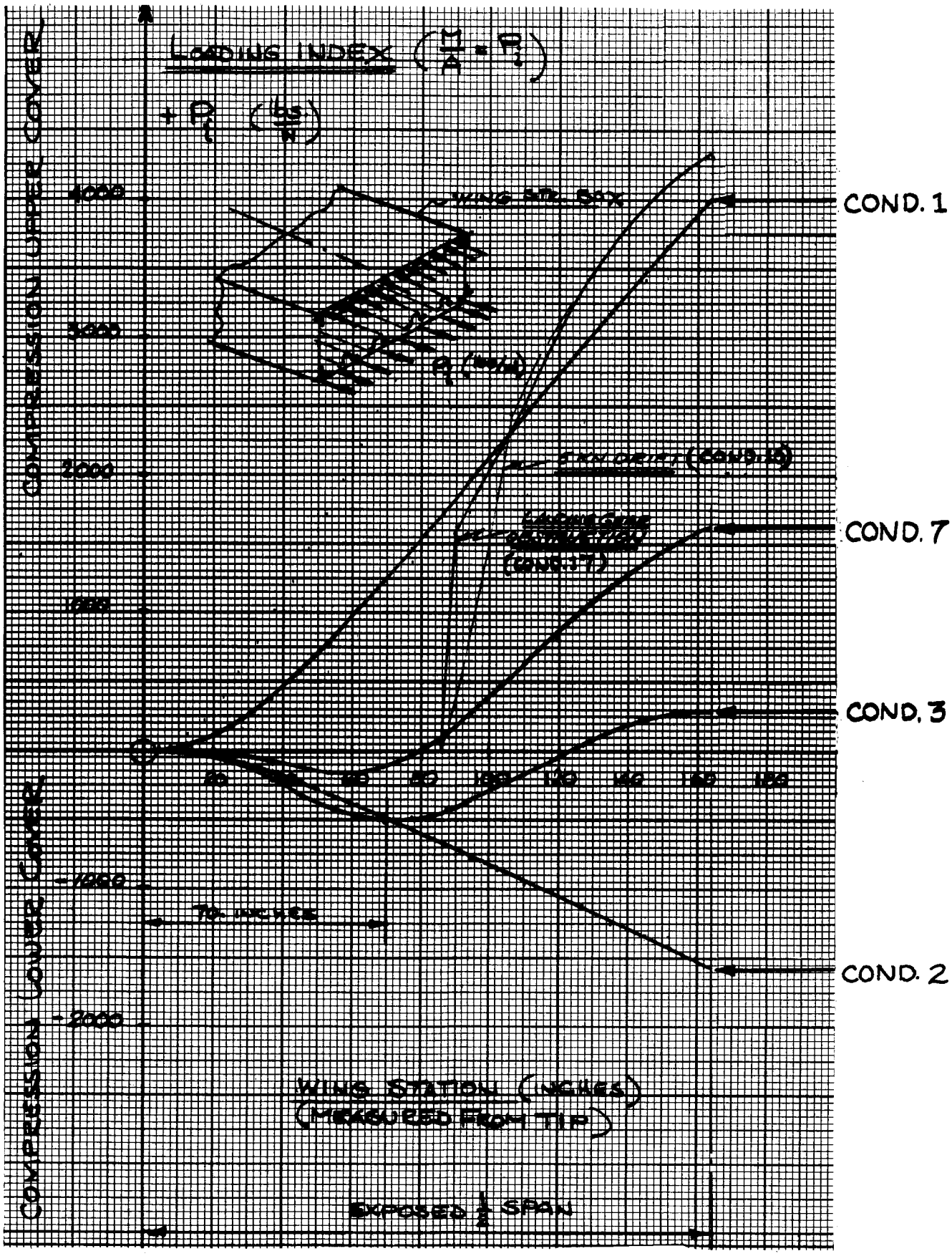


Figure VII-20. Wing Cover Loading Index

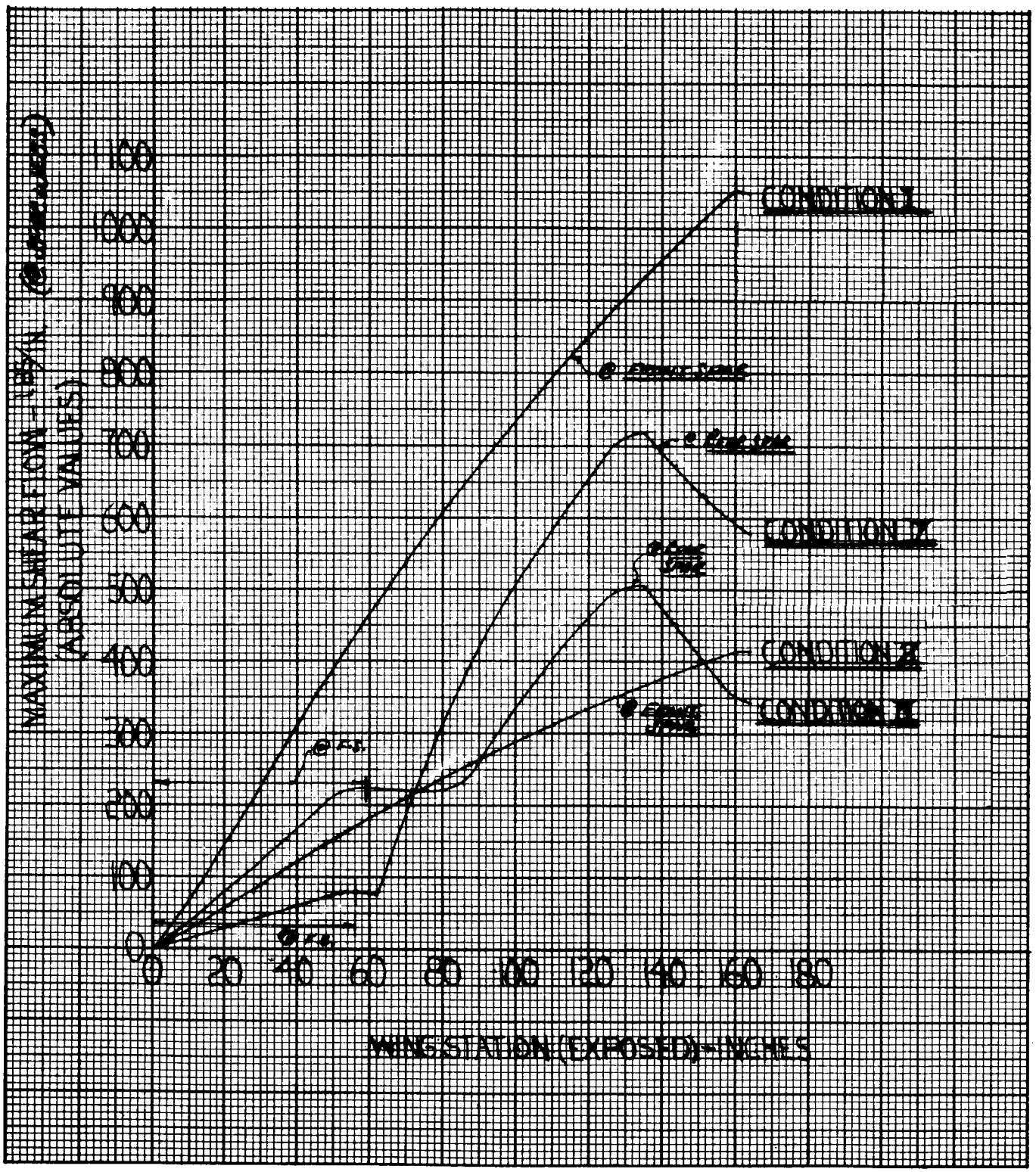


Figure VII-21. Maximum Shear Flows - Flight Conditions

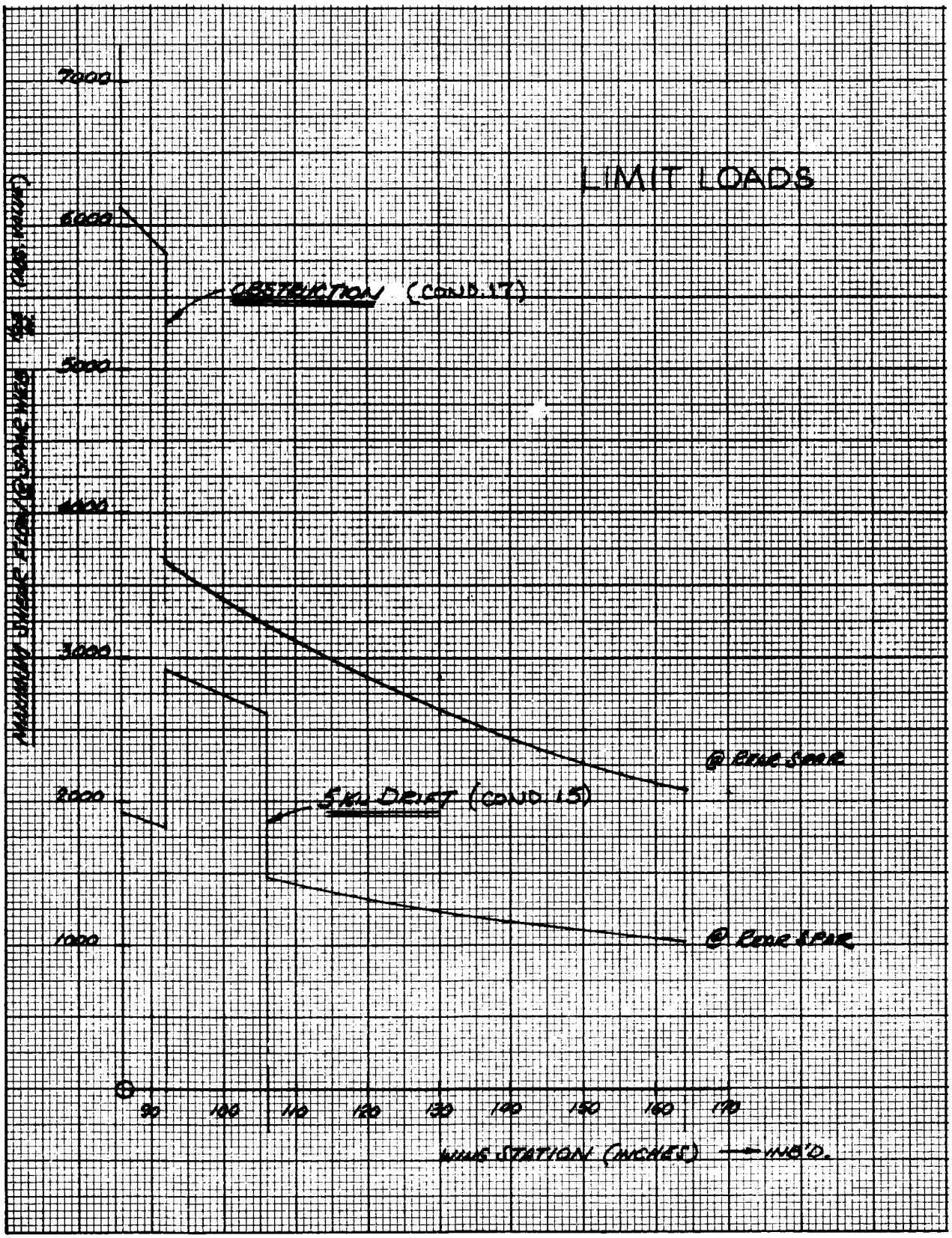
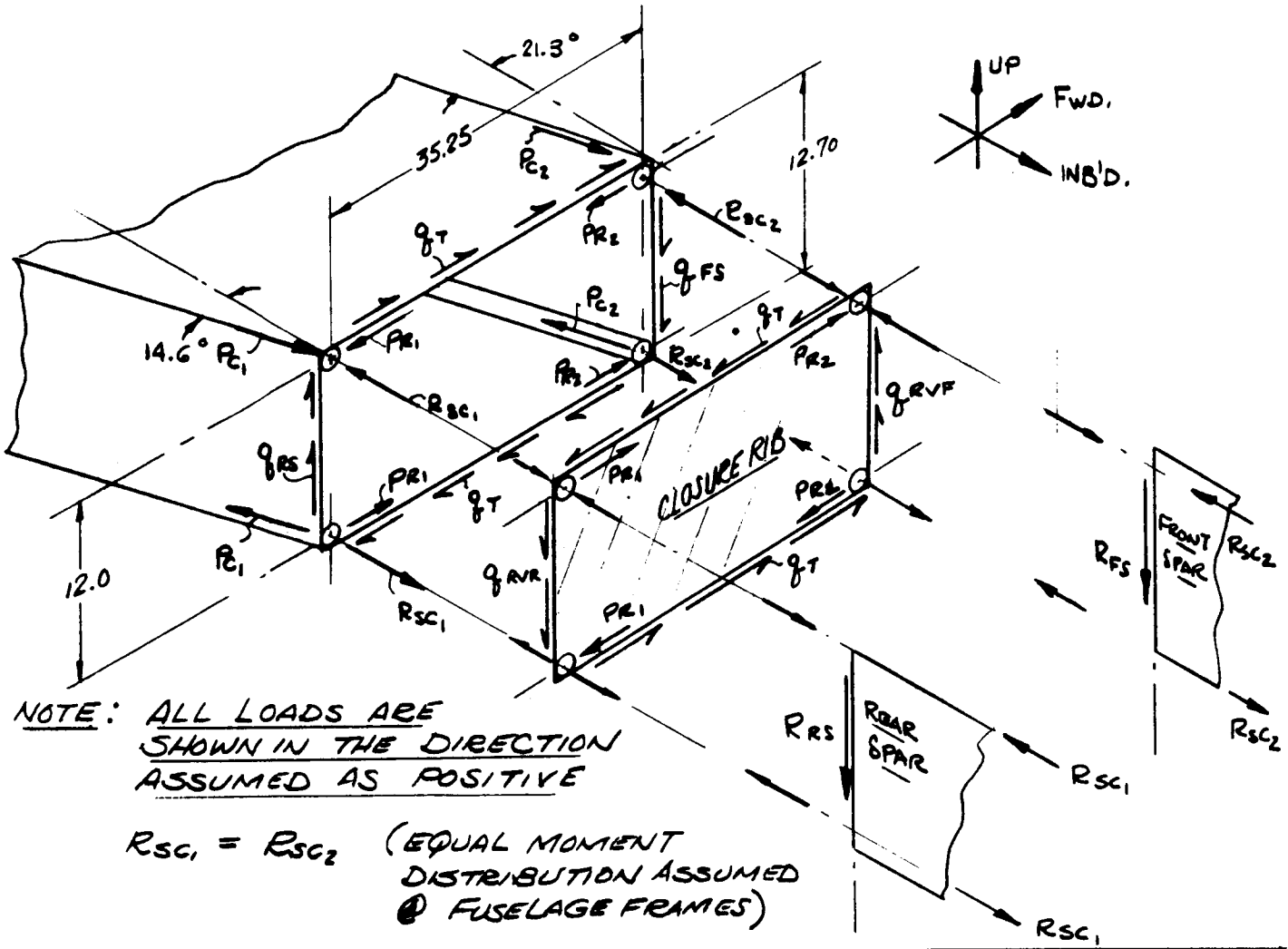


Figure VII-22. Maximum Shear Flows - Landing Conditions



NOTE: ALL LOADS ARE SHOWN IN THE DIRECTION ASSUMED AS POSITIVE

$R_{SC_1} = R_{SC_2}$ (EQUAL MOMENT DISTRIBUTION ASSUMED @ FUSELAGE FRAMES)

CONDITION	q_{RS} lbs./IN	q_T	q_{FS}	P_{C_1} & P_{C_2}*	q_{RVR} lbs./IN	P_{R_1} (lbs.)	P_{R_2} (lbs.)	q_{RVF} lbs./IN	R_{RS} (lbs.)	R_{FS} (lbs.)
1	2709	1400	92	97533 101280 R_{SC} =	241 94384 lbs.	24585	36735	241	31340	1887
2	-1065	-550	-36	-38328 -39800 R_{SC} =	-95 -37091	-9661	-14436	-95	-12317	-742
7	+1415	939	463	34740 36074 R_{SC} =	-356 33618.	8757	13084	-356	22486	-10385
3	508	470	432	2090 2170 R_{SC} =	-436 2022.	527	787	-436	11995.	-11018

Figure VII-23. Summary of Ultimate Wing Support Loads

LANDING GEAR LOADINGS:

CONDITION	q_{RS} lbs/IN	q_T	q_{FS}	P_{C1} & P_{C2}	q_{RVR} lbs./IN	P_{R1} (lbs.)	P_{R2} (lbs.)	q_{RVF} lbs./IN	R_{RS} (lbs.)	R_{FS} (lbs.)
15	3108.	2045.	982.	96597. 100308.	-420.	24349.	36383.	-420	44804	-17804
17	4264.	3201	2138.	81481. 84611.	-1832	20539.	30689.	-1832.	77420	-50420

Figure VII-23 (Cont'd)

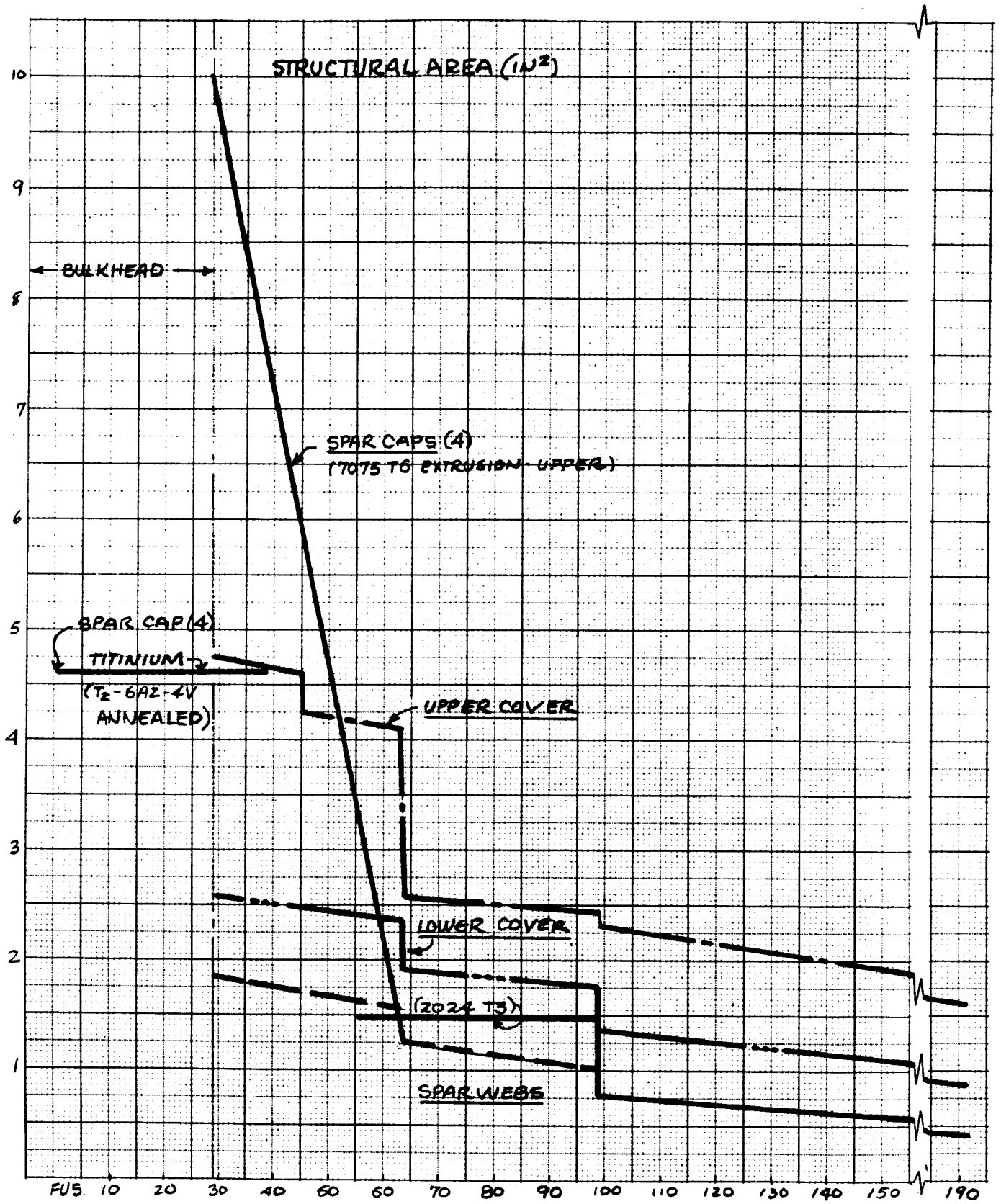


Figure VII-24. Wing Structural Area Requirements

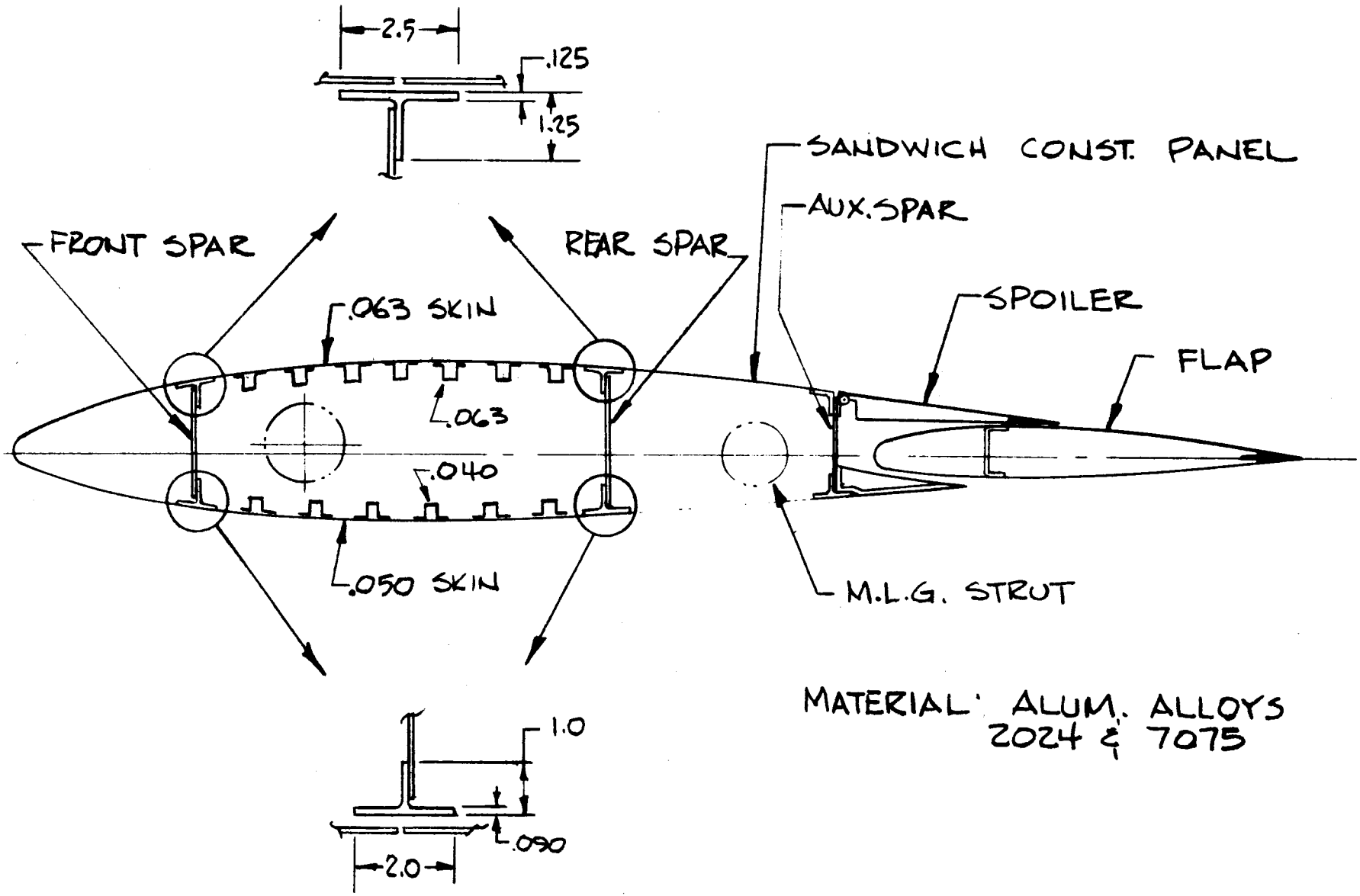


Figure VII-25. Typical Wing Section, Inboard

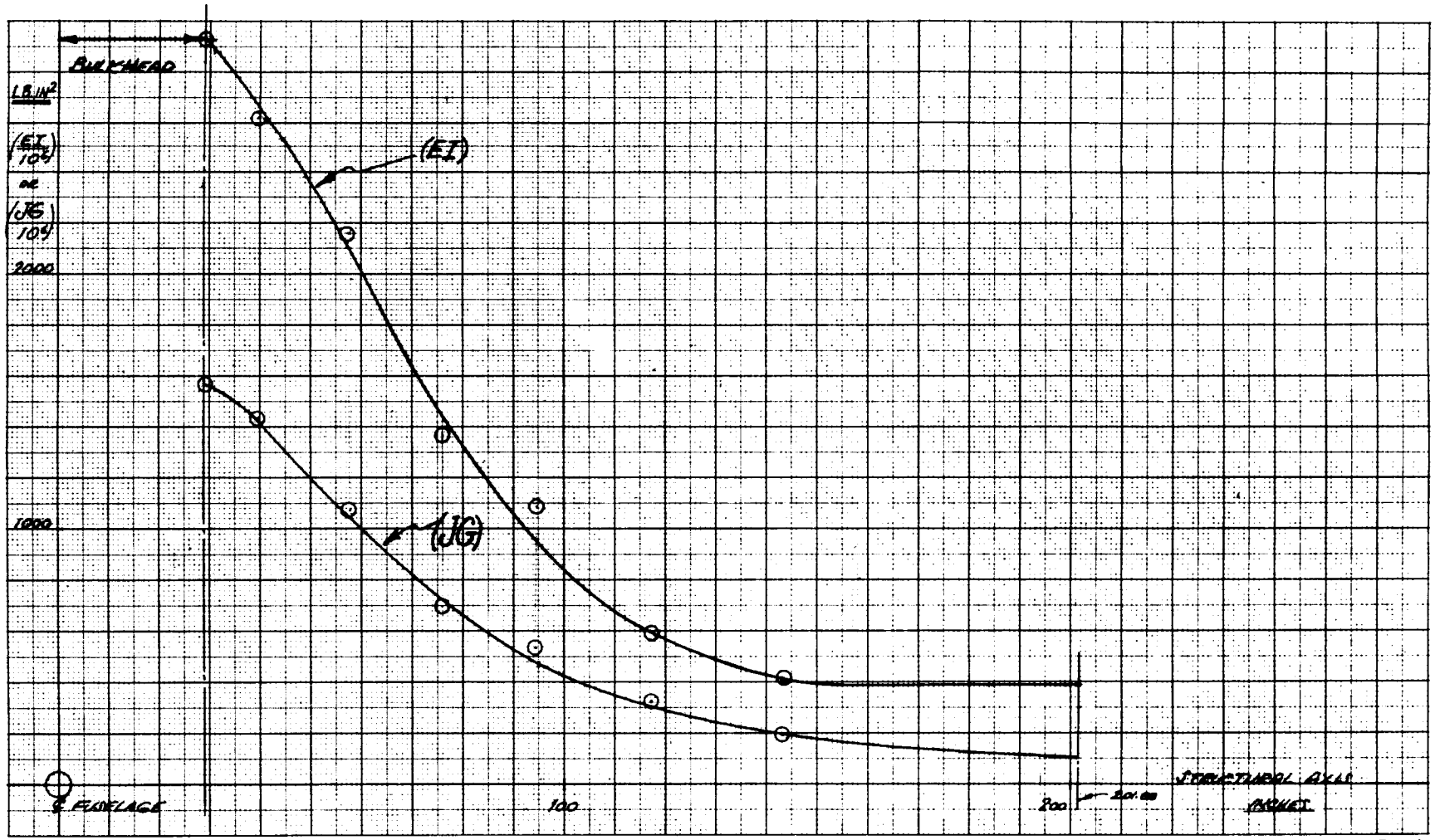


Figure VII-26. Wing Bending and Torsional Stiffnesses

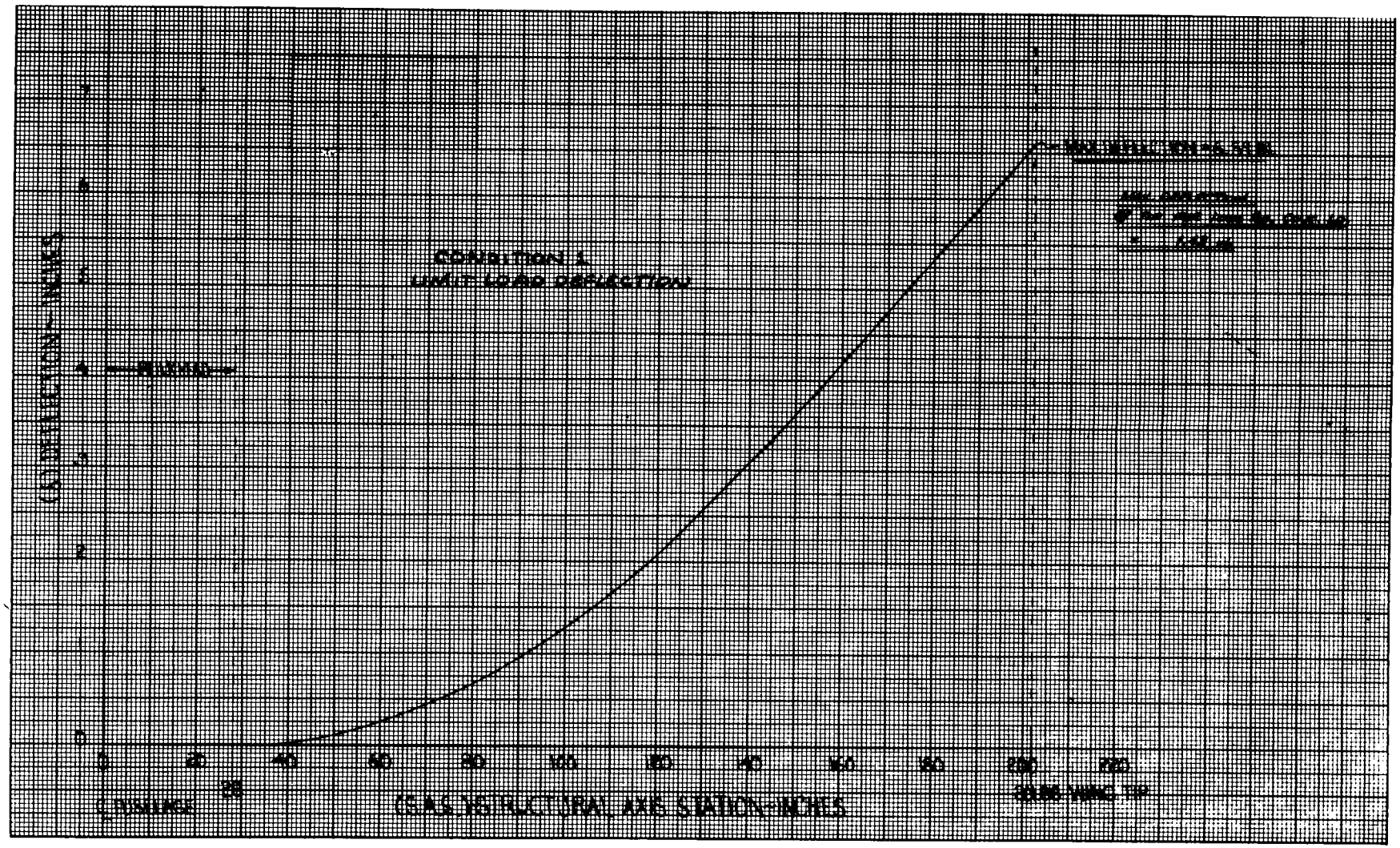


Figure VII-27. Wing Deflection

I. HORIZONTAL TAIL ANALYSIS

All of the flight conditions given in Section D were considered in the design of the horizontal tail. The critical flight conditions were Condition 4 (maximum up load) and condition 7 (maximum down load). These loads are approximately equal and for expediency, the load of Condition 7 was assumed to be reversible. The resulting design ultimate bending moments, shear and torques are shown in Figure VII-28. Figure VII-29 shows the corresponding loads on the carry through torque tube and also a strength check of the tube. Figures VII-30 through VII-32 show the structural area requirements of the surface and the torque tube. Bending and torsional stiffnesses were calculated for flutter analyses. These are presented in Figure VII-33.

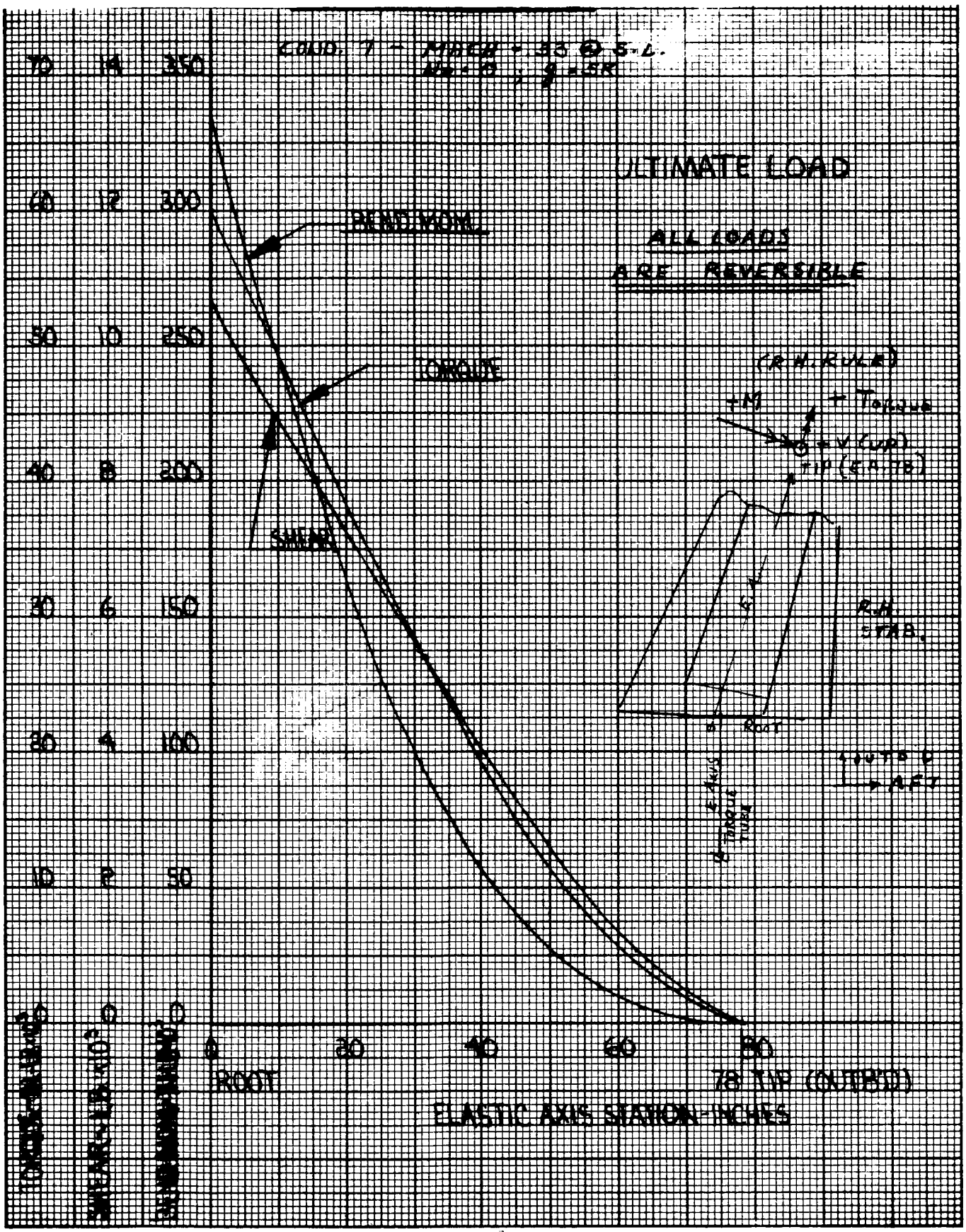
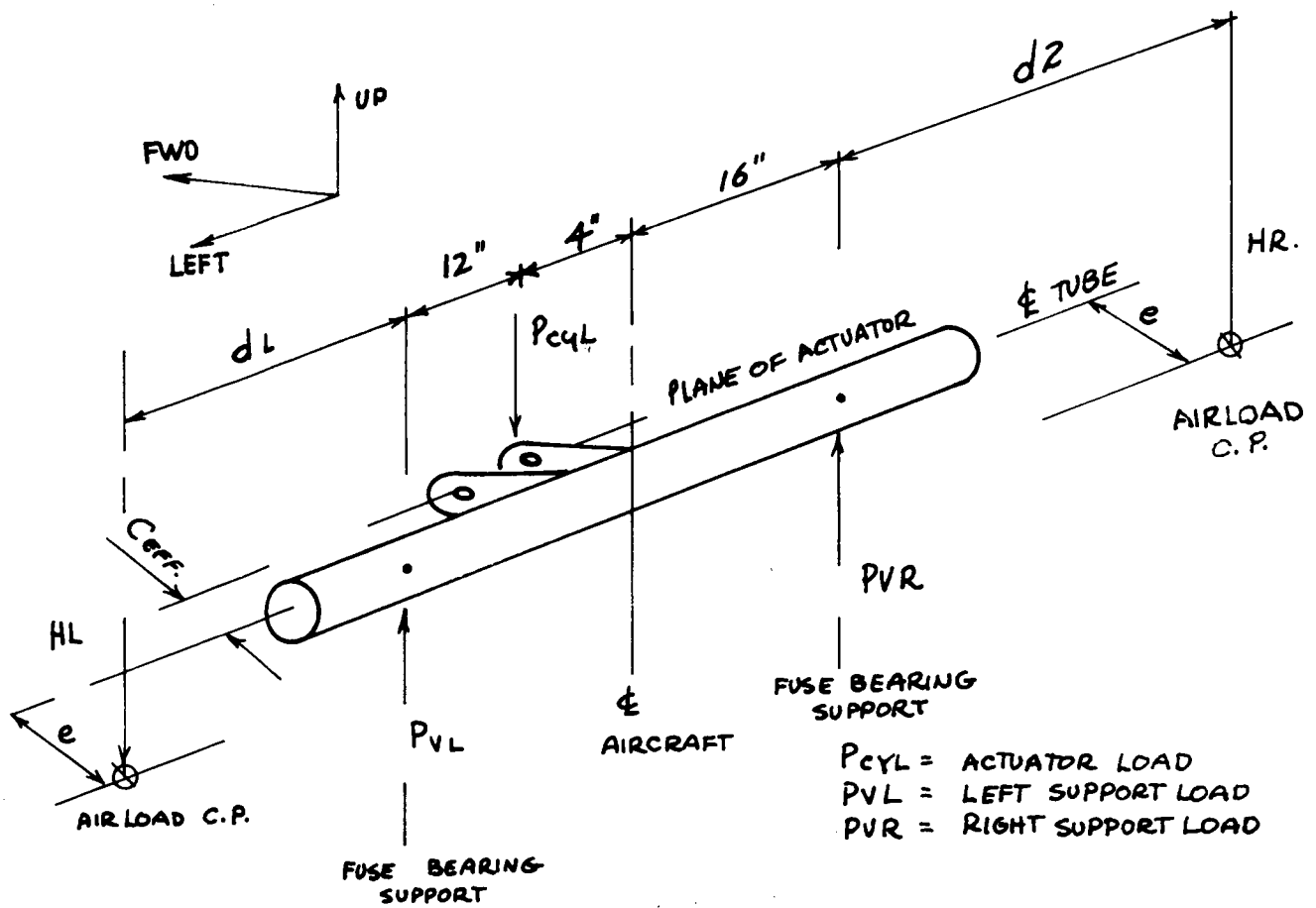


Figure VII-28. Horizontal Tail Bending Moment, Shear and Torque



CONDITION*	H_R lb.	H_L lb.	d_r in.	d_L in.	C_{eff} in.	e in.	P_{cyl} lb.	P_{VL} lb.	P_{VR} lb.
57 1/2% Left	± 7560	± 10640	33	33	5.5	0.5	± 1655	± 14850	± 5005
57 1/2% Right	± 10640	± 7560	33	33	5.5	0.5	± 1655	± 5418	± 14437

* Maximum Tail Loads Distributed 57 1/2% and 42 1/2 Per Side

Torque Tube 4.25" O/D, 0.25" Wall $I = 6.30$ In.^{4}, $J = 12.60$ In.^{4}

Material 4340 Steel FTU = 160-180 KSI

Max. Torque = 10640 x .50 = 5320 In. Lb, Max+ Moment = 10640 x 33 = 351120 In. Lb

Max. Sheal Stress $f_s = \frac{5320 \times 2.125}{12.60} = 0.9$ KSI Negligible

Max. Bending Stress $= \frac{351120 \times 2.125}{6.30} = 118.4$ KSI

Allow Bending Stress $F_{BRU} = 210.0$ KSI Strength Ample, But Wall Thickness Maintained at 0.25" for Stiffness

Figure VII-29. Horizontal Tail Torque Tube Loads and Analysis

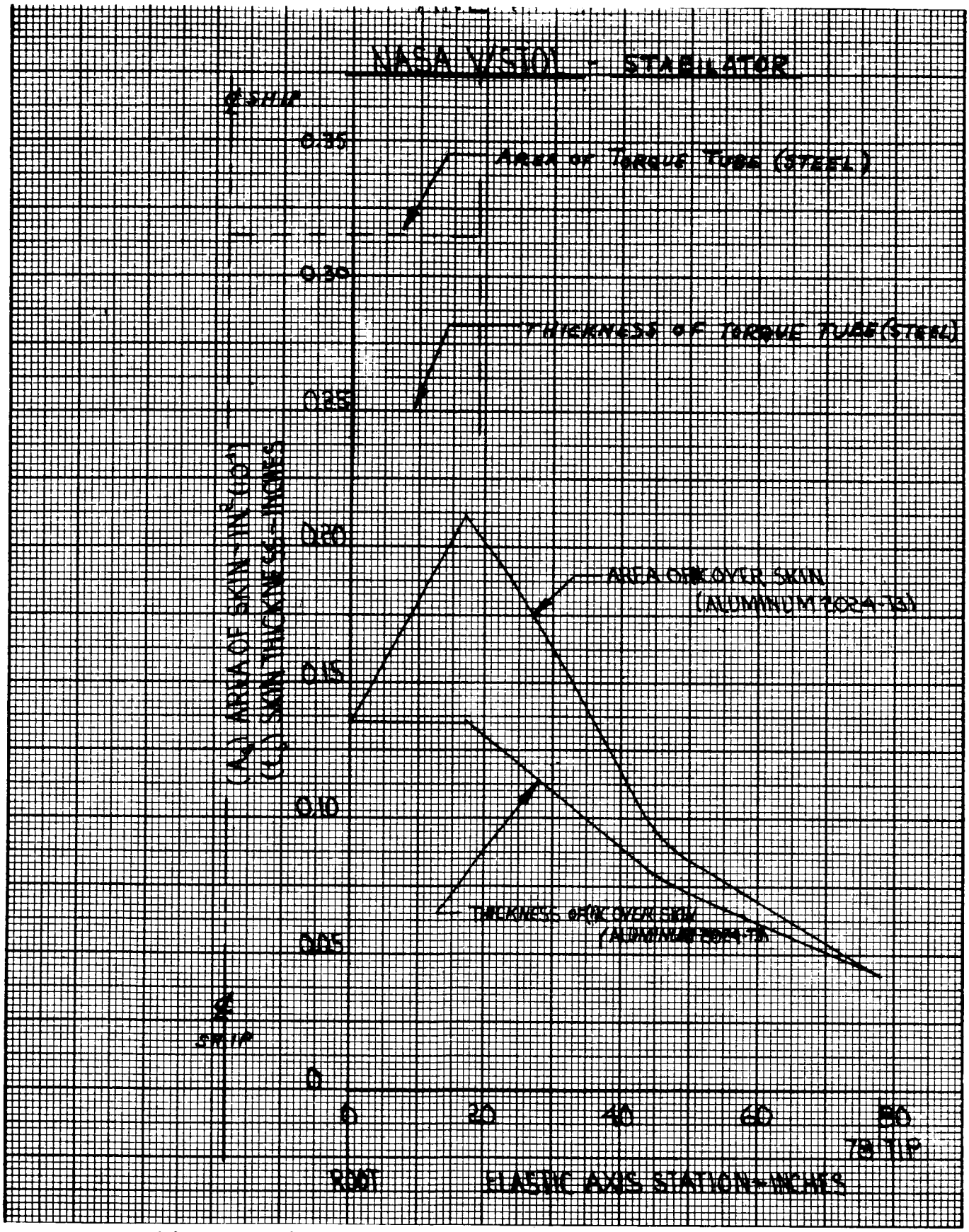


Figure VII-30. Horizontal Tail-Required Skin Thickness and Area

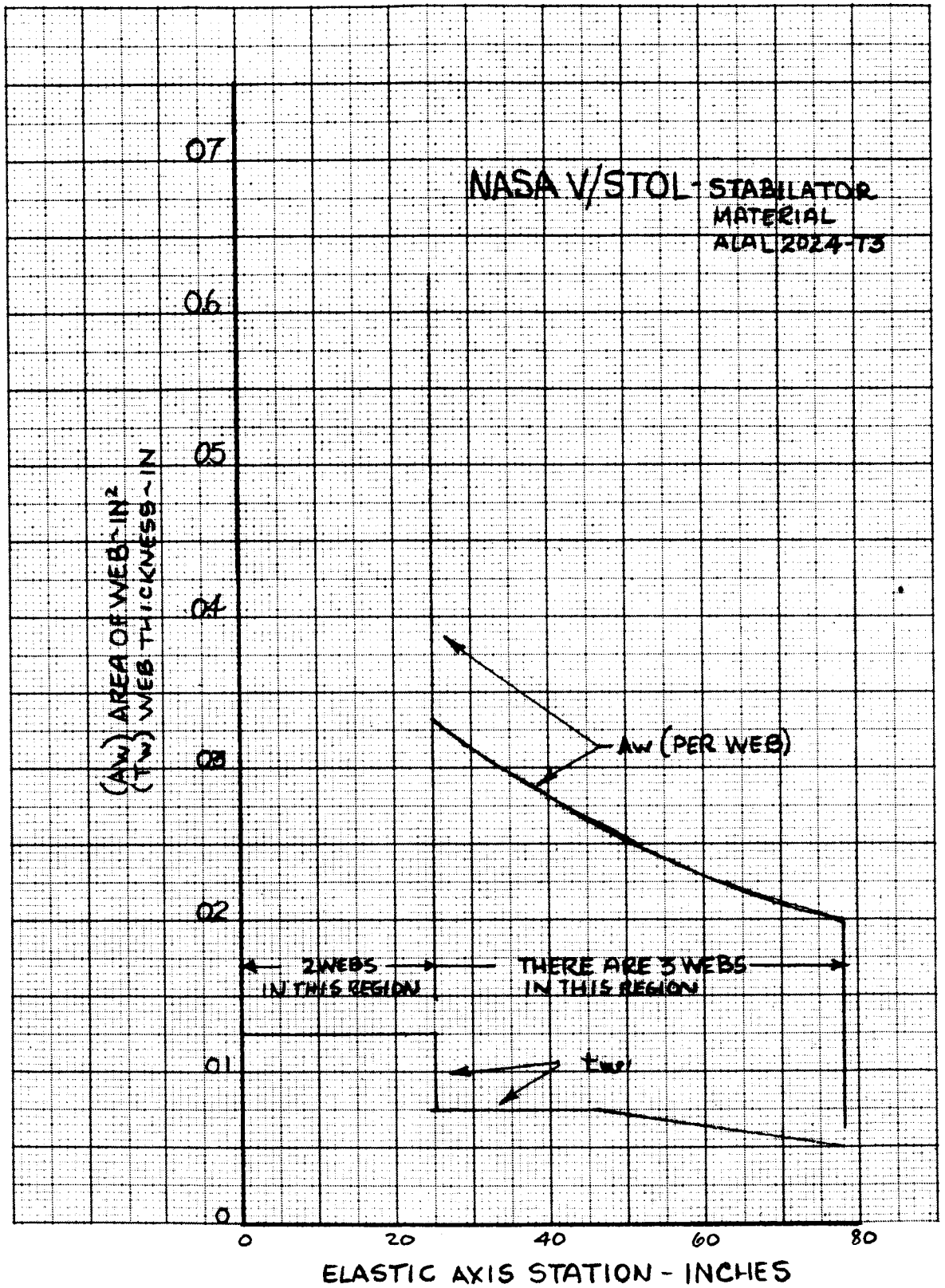


Figure VII-31. Horizontal Tail Required Web Thickness and Area

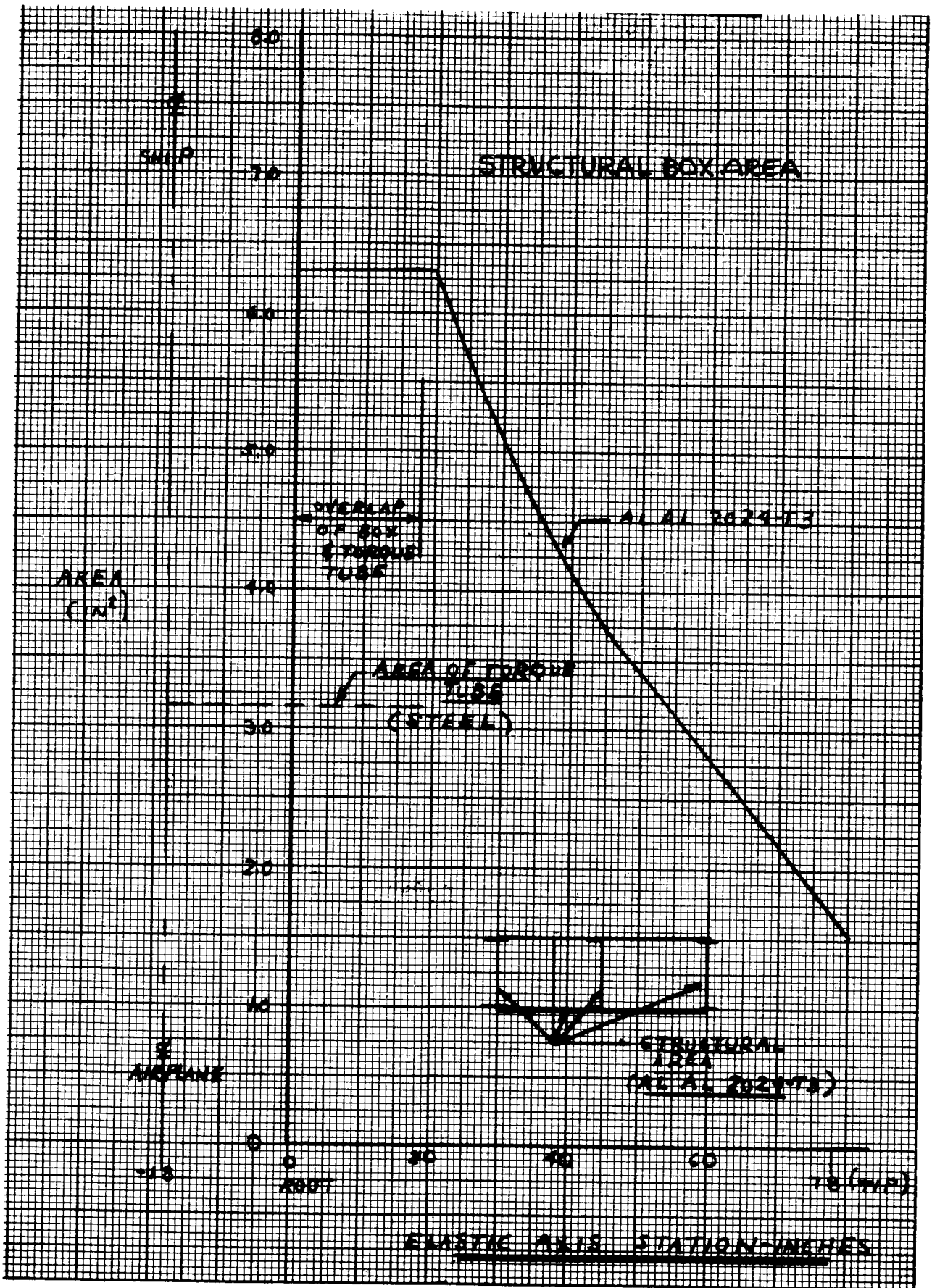


Figure VII-32. Horizontal Tail Required Box Area

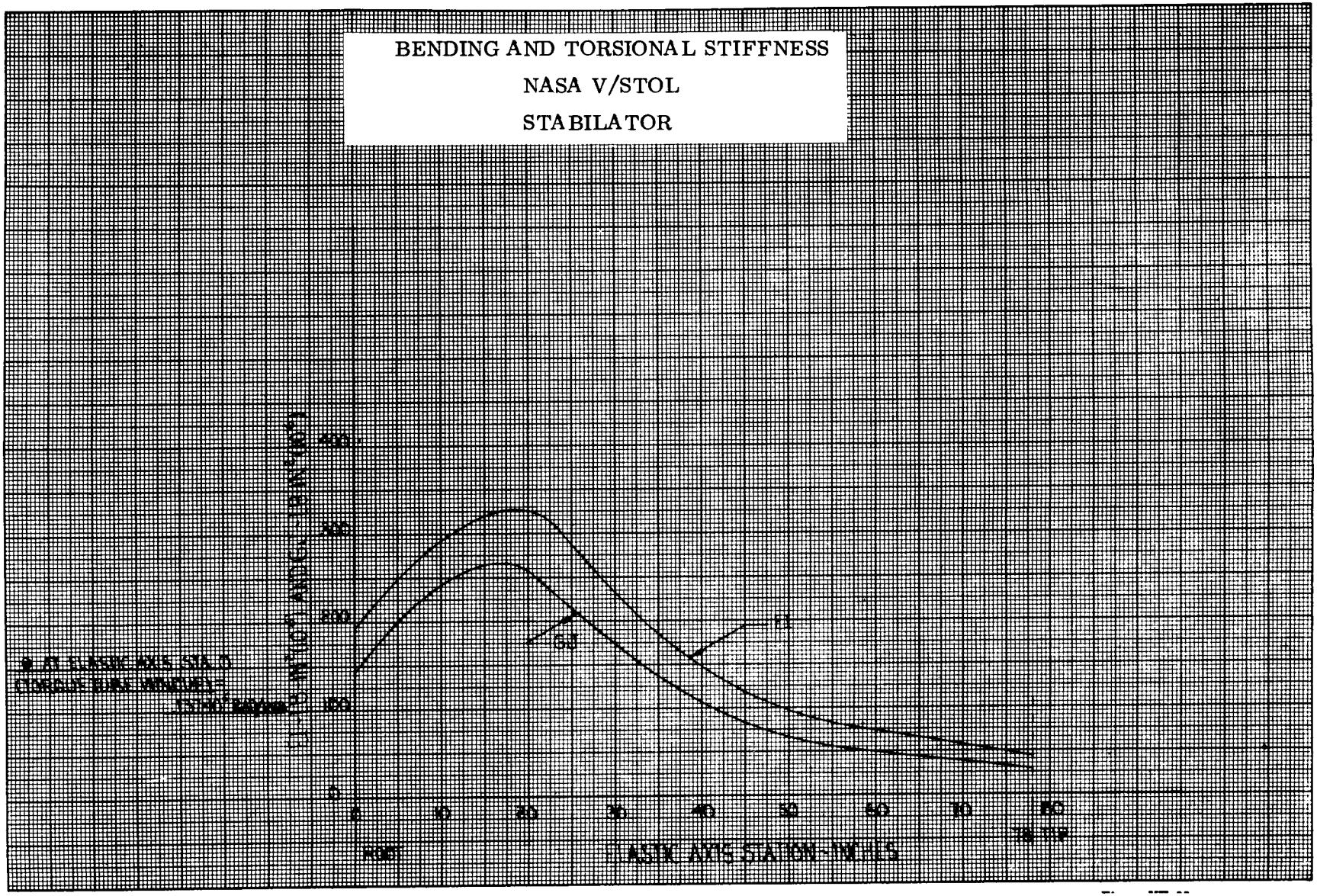


Figure VII-33

J. VERTICAL TAIL ANALYSIS

Two conditions were analyzed for the design of the vertical fin. These were Condition 8 (steady-state sideslip) and Condition 9 (maximum sideslip with maximum rudder). Shear moments and torques for these two conditions are presented in Figures VII-34, VII-35 and VII-36 respectively. The spanwise skin cover and web thickness requirements are shown in Figure VII-37. The fin loads are transferred into the fuselage through attachment fittings to fuselage frames at the front and rear spars. The loads applied to the fuselage frames are shown in Figure VII-38. Bending and torsional stiffness plots for the fin are presented in Figures VII-39 and VII-40 respectively.

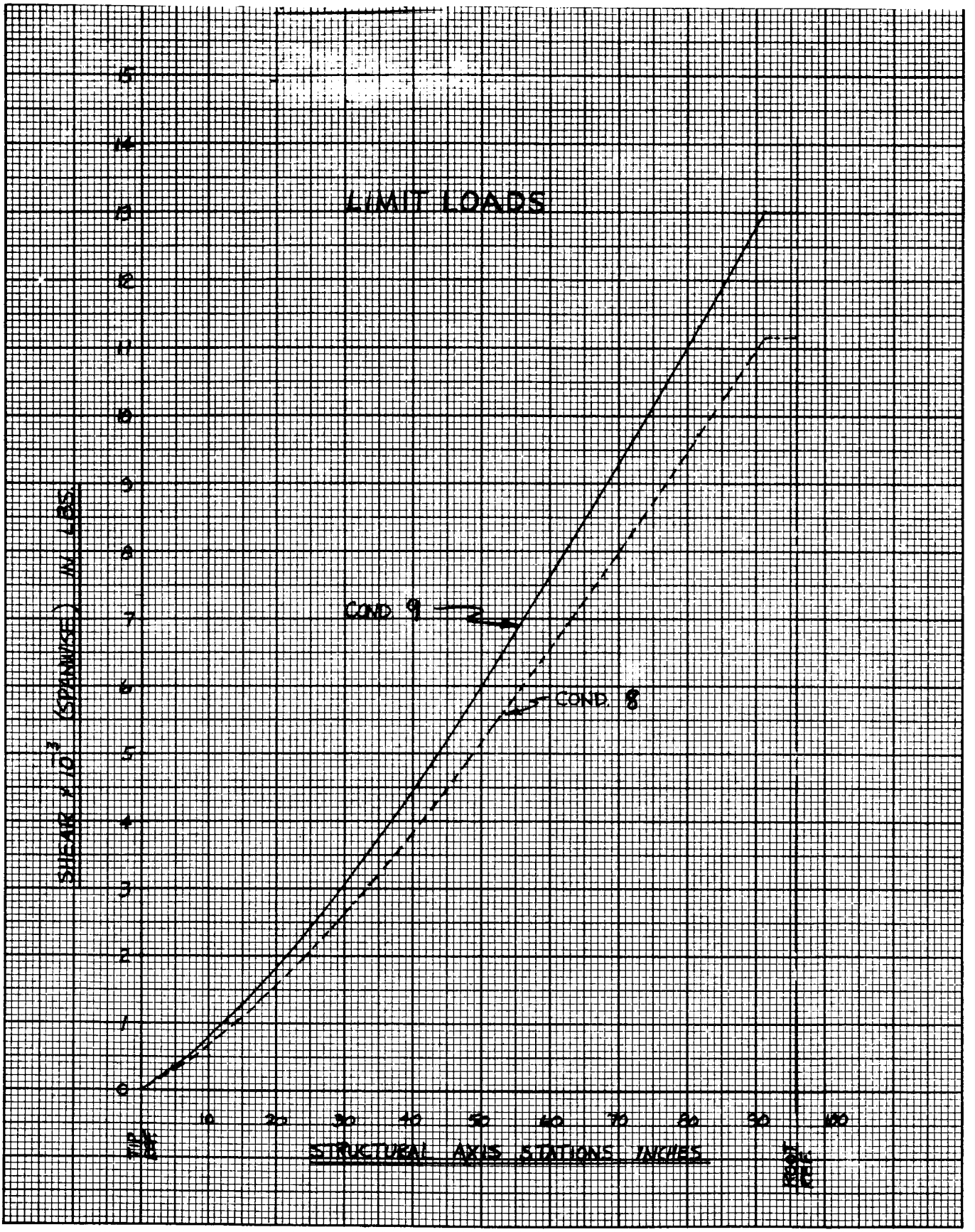


Figure VII-34. Vertical Fin Shears

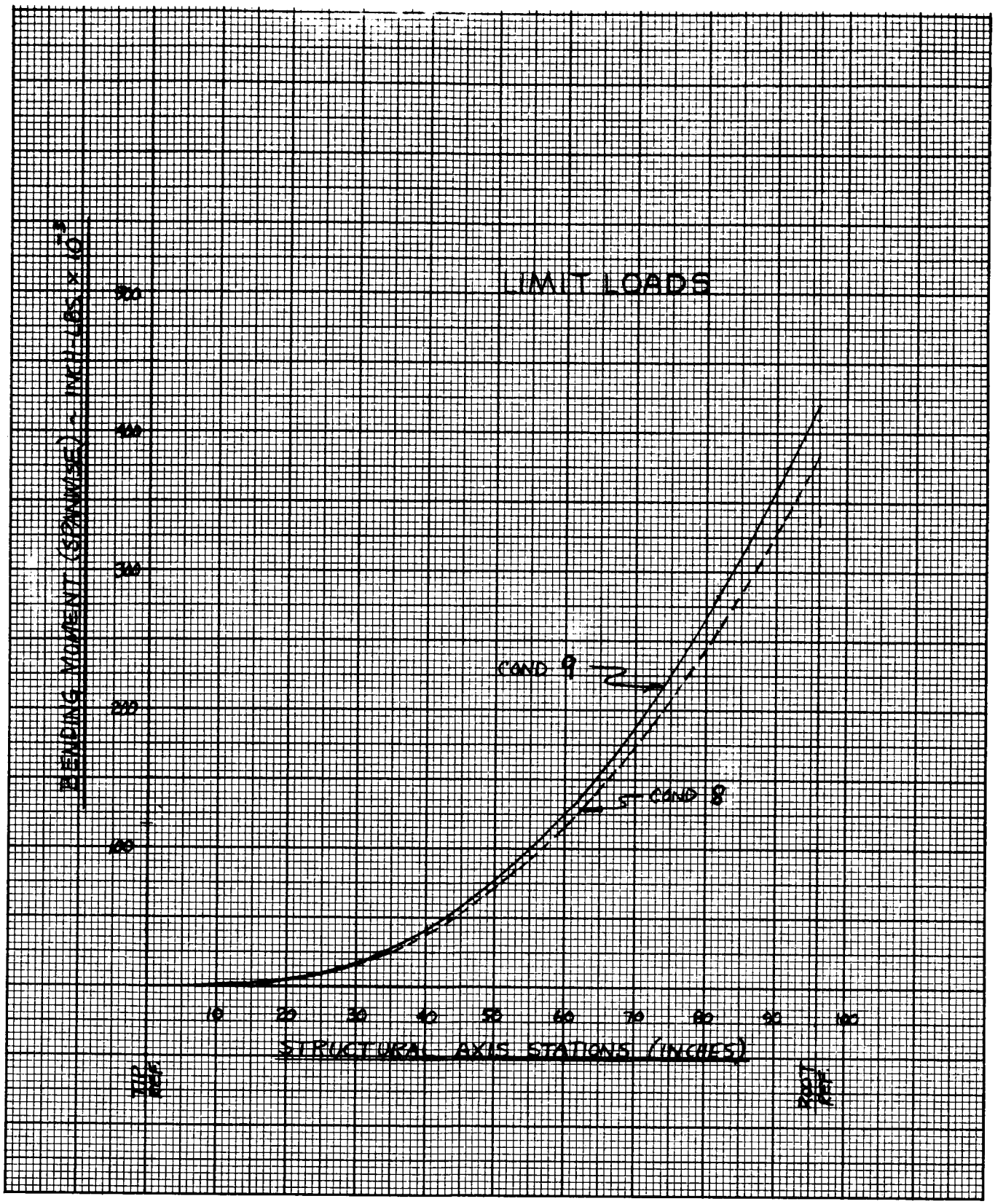


Figure VII-35. Vertical Fin Bending Moments

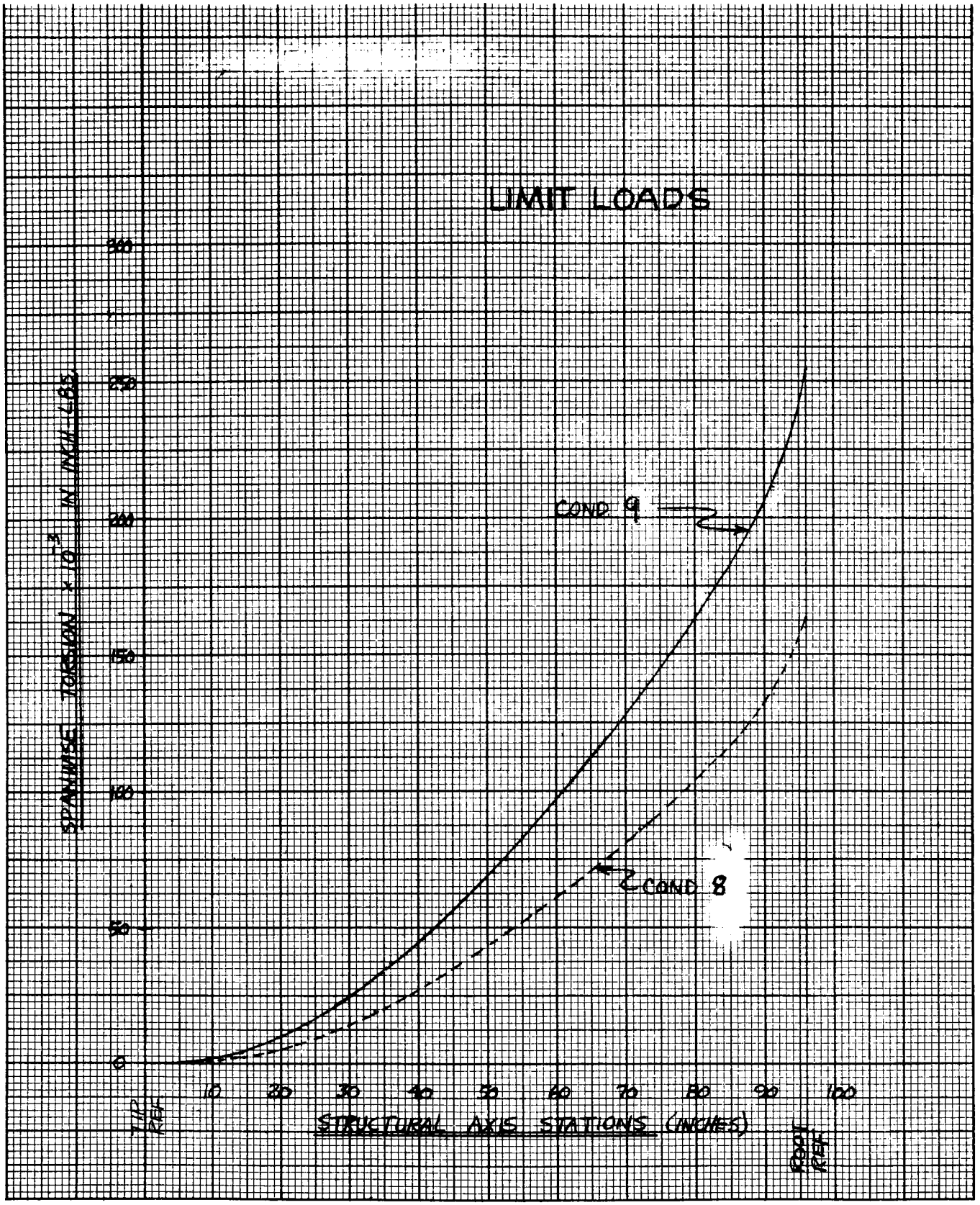


Figure VII-36. Vertical Fin Torques

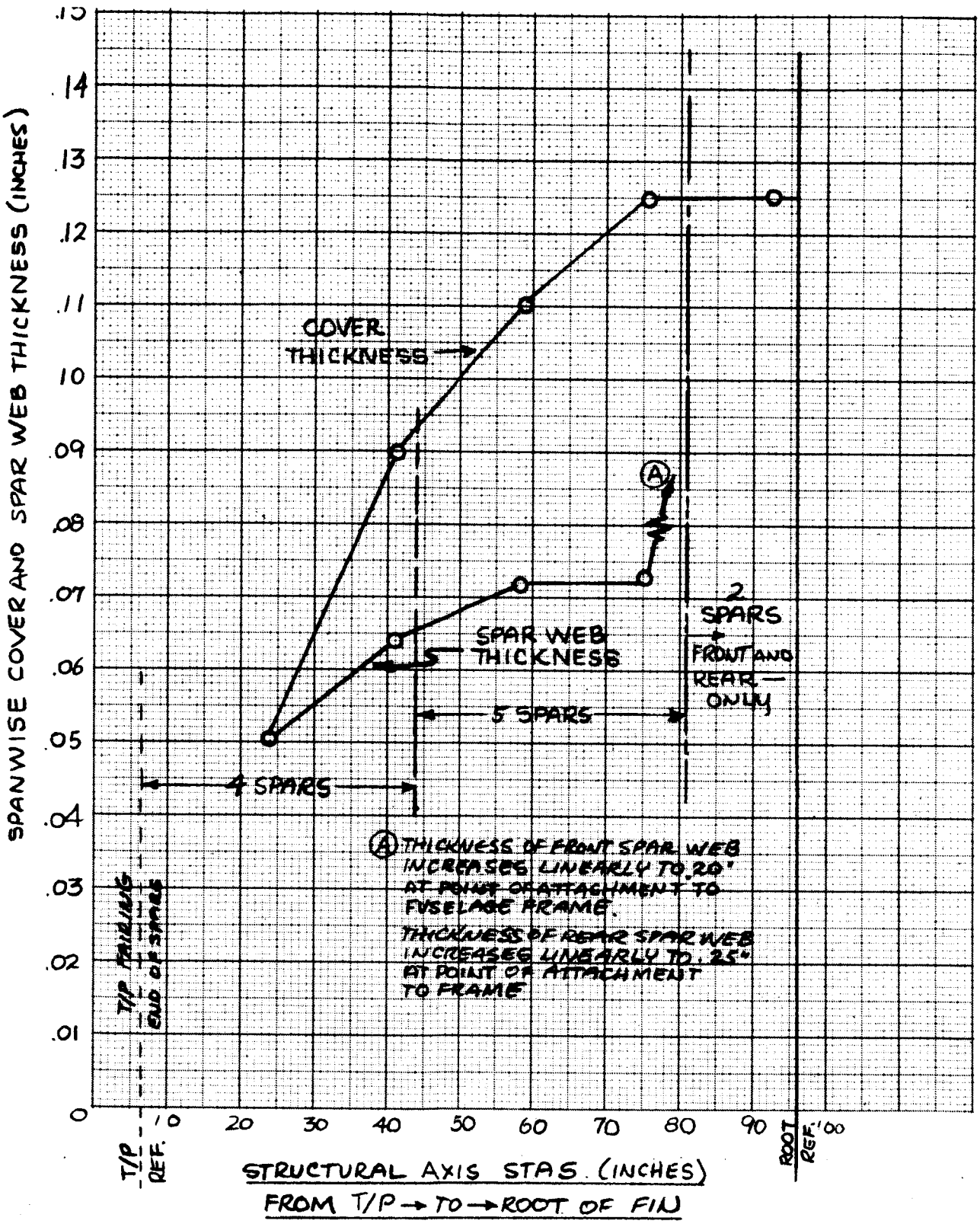
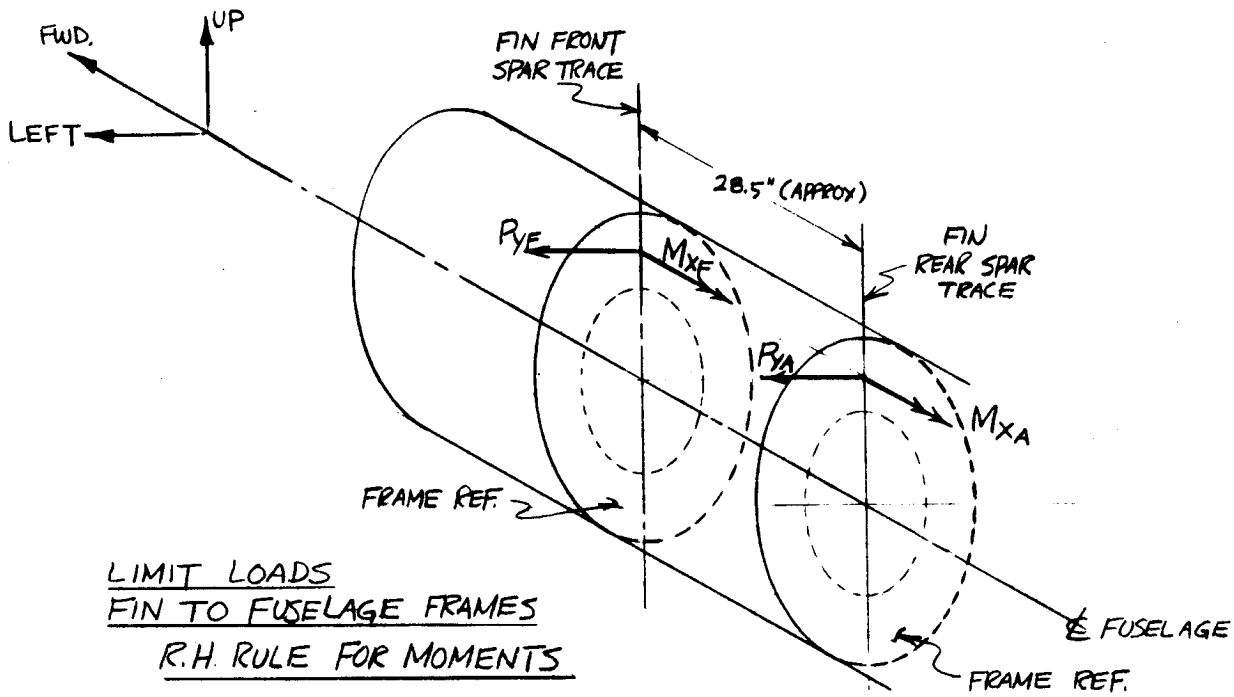


Figure VII-37. Vertical Fin Skin and Web Thickness Requirements



	LBS. P_{YF}	LBS. P_{YA}	IN-LBS. M_{XF}	IN-LBS. M_{XA}
COND. 8	5910	5390	211000	220500
COND. 9	9240	3760	332600	155000

NOTE: LOADS SHOWN ARE FOR FLIGHT CONDITIONS WHERE RESULTANT FIN LOAD IS TO THE LEFT. RESULTANT FIN LOAD COULD ALSO ACT TO RIGHT, IN WHICH CASE ABOVE LOADS ARE REVERSED. USE R.H. RULE FOR MOMENTS

Figure VII-38. Fuselage-Fin Attachment Limit Loads

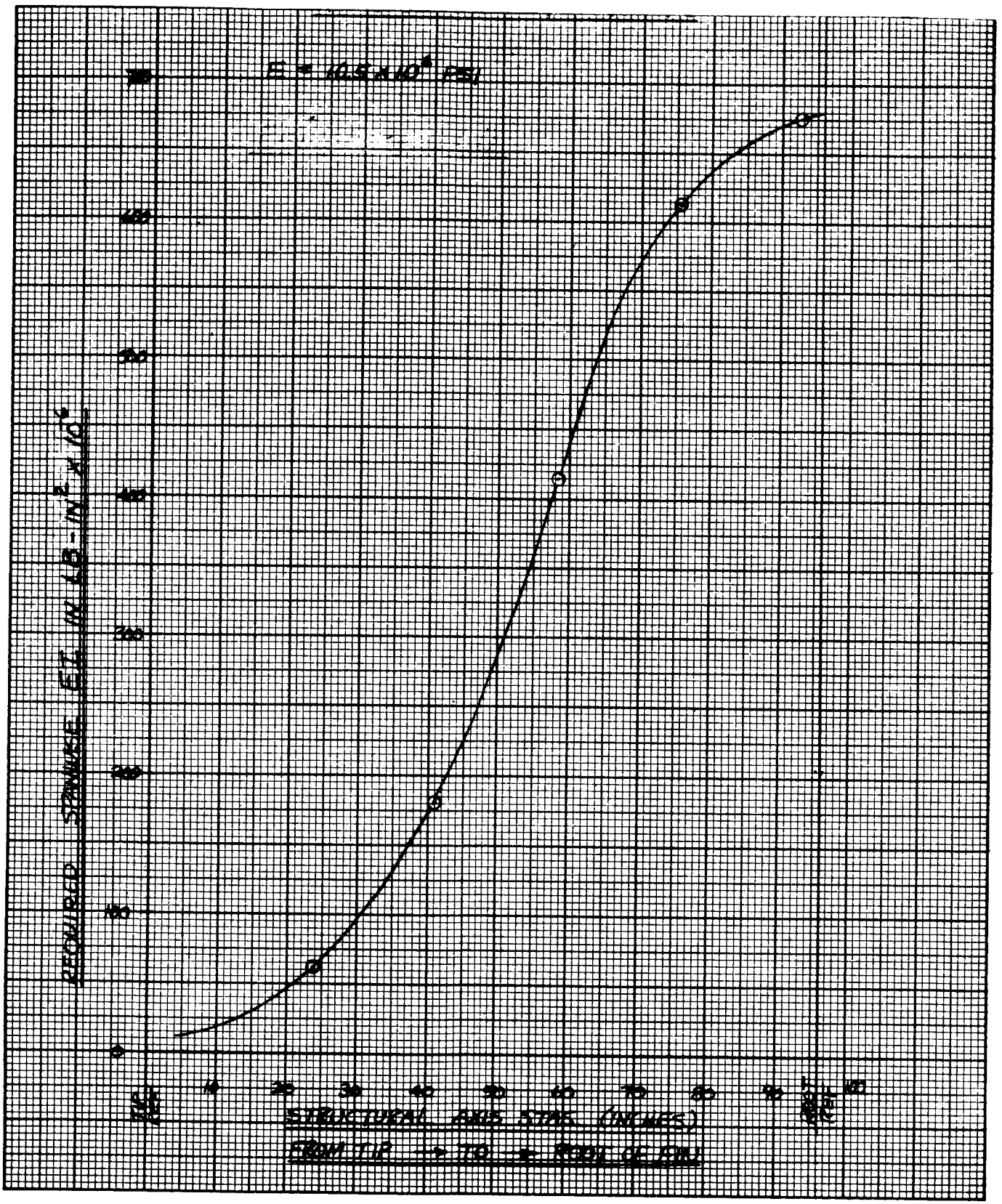


Figure VII-39. Fin Bending Stiffness

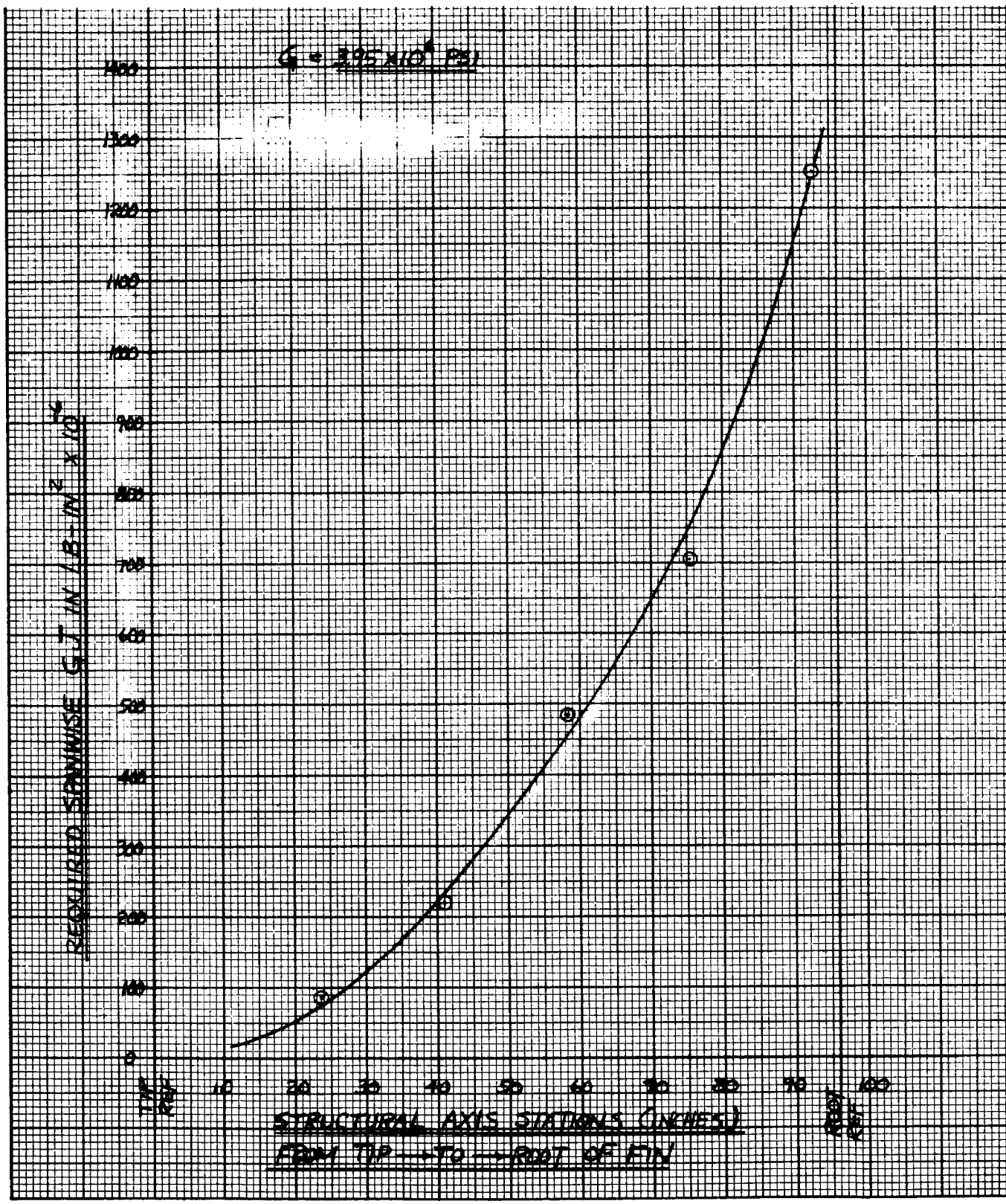


Figure VII-40. Vertical Fin Torsional Stiffness

K. FUSELAGE ANALYSIS

Fuselage shears and moments were computed for all of the conditions shown in Sections D and E except for Landing Conditions 16, 18, and 19 which were considered noncritical by comparison with Condition 15. A listing of the conditions analyzed is given in Table VII-6. This table further defines the landing conditions of Section E according to the direction of the landing loads in the horizontal plane. The shear and moment calculations were performed utilizing a computer program (Reference 2).

In formulating the flight conditions, the wing loads are introduced into the fuselage as wing spar loads and the tail loads are introduced as shears and concentrated moments at the horizontal tail attachment bulkhead. The body loads associated with the wing and tail loads were introduced as distributed loadings along the fuselage.

The dead weight loadings were based on a panel point weight distribution.

For conditions involving both side and vertical loads, shears and moment were computed for the side and vertical loading configurations separately and the resulting distributions were combined in the detail analysis as required.

Results of the shear and moment calculations are given in Figures VII-41 through VII-43 in the form of envelopes of maximum vertical and side shear and bending moments.

Landing Condition 17 (vertical plus obstruction) was generally critical for vertical bending between F.S. 120 and 325. For the rear fuselage between F.S. 360 and 440, the maximum down tail load (Condition 7) produced the maximum bending moments. At the rear spar bulkhead F.S. 281.5 where the maximum fuselage bending occurs, the bending moments from the maximum flight condition (Condition 7) were 71% of the design values obtained from Condition 17A. Further details of the critical loadings are presented in the analysis results discussed later.

The fuselage structure in the engine bay consists of the two body side walls and a center beam, together with the top of the fuselage and nacelle. Vertical shear and bending moments are resisted by the three vertical beams while side shear and moment are resisted by the top fuselage and nacelle structure. Fuselage torsion in this region is transmitted by differential bending in the fuselage sidewalls.

An analysis was performed to determine the distribution of internal loadings at the wing support bulkheads due to vertical and side bending loads on the fuselage.

The vertical shears and moment are distributed to the three vertical beams as mentioned above. Since side shear is resisted in the top of the fuselage and nacelle, side loads are transferred up from the Horizontal Reference Line to the upper fuselage with the torque from this transfer being resisted by differential bending of the fuselage side walls as is primary fuselage torque. The side moment at the forward end of the engine section is resisted by couple loads in the upper fuselage longerons. The side shear is divided equally in the upper fuselage and nacelle skin panels on each side. Side shear introduced between the forward end of the engine section and a given section being analyzed is integrated to give moment which is resisted equally by differential bending of the two fuselage-nacelle skin panels and longerons. These incremental shears produce incremental torques about the shear center of the side load resisting structure, which torques are also resisted by differential bending in the vertical side walls.

A typical engine bay structural section is shown in Figure VII-44. The primary longerons, members A, B, E, F and G are indicated. Results of the analysis of engine bay sections at fuselage stations 247 and 281.5 are given in Figures VII-45 and VII-46 respectively. Only the maximum loads in a given direction followed by the design condition number in parenthesis are quoted in these figures. In the analysis of the section these loads were combined with the corresponding loads in the other directions as appropriate. The material is general Ti 6AL-4V annealed. Material properties were based on 100 hours exposure at the temperatures

given in Paragraph G. Landing condition loadings were assumed to occur at temperature.

Figure VII-47 shows the analysis results of a typical section in the aft fuselage. This section is taken at F.S. 384.5 which is at the forward end of the electronics bay. The large access door between the lower longerons is designed to carry primary shear loads only. The maximum design loadings in each direction are indicated in the figure. These occurred in Conditions 4, 7 and 9. It will be noted that a light gage conventional aluminum alloy structure is sufficient to carry these loads.

TABLE VII-6. LOADING CONDITIONS USED FOR FUSELAGE SHEARS AND MOMENTS

Symmetric Flight Conditions

Condition	Mach No.	Load Factor n_z	Flaps	Pitching Acceleration
1	.7	3.75	Up	-
2	.7	-1.5	Up	-
3	.33	- .59	Dwn.	-
4	.4	3.75	Up	- 5 rad/sec
5	.7	1.88	Up	5
6	.7	-1.0	Up	5
7	.33	0	Dwn.	5

Asymmetric Flight Conditions

Condition	Mach No.	Load Factor n_y	Sideslip Angle-Degrees	Rudder Angle-Degrees
8 (Steady Sideslip)	.7	1	9.5	0
9 (Max. Rudder Defl.)	.7	1	15.2	10.

Landing Conditions

Condition	Description
11	Vertical Landing: 2-Point
12	Vertical Landing: 3-Point
13	Conventional Landing: Spin-Up
14	Conventional Landing: Spring Back
15S1	5 Knot Drift - Side Load/Main Gear
15S2	5 Knot Drift - Side Load/Nose Gear
17A	Vertical Obstruction: Aft Load/Main Gear
17B	Vertical Obstruction: Fwd. Load/Main Gear
17C	Vertical Obstruction: Aft Load/Nose Gear
17D	Vertical Obstruction: Fwd. Load/Nose Gear

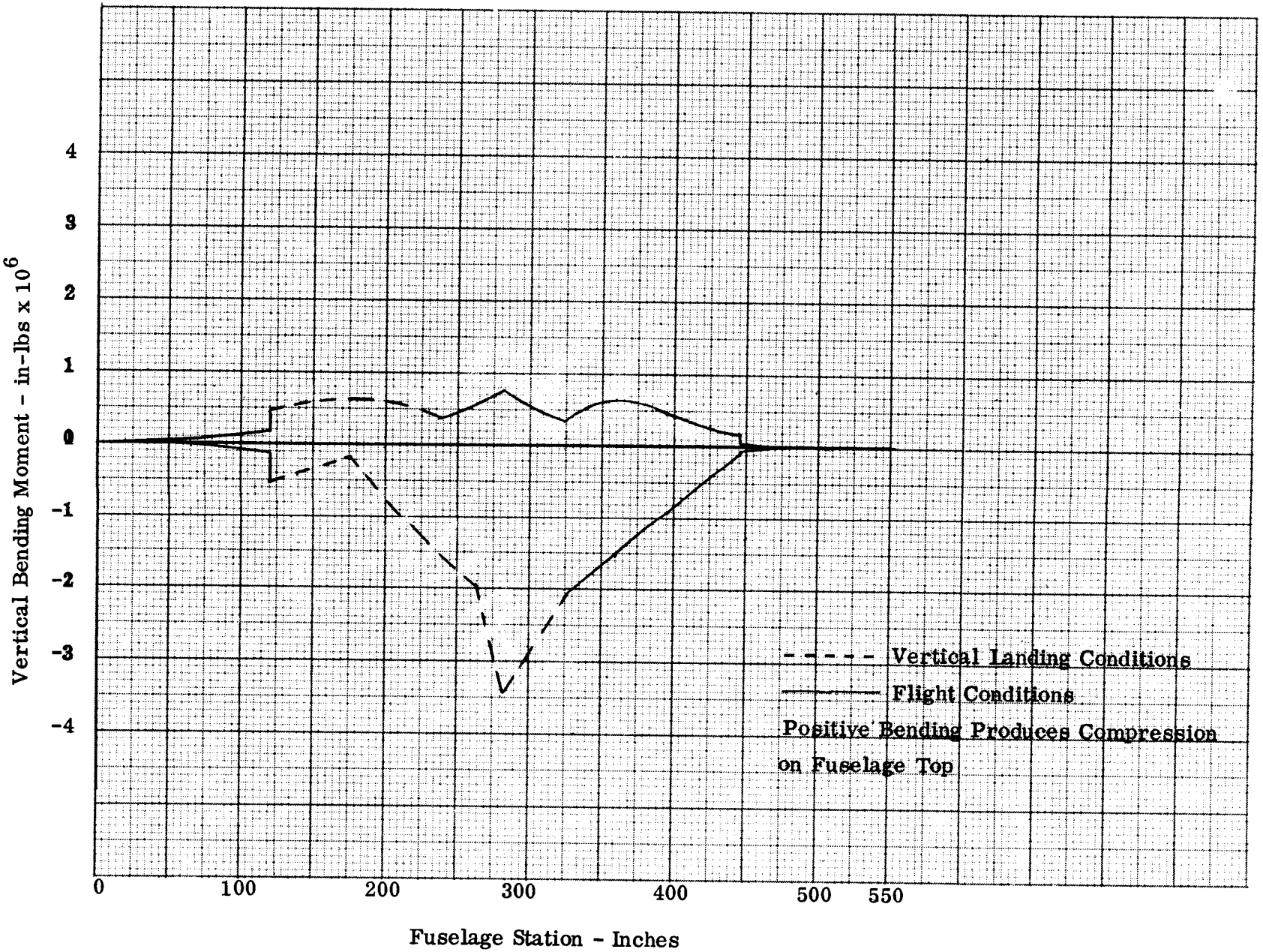


Figure VII-41. Fuselage Vertical Bending Moment Envelope

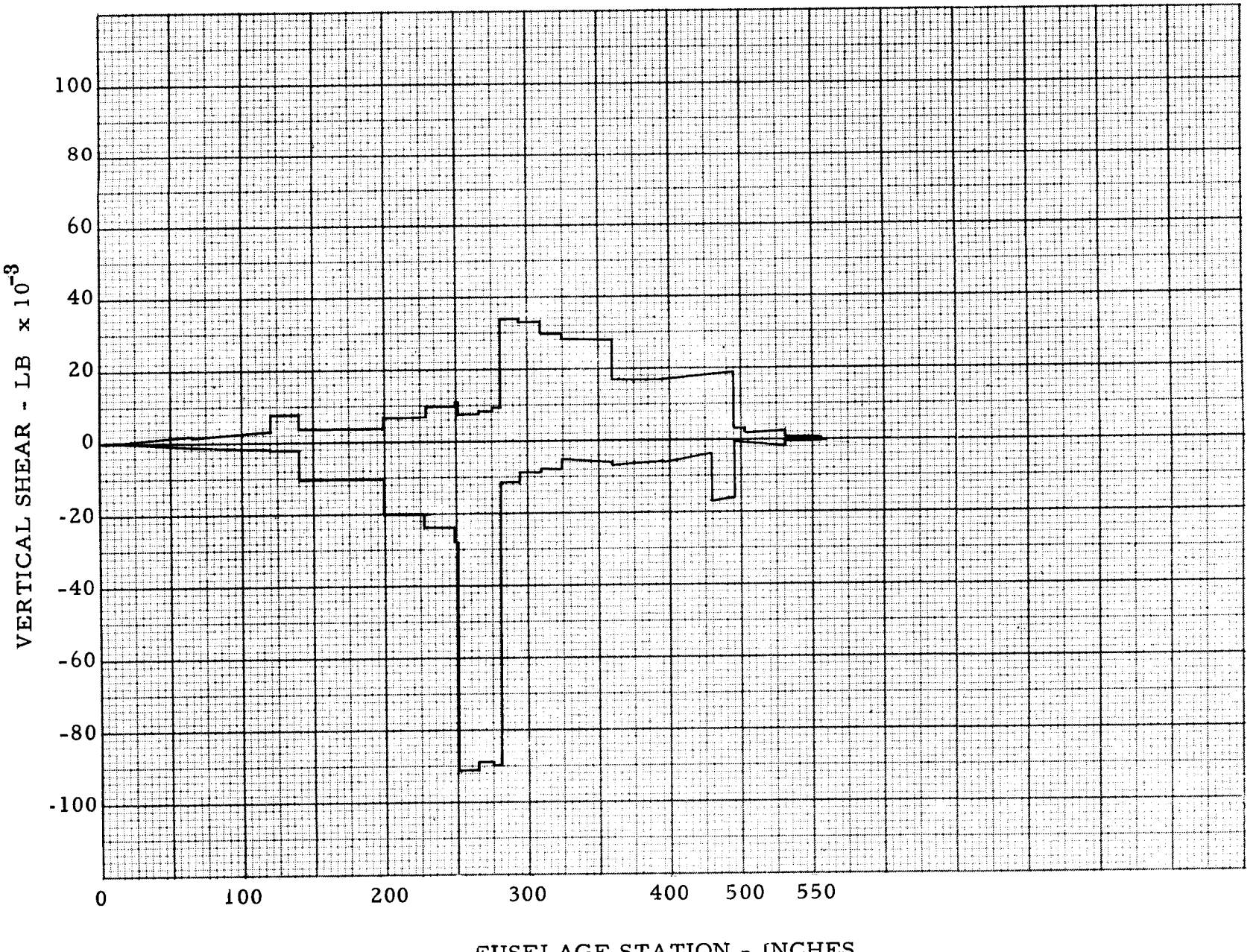


Figure VII-42. Fuselage Vertical Shear Envelope

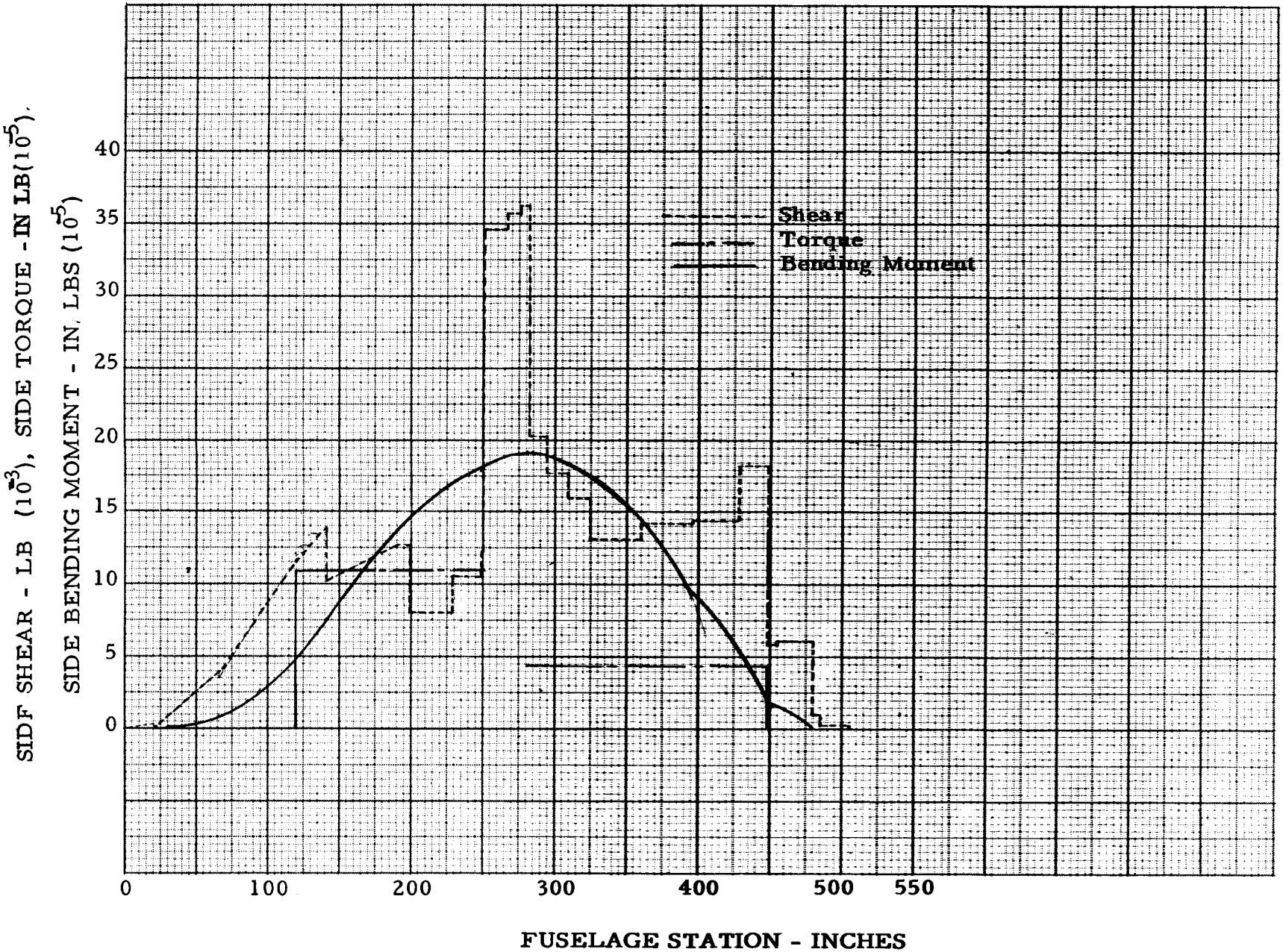


Figure VII-43. Fuselage Side Shear, Torque and Bending Moment Envelopes

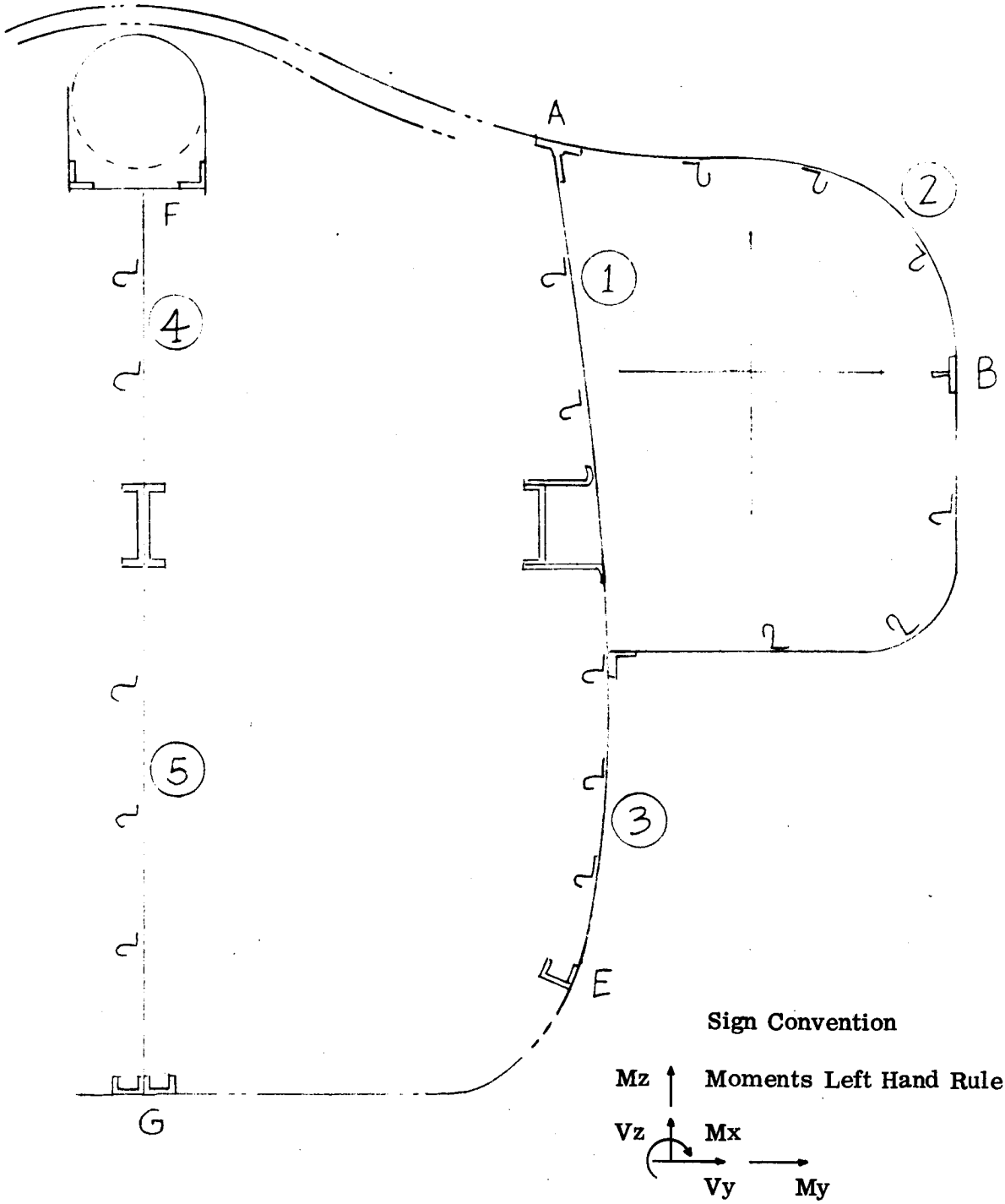


Figure VII-44 Typical Section - Engine Bay

Maximum External Loads (Ult)

V_z = -28350 lb Fwd (1)
-91220 lb Aft (17A)

M_y = -1807910 in lb (17B)

V_y = -12280 lb Fwd (15S1)
-34690 lb Aft (15S1)

M_z = 1801000 in lb (9)

M_x = 43690 in lb (nose gear side load - 15S2)

Maximum Longeron Loads

Point	A	B	E	F	G
Load (lb)	+29590 -25450	+4940	-9840	+7870	-7850
Effect Area (in^2)	0.313	0.112	0.129	0.234	0.151

Stringers Max Load = -3620 lb Effect. Area = .072 in^2

Skin Shears

Item	Forward			Aft		
	Side Walls	Center Wall	Nacelle Top	Side Walls	Center Wall	Nacelle Top
Av. shear (lb/in)	250	31	308	810	102	870
Skin thickness (in)	.032	.025	.032	.050	.025	.050

Material Ti 6 Al - 4V Annealed

Figure VII-45. Analysis Results - Fuselage Station 247

Maximum External Loads (Ult)

V_z = -90140 lb Fwd (17A)
+34000 lb Aft (17A)

M_y = -3,497,300 in lb (17A)

V_y = -36220 lb Fwd (15S1)
-20340 lb Aft (15S1)

M_z = 1,929,600 in lb (9)

Maximum Longeron Loads

Point	A	B	E	F	G
Load (lb)	-33200 +51600	+31200	-22700	+7870	-7850
Effect Area (in^2)	0.553	0.280	0.287	0.234	0.151

Stringers

Max. Stringer Load = 5500 lb
Effect. Area = 0.105 in^2

Skin Shears

Item	Forward			Aft		
	Side Walls	Center Wall	Nacelle Top	Side Walls	Center Wall	Nacelle Top
Av. Shear (lb/in)	800	101	905	302	38	505
Skin Thickness (in)	.050	.025	.050	.032	.025	.032

Material Ti 6Al - 4V Annealed

Figure VII-46. Analysis Results - Fuselage Station 281.5

Maximum External Loads

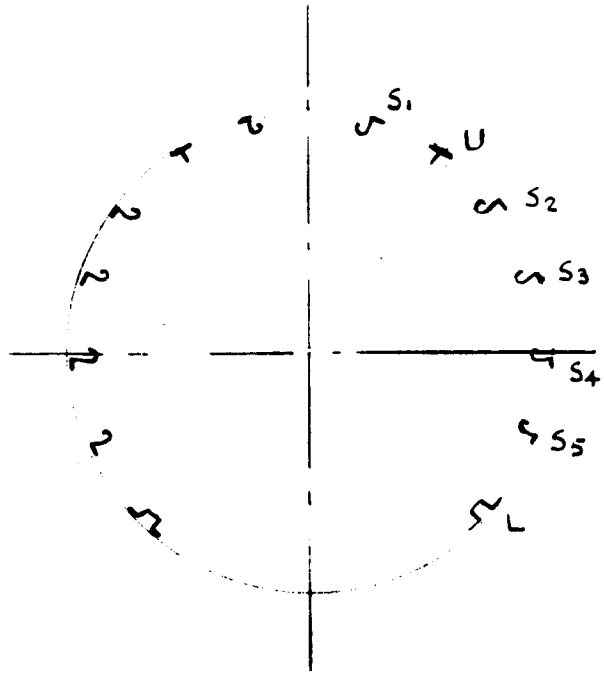
V_z = 17000 lb (9)

V_y = 14200 lb (7)

M_y = -1.1×10^6 in lb (7)

$+0.56 \times 10^6$ in lb (4)

M_z = 1.14×10^6 in lb (9)



Section F.S. 384.5

Maximum Longeron and Stringer Loads

Point	U	L	S1	S5
Load (lb)	+14500	+15400	+4980	+7150
	-4500	-13100	-2030	-8960
Effect. Area (in^2)	.244	.316	.0909	.1587

Skin

Av. Shear Flow = 297 lb/in Skin Thickness = .032 in

Materials Aluminum alloys 2024 T3 and 7075 T6

Figure VII-47. Analysis Results - Fuselage Station 384.5

L. LANDING GEAR ANALYSIS

A complete detailed stress analysis was done on the main gear to substantiate the design and justify the weight estimate. The vertical loads used were those originally derived for a rigid structure support which are approximately 11% higher than the final values presented in Paragraph E, hence the results presented here are conservative.

A summary of the main gear stress analysis results are presented in Figure VII-48. The strut sections were calculated for a 220 KSI heat treat steel and a 260 KSI heat treat steel to determine the effect on weight and cost. The use of a 260 KSI heat treat steel resulted in a weight saving of approximately 30 pounds for two struts.

Figure VII-49 presents a sketch of the main gear indicating a sign convention for loads. The ultimate loads applied to the wing support structure by the main gear trunnion are given in Table VII-7.

The following data are presented on the nose gear analyses.

1) Figure VII-50 - showing a sketch of the nose gear

2) Table VII-8 - presenting nose gear trunnion loads applied to the fuselage structure

3) Table VII-9 - presenting nose gear drag brace loads applied to the fuselage structure

MODE	CONDITION		GROUND LOADS P_z (LBS) UP(+) DOWN(−)	P_x (LBS) AFT(+) FWD(−)	P_H * (LBS) RIGHT(+) LEFT(−)	△ TRUNNION LOADS $_zP_T$ (LBS) UP(+) DOWN(−) $_zP_{TF}$	$_zP_{TA}$	$_xP_T$ (LBS) AFT(+) FWD(−)	$_HP_T$ (LBS) RIGHT(+) LEFT(−) $_HP_{TF}$	$_HP_{TA}$	SIDE BRACE LOADS △ $_zP_{SB}$ (LBS) UP(+) DOWN(−)	$_HP_{SB}$ (LBS) RIGHT(+) LEFT(−)
VERTICAL	2-POINT		30,000	—	—	12,000	18,000	—	—	—	—	—
VERTICAL	3-POINT		26,025	—	—	10,400	15,625	—	—	—	—	—
CONVENTIONAL	SPIN-UP		17,700	12,525	—	-35,100	52,800	12,525	—	—	—	—
CONVENTIONAL	SPRING-BACK		17,370	-11,895	—	47,000	-29,630	-11,895	—	—	—	—
VERTICAL	** OBSTRUCTION (INB'D-OUTB'D)	A	30,000	—	15,000	-250	7950	—	-27,500	-18,500	22,300	61000
		B	30,000	—	-15,000	29,300	28,000	—	27,500	18,500	-22,300	-61,000
CONVENTIONAL	DRIFT		LESS CRITICAL THAN CONDITION "OBSTRUCTION" INB'D OUT'D									
VERTICAL	** OBSTRUCTION (AFT-FWD)	A	30,000	15,000	—	-52,000	82,000	15,000	—	—	—	—
		B	30,000	-15,000	—	76700	-46,700	-15,000	—	—	—	—
VERTICAL	(** 5 KNOT DRIFT)	*** A	30,000	—	21,000	-4200	4800	—	-36,000	-23,900	+29,400	+81,000
		*** B	30,000	—	-21,000	28,100	31,300	—	36,000	23,900	-29,400	-81,000

* P_H Acts at Ground Line

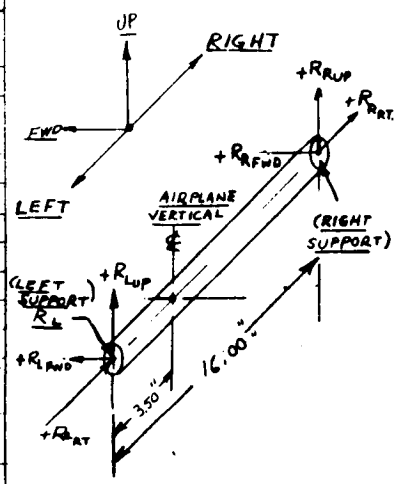
** Where Two Conditions Exist (A&B) Loads Act Inb'd or Outb'd - Aft or Fwd Respectively

*** This Condition is Used Only for Wing Design

Effect of Retracting Cylinder Loads not Shown Note: See Figure VII-49 for Load Notation

Table VII-7. Main Gear Ultimate Loads Applied to Wing

MODE	CONDITION-TYPE	CONDITION	V** (Lbs.) UP (+) DOWN (−)	S (Lbs.) RIGHT (+) LEFT (−)	D (Lbs.) FWD (+) AFT (−)	R_R (Lbs.) UP (+) DOWN (−)	R_R RIGHT (+) LEFT (−)	R_R FWD (+) AFT (−)	R_L (Lbs.) UP (+) DOWN (−)	R_L RIGHT (+) LEFT (−)	R_L FWD (+) AFT (−)
VERTICAL	3-POINT	1-A	8550_{LW}	—	—	-940	—	-495	11900	—	-1775
		1-B	8550_{RW}	—	—	5510	—	-495	4950	—	-1775
CONVENTIONAL	SPIN-UP	2-A	5025_{LW}	—	-2745	-1125	—	1860	1500	—	1895
		2-B	5025_{RW}	—	-2745	1775	—	-205	-2200	—	3960
CONVENTIONAL	SPRING-BACK	3-A	4680_{LW}	—	3525	1280	—	-3000	12800	—	-11725
		3-B	4680_{RW}	—	3525	4830	—	-375	9250	—	-7300
VERTICAL	5 KNOT DRIFT	4-A	8550_{LW}	5985	—	28460	5985*	-2175	-18000	5985*	-95
		4-B	8550_{RW}	5985	—	34910	5985*	-2175	-24450	5985*	-95
		4-C	8550_{LW}	-5985	—	-30340	-5985*	1185	40800	-5985*	-3455
		4-D	8550_{RW}	-5985	—	-23840	5985*	1185	34350	5985*	-3455
VERTICAL	FWD. OBSTRUCTION	5-A	8550_{LW}	—	-5985	-3940	—	4150	190	—	4565
		5-B	8550_{RW}	—	-5985	2400	—	-335	-6150	—	9050
VERTICAL	AFT. OBSTRUCTION	6-A	8550_{LW}	—	5985	2150	—	-5150	22600	—	-20135
		6-B	8550_{RW}	—	5985	8550	—	-715	16200	—	-12600
—	Δ- UNIT (1000# COMP.) RETRACTING CYL. LOADS	—	—	—	—	65	—	-75	-110	—	671



SIGN CONVENTION

* LOAD ACTS AT EITHER R_R OR R_L

** SUBSCRIPT (R.W.)–LOAD ACTS ON RIGHT WHEEL, SUBSCRIPT (L.W.)–LOAD ACTS ON LEFT WHEEL

Δ- RETRACTING CYLINDER LOADS SHOULD BE ADDED TO R_R & R_L OF OTHER CONDITIONS WHERE THEY AGGRAVATE (INCREASE) LOADS

Table VII-8. Nose Gear Trunnion Ultimate Loads Applied to Fuselage

MODE	CONDITION-TYPE	CONDITION	GROUND LOADS V* (LBS.) UP(+) DOWN(-)	GROUND LOADS S (LBS.) RIGHT(+) LEFT(-)	GROUND LOADS D (LBS.) FWD(+) AFT(-)	P_D-DRAG BRACE LOADS TOTAL (LBS.) TENSION(+) COMP.(-)	P_D-DRAG BRACE LOADS COMPONENTS (LBS) UP(+) DOWN(-)	P_D-DRAG BRACE LOADS COMPONENTS (LBS) FWD(+) AFT(-)
VERTICAL	3-POINT	1-A	8550_{LW}	—	—	-2960	-1910	+2270
		1-B	8550_{RW}	—	—			
CONVENTIONAL	SPIN-UP	2-A	5025_{LW}	—	-2745	+8450	+5450	-6500
		2-B	5025_{RW}	—	-2745			
CONVENTIONAL	SPRING-BACK	3-A	4680_{LW}	—	3525	+14600	-9400	+11200
		3-B	4680_{RW}	—	3525			
VERTICAL	5 KNOT DRIFT	4-A	8550_{LW}	5985	—	-2960	-1910	+2270
		4-B	8550_{RW}	5985	—			
		4-C	8550_{LW}	-5985	—			
		4-D	8550_{RW}	-5985	—			
VERTICAL	FWD. OBSTRUCTION	5-A	8550_{LW}	—	-5985	+19200	+12300	-14700
		5-B	8550_{RW}	—	-5985			
VERTICAL	AFT OBSTRUCTION	6-A	8550_{LW}	—	5985	-25200	-16200	+19300
		6-B	8550_{RW}	—	5985			

*SUBSCRIPT (R.W.)~LOAD ACTS AT RIGHT WHEEL, (L.W.)~LOAD ACTS AT LEFT WHEEL

Table VII-9. Nose Gear Drag Brace Ultimate Loads Applied to Fuselage

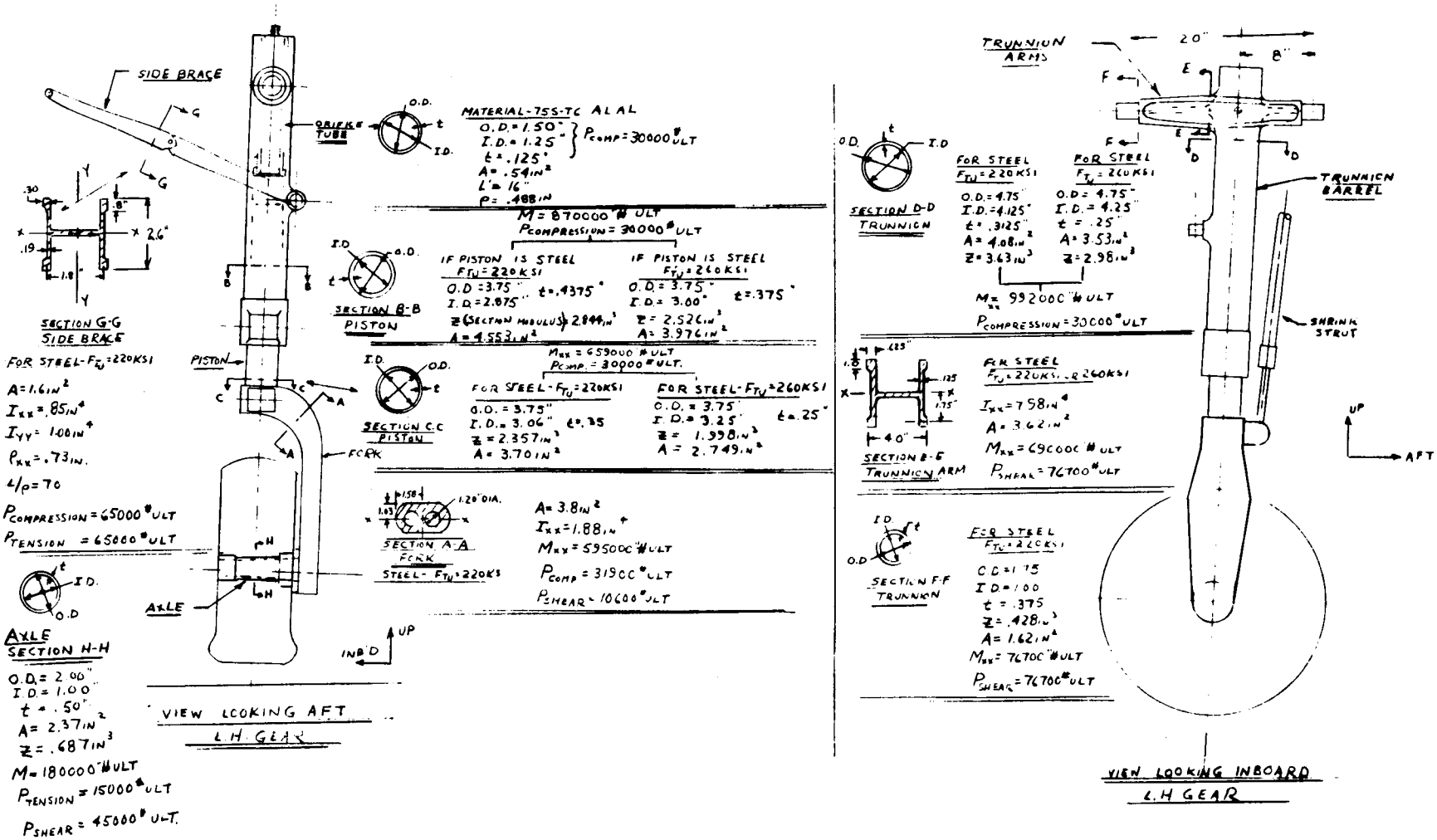


Figure VII-48. Results of Main Gear Stress Analysis

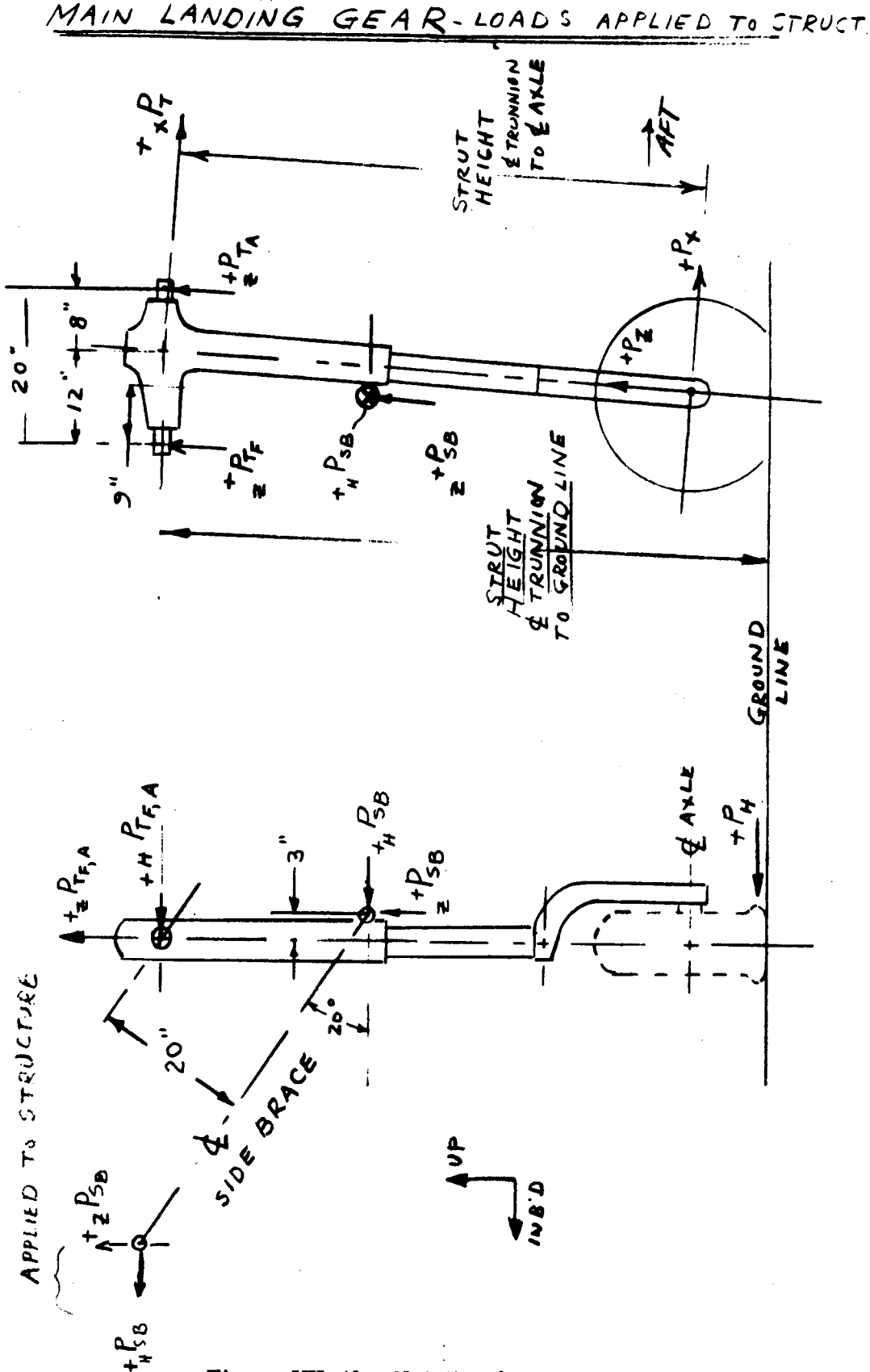


Figure VII-49. Notation for Main Gear Loads

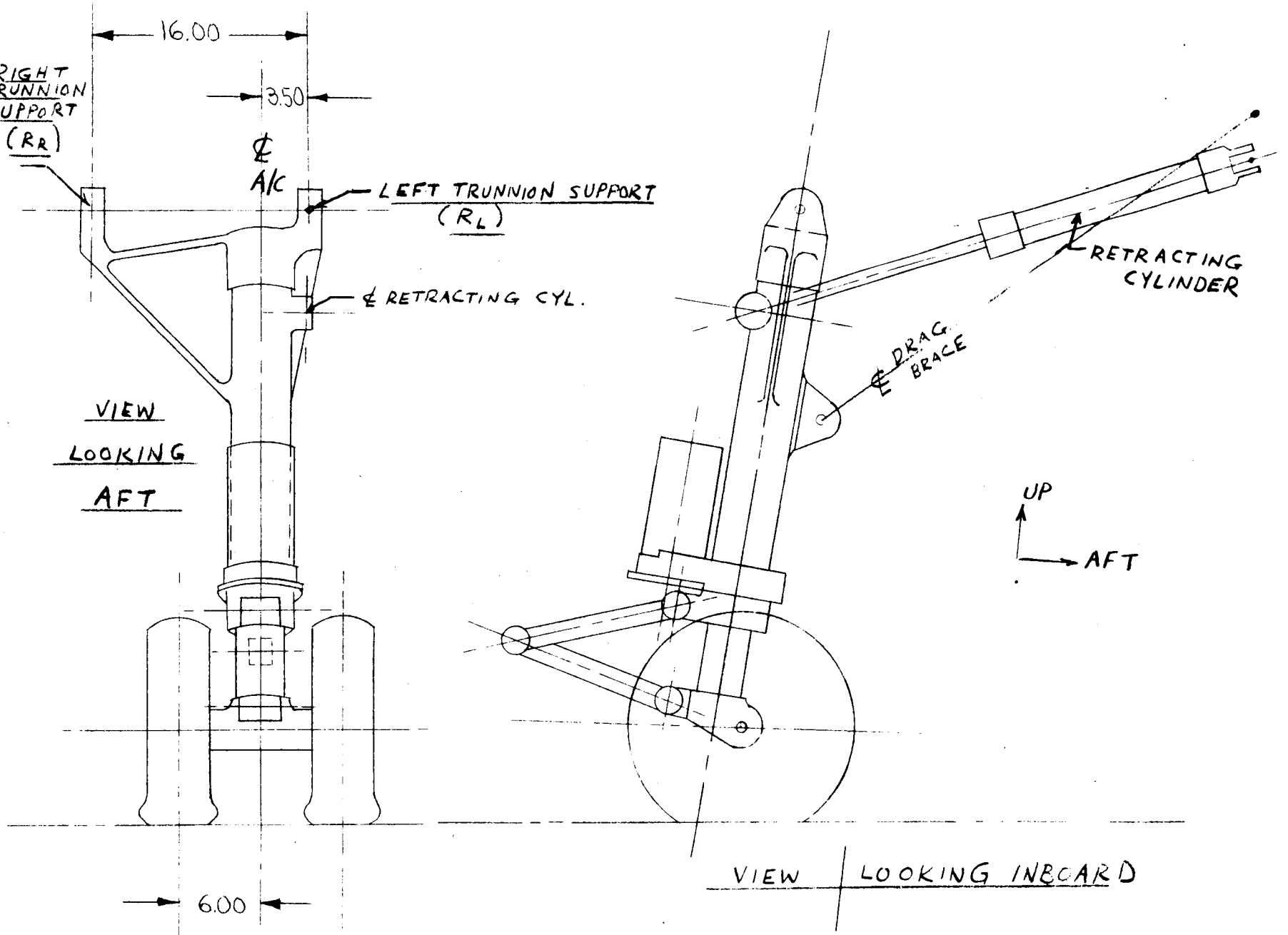


Figure VII-50. Sketch of Nose Gear

M. FATIGUE

A flight fatigue analyses was done for range of K_F values (ratio of fatigue strengths of plain and notched specimens) to establish the value of K_F at which the static allowable stress could be used for design and also meet the required fatigue life. The load spectrum included gust, flight maneuver and landing load occurrences for the NASA required service life of 300 hours with a scatter factor of 4. Estimation of gust intensity and frequency were taken from Reference 3. The spectrum is shown in Table VII-10. The fatigue damage in aluminum alloy 2024 T3 clad sheet was estimated using Miner's Rule. The results are presented in Figure VII-51. This shows that the fatigue spectrum does not entail any penalty on the static strength design below values of K_F equal to 2.75. This value of K_F covers most of the stress concentrations likely to be obtained in normal design and hence it can be concluded that the required service life will not entail any significant structural penalty for this aircraft.

TABLE VII-10 FATIGUE SPECTRUM

FLIGHT FATIGUE SPECTRUM - MANEUVER PLUS GUST

300 Hours with a Scatter Factor of 4.0

Limit Load Factor = 3.75g

Normal Load Factor n_z	Occurrences
1.31	171866
1.67	18480
2.06	3880
2.44	1224
2.81	364
3.19	120
3.56	36
3.94	12
4.31	4
4.69	2

TABLE VII-10 (Cont.)

LANDING FATIGUE SPECTRUM

300 Hours with Scatter Factor of 4.0

4 Landings per Hour = 4800 Cycles

Normal Load Factor n_z	Occurrences
.68	624
.72	960
.79	1103
.89	936
1.02	610
1.18	322
1.35	145
1.56	61
1.80	24
2.07	10
2.36	3
2.68	1
3.03	1

EFFECT OF FATIGUE SPECTRUM ON ALUMINUM ALLOY 2024 T3 CLAD SHEET

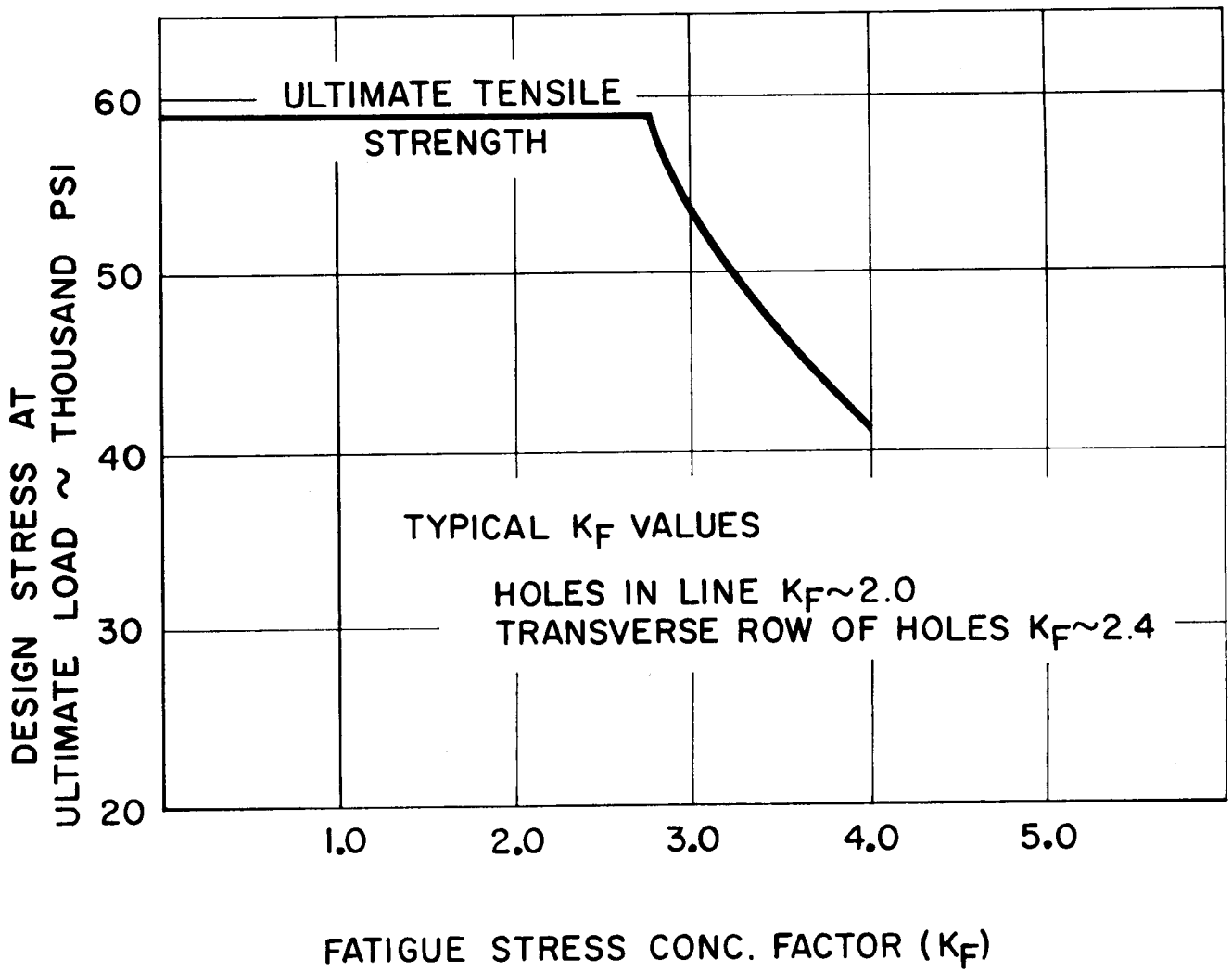


Figure VII-51

N. FLUTTER

1. General

In order to insure that the stiffness levels in the wing, horizontal tail, and vertical tail are sufficient to preclude the occurrence of flutter throughout the operating range of the airplane, three dimensional uncoupled mode flutter analyses were performed using computed incompressible air forces corrected for finite span effects. Computations of the inertia and zero forces as well as the solutions for the related eigenvalues and eignevectors were all performed on the IBM 7094 computer. The results indicate that the aircraft will have ample flutter margin throughout the flight regime.

2. Wing Analysis

The wing was analyzed for the generally more critical symmetric case. Degrees of freedom considered are uncoupled elastic wing first bending, torsion, and second bending, plus rigid airplane translation and pitch for which computed uncoupled frequencies are respectively, 10.4, 30.9 and 39.0 cps. The results of the analysis are presented in Figure VII-52 which shows that the wing incompressible flutter speed is 760 knots at sea level. Figure VII-53 displays the airplane speed envelope and the flutter boundary for the wing (which is also the flutter boundary for the airplane since the tail surfaces have higher flutter speeds). It shows that the minimum margin is well in excess of the amount required.

3. Horizontal Tail Analysis

The analysis of the horizontal tail is based on the first and second panel bending, panel torsion, and panel pitch degrees of freedom, in which pitch frequencies of 10, 20, 30, 40, cps and infinity were considered. Computed uncoupled first and second panel bending and panel torsion frequencies are respectively 42.1, 179.0 and 86.0 cps. The results of the analyses for the various pitch frequencies considered are presented in Figure VII-54 which indicates a minimum incompressible flutter speed of 1120 knots at sea level.

4. Vertical Tail Analysis

The vertical tail analysis is based on the first and second panel bending and the panel torsion degrees of freedom. Computed uncoupled frequencies are respectively 32.6, 121.0, 92.5 cps. Analysis results are presented in Figure VII-55 which indicates a vertical tail incompressible flutter speed of 1580 knots. To determine proper levels of rudder rotational stiffness and mass balance distribution, further analytical work incorporating fin-rudder elastic modes and antisymmetric fuselage modes will be required.

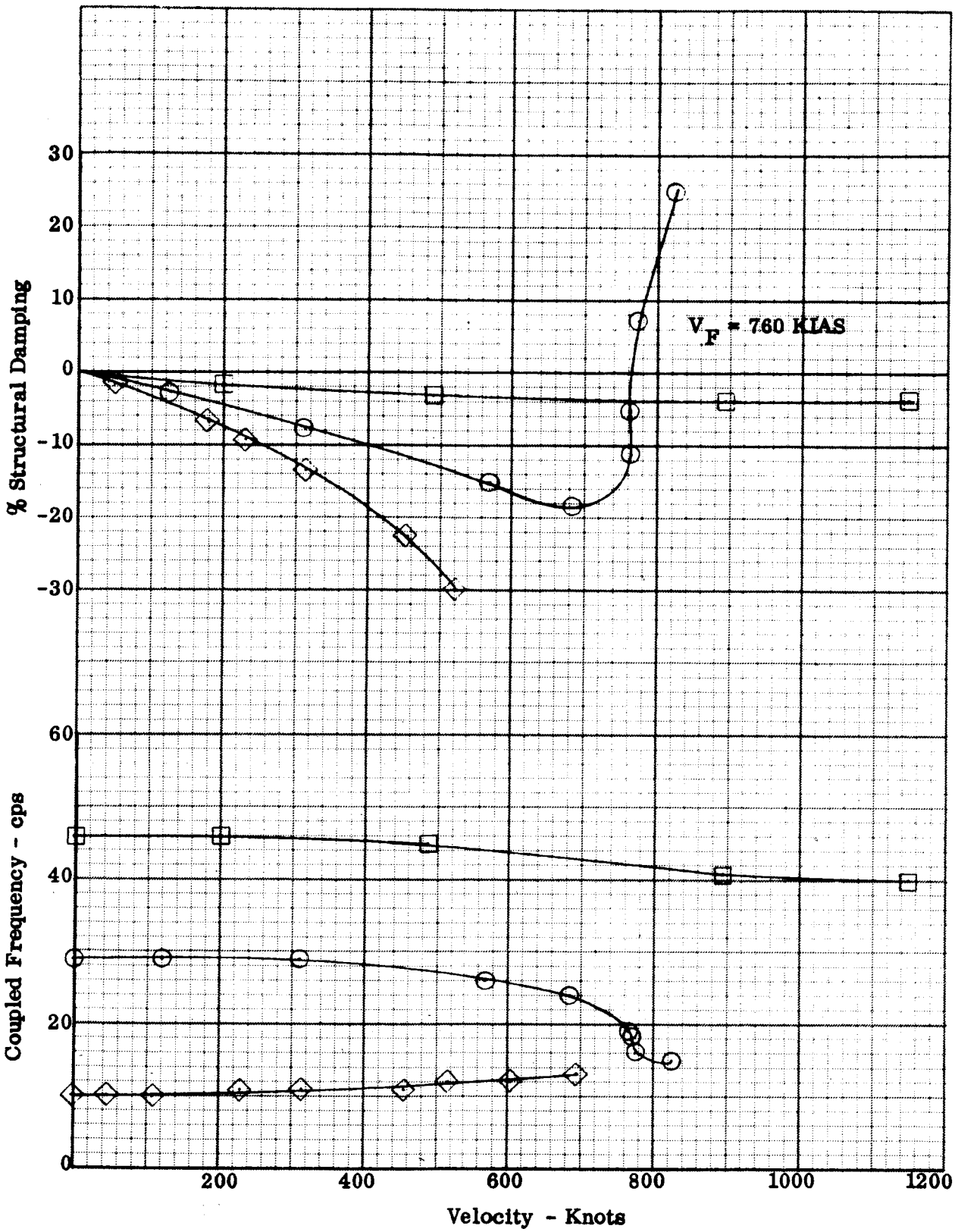


Figure VII-52. Wing Flutter Results
Velocity vs Damping and Velocity vs Frequency

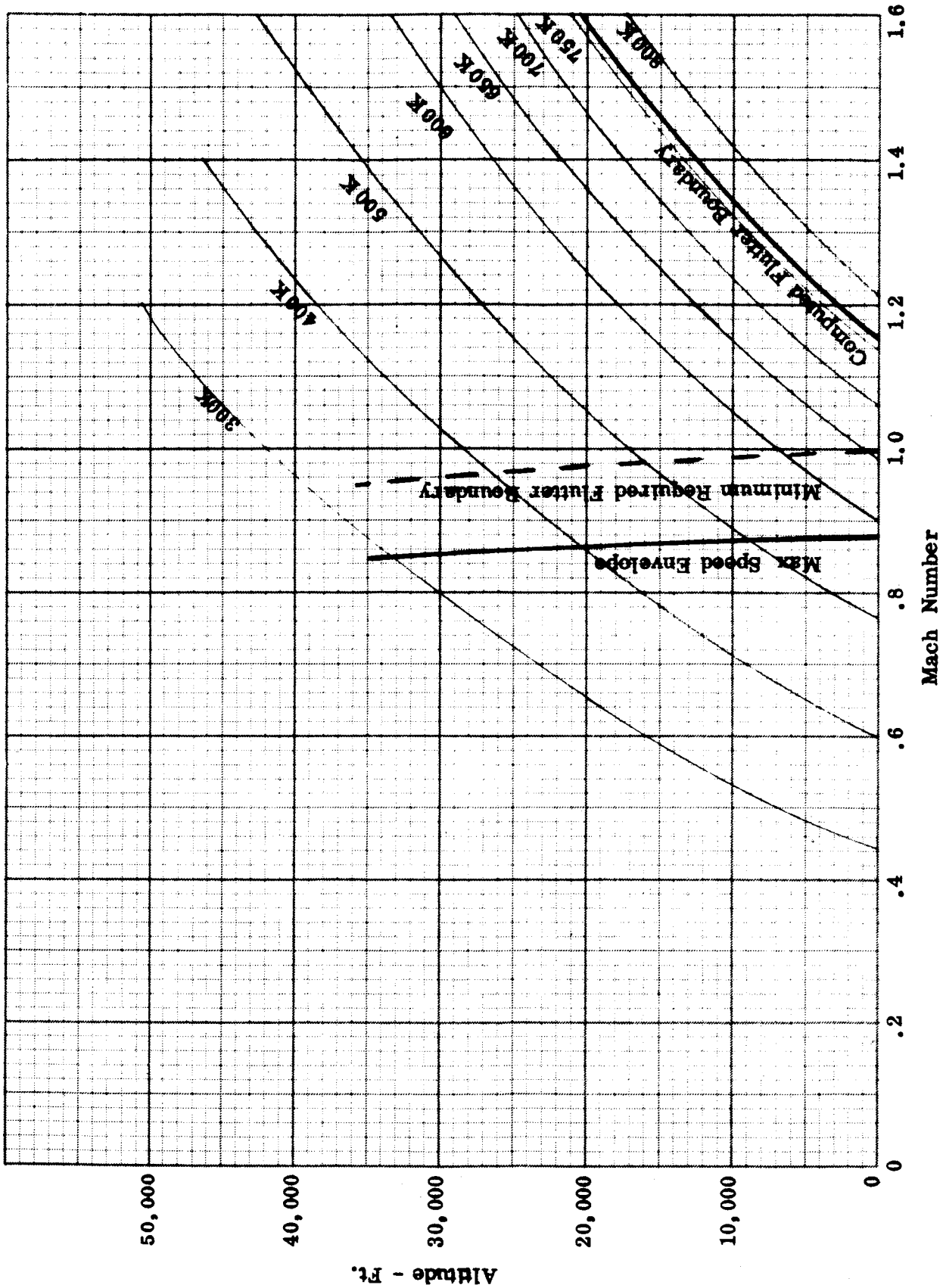


Figure VII-53. Aircraft Flutter Envelope

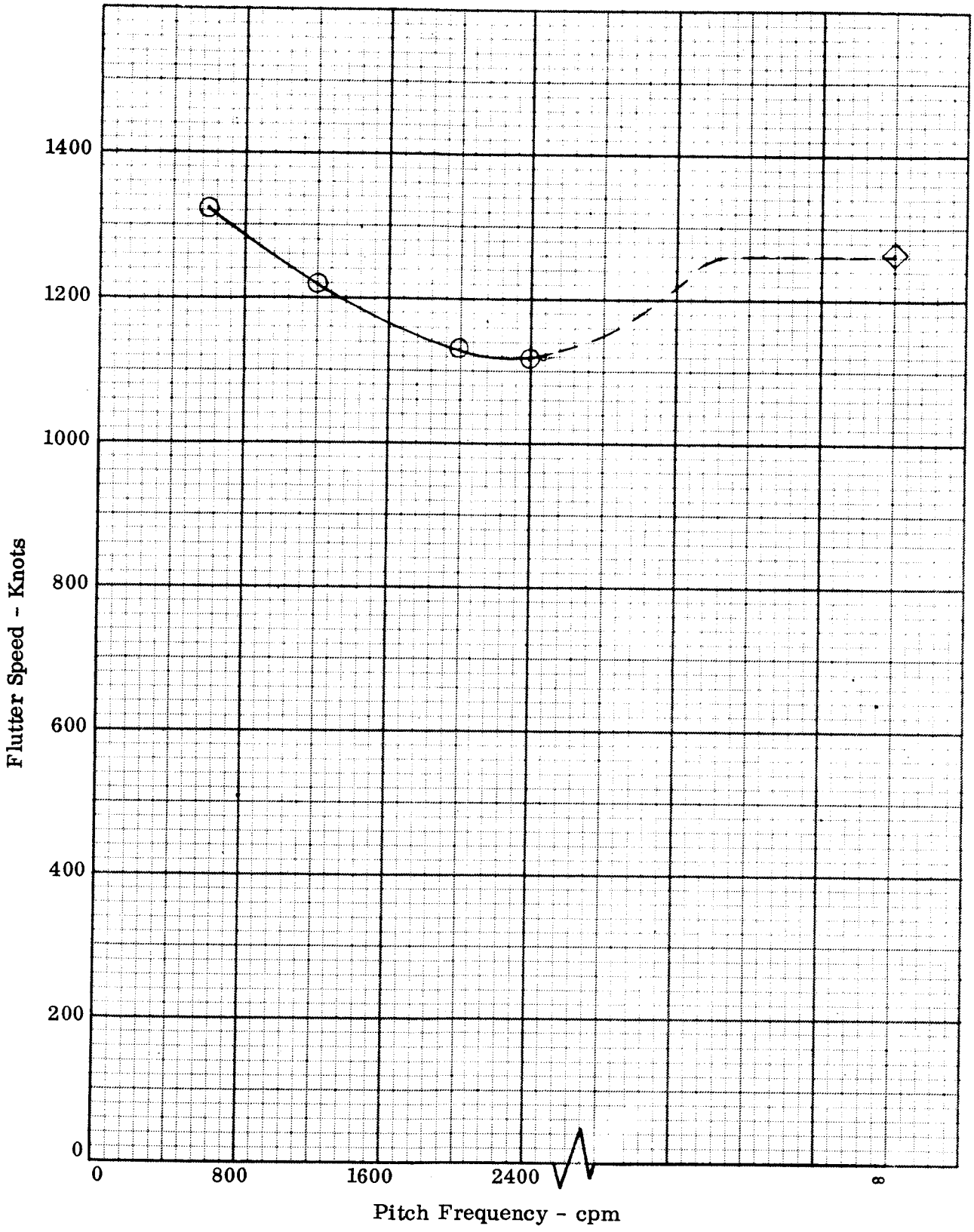


Figure VII-54. Horizontal Tail Flutter Results Flutter Speed vs. Horizontal Tail Pitch Frequency

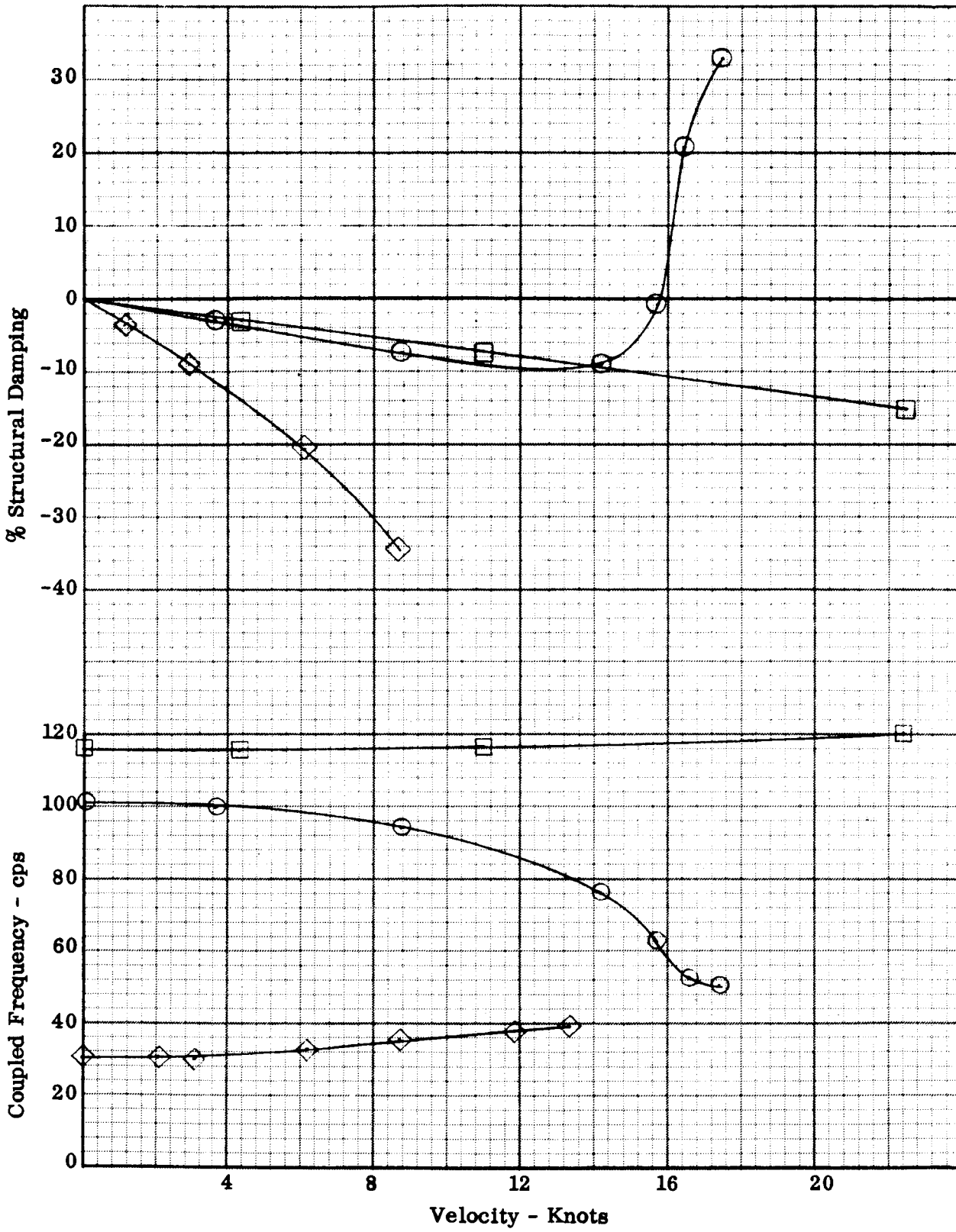


Figure VII-55. Vertical Tail Flutter Results - Velocity vs. Damping and Velocity vs. Frequency

STRUCTURES - CONCEPT J

A. FLIGHT LOADS

Concept J is a modified stretched Sabreliner. According to North American Aviation this aircraft is designed to a limit flight load factor of 3.0 and -1.0 at a weight of 20000 lb. The estimated V-n diagram (based on data for the T39A) is shown in Figure VII-56. It is recommended that Concept J be designed to this flight envelope at a design gross weight of 19760 lb. This will require analyzing the entire aircraft for the revised loading distributions and reinforcing where necessary. To avoid an ultimate static test and satisfy the requirements of MIL-S-5711 (USAF) new structures should be designed to 1.25 times the revised design loadings and proof tested to these loads. The entire fuselage center section (fuselage stations 190 to 340) and the wing carry through will be essentially new structure.

B. GROUND LOADS

North American Aviation has stated that the stretched Sabreliner has been drop tested to a sink speed of 10 ft/sec at a landing weight of 17500 lb. The original T-39 design is designed to 10 ft/sec at a landing weight of 13000 lb. The corresponding ground load factor for the T-39 design is 2.68 and with 1.0g wing lift the aircraft limit load factor is 3.68. North American has stated that the gear for the stretched Sabreliner is the same as the T-39 (apart from larger brakes) and that increased landing capability obtained in the drop test of the stretched Sabreliner was due primarily to the energy absorbed in the structure. Based on this data it is estimated that the permissible ground load factor at a design landing weight of 19760 lb is

$$N_G = 2.68 \left(\frac{13000}{19760}\right) = 1.76$$

It was also estimated that the energy absorbed by the Sabreliner structure is approximately equivalent to an increase in stroke of approximately 3.0 inches. Using this information, the landing gear characteristics for Concept J were

investigated by means of the following relation, which considers kinetic, potential and dissipative energies involved in the landing gear operating cycle: -

$$W \left(\frac{V^2}{2g} + D + S \right) = L (D + S) + F \quad (\eta_T D + \eta_H S)$$

where:

W	=	design landing weight
V	=	sink speed
g	=	acceleration due to gravity
D	=	tire deflection
S	=	hydraulic cylinder stroke
L	=	Aerodynamic lift or engine vertical thrust
F	=	maximum ground load
η_T	=	tire efficiency
η_H	=	hydraulic cylinder efficiency

It was found that for the new gear design of Concept J the following landing capability is obtained.

Landing Mode	Ground Reaction Load Factor	Airplane Load Factor	Sink Speed (ft/sec)
VTO	1.76	2.43	11.3
CTOL	1.60	2.60	12.0

Local reinforcement of the wing structure will be required to meet the VTOL side load requirements.

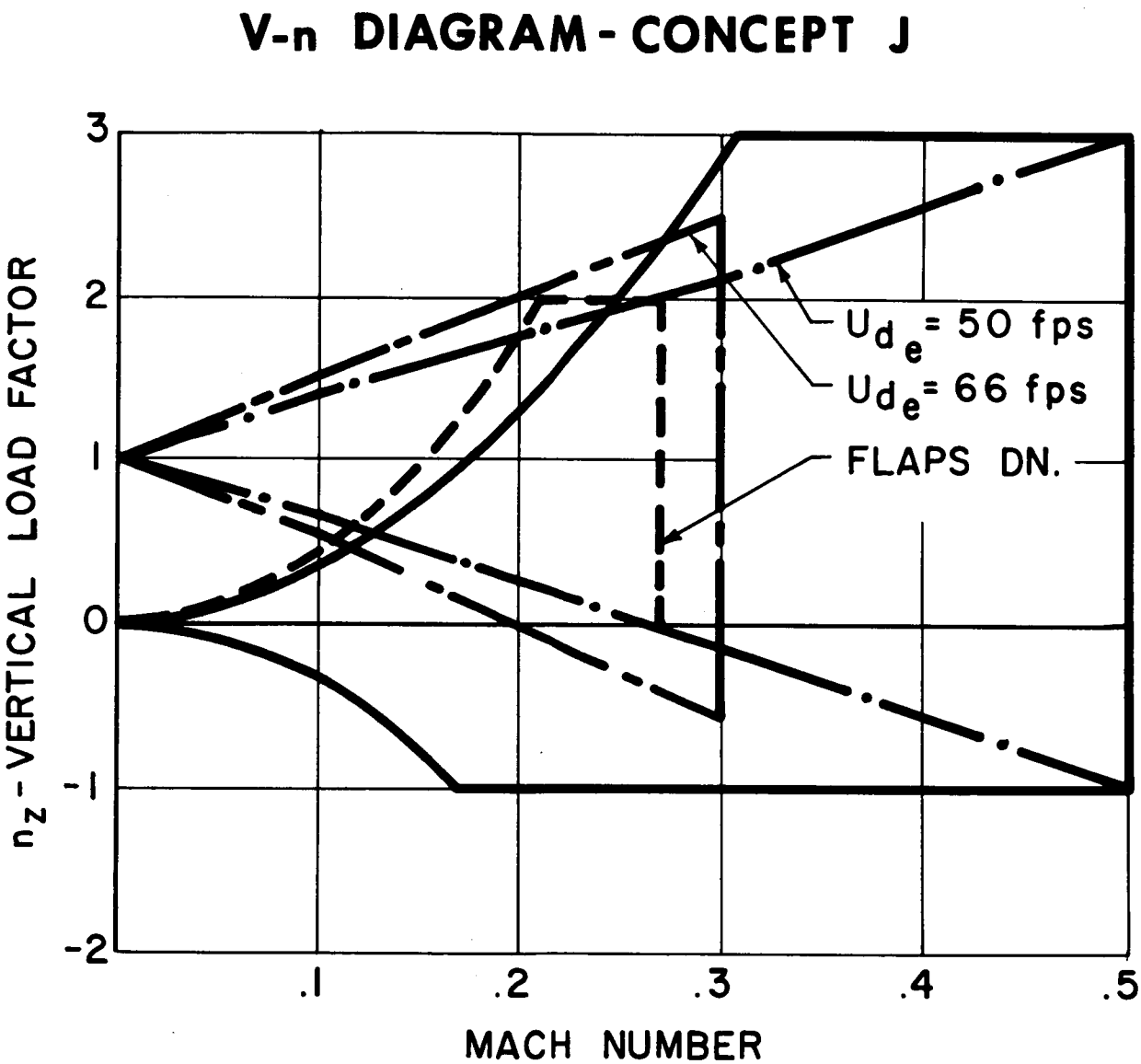


Figure VII-56

REFERENCES

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2. C. Meissner, R. Dudley; The Solution of Static and Dynamic Shears and Moments Utilizing Digital Computing Facilities; Republic Aviation Corporation Report, RAC 813-1, June, 1963.

3. H. Press, R. Steiner; An Approach to the Problem of Establishing Severe and Repeated Gust Loads for Missile Operations; NACA TN 4332.

SECTION VIII

ELECTRICAL

A. GENERAL

The major subsystems described in this section consist of communications, navigation, antenna, and electrical power. Figure VIII-1 provides a block diagram of the combined communications-navigation subsystems and Figure VIII-2 is a block diagram of the electrical power subsystem. Figure VIII-3 summarizes the recommended components of the communications-navigation subsystems as well as size and weight data. Descriptions and specifications for the major components of all subsystems are provided below.

B. COMMUNICATIONS SUBSYSTEM

1. VHF Transceiver

The VHF Transceiver, Antenna, and Control Panel operate over the frequency range from 116.0 to 149.95 MHz (megacycles) providing 680 channels spaced at 50 KC increments for air-to-air or air-to-ground voice communications. The range of this transceiver is limited to the line of sight and approximates 45 miles at a 1,000 ft altitude.

The specific set recommended for Configuration C is the Collins 618M-1A VHF Transceiver which combines both the receiver and transmitter sections within a single unit. As a result, many of the mechanical and electrical subcomponents associated with separate units have been eliminated and both a reduction in weight and improvement in reliability realized. The almost exclusive use of transistorized circuitry further enhances the reliability of this unit. Front panel test provisions on the transceiver unit provide an extensive trouble shooting capability without requiring equipment removal. Complete accessibility through dust cover removal and semi-modular construction have also been incorporated in the design of this unit to simplify maintenance. Dynamic squelch and audio limiting are two important features that assist in reducing pilot work load. Dynamic squelch is provided by a separate sensing circuit which automatically adjusts the squelch as a

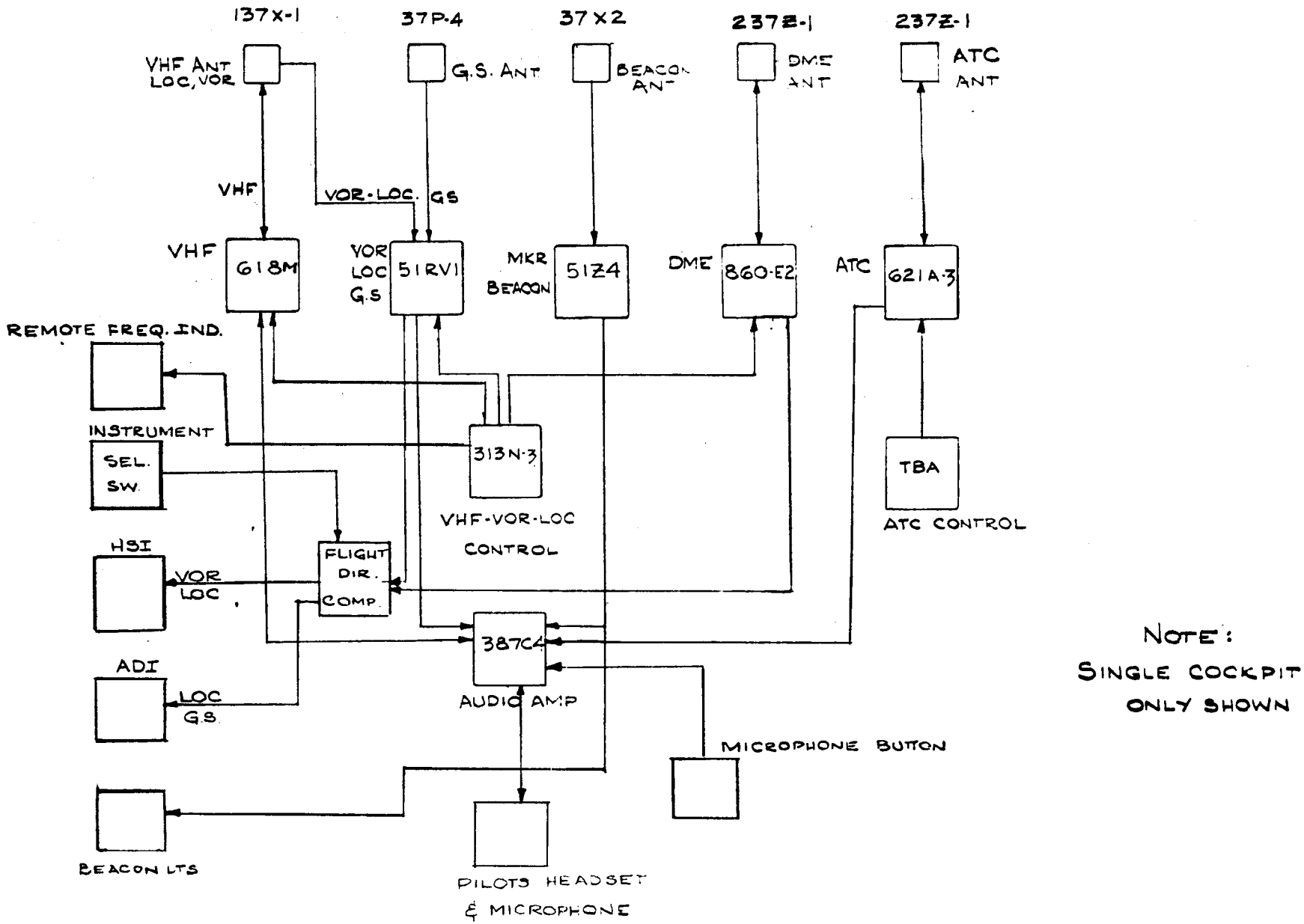


Figure VIII-1. Communication-Navigation Subsystem, Block Diagram

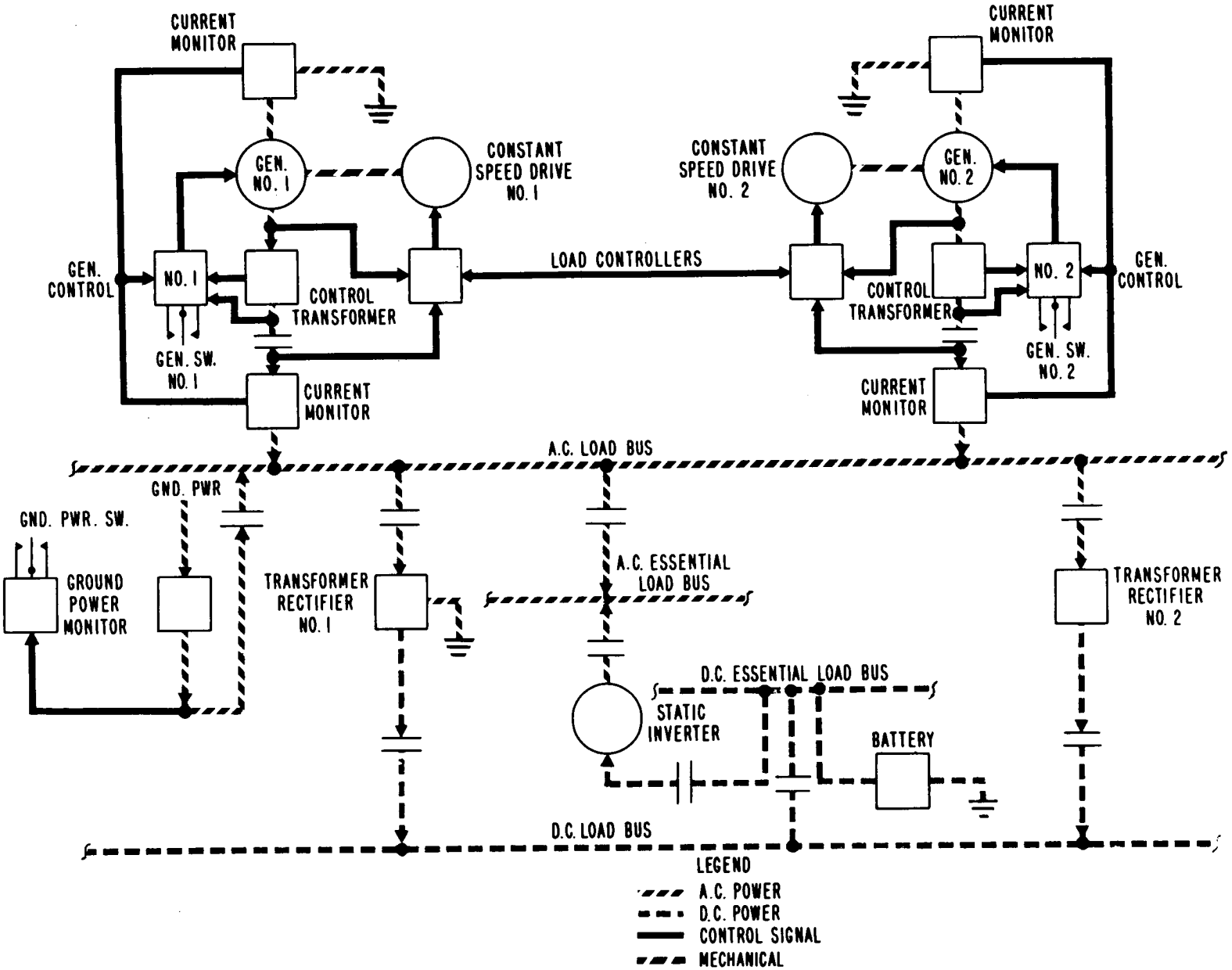


Figure VIII-2. Electrical Power System

COMMUNICATIONS SUBSYSTEM

	NEW DESIGN (C)	WT (LB)	VOL ($IN.^3$)
VHF TRANSCEIVER	618M-1A TRANSCEIVER	19.0	469
	313N-3 CONTROL (2)	2.0	65
	137X-1 ANTENNA	6.0	N/A*
	390Y-2 SHOCKMOUNT	1.7	72
	TOTAL	28.7	606
ATC TRANSPONDER	621A-3 TRANSPONDER	25.2	732
	TBA CONTROL (2)	1.5	48
	350E-3D SHOCKMOUNT	2.1	96
	237Z-1 L-BAND ANTENNA	0.2	N/A*
	TOTAL	29.0	876
AUDIO CONTROL UNIT	387C-4 AUDIO AMPL (2)	1.6	65
	TOTAL	1.6	65
	UNIT TOTALS	59.3 LB	1,547 $IN.^3$
	TOTALS ADJUSTED FOR CONTROLS DIFFERENCES	64.4 LB	1,735 $IN.^3$

*EXTERNALLY MOUNTED

NAVIGATION SUBSYSTEM

	NEW DESIGN (C)	WT (LB)	VOL ($IN.^3$)
COMPASS SYSTEM	C702205002 GYRO COMPASS & AMPL	9.2	425
	DT-173/AJN FLUX VALVE	3.0	50
	COCKPIT CONTROL (2)	1.0	40
	TOTAL	13.2	515
VOR-ILS LOCALIZER RECEIVER	137X-1 ANTENNA	N/A*	N/A**
	51RV-1 VOR-LOC-GS RECV'R	18.5	473
	390Y-1 SHOCKMOUNT	1.7	130
	TOTAL	20.2	603
ILS GLIDE SLOPE RECEIVER	37P-4 ANTENNA	0.7	N/A**
	PART OF 51RV-1 RECV'R	N/A*	N/A*
	TOTAL	0.7	N/A
MARKER BEACON RECEIVER	51Z-4 MARKER BEACON RECV'R	3.3	178
	37X-2 ANTENNA	0.3	N/A**
	390R-1 SHOCKMOUNT	0.6	55
	TOTAL	4.2	233
DISTANCE MEAS EQUIP OR TACAN	237Z-1 ANTENNA	0.2	N/A**
	313N-3 CONTROL (2)	N/A*	N/A*
	860E-2 TRANSCEIVER	36.0	732
	350E-3D SHOCKMOUNT	1.5	200
	TOTAL	37.7	932
	UNIT TOTALS	76.0 LB	2,283 $IN.^3$

* INCLUDED IN COMMUNICATIONS SUBSYSTEM

** EXTERNALLY MOUNTED

FIGURE VIII-3

function of the prevailing background noise level and effectively limits reception to voice intelligence only. The audio limiting circuit provides an approximately constant modulation independent of the voice characteristics of the speaker or the type of microphone used. The 618M-1A Transceiver also provides for data transmission and reception although this function has not as yet been fully explored for its possible flight test applications.

The control panel (Collins P/N 313N-3D) associated with this transceiver provides a complete tuning capability for almost all Comm.-Nav. equipment. As a result, power, volume, and frequency controls for VHF, VOR, Localizer, Glide Slope, and DME are all centralized on a single cockpit panel. Glide slope receiver frequency is automatically selected as a function of the desired localizer frequency by means of a ganged control located under the localizer frequency selector knob. Controls for VOR and ILS are also provided on the panel face to permit the testing of these units prior to flight without the use of auxiliary ground support equipment. Ganged controls are employed to minimize the required panel size.

SPECIFICATIONS FOR COLLINS 618M-1A TRANSCEIVER

Size: Short 1/2 ATR Case — Weight: 19.0 pounds
Ambient Temperature Range: -54°C to +55°C Continuous, or +71°C for 30 minutes

Altitude: Sea level of +45,000 feet

Shock: 15g operational, 30g impact nonoperational

Frequency Tuning: Collins 313N series control

Receive Characteristics:

Sensitivity: Not more then 3uv input for 6db S+N/N

Selectivity: At 50kc channel spacing, 40kHz minimum at 6db points, 58kc at 60db points, NMT 3db ripple

Spurious Response: All spurious responses including images down at least 60 db.

Squelch: Noise silencer operating from ratio of carrier noise level to noise level.

AGC Characteristics: Less than 3 db variation from 5-100,000 μv
Less than 2 db variation from 100,000 - 300,000 μv

Audio Output Level: At least 100mw with 4μv input and 30% modulation at 1kHz

Data Selcal: 0.5v output with 1,000μv signal modulated 30% at 1kHz

Audio Harmonic Distortion: NMT 7% each 30% modulation, NMT 29% each 85% Mod.

Input Power: 53 watts at 27.5 volts

Transmit Characteristics:

Power Output: 25 watts minimum, 30 watts nominal

Duty Cycle: Continuous at 90% modulation

Output Impedance: Unbalanced into a 52 ohm load

Frequency Stability: ±.005% from -55°C to +55°C

Harmonic and other Spurious: With a 52 ohm load Harmonics are down 60db, and all others are down 75db

Modulation Characteristics: 90% over frequency range with NMT 0.25 volts RMS 1000 Hz input signal

Noise Level: Over-all noise level at least 40db below 90% modulation

Input Power: 250 watts with 90% modulation at 27v dc.

2. Audio Amplifier (Intercom)

The unit recommended for this function is a panel mounted transistorized 387C-4 audio amplifier and control which provides audio switching functions for all applicable elements in a centralized location for each crew station. Although the input capabilities of this unit are in excess of that required by the anticipated avionic compliment, the cost for a special customized unit can not be justified. This excess capability also provides a high degree of flexibility should additional audio equipment (i.e., a voice recorder) be added at a later date. The individual elements serviced by this unit are: VHF, VOR, Marker Beacon, DME, and ATC (crew station and ground crew intercoms are integral to this unit).

SPECIFICATIONS FOR THE COLLIN 387C-4 AUDIO CONTROL UNIT

Size: 5-3/4"W x 1-7/8"H x 4-11/16"D (including connector)

Weight: 1.6 pounds

Environment: -55°C to +55°C continuous, +71°C short period
99 to 100% humidity at 50°C for 48 hours

Altitude: Sea level to 55,000 feet

Vibration: RTCA paper 100-54/D0-60, 0.06da, 10-55 Hz in each plane

Shock Conditions: 3-15g shock, 1-10ms-30g shock

Input Power: 27.5 V dc and 0.28 amps (7.7 watts)

Output Power: 100mw

Frequency Response: Room conditions - ± 1 db from 300 to 5,000 Hz with a ± 3db roll off at 200 to 6000 Hz (1,000 cps reference).

Noise Level: At least 40 db below nominal output

AGC Characteristics: Limiting ratio - output will not increase more than 3db for a 20db increase in input signal

Cross Talk - More than 60 db isolation between inputs

3. Air Traffic Control (ATC) Transponder

The ATC Transponder subsystem consists of a control panel, transponder (receiver and transmitter) and antenna, and provides a selected coded pulse reply on any of three basic modes in response to a valid main lobe ground interrogation signal. By monitoring the aircrafts reply on a display indicator, the air traffic controller can identify a given aircraft and rapidly determine its position relative to the ground station.

A condensed circuit description for the Collins 621A-3 ATC transponder is presented as follows:

The ground interrogation signal received at the antenna is passed through a filter, directional coupler, preselector and mixer where it is heterodyned with the local oscillator frequency to generate a 60 MHz I. F. This signal is then amplified and detected and the resultant video processed in the decoder.

The video signal is then analyzed to determine if the interrogation is a main lobe or side lobe signal. For each valid main lobe interrogation a trigger pulse, depending on the mode of operation (i. e. a, b, or c) is applied to the encoder to initiate the proper reply code. A suppression signal generated within the VHF/AM transceiver will disable the encoder during transmission time, and conversely a decoder pulse will disable the VHF transceiver while the transponder is transmitting.

The reply code from the encoder is then applied to the modulator to control the transmitters RF output. If a mode "C" interrogation is received, the reply code from the encoder will be supplied by the altitude coding wires from an altitude digitizer.

An automatic overload control (AOC) voltage is generated within the encoder to control the IF amplifier and limit the number of replies generated. The preselector prevents the high power transmitter replies from damaging the mixer crystals. The directional coupler is used to sample the interrogation reply and for purposes of injecting a test signal into the receiver. During normal operation an RF transmitted sample is sent to the filter and detector and, if of proper frequency and amplitude, will provide the pilot with visual and aural indications of transponder operation. For test purposes, a push to test button on the ATC control panel can be depressed and both visual and aural indications of proper operation noted.

SPECIFICATIONS FOR COLLINS 621A-3 AIR TRAFFIC CONTROL TRANSPONDER

Transmitter Power Output: 500 watts

Transmitted Frequency: 1090MHz ± 2.5 MHz

Receiver Frequency: 1030 MHz Sensitivity: -74dbm

Receiver Selectivity: ± 3 MHz at 3db points. Greater than 60db attenuation at ± [illegible]

Reply Code Capability: 4096 reply codes on all four modes

Replay Code Pulse Spacing Tolerance: ±0.1 sec. for any pulse with respect to the first framing pulse

Modes of Interrogation: A, B, C, and provisions for D (8, 17, 21 and 25 μ sec. spacing)

Sidelobe Suppression: Both 3 pulse (FAA) and 2 pulse (IACO) sidelobe suppression

Weight: 25.2 pounds without shockmounts (23.1 with mounts) Size: 1/2 ATR long

Input Power: 70VA at 115V. 300 to 1,000 Hz - No. D.C. required

Environmental Conditions

Altitude: 0 to 45,000 feet

Temperature: -54°C to +55°C continuous

Vibration: 0.29 inch total excusion from 10 to 55 Hz or a maximum of 2 g's. Constant acceleration of 1.5g for 55 to 1000 Hz

Applicable Specifications: TSO-C74 and ARINC characteristics 532D

C. NAVIGATION SUBSYSTEM

1. Compass System

The Kearfott C702205002 directional gyro proposed is a two degree of freedom instrument for determining azimuth direction from a gyro spin axis heading reference maintained level in a plane parallel to a tangent to the earths surface. This unit can also be slaved to the DT-173/AJN flux valve which serves as a magnetic reference. Inner axis gimbal freedom is ±85° and outer axis gimbal freedom is 360°. Roll stabilization is employed to minimize roll gimbal errors and their resultant apparent azimuth error.

SPECIFICATIONS FOR C702205002 DIRECTIONAL GYRO

Excitation: 115 volts, 400 Hz	
Rotor Open Circuit Z: 2300/66.5° ohms	Pickoff
Output Voltage: 18.2 ± .075 volts	
Max. Error from E.Z.: 10 Min.	
Torquer: D.C. (120 volts max.)	
Power: 2.5 watts	Torquer
Torquing Rate: 50°/min. max.	
Power: Start - 170 ma - 20 watts	
Run - 130 ma - 7 watts	
Runup Time: 6 min.	Motor
Angular Momentum: 8×10^6 GM CM^2/sec.	
Excitation: Internal	
Leveling Rate: 3± 1°/min.	Sensor
Size: 8.25L x 7.25 W x $6^{21}/32$ H (inches)	
Weight: 9.2 pounds	
Temp: -54°C to + 71°C	
Shock: 15 g's	
Vibration: 0.20 In. DA at 5 to 50 Hz 3 g's at 50 to 1000 Hz	

2. VHF Omnidirectional Range (VOR) and Localizer and Glide Slope Receiver

The VOR receiver covers the odd frequency range from 118.0 to 117.9 at 50kHz intervals and is used to provide aircraft bearing to a VOR ground station. ILS localizer frequency covers the same range as VOR but uses even 50kHz intervals and provides a signal corresponding to aircraft lateral deviation from localizer beam center. A single receiver can be utilized for both these functions due to the identical frequency range involved, however, glide slope operation in the 329.3 to 335.0 megacycle range requires the use of a second receiver. The selection of a localizer frequency automatically tunes the glide slope receiver to the correct paired frequency. Both these receivers, as well as a manual and automatic VOR instrumentation unit, are contained within a single chassis assembly recommended for this vehicle and designated by the manufacturer (Collins) as the 51 RV-1 Navigation/ILS receiver.

As with all electronic elements considered for this program, this unit features a solid state design and does not employ any moving parts with the exception of a function relay. Built-in self test has been designed into this unit for both VOR and ILS. The VOR function allows the pilot to check manual and automatic VOR instrumentation from the cockpit using the 313N-3D control panel on the right hand console. Similarly, ILS (Loc. and G.S.) accuracy may also be checked using this same cockpit control panel.

SPECIFICATIONS FOR THE 51 RV-1 NAVIGATION/ILS RECEIVER

General

Power: 90 watts maximum at 26V AC for both VOR and ILS

Ambient Temperature: -54° to +55°C continuous and 30 min. at 71°C for short term operation

Altitude: S.L. to 45,000 ft.

Size: 1/2 ATR short case (4.9 In. W, 12.6 In. L, 7.6 In. H)

Weight: 18.5 pounds

VOR-Loc. Performance

Freq. Range: 108.0 - 117.95 Mc. in 50 kHz increments

Sensitivity: Audio Channel - Not more than 3μv for 6db s/n/N ratio
VOR Channel - Not more than 3μv input for fully masked flag and satisfactory Nav. performance

AGC Characteristics: 3 db variation from 5 to 50,000 μv

Selectivity: 40 kHz minimum at 6db points, 68kHz at 60 db points and not more than 3db passband ripple

Spurious Response: All spurious responses including images are at least 70 db down

Gain: A 5uv signal, modulated 30% with a 1000 ohm load provides a 100 mW audio output into a 600 ohm load

Frequency Stability: 0.004% from +10°C to +55°C
0.005% from -55°C to +71°C

Nav. Outputs: 5 deviation indicator loads, 4 flag loads, 2 to - from loads, and 2 RMI pointer loads

Bearing Accuracy: ±0.25° under standard conditions, or
±0.8° over environmental extremes
defined in RTCA paper D0114

Glide Slope Performance:

Frequency Range: 329.0 - 335.0 in 300kHz increments

Sensitivity: No more than 20μv input with a 2 db audio ratio for 47μa (course sensitivity control set at 78μa)

Selectivity: 140kHz minimum at 6db points, 450 kHz maximum at 60 db points

Spurious Response: All spurious responses including images are at least 60db down in the frequency range of 0.09 - 1500MHz

AGC Characteristics: Adjustable audio AGC to vary audio rise from +5 to -15% when the input is increased to 700 to 14,000μv.

Frequency Stability: 0.01% from -55 C to +71 C

Navigation Outputs: 5 deviation loads and 4 flag loads.

3. Marker Beacon

This unit is a solid-state single-frequency receiver (tuned at 75 MHz) designed to provide a sharp aural tone as well as trigger the cockpit beacon light on passage over airways and ILS outer and middle markers. Self test is incorporated within the Collins 51Z-4 unit recommended to permit cockpit receiver checkout.

SPECIFICATIONS

Size: 1/4 ATR short case (2 1/4 In W, 3 3/16 In H. 12 9/16 In L)

Weight: 3.3 pounds

Mount: 390 R-1 Shockmount

Power: 0.4 amps intermittent at 27.5 vdc, 0.16 amps continuous

Sensitivity: 200 to 200,000 μv.

Hi-Lo Sensitivity: Adjustable between 200 and 200,000 μv

Selectivity: 40 kHz at 6 db, and 60 db at 250 kHz

Frequency: 70 MHz

Altitude: S. L. to +55,000 ft.

Output: Will drive two sets of #47 lamps or 3 single #47 lamps

4. Distance Measuring Equipment (DME)

DME performs a search function to the ground station selected as designated by the frequency set in on the left hand control panel and tracks the reply to provide aircraft to station slant range. The unit recommended for this function is the Collins 860E-2 which combined with VOR bearing determines aircraft position on a continuous basis. Employing solid state circuit design, except for the RF power amplifiers, this unit is equipped for split channel operation resulting in a total of 250 available channels. Anti-echo provisions have also been incorporated to eliminate false tracker lock-on to reflections existing in the interrogation path. This feature relies on the fact that echo signal distance always occurs beyond the range of the desired station. For short range signals where the echo problem is more severe due to signal strength, the search cycle is active only on the outbound direction. If lock-on is lost at short ranges, lock-on during the retrace to zero is inhibited and only resumed on the outbound search. Automatic memory provisions are also included based on last valid data. Since doppler velocity will be available, the velocity memory option will be provided to permit range tracking during short period signal lost.

860-2 DME SPECIFICATIONS

Transmitter Frequency Range: 1025-1150 MHz

Receiver Frequency Range: 962-1213 MHz

Selection Time: 9 seconds

Transmitter Output: 1 KW nominal Stability: 0.007%

Receiver Sensitivity:	-90 dbm
Range Capability:	Up to 195 miles
Range Accuracy:	±0.1 nautical miles from 0 to 5 miles ±0.2 nautical miles from 5 to 195 miles
Search Time:	15 seconds maximum for long range station up to 195 nautical miles
Dynamic Tracking:	Up to 2,000 knots
Audio Output:	100 mw minimum into a 600 ohm load
Power:	85 VA at 115 volts, 380 to 420 Hz and 1.5 amps at 27.5 vdc
Cooling:	Self-contained
Size:	1/2 ATR Long Case
Weight:	36 pounds

D. ANTENNA SUBSYSTEM

Because all unit involved in the two subsystems are either individual or combined receivers or transmitters, the antenna system for each unit was studied to assure optimum operation. As part of this effort, drag effects were also considered and trade-off studies conducted concerning the cost increment associated with internal (flush mounted or recessed zero drag types) versus exterior mounted units. For Configuration C, a substantial saving was evident for the exterior antenna arrangement for a minimal drag penalty. Refer to Figures VIII-4, -5, and -6 for antenna data on basic parameters, installation trade-off results, and aircraft location.

E. ELECTRICAL POWER GENERATION

Based on numerous in-house studies of high reliability electrical power generation systems, most recently typified by the effort conducted during the US/FRG V/STOL program, both a redundant isolated system and paralleled generator system offer attractive advantages. Of these two however, the paralleled A.C. system, consisting of a pad-mounted unit on each of the aircraft's two cruise engines, offers the most promise in regards to maximum reliability at a reasonably low cost and weight penalty and has been choosen for this vehicle. Advantages afforded by a system of this nature are: better load division and fault protection, reduced voltage and frequency transients, lower wave form distortion due to non-linear loads, elimination of beat frequency problems associated with independent generators, and

ANTENNA P/N	137X-1 (COLLINS)	237Z-1* (COLLINS)
FUNCTION	VHF, VOR, LOC OPER	ATC & DME OPERATION
MAX. DIM (LWD-IN)	25 x 11 x 25	2 x 4 x 5
FREQ. COVERAGE (mc)	108-118 & 118-136	960 TO 1215
SWR - IMPED.	>2:1 >5:1	
MAX. ALLOW. FORCE	0 G M. $\theta = 5°$ & $\psi = 5°$	NOT SPECIFIED
WEIGHT (#'s)	6.0	0.25
POLARIZATION OR PATTERN	HORIZ. FOR VOR & LOC. VERT. FOR VHF OPER.	1/4 λ MONOPOLE MTD. ON A GROUND PLANE
DRAG (ANTENNA ONLY)	2.7 #'s AT 217 KNOTS S.L AND θ, ϕ & $\psi = 0°$	NEG
RECOMMENDED LOCATION	TOP OF FUSELARGE	REAR UNDERSIDE OF FUS.

NAVIGATION SUBSYSTEM

ANTENNA P/N	37P-4 COLLINS	37X-2 COLLINS
FUNCTION	G. SLOPE RECEPTION	MARKER BEACON RECEPT.
MAX DIM (W H D-IN)	4.5 x 6 x 2	2 x 2 x 12
FREQ. COVERAGE	329 TO 336 MH_z	75 MH_z
SWR - IMPEDANCE	>2:1 52 OHMS = Z_T	>3:1 52 OHMS = Z_T
MAX ALLOW. FORCE	NOT SPECIFIED	NOT SPECIFIED
WEIGHT (#'s)	1.0	0.7
POLARIZATION	HORIZONTAL	HORIZONTAL
DRAG	NEGLIGIBLE	3.5 OZ AT 347 KNOTS
RECOMMENDED LOCATION	NOSE AREA FACING FORWARD	FWD UNDERSIDE OF FUS

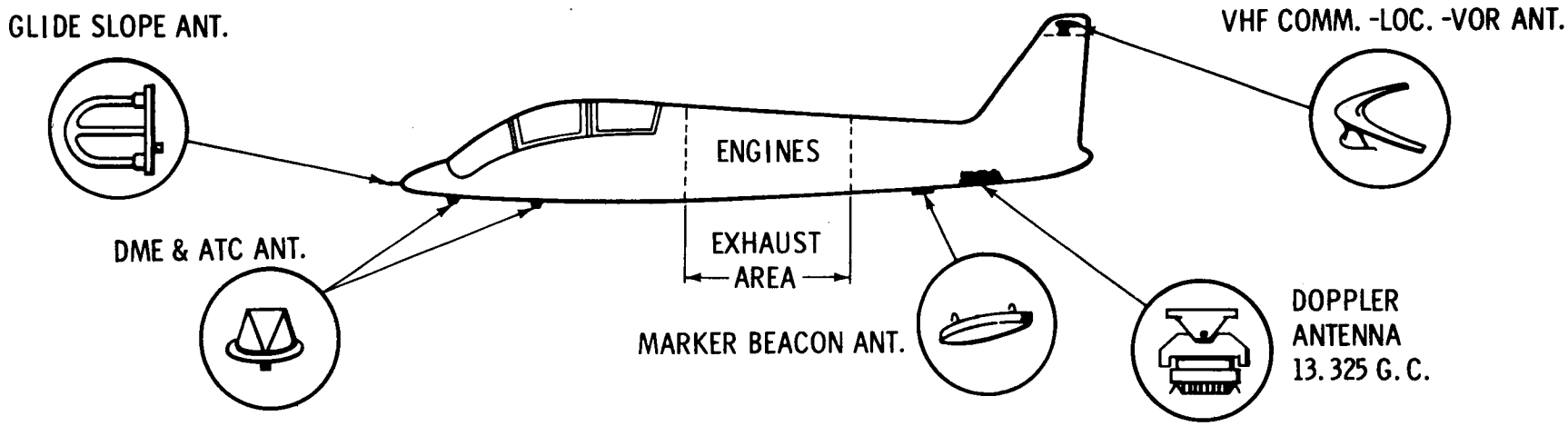
Figure VIII-4. Antenna Parameters Communications Subsystem

FUNCTION	MFGR'S. P/N	ACQUISITION COST	WT.	DRAG (ANTENNA ONLY)	ESTIMATED INSTALLATION COST
VOR-LOC-VHF COMM.	COLLINS 137X-1	249.00	6.0	(FULLY EXPOSED) 2.7 #'s* AT 217K, SL, α=0	2600.00
ATC AND DME	(2 REQ'D) COLLINS 237E-1	63.00 EA. 126.00	.25 EA. 0.5	NEGLIGIBLE	400.00
GLIDE SLOPE	COLLINS 37P-4	142.00	1.0	NEGLIGIBLE	400.00
MARKER BEACON	COLLINS 37X-2	(EST.) 150.00	0.7	NEGLIGIBLE	400.00
TOTALS	5	$667.00	#'s 8.2	~30 #'s	$ 3,800.00
VHF COMM.	COLLINS 37R-2A	150.00	3.0	0	2400.00
ATC AND DME	DORNE AND MARGOLIN NI-7(2)	200.00 EA. 400.00	1.3 EA. 2.6	0	2400.00
MARKER BEACON	TRANSCO 2279	150.00	1.8	0	1600.00
LOC., VOR GLIDE SL.	DORNE & MARGOLIN DMN 9	400.00	1.5	0	5400.00
TOTALS	5	$1,100.00	#'s 8.9	0 #'s	$ 11,800.00

* IF THIS ANTENNA IS SEMI ENCLOSED WITHIN A TAIL CAP RADOME AS RECOMMEND ITS ASSOCIATED DRAG WILL DROP TO LESS THAN 1 POUND.

Figure VIII-5. External Vs. Internal Antenna Cost, Weight & Drag Comparisons

CONCEPT C



ANTENNA SUMMARY		
FUNCTION	P/N	FREQUENCY
VHF COMM. LOCALIZER VOR	137X-1	118-135.9 MH_z AND 108-117.95 MH_z
MARKER BEACON	37X-2	75 MH_z
GLIDE SLOPE	37P.4	329-336 MH_z
DME AND ATC	237Z-1	960-1215 MH_z

Figure VIII-6. Communication-Navigation & Doppler Subsystem Antennas

lower susceptibility to shielding and wiring routing problems in sensitive equipment. Variable speed constant frequency units have been considered for this application; however, the development problems and technical risk involved do not warrant its inclusion in this program. Figure VIII-2 presents a block diagram of this paralleled generator system arrangement including D.C. power sources.

1. A. C. System

Two identical parallel connected air cooled 15 KVA, brushless generators whose output characteristics are in accordance with MIL-STD-704 supply primary A. C. power, as well as that required for transformer rectifier (TR) conversion to D. C. These four-wire, three-phase units, each capable of supplying the aircraft's total load, are driven from separate conventional constant speed drive gear boxes associated with the aircraft's two cruise engines, and are self-excited to eliminate the necessity for external ground or battery power during voltage build-up. Automatic generator paralleling is also provided when the proper condition between each unit is satisfied. Protective circuitry for over and under generator voltage and excitation, under frequency, excessive differential current, and cycling is incorporated. Generator and transformer rectifier warning lights as well as all necessary cockpit controls are included on the right-hand slope panels.

2. D. C. Power

Paralleled T.R. units excited from the aircraft's A. C. bus will normally supply all D. C. power. Either unit can be supplied from either generator and is sized to supply the total D. C. load. A Yardney silver zinc battery will also be installed to supply power prior to T.R. cut-in on engine start and for emergency essential D. C. power. A. C. power on engine start (instruments, etc.) or flight emergency will be provided by 2500 VA rotary inverter driven off the aircraft's battery.

The sizing of all major units associated with this subsystem was based on a preliminary load analysis in accordance with MIL-E-7016C and 7017, and is contained with memo P.D. 610-120-313 dated 15 May 1967.

Figures VIII-7 and -8 present the results of the condition-time interval-load analysis for both A. C. and D. C. generating systems and reflect the calculated loads in each instance (considering derating and all other factors pertinent to each electrical component) and the generating capacity available. In each instance, it can be seen that the KVA or ampere generating capacity is at least twice the calculated load demand, verifying the ability of the redundant system proposed to adequately satisfy all A. C. and D. C. power requirements under a single generator or transformer rectifier out condition.

For ground maintenance, preflight testing, load anchor, and engine start, a 15 KVA, 115/200 volts, 3-phase, 380 to 400 Hz external power source is recommended.

PRELIMINARY SPECIFICATION - ROTARY INVERTER

Type: 32B56-7 Mfgr. Bendix

Voltage:Output: 115 volts ±2.3 volts, 3-phase

Frequency: 400 Hz ±10 Hz

Overload: 150% for 5 minutes or 200% for 15 seconds

Volt Amperes: 2500

Weight: 43 pounds

Short Circuit Current: Exceeds 250%

Input Voltage: 26 to 29 vdc

Ampere Range: 160 to 175

Amperes at 27.5 vdc: 165

Applicable MIL Specs.: MIL-I-7032D

Applicable Drawings: MS25097-1 and MS17504-1

PRELIMINARY SPECIFICATION - GENERATOR (G1 AND G2)

Type: 28B141-15 Mfgr. Bendix

Rating (KVA): 15

Speed Range: 7600 - 8400 rpm

Overspeed: 11,000 rpm

Maximum Speed for regulation: 10,000 rpm

Overload Capacity: 2 minutes - 22.5 KVA
5 seconds - 30.0 KVA

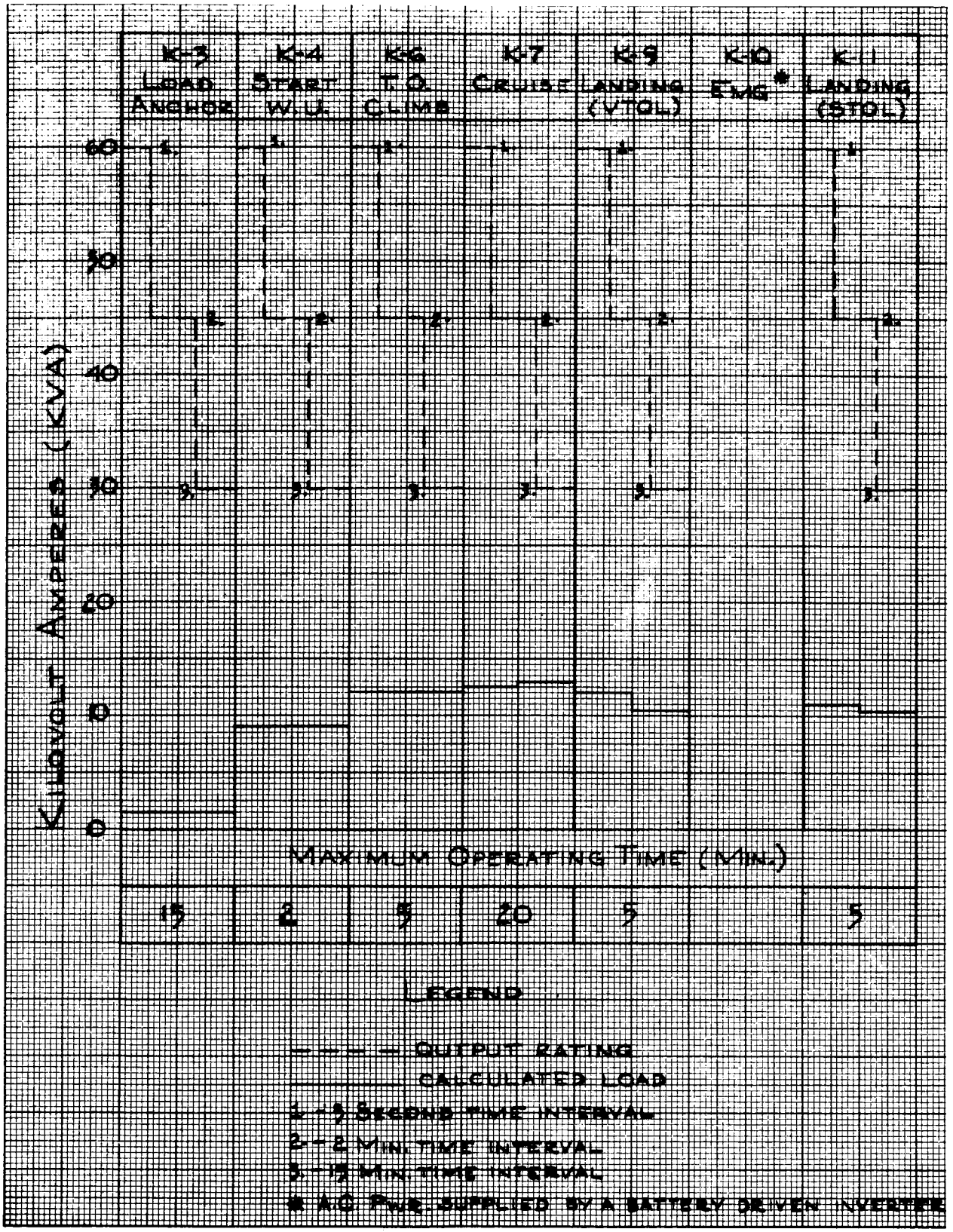


Figure VIII-7. A.C. Generating System

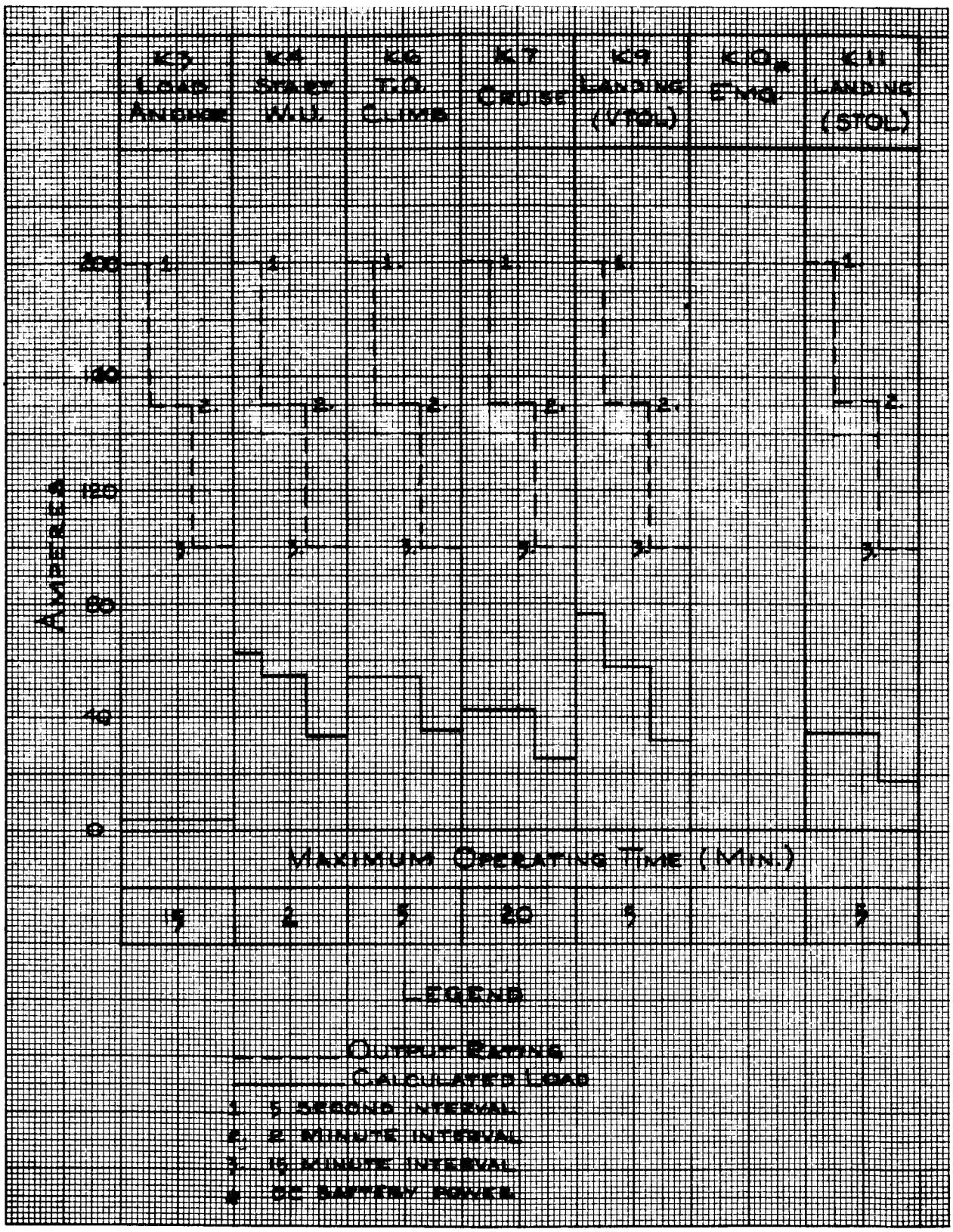


Figure VIII-8. D.C. Generating System

PRELIMINARY SPECIFICATION - TRANSFORMER RECTIFIER (TR 1 AND TR 2)

Number of Units: 2 Mfgr. Westinghouse

Rating: 50 amperes

Voltage Out: 28 vdc Voltage In: 115 V, 3ϕ, 400 Hz

Interval Rating: 75 amps for 5 minutes or 100 amps for 1 minute

Voltage Regulation: 26 to 29 vdc

PRELIMINARY SPECIFICATION - BATTERY

Type: Silver zinc Mfr. Yardney Electric

Rating: 50 ampere hours

Voltage: 28 vdc

Interval Rating: 200 amps for 15 minutes

F. AIRCRAFT SENSORS AND DISPLAYS

Studies concerning aircraft sensory equipment for the basic aircraft flight instrumentation were conducted and closely coordinated with VSS requirements to assure compatibility where a common sensor was involved. Items investigated included the vertical reference, gyromagnetic compass, air data and doppler velocity units, as well as miscellaneous vanes and probes. Efforts were then directed toward integrating these sensory units as well as the radio navigation aids previously mentioned into the appropriate cockpit displays and instrumentation.

As a starting point for this effort, a standard Air Force instrument package consisting of an altitude group, airspeed group, attitude director indicator (ADI), and horizontal situation indicator (HSI) were integrated on a preliminary basis with their associated sensory equipment and radio navigation aids, as applicable. Although providing the desirable features of a centralized instrument grouping, these units require refinements to improve their suitability for V/STOL operation particularly under IFR conditions (real or simulated) where pilot scan rates approach exceptionally high levels. Although industry as well as government funded studies have not fully investigated the area of displays and instruments relative to the unique characteristics of V/STOL aircraft, certain important factors have been determined which should be incorporated in this instrument grouping. As an example of some of these changes, the ADI should employ a three dimensional sphere with azimuth marking and include a fine pitch attitude tape graduated in one degree increments. Radar altitude should be included within the altitude grouping.

Flight path angle should be considered within this same grouping rather than in the airspeed group. Automatic low-limit cut-out should be considered for parameters that cannot be accumulatively monitored in this area and cause distracting instrument scale or symbol excursions resulting in negative information. Variable damping should be included as a function of altitude for the radar altimeter scale.

In addition to the integrated instrument approach, serious consideration should be given to a heads-up display as a transitional aid for IFR operation to permit rapid pilot acclimation to visual operation as breakout occurs. Using a CRT and optical projection techniques, all the information required for guidance and flight path control can be presented on a transparent combining glass installed above the main instrument panel.

Concerning hover displays, a cross track, a long track, and vertical velocity instrument is proposed to display the data derived from a Doppler sensor. A set of cross pointers are provided for the horizontal velocities while a small vertical scale at the left hand side of this instrument displays vertical velocity.

SECTION IX

HYDRAULIC-MECHANICAL SUBSYSTEMS

A. FLIGHT CONTROL SYSTEM

1. Primary Flight Controls

a. Aerodynamic Flight Controls - Concept "C"

Each pilot will be provided with a complete set of primary flight controls consisting of a conventional control stick for lateral and longitudinal control, a pair of rudder pedals for directional control, exhaust vectoring levers, and engine throttles. The aerodynamic controls will be designed and installed to conform to the requirements of specification MIL-F-9490C, Type III systems (irreversible power operation). Engine throttles and vectoring levers will be designed and installed substantially in accordance with the requirements of HIAD. A schematic of the aerodynamic and reaction controls is shown in Figure IX-1 and the general arrangement in Figure IX-2.

The safety pilot's stick and rudder pedals will be mechanically connected to power actuators which will be used to drive the aerodynamic surfaces and reaction control nozzles.

The tandem power actuators will provide the necessary force to move the surfaces against all air loads. In the event of failure of either hydraulic system, the actuators will provide half their total hinge moment capability but still capable of moving the surfaces against all normal maneuvering loads.

The safety pilot's artificial "feel" forces will be provided by mechanical devices which incorporate helical springs, replaceable cams, and electric trim motors. The "feel" devices will be located close to the power actuators to reduce play between these components and thereby minimize error in neutral positioning of controls and surfaces.

Series differential actuators (SDA) will be installed between the "feel" devices and the mechanical input to the power actuators for operation by the stability augmentation system which is to be provided. The installation of the SDA as a series link will prevent SAS inputs from producing movement of the safety pilot's controls (stick "talk"). Separate SDA'S will be provided for the pitch reaction nozzles. One unit will be installed between the stabilator horn and aft nozzle to eliminate the effects of surface inertia and increase the response of this actuator to SAS inputs. Another unit will be installed between the safety pilot's stick and forward nozzle to provide means for actuation of this nozzle from the SAS. The SDA'S will be dual redundant, electrically and hydraulically, to ensure operation of the stabilization system in the event of a single hydraulic or electrical system failure.

The evaluation pilot's controls will be electrically connected to the surface power actuators through the electro-hydraulic parallel command actuators. In this fly-by-wire mode of operation, the entire control system moves and thereby enables the safety pilot's controls to follow the evaluation pilot's electrical inputs. The parallel actuator's power will be limited to enable over-powering by the safety pilot's controls.

The functions of parallel command, series damping, and power actuation will be provided by separate units rather than by customized integrated packages as the former are readily available and the latter require extensive design and development effort. This concept is in keeping with "the program philosophy of using state-of-the-art, fully qualified hardware to the maximum extent feasible for minimum cost and delivery time".

The evaluation pilot's artificial "feel" devices will be similar to the safety pilot's devices, except that they will also include electrically actuated break-out force adjustment and force gradient changer mechanisms. This will enable the evaluation pilot to vary feel force and force gradient in all modes of flight.

The evaluation pilot's controls will be provided with a separate linkage system which can be mechanically connected on the ground to the safety pilot's stick and rudder pedals for specific missions. In addition, space provisions will be made for the installation of a side-stick controller in the evaluation pilot's cockpit. The side-stick will be mechanically connected to the evaluation pilot's "feel" devices and will track the center stick only in the fly-by-wire mode of operation. The side-stick will be automatically disengaged when the evaluation pilot's controls are mechanically connected to the safety pilot's controls.

Control during hovering and low-speed flight will be effected through reaction control nozzles installed at the aircraft's nose, tail, and wing tips. The roll and yaw nozzles will be connected to their respective aileron and rudder horns. The aft pitch nozzle will be connected to the stabilator horn in series with a damping actuator, and the forward pitch nozzle will be connected to the safety pilot's stick in series with a similar actuator as previously described. The nozzle controls will not be disengaged from the aerodynamic controls during conventional flight modes in order to eliminate additional mechanisms which would decrease reliability, add weight, and increase costs and maintenance.

The reaction nozzles will be designed to meet the installation requirements of linearity between thrust and input motion, low operating forces, low weight, and low wear for long, maintenance-free life. These requirements are best met by the cylindrical configuration which incorporates rectangular orifices and eliminates the need for dynamic seals and surfaces in sliding contact.

b. Concept "J" Flight Controls

The T-39's primary flight controls will be redesigned for dual power operation. The power, series, and parallel actuators will be located close to the rudder, elevator, and ailerons, to enable maximum utilization of the existing dual mechanical controls. The control wheels will be replaced by independently operated control sticks, and the two sets of rudder pedals will be redesigned to permit independent operation of the rudder controls. The evaluation

pilot's "feel" devices will be located in the nose of the aircraft. Existing rudder, stabilizer, and aileron trim controls will be utilized to the maximum extent to minimize redesign.

c. Exhaust Vectoring Controls

The engine exhaust nozzles will be hydraulically powered and mechanically interconnected as shown schematically in Figure IX-3. Actuation of the pilot's gross vector lever will position the tandem power actuator which, in turn, applies the positioning signal simultaneously to all the nozzle actuators. The mechanical input to the power actuator can be applied in any one of three ways:

1) Mechanically through the safety pilot's gross vector lever.

2) Electrically through the pivoting knobs provided on the lift engines master power lever.

3) Electrically through the evaluation pilot's vector lever.

Dual hydraulic redundancy will be provided for the power actuator and split hydraulic power will be provided for the engine nozzle actuators so that loss of any one hydraulic system will not result in complete loss of the vectoring controls.

d. Engine Throttle Controls

The engine throttle control system to be provided will be arranged as shown schematically in Figure IX-4. Each pilot will be provided with three engine power control levers. Two levers will be used for the control of lift-cruise engine power, and a third lever will be used for the control of all lift engine power. The lift-cruise engine control systems will be manually powered and the lift engine systems hydraulically powered. The operating loads on each lever will be limited to a maximum of 7.5 pounds to comply with HIAD requirements.

Rotary electromechanical actuators will be provided in each control system for the electrical mode of operation. The actuators will be clutch engaged when energized, and their output power will be limited to permit overpowering by the pilots' mechanical inputs.

The separate throttle levers for the lift-cruise engines permit individual starting and checkout of each engine on the ground. A linear electromechanical actuator will be provided for each lift engine to enable trimming in flight as well as individual starting and checkout on the ground as explained in Section X. As the output of this actuator is additive to that of the master lift engine power lever, a load limiter is provided to prevent overloading of actuator and engine lever.

2. Secondary Flight Controls

a. Wing Flaps - Concept "C"

The partial-span Fowler-type flaps to be provided at the trailing edge of the wing will be actuated by an electromechanical system. The flap in each wing will be supported by and extend and retract on a pair of track and roller assemblies. Each flap will be operated by a pair of screw-jack actuators. The screw-jack actuators will be driven by flexible drive shafts which will transmit the power from two reversible electric drive motors through a set of gear boxes, angle drives, and tee drives. The flexible shaft system will also function to interconnect and synchronize the four screw jacks. A fail-safe system will be provided to stop actuation if the two flaps are not synchronized within set limits.

The electric drive motors will be powered by the primary DC bus and protected by the T.E. Flap circuit breaker. Either motor will be capable of driving the flap system, and each will be protected against jam loads by a current limiter.

Flap position is controlled by the flaps handle to be provided on the throttle quadrant in each cockpit. The electrical control system will also

provide for automatic retraction of the flaps at flight speeds above a set maximum. A flap position indicator will be provided in each cockpit. The indicating system will be powered by the secondary DC bus and protected by a flap position indicator circuit breaker.

b. Concept "J" Wing Flaps

For Concept "J", the electromechanical wing flap actuating system existing in the T-39 will be retained.

c. Diverter Valves

The lift-cruise engine diverter valves will be controlled by a solenoid-operated, four-way, three-position selector valve. The diverters will be operated by actuating a three-position switch located on a panel immediately forward of the throttle quadrant. When the switch is placed in the "cruise" position, the solenoid valve is de-energized to direct hydraulic pressure to operate the diverters to the "cruise" position. When the switch is placed in the "lift" position, the valve is energized to direct hydraulic pressure to operate the diverters to the "lift" position. With the switch in the "intermediate" position, the valve is energized to block all valve ports and hold the diverters at the desired intermediate position. The intermediate position of the diverters will be used to effect thrust decay and maximize cruise engine bleed air horsepower during air start of the lift engines.

d. Engine Inlet and Exhaust Doors

Engine inlet and exhaust doors operation will be controlled by a solenoid-operated control valve. A check valve will be installed in the pressure port of the valve to prevent interflow. Prevention of interflow will effect a "fail-open" condition for the inlet doors in the event a loss of hydraulic pressure occurs after the doors are placed in the "open" position.

B. HYDRAULIC SYSTEM

1. General Description

Two independent hydraulic systems, designated as Systems No. 1 and No. 2, will be provided. The systems, shown in Figure IX-5, will be designed and installed in accordance with MIL-H-5440C. Both will operate at 3000 psi pressure, at Type II system temperatures (-65°F to +275°F) and with MIL-H-5606B hydraulic fluid. System No. 1 will be powered by a pump driven by the left cruise engine. System No. 2 will be powered by an identical pump driven by the right cruise engine.

System No. 1 will be austere and will power all primary flight controls and one utility subsystem as listed below.

a) One side of each rudder, stabilator, spoiler, and aileron tandem power actuator

b) One side of each rudder, stabilator, and aileron tandem series differential actuator

c) One side of each pitch reaction jet tandem series differential actuator

d) One side of the lift engines throttle tandem power actuator

e) One side of the vector controls tandem power actuator

f) Full power to the aft fuel tank boost pump

System No. 2 will power all primary flight controls and all utility subsystems as listed below.

a) One side of each rudder, stabilator, spoiler, and aileron tandem power actuator

b) One side of each rudder, stabilator, and aileron tandem series differential actuator

c) One side of each pitch reaction jet tandem series differential actuator

d) One side of the lift engines throttle tandem power actuator

e) One side of the vector controls tandem power actuator

f) Full power to the three parallel actuators used in the primary flight controls system

g) Full power to the forward fuel tank boost pump

h) Full power to the lift engines inlet and exhaust doors

i) Full power to the engine diverter valve actuating system

j) Full power to the landing gear retraction system

k) Full power to the wheel brakes system

l) Full power to the nose wheel steering system

The utilization of dual powered flight controls will enable control of the aircraft in the event of failure of one of the cruise engines or hydraulic systems.

2. Pumps

Constant-pressure, variable-delivery, piston pumps will be used in both systems as the operating characteristics of this type pump satisfactorily meets the requirements of high-response servosystems. The two pumps will be identical to reduce spares and installation costs and to ease maintenance.

3. Operating Temperature

The systems will be designed to operate at Type II system temperatures as the aircraft's use of several engines will expose a greater than normal area of the hydraulic system to high engine ambients. It is anticipated that sufficient cooling of the hydraulic fluid will be effected through the constant circulation required to meet the null leakage requirements of the several

servoactuators. In addition, Type II, 3000 psi systems are representative of the current state-of-the-art and will ease the task of acquiring fully qualified components.

4. Reservoirs

Identical piston pressurized (bootstrap) reservoirs will be provided for each system to reduce manufacturing costs and simplify maintenance. Pressurization of the pump inlet by use of this type reservoir eliminates aeration and foaming of the hydraulic fluid and thereby enhances the performance of the servosystems. A small accumulator will be provided in the bootstrap pressure line to forestall cavitation when sudden demands are placed on the pump. The reservoirs will be provided with an over-fill relief and bleed valve, reservoir fill coupling, and a fluid level gage.

5. Landing Gear System

The landing gear will be provided with mechanical "up" and "down" locks. A solenoid operated selector valve will be provided for normal gear retraction and extension. An air-oil accumulator will be utilized to store pressurized fluid for emergency gear extension. A manually operated control valve will be provided for emergency extension of the gear. The accumulator will be provided with an air pressure gage and air charging valve. The pressure gage will be located so that it can be read by the pilots before they enter the cockpits.

6. Wheel Brake System

The wheel brake system to be provided will be designed in accordance with MIL-B-8584B Type V systems which utilize a combination master brake cylinder and power brake valve. Normal braking will be effected by metering of System No. 2 hydraulic pressure through the power brake valves and emergency braking will be accomplished by the generation of pressure in the master brake cylinder portion of the unit. A pull handle will be provided in each cockpit for the application of emergency brake pressure. Separate emergency brake pressure lines will be used to connect the master brake cylinders and wheel brakes.

A shuttle valve will be provided at each main wheel to isolate the emergency and normal brake pressure lines. The parking brake system will consist of an air-oil accumulator and parking brake valve. Pull handles will be provided in each cockpit for the application of parking brakes. The brake pedals in both cockpits will be mechanically connected to the same combination brake units.

7. Nose Wheel Steering System

The nose wheel steering system to be provided will be of the closed loop servo type. Toe pressure on the rudder pedals will generate an electrical error signal which is amplified and relayed to an electro-hydraulic servovalve. The servovalve directs pressure flow to the selected steering cylinder port. The wheel turns until the error signal is balanced by a feedback signal. The steering system will operate only when the landing gear "down" line is pressurized and the weight of the airplane is on the landing gear. The system is pressurized and de-pressurized by depressing the steering switch on the control stick grip. The steering switch operates a three-way two-position solenoid-operated valve installed in the landing gear "up" and "down" lines. When the valve is energized, gear "down" line pressure is admitted to the steering servovalve; when de-energized, the pressure is blocked.

8. Fuel Boost Pump System

Hydraulic power for the hydraulic motor driven fuel boost pumps will be divided between the two hydraulic systems in order to effect equal loading on the identical hydraulic pumps powering each system. In addition, this arrangement achieves the greatest possible redundancy in the event of a failure of either a cruise engine or a hydraulic system. A solenoid shut-off valve will be provided for ground maintenance check. Run-away of the hydraulic motor will be prevented by the installation of a flow control valve (restrictor) in the pressure line between the motor and valve.

9. Lines and Fittings

Swivel fittings will be utilized in lieu of flexible hoses where the actuator installation will permit in order to reduce system weight and decrease

fire hazard. Stainless steel tubing will be used in all engine compartments to reduce fire hazard. Aluminum alloy tubing will be used in other areas to reduce overall system weight. Tube fittings will be the MS flared type.

10. Firewall Shut-Off Valves

A normally open, solenoid-operated, fire-wall shut-off valve will be installed in the pump inlet line to prevent spread of fires occurring in the cruise engine compartments. The valve will be energized to the "close" position when the affected cruise engine fire extinguishing handle is pulled.

11. Service Couplings

Self-sealing quick-disconnect couplings will be provided to facilitate pump or engine removal. In addition, each system will be provided with self-sealing service couplings to enable ground operation of either system by external hydraulic power sources.

12. Concept "J" Hydraulic Systems

The T-39 airplane is presently provided with a single hydraulic system which operates at 3000 psi pressure and with MIL-H-5606 fluid. In early models, the system is powered by an electric-motor-driven, constant-displacement pump. In later models, two engine-driven, variable-delivery pumps, operating in parallel, are used to pressurize the system. For Concept "J", a second hydraulic system would be provided in the T-39 to meet the requirement for dual power-operated flight controls. Each system would be pressurized by an engine-mounted variable-delivery pump. The existing bleed air pressurized reservoir would be replaced by an airless "bootstrap" reservoir.

The following subsystems would be retained as they already comply with the requirements of the work statement:

a) Landing Gear Actuation

b) Landing Gear Mechanical "Up" and "Down" Locks

c) Emergency Landing Gear Extension

d) Nose Wheel Steering

e) Wheel Brakes

f) Emergency Brakes

g) Parking Brakes

The new subsystems to be added for V/STOL operation (flight controls, engine throttles, exhaust vector, fuel pumps, diverter valves, and engine inlet and exhaust doors) would be designed substantially in accordance with the subsystems proposed for Concept "C".

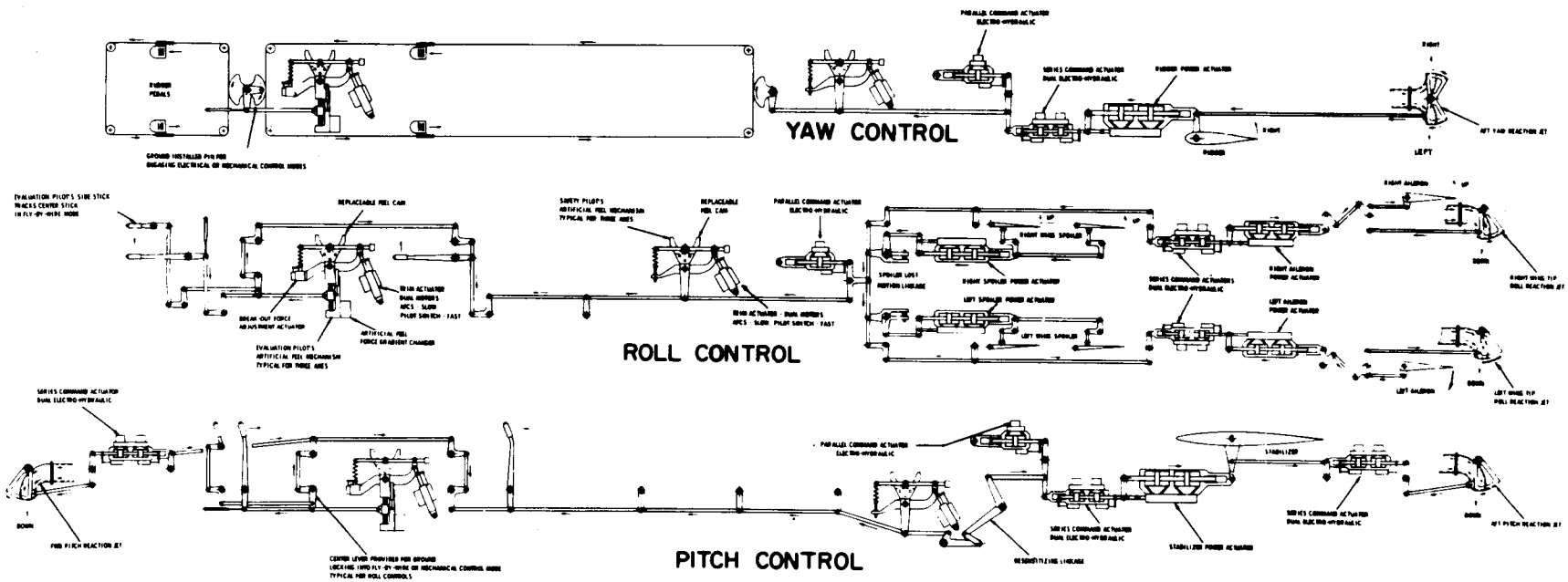


Figure IX-1

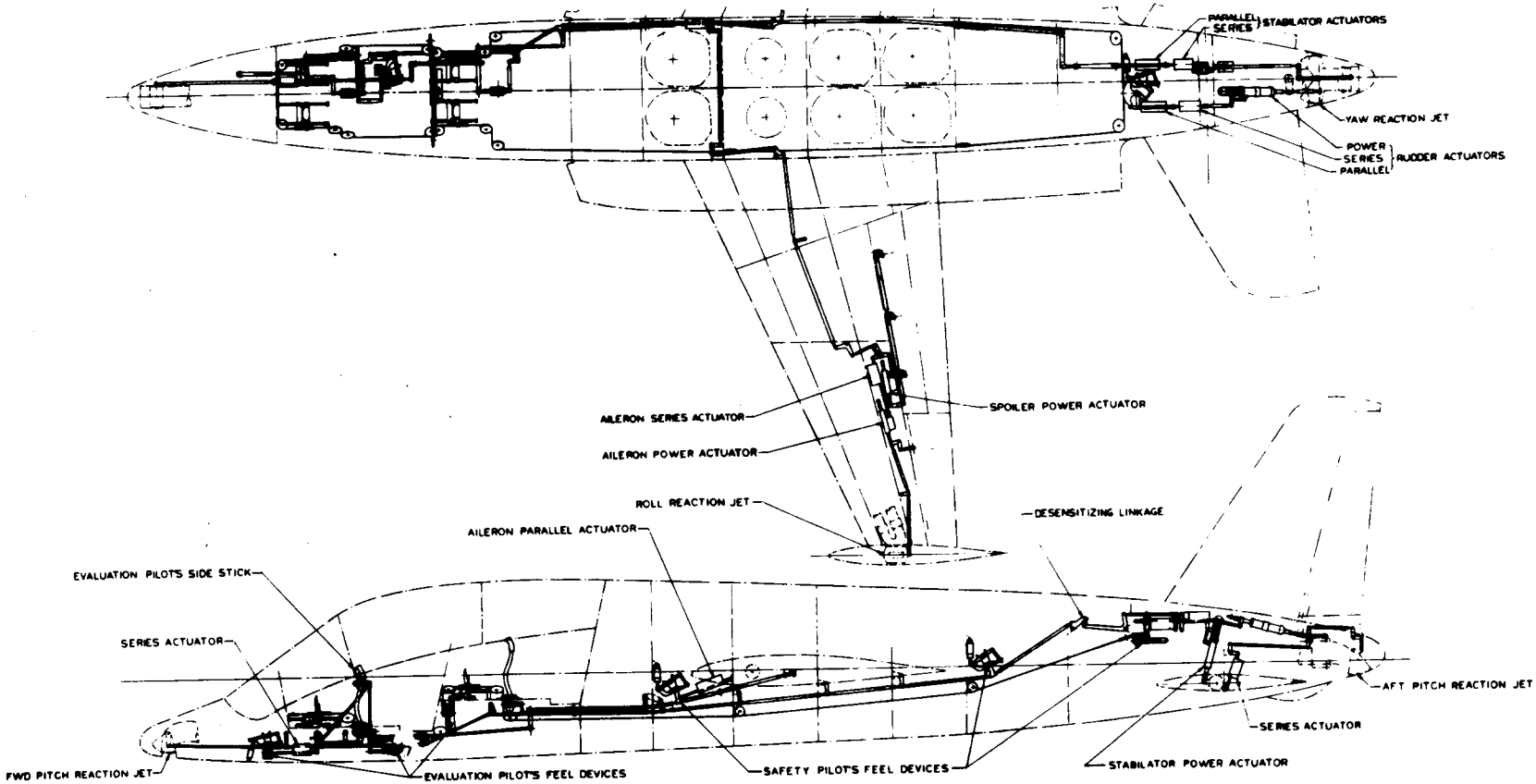

PARALLEL
SERIES
STABILATOR ACTUATORS
YAW REACTION JET
POWER
SERIES
PARALLEL
RUDDER ACTUATORS
SPOILER POWER ACTUATOR
AILERON SERIES ACTUATOR
AILERON POWER ACTUATOR
ROLL REACTION JET
DESENSITIZING LINKAGE
AILERON PARALLEL ACTUATOR
EVALUATION PILOT'S SIDE STICK
SERIES ACTUATOR
AFT PITCH REACTION JET
SERIES ACTUATOR
FWD PITCH REACTION JET
EVALUATION PILOT'S FEEL DEVICES
SAFETY PILOT'S FEEL DEVICES
STABILATOR POWER ACTUATOR

Figure IX-2

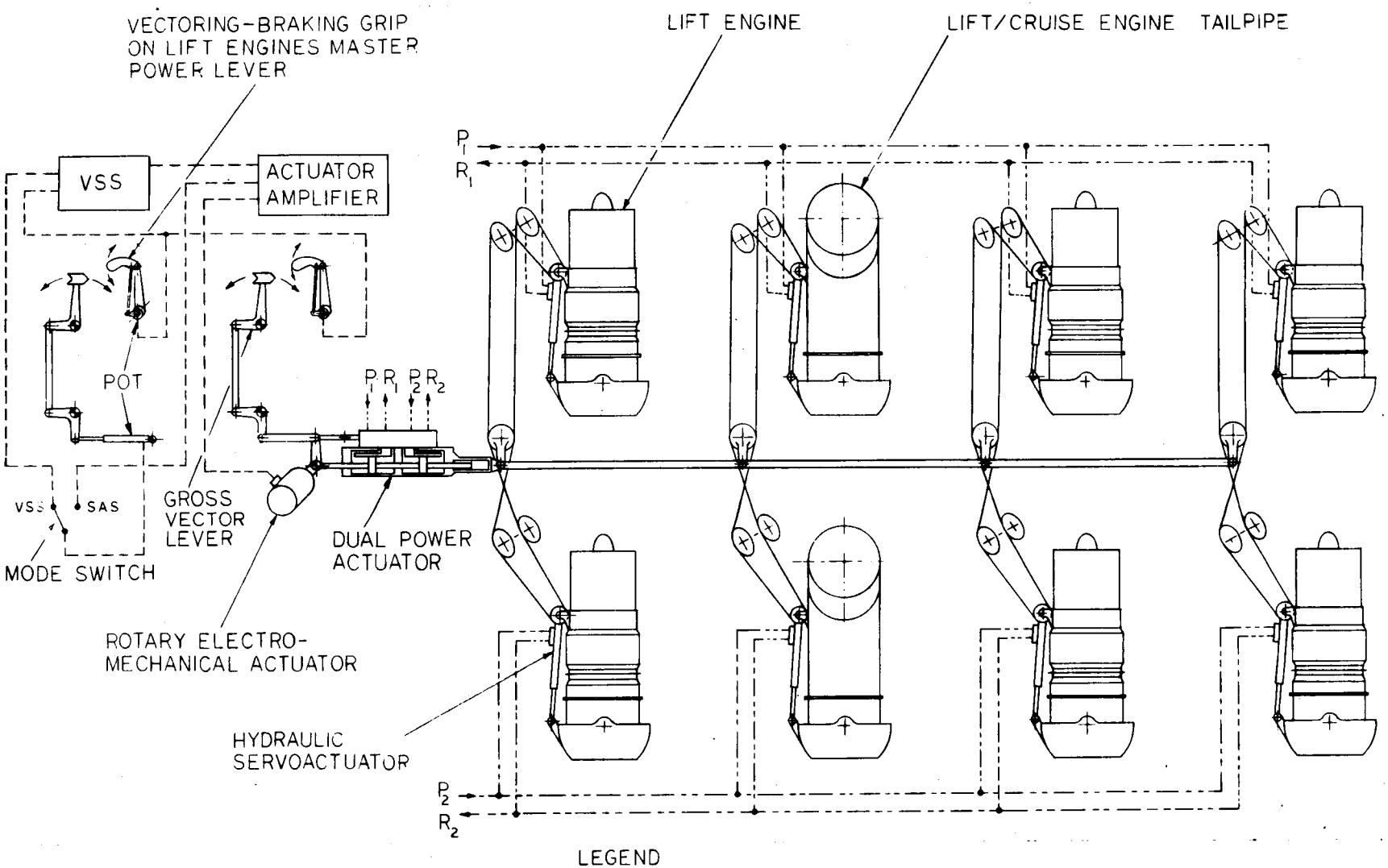

VECTORING-BRAKING GRIP ON LIFT ENGINES MASTER POWER LEVER
LIFT ENGINE
LIFT/CRUISE ENGINE TAILPIPE
VSS
ACTUATOR AMPLIFIER
POT
VSS
SAS
MODE SWITCH
GROSS VECTOR LEVER
ROTARY ELECTRO-MECHANICAL ACTUATOR
DUAL POWER ACTUATOR
HYDRAULIC SERVOACTUATOR
P1 R1 P2 R2
P1
R1
P2
R2
LEGEND
VSS VARIABLE STABILITY SYSTEM
---- ELECTRICAL SIGNAL
—·-·— HYDRAUALIC LINES
FIXED PIVOT
P1 HYDRAULIC PRESSURE-SYSTEM NO. 1
R1 HYDRAULIC RETURN-SYSTEM NO. 1
P2 HYDRAULIC PRESSURE-SYSTEM NO. 2
R2 HYDRAULIC RETURN-SYSTEM NO. 2

Figure IX-3

ENGINE THROTTLE CONTROLS - CONCEPTS C AND J

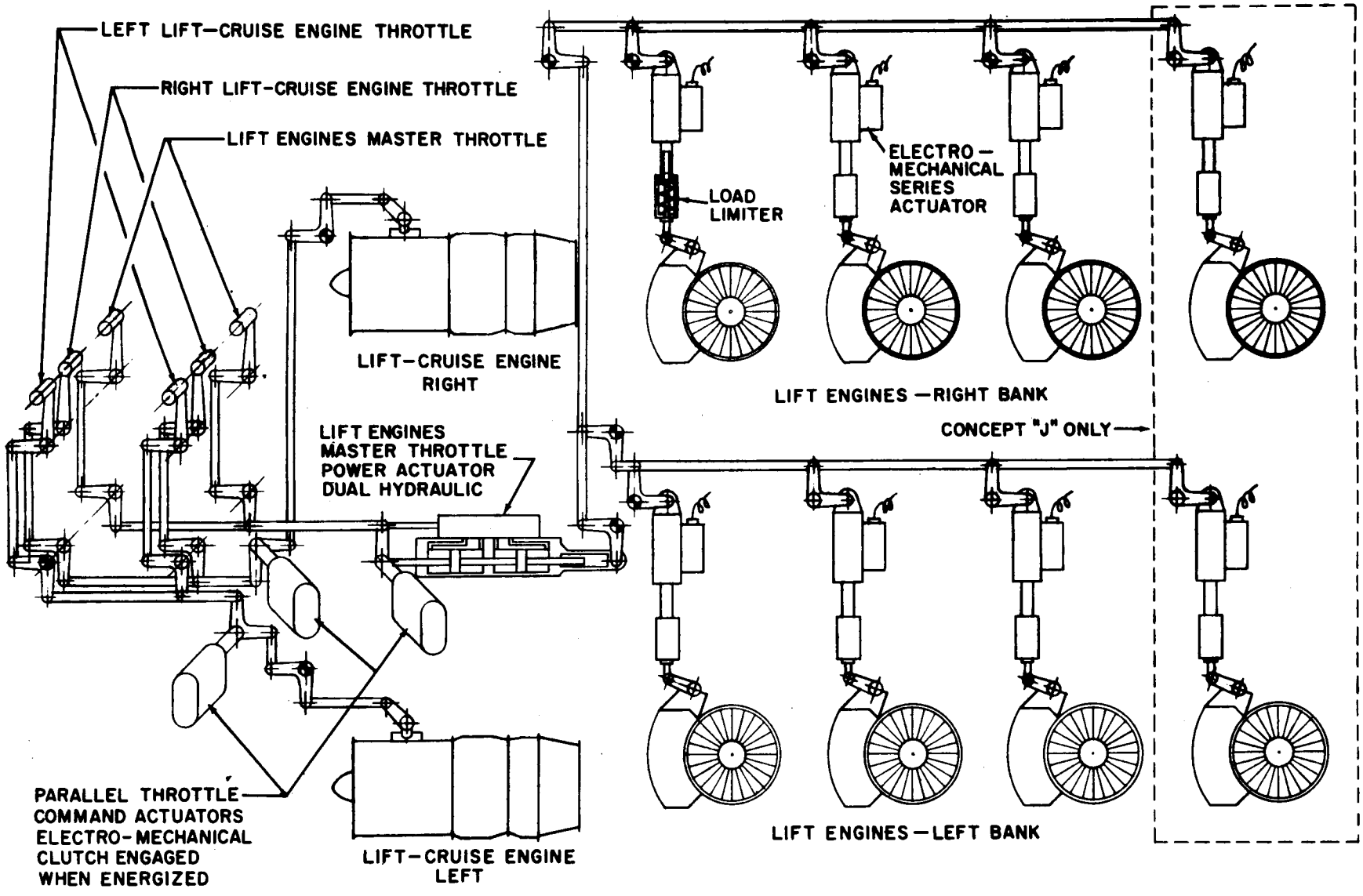


Figure IX-4

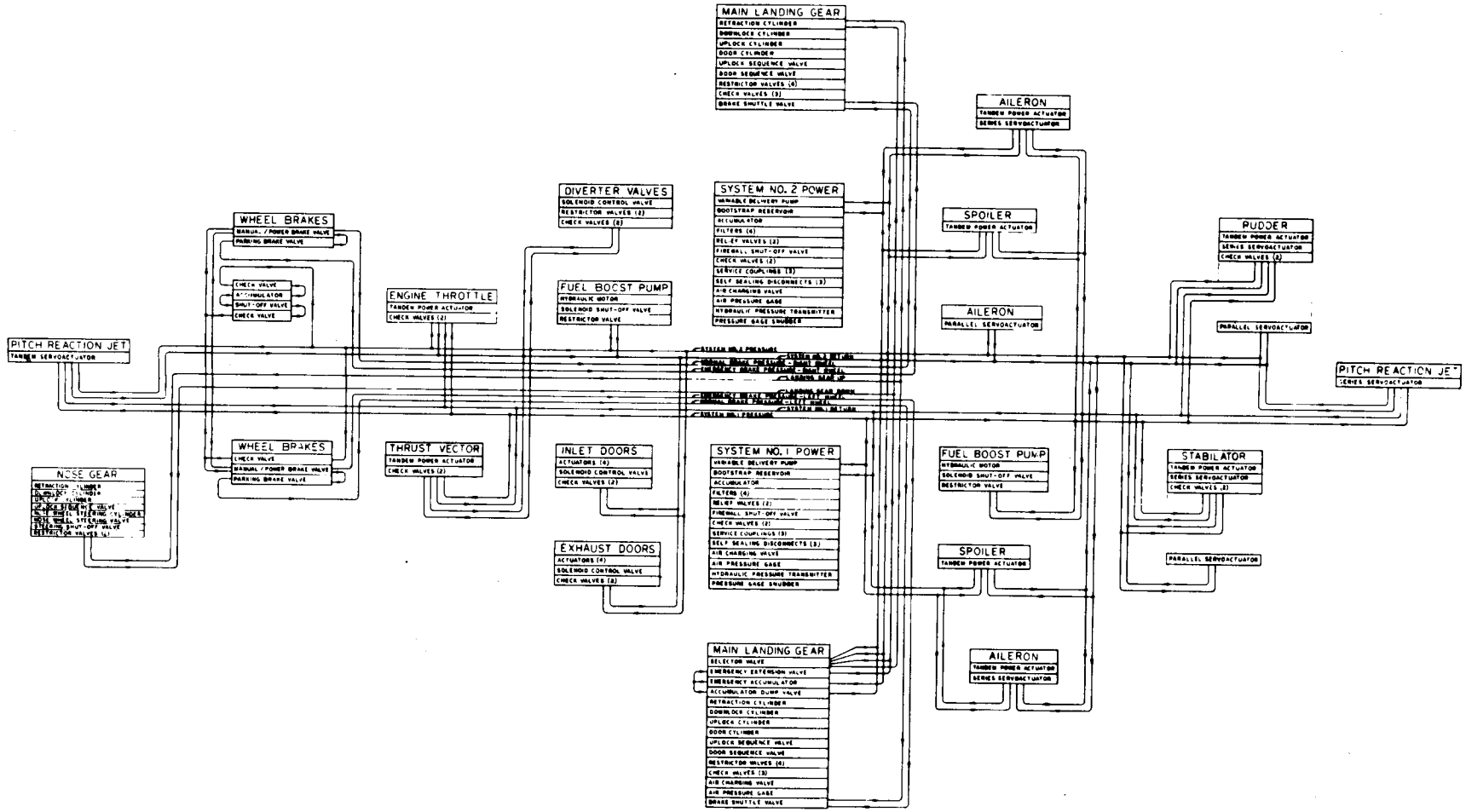

MAIN LANDING GEAR
RETRACTION CYLINDER
DOWNLOCK CYLINDER
UPLOCK CYLINDER
DOOR CYLINDER
UPLOCK SEQUENCE VALVE
DOOR SEQUENCE VALVE
RESTRICTOR VALVES (4)
CHECK VALVES (3)
BRAKE SHUTTLE VALVE
AILERON
TANDEM POWER ACTUATOR
SERIES SERVOACTUATOR
DIVERTER VALVES
SOLENOID CONTROL VALVE
RESTRICTOR VALVES (2)
CHECK VALVES (8)
SYSTEM NO. 2 POWER
VARIABLE DELIVERY PUMP
BOOTSTRAP RESERVOIR
ACCUMULATOR
FILTERS (4)
RELIEF VALVES (2)
FIREWALL SHUT-OFF VALVE
CHECK VALVES (2)
SERVICE COUPLINGS (3)
SELF SEALING DISCONNECTS (3)
AIR CHARGING VALVE
AIR PRESSURE GAGE
HYDRAULIC PRESSURE TRANSMITTER
PRESSURE GAGE SNUBBER
SPOILER
TANDEM POWER ACTUATOR
RUDDER
TANDEM POWER ACTUATOR
SERIES SERVOACTUATOR
CHECK VALVES (8)
WHEEL BRAKES
MANUAL/POWER BRAKE VALVE
PARKING BRAKE VALVE
CHECK VALVE
ACCUMULATOR
SHUT-OFF VALVE
CHECK VALVE
ENGINE THROTTLE
TANDEM POWER ACTUATOR
CHECK VALVES (2)
FUEL BOOST PUMP
HYDRAULIC MOTOR
SOLENOID SHUT-OFF VALVE
RESTRICTOR VALVE
AILERON
PARALLEL SERVOACTUATOR
PARALLEL SERVOACTUATOR
PITCH REACTION JET
TANDEM SERVOACTUATOR
PITCH REACTION JET
SERIES SERVOACTUATOR
WHEEL BRAKES
CHECK VALVE
MANUAL/POWER BRAKE VALVE
PARKING BRAKE VALVE
THRUST VECTOR
TANDEM POWER ACTUATOR
CHECK VALVES (2)
INLET DOORS
ACTUATORS (4)
SOLENOID CONTROL VALVE
CHECK VALVES (2)
SYSTEM NO. 1 POWER
VARIABLE DELIVERY PUMP
BOOTSTRAP RESERVOIR
ACCUMULATOR
FILTERS (4)
RELIEF VALVES (2)
FIREWALL SHUT-OFF VALVE
CHECK VALVES (2)
SERVICE COUPLINGS (3)
SELF SEALING DISCONNECTS (3)
AIR CHARGING VALVE
AIR PRESSURE GAGE
HYDRAULIC PRESSURE TRANSMITTER
PRESSURE GAGE SNUBBER
FUEL BOOST PUMP
HYDRAULIC MOTOR
SOLENOID SHUT-OFF VALVE
RESTRICTOR VALVE
STABILATOR
TANDEM POWER ACTUATOR
SERIES SERVOACTUATOR
CHECK VALVES (8)
NOSE GEAR
RETRACTION CYLINDER
DOWNLOCK CYLINDER
UPLOCK CYLINDER
NOSE WHEEL STEERING VALVE
STEERING SHUT-OFF VALVE
RESTRICTOR VALVES (4)
EXHAUST DOORS
ACTUATORS (4)
SOLENOID CONTROL VALVE
CHECK VALVES (8)
SPOILER
TANDEM POWER ACTUATOR
PARALLEL SERVOACTUATOR
MAIN LANDING GEAR
SELECTOR VALVE
EMERGENCY EXTENSION VALVE
EMERGENCY ACCUMULATOR
ACCUMULATOR DUMP VALVE
RETRACTION CYLINDER
DOWNLOCK CYLINDER
UPLOCK CYLINDER
DOOR CYLINDER
UPLOCK SEQUENCE VALVE
DOOR SEQUENCE VALVE
RESTRICTOR VALVES (4)
CHECK VALVES (3)
AIR CHARGING VALVE
AIR PRESSURE GAGE
BRAKE SHUTTLE VALVE
AILERON
TANDEM POWER ACTUATOR
SERIES SERVOACTUATOR

Figure IX-5

SECTION X

COCKPIT ARRANGEMENT

A. CREW STATION DESIGN

1. General

For Concept "C", the vehicle provides for two crew members in a tandem-seating arrangement. The forward crew station will be occupied by the "Evaluation Pilot" and the aft crew station will be occupied by the "Safety Pilot". Each crew station will be equipped with controls and displays necessary for vehicular flight control. Access to the crew stations is accomplished through two aft-hinged canopies which will open sufficiently to permit convenient ingress and egress with normally used flight apparel.

For Concept "J", the vehicle provides for two crew-members in a side-by-side seating arrangement. The left crew station will be occupied by the "Evaluation Pilot" and the right crew station will be occupied by the "Safety Pilot". Access to the crew stations is accomplished through two side opening canopies, hinged at top center.

Concept "J" is a modified version of the T-39 airframe which will necessitate extensive redesign of the entire crew compartment to meet the NASA V/STOL aircraft requirements. The primary reason for this redesign, however, is to provide an adequate and reliable crew escape capability. This will generally require:

- A new windshield and canopy
- A new aft bulkhead
- Ejection seats
- Control sticks in lieu of control wheels
- Revision of structural members and panels, consoles, and pedestals for the escape clearance envelope

Anticipating the structural design requirements to incorporate the new windshield/canopy, it is difficult to assure adequate hand clearance for the evaluation pilot's operation of engine power controls.

2. Geometric Configuration

The geometric configuration of the control/display mounting surfaces is designed to comply with basic human factors principles for optimum visual reference to and/or access to those control/display elements provided for aircrew operative and monitoring functions. The mounting surfaces of instrument panels, consoles, and the pedestal are canted to be normal to sight lines from the normal eye position, insofar as this is practical. With such configuration, parallax in reading displays and obscurity of identifying nomenclature adjacent to control knobs is minimized.

The established positions of the instrument panels, the consoles, and the pedestal provide unrestricted leg movement for actuation of rudder pedals or wheel braking and comply with HIAD (Handbook of Instructions for Aircraft Designers, AF Manual AFSCM 80-1) requirements for an ejectable-seat-type escape envelope. Primary flight controls such as rudder pedals, control stick and engine power levers are dimensionally located from the Neutral Seat Reference Point (NSRP) in compliance with HIAD drawing AD-1.

3. Arrangement of Controls and Displays

The locations of controls (other than primary flight controls) and displays in the crew stations have been established by human factors analysis. In addition to a task analysis which takes into consideration the conditions of usage (i.e., critical VTOL phase, conversion phase, or cruise phases of flight), priority of usage, and frequency of usage, other determining factors for location of controls and displays were physiology of the human body, seating restraints, and flight safety.

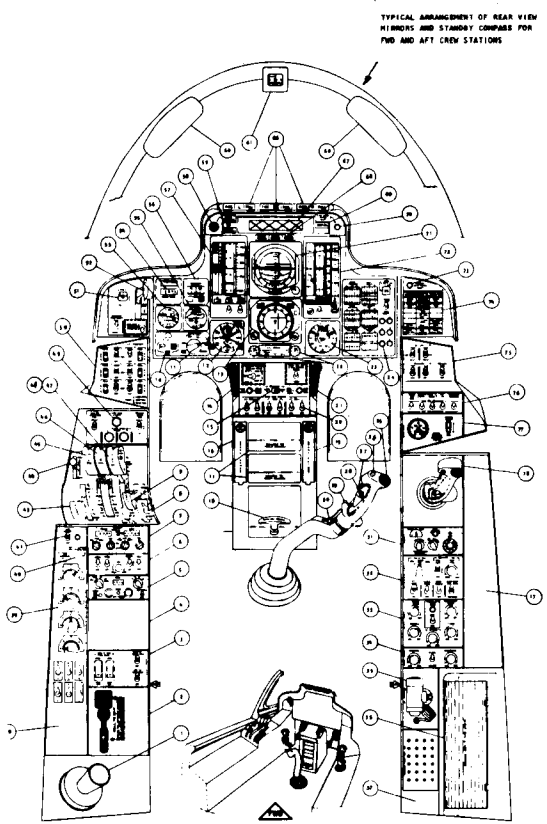
Controls employed in the critical phases of flight have been located for left-hand operation to preclude potential hazards to flight safety which would exist by switching hands for gripping the flight control stick. The general controls falling into this category are those employed for thrust vectoring, induction system, airstarting of engines, landing gear operation, secondary flight controls of an emergency nature, and variable stability system (VSS) engagement. A specially configured switch for emergency disconnect of the VSS has been

placed on the control stick column immediately below the grip so that it is accessible for actuation by either hand.

The arrangement of controls and displays in the crew stations which is depicted in Figure X-1 for Concept "C" and in Figure X-2 for Concept "J" is substantially in accordance with the requirements of HIAD. It must be realized that the HIAD can only be considered as a "guide" to design criteria based on experience factors and general technology influencing flight safety and crew station standardization insofar as practical. There are certain areas where the HIAD requirements cannot be logically applied to some of the unique control/display features being considered in the NASA V/STOL crew station design. This is exemplified by a stated requirement, "Control all VTOL aircraft lifting power, regardless of the powerplant configuration, by a collective-type lever." (Ref. HIAD, Part C. 2, Paragraph 2.13.2.) The Republic proposed propulsion and vectoring controls concept undeniably violates the stated requirement which is considered a premature attempt at standardization of V/STOL controls without evaluation of unique concepts. There are other areas where conflict will be evident, and Republic feels that deviations from stated requirements are justified for efficiency of task execution and safety of flight in V/STOL-type aircraft.

For Concept "C", standby flight instruments are provided in the aft crew station only for the "Safety Pilot," and the variable stability system (VSS) controls (except for emergency VSS disconnect) are provided in the forward crew station only for the "Evaluation Pilot."

For Concept "J," it can be generally stated all critical controls and displays needed for individually controlling vehicular flight and landing requirements are duplicated in both crew stations. The variable stability system (VSS) controls alone (except for emergency VSS disconnect) are provided in the left crew station only for the "Evaluation Pilot." Standby flight instruments, communications and navigation controls, engine starting switches, selective lift engine operation controls, fuel system controls, engine condition indicator, electrical power switches, environmental controls, exterior lighting controls, and canopy jettison control for ground escape are centrally located on the main instrument panel or on the center console where they can be seen and operated by either crew member.



LEGEND

1. PILOT'S RELIEF BOTTLE (PT. CAP.
2. AUX. CANOPY JETTISON CONTROL
3. FUEL MANAGEMENT CONTROL PANEL
4. RESERVED FOR FUTRUE CONTROL PANELS
5. A.T.C. CONTROL PANEL
6. AUDIO CONTROL PANEL
7. COMM/NAV CONTROL PANEL
8. ADJUSTABLE STOP FOR PRE-SELECTING THRUST VECTOR NEEDED FOR STOL
9. THRUST VECTORING CONTROL
10. LAND GEAR CONDITION INDICATOR
11. INSTR. MODE SELECT SWITCH
12. CLOCK
13. HORIZONTAL SITUATION INDICATOR
14. EMERG GEAR EXTENTION CONTROL
15. YAW TRIM SWITCH
16. SPARE LAMPS STOWAGE CASE
17. RESERVED FOR ELECT. CKT BKRS OR FUSES
18. PARKING BRAKE CONTROL
19. SPARE FUSES STOWAGE CASE
20. STAB AUG AND AUX FLT CONTROL PANEL
21. EMERG BRAKE CONTROL
22. HYDRAULIC PRESSURE INDICATOR
23. FUEL QUANTITY INDICATOR
24. ENGINE CONDITION INDICATOR
25. PITCH AND ROLL TRIM SWITCH
26. FUNCTION NOT ASSIGNED
27. STAB AUG. SYS. EMERG DISC. SWITCH
28. COMM. "HIC" SWTICH
29. NOSE WHEEL STEERING SWITCH
30. VAR. STAB. SYS. EMERG DISC. SWITCH
31. COMPASS CONT. PANEL
32. ENVIRONMENTAL CONTROL PANEL
33. INTERIOR LIGHTING CONTROL PANEL
34. EXTERIOR LIGHTING CONTROL PANEL
35. MOVEABLE UTILITY LIGHT
36. FLIGHT DATA AND MAP STOWAGE CASE
37. INTERIOR LIGHT LEVEL BALANCE CONTROL TRIMMING RESISTORS PANEL
38. RESERVED FOR VAR. STAB SYS. CKT BKRS
39. VAR. STAB. SYS FUNCTION SELECT SWS.
40. VAR STAB SYS ENGAGE SWITCH
41. VAR STAB SYS POWER SWITCH
42. BAIL-OUT SIGNALLING SWITCH
43. FLAP HI-LIFT CONTROL
44. POWER LEVER FRICTION CONTROL
45. LEFT CRUISE ENGINE POWER CONTROL
46. RIGHT CRUISE ENGINE POWER CONTROL
47. LIFT ENGINES MASTER POWER CONTROL STOWABLE WHEN NOT IN USE
48. VECTORING BRAKING CONTROL-FOR CRITICAL 65 TO 90 DEGREES RANGE FROM HORIZONTAL GRIP MOVES UPWARD OR DOWNWARD THRU LIMITED SECTOR WITH DETENTED POSITION FOR HOVER
49. ENGINES START AND INDUCTION CONTROL PANEL
50. LIFT ENGINES OPERATION CONTROL PANEL
51. ASSOC. LANDING CONTROL PANEL
52. LAND GEAR CONT-WITH INTEGRAL GEAR UNSAFE WARN LIGHT
53. FLAP HI-LIFT POSITION INDICATOR
54. COMM. FREQUENCY INDICATOR
55. AVERAGE THRUST VECTOR ANGLE INDICATOR
56. SIDE-SLIP INDICATOR
57. AIRSPEED/MACH INDICATOR
58. FIRE EXTINGUISH AGENT DISCHARGE SWITCH
59. BAIL-OUT AND CANOPY UNLOCKED WARN LIGHTS
60. REAR-VIEW MIRROR
61. STANDBY MAGNETIC COMPASS
62. STANDBY AIRSPEED INDICATOR
63. STANDBY ATTITUDE INDICATOR
64. STANDBY TURN AND SLIP INCIATOR
65. STANDBY ALTIMETER
66. FIRE EXTINGUISH ARMING CONT-WITH INTEGRAL FIRE WARN LIGHT
67. PARA-VISUAL DISPLAY OF ANGLE-OF-ATTACK AND SIDE-SLIP
68. AIRWAY, MIDDLE, AND OUTER MARKER BEACON LIGHTS
69. MASTER CAUTION LIGHT
70. FIRE WARN SYS TEST SWITCH
71. ATTITUDE DIRECTOR INDICATOR
72. ALTITUDE VERTICAL VELOCITY INDICATOR
73. MASTER CAUTION RESET SWITCH
74. CAUTION ANNUNCIATOR LTS PANEL
75. ELECT. POWER CONTROL PANEL
76. SYS. MONITORING TAKE-OVER CONTROL PANEL
77. OXYGEN CONTROL PANEL
78. SIDE-STICK FLIGHT CONTROLLER

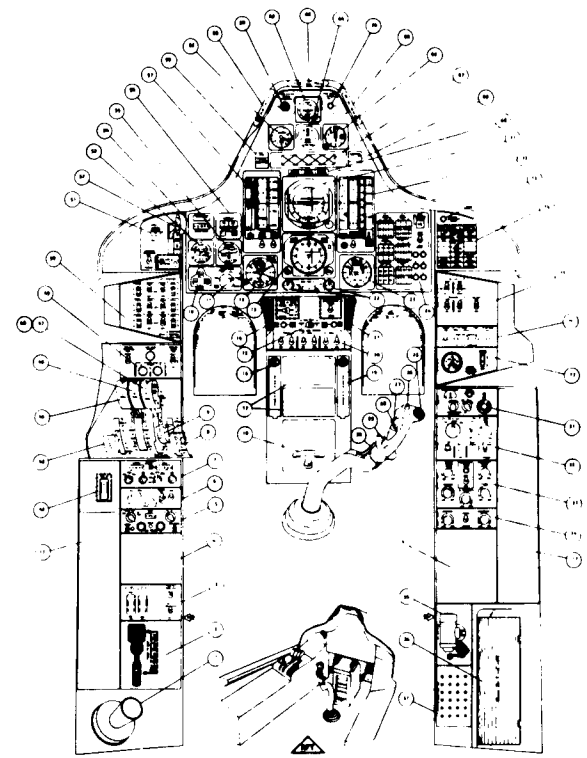


Figure X-1. Arrangement of Controls and Displays Crew Stations Concept "C"

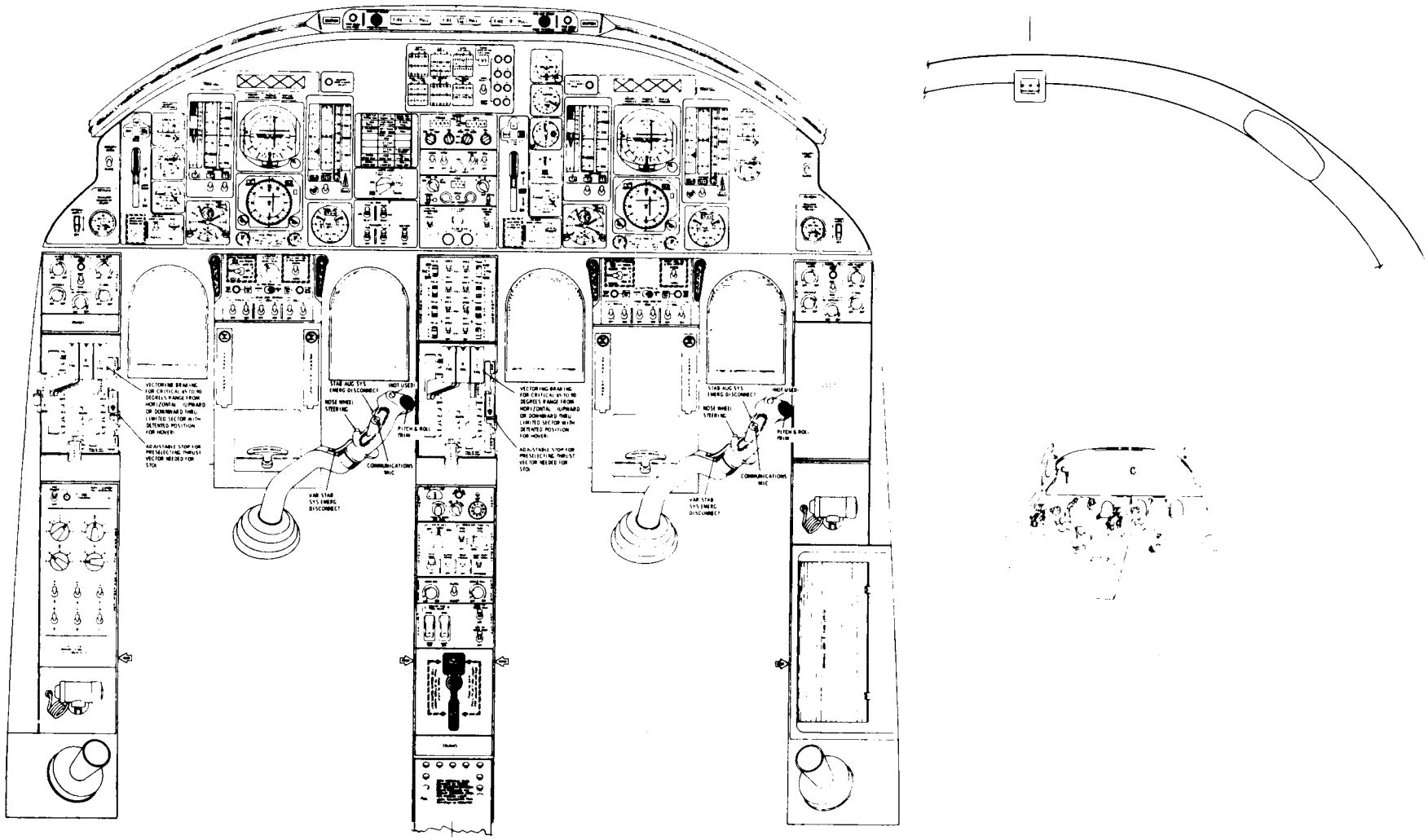


Figure X-2. Arrangement of Controls and Displays Concept "J"

4. External Vision

For Concept "C" the necessity for maximum possible external vision has been taken into consideration in establishing aircraft nose section contours, location of structural members for the crew compartment and its enclosure (i.e., canopy and windshield), and the upper edge contour of the instrument panels with their glare shields.

For Concept "J", modification of an existing airframe limits the external vision. Although the external vision normally available in the T-39 can be improved upon in the process of redesigning the crew compartment area, the vision requirements of the NASA V/STOL work statement cannot be met. As a more comprehensive study is required to define the maximum vision that may result through rework, further discussion of external vision for Concept "J" is omitted from this report, and the following discussion pertains to external vision provided in the new airframe proposed for Concept "C".

For the forward crew station, the use of primary flight indicators similar to the AF integrated instrument group and the introduction of the peripheral command indicator used for a "para-visual" display of angle-of-attack and side-slip inherently establish a basic height for the central portion of the main instrument panel. Considering the best compromise between requirements for ejection clearance envelope, visual reference to instruments, and external vision, the upper edge of the instrument panel glare shield becomes the critical factor governing the downward vision over-the-nose.

The forward crew member will be provided with 18.5° downward vision from 0° azimuth up to 15° azimuth and then a sharp increase to 30.2° downward at 15° azimuth, from which point the vision angle gradually increases to a maximum of 37.5° downward at 75° azimuth and then decreases so that 32.2° downward vision is available at 110° azimuth.

The forward crew member's external vision will be substantially reduced to 10.7° downward from 0° azimuth through 16.2° azimuth at such time that an advanced pictorial display (anticipatory development item of 17 inches wide by 18 inches high by 15 inches deep, approximate dimensions) is substituted for the AF-type integrated flight instruments.

The aft crew member will be provided with 5° downward vision over the top edge of the forward crew station seat headrest, i.e., at 0° azimuth from which the vision angle continually increases to 27° downward at 30° azimuth, then to a maximum of 37.5° downward at 80° azimuth, and then decreases so that 33.5° downward vision is available at 110° azimuth. The aft crew member will be able to see 11° below the horizon over the centerline of the fuselage nose by merely shifting his head laterally.

The external visual range afforded both crew members is depicted in Figure X-3 which plots elevation vision angles at azimuth angular positions from 0° through 180°, based upon the normal or "design" eye position. A separate table in Note 3 of Figure X-3 indicates comparative increase in downward vision when the crew members take advantage of the ability for considerable lateral head movement (to the extent afforded by the inner surface of the canopy glass) for maximum view of the terrain below the aircraft.

5. Escape Provisions

Each crew station for Concept "C" and Concept "J" is equipped with an ejectable seat designed for safe escape from zero altitude, and zero velocity to maximum altitude and airspeed performance capability of the vehicle. However, it must be realized that there is a risk factor involved during hovering conditions in close proximity to the ground where ejection at unusual attitude may not permit a trajectory for safe recovery of the aircrew. The escape system provides for automatic jettisonning of the canopies immediately prior to ejection of the seats and crew members; and although highly reliable cartridge-actuated devices will be employed, the crew can safely eject through the canopy in the event of its failure to be jettisoned.

A more comprehensive discussion of the escape system is provided in Paragraph C.3.

6. Cabin Environment

The crew compartment for Concept "C" and Concept "J" will be air conditioned for comfort. Although cabin pressurization will not be provided, the

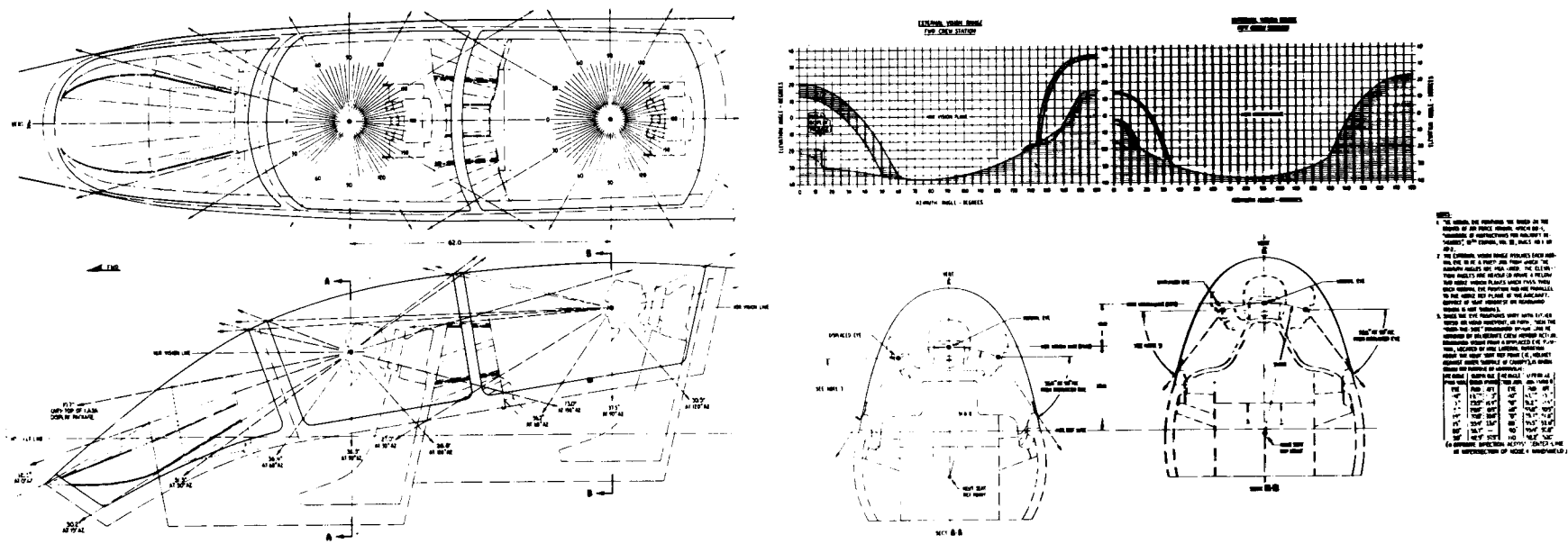


Figure X-3. External Vision Range
Crew Stations Concept "C"

compartment will be sealed (inflatable seal at the edge of each canopy) to prevent introduction of dust and gaseous irritants during VTOL flight phases. A more comprehensive discussion of the Environmental Control System is contained in Section IV.

Supplemental breathing oxygen will be provided for each crew member, sufficient for a duration of 34 minutes. A description of the oxygen system is contained in Paragraph C. 2.

Artificial illumination for controls and displays will provide for aircraft operations at night or low ambient light conditions. The brightness levels for illumination will be independently controllable at each crew station. A discussion of the lighting system is contained in Paragraph C. 8.

B. CONTROLS AND DISPLAYS

1. Flight Control

a. Primary Flight Controls

Basic control of aircraft roll and pitch attitudes will be accomplished by a conventionally located stick column equipped with the AF standard Type B-8A grip. Stick travel will be within the limits prescribed by HIAD drawing AD-2. The stick in the forward or left crew station will be capable of controlling movement of the aerodynamic surfaces and reaction jets either hydromechanically or electro-hydraulically through a "fly-by-wire" system. The stick in the aft or right crew station will provide hydromechanical control only and override the forward or left stick when operating in the "fly-by-wire" mode.

For Concept "C", a separate "side-stick" flight controller is provided in the right console of the forward crew station only for "fly-by-wire" operation, and its control can also be over-ridden by the stick in the aft crew station. For Concept "J", the "side-stick" will not be provided,as its optimum location for the Evaluation Pilot's use, based on human factors and safety of flight considerations, is the same location that must be occupied by the Safety Pilot's engine power controls.

Aircraft yaw control will be accomplished by conventionally located rudder pedals, which will be of a configuration identical to or similar to the "sliding" type pedal assemblies employed on the F-105 series aircraft. Pedal travel for control purposes will be in accordance with HIAD drawing AD-1. The pedals will be adjustable to accommodate crew members of different physical stature by means of a spring-loaded lever and pawl, at the outboard side of each pedal, which engages any one of several slotted positions in the adjustment link. The forward crew station of Concept "C", and both left and right crew stations of Concept "J", will be provided with the full adjustment range specified by HIAD drawing AD-1; however, the aft crew station of Concept "C" will have a reduced adjustment range (penalty for taller pilot) because of limitations imposed by crew compartment dimensions. The pedals in the forward or left crew station will be capable of movement of the aerodynamic

surface and reaction jets either hydromechanically or electro-hydraulically through a "fly-by-wire" system. The pedals in the aft or right crew station will provide hydromechanical control only and override the forward or left pedals when operating in the "fly-by-wire" mode.

A more comprehensive description of the primary flight control system is provided in Section IX.

b. Secondary Flight Controls

A control for extension and retraction of the high-lift devices (flaps and/or slats) will be located on the left console for Concept "C", and on the left outboard and center consoles for Concept "J", in the vicinity of the engine power levers. A flap-position indicator will be located on the main instrument panel at the left side of the primary flight instrument group.

Control of pitch trim and roll trim will be accomplished by the "coolie-hat" switch located on the stick grip. An emergency pitch trim override switch will be provided, located on the pedestal. Control of yaw trim will be accomplished by a switch located on the pedestal. Trimming the aerodynamic surfaces for take-off conditions can be accomplished by a push-button-type switch located on the pedestal, adjacent to which an electromechanical display element will be provided to indicate "TRIM OK".

Emergency override of the automatically positioned pitch and roll mechanical advantage shifter and yaw limiter will be accomplished by a switch located on the pedestal. Fault signal lights, one for each of the axes, will be included on the Caution Annunciator Panel.

Control of the Stability Augmentation System (SAS) will be accomplished by six solenoid hold-in-type toggle switches located on the pedestal. Because of system redundancy, two switches each are employed for the pitch, roll, and yaw axes. SAS fault signal lights, one for each of the axes, will be included on the caution annunciator panel. Testing the system can be accomplished by a push-button-type switch also located on the pedestal, adjacent to which an electromechanical display element will be provided to indicate

"SAS OK". Emergency disconnect of the SAS will be accomplished by the "trigger" switch located on the stick grip.

Controls for the Variable Stability System (VSS) will be located on the left console in the forward crew station only for Concept "C", and on the left outboard console in the left crew station only for Concept "J". A lever-lock-type toggle switch will be used for application of system power and a small light adjacent to the switch will illuminate when VSS power is on. A push-pull-type switch having a solenoid hold in feature will be used for system engagement and an advisory signal light adjacent to the switch will be provided to indicate "VSS ENGAGED". Variable functions will be introduced by means of four rotary switches and six toggle switches which will control system operation through the use of pre-programmed "patch-boards". Identification of the VSS function switches and their respective selectable positions will be accomplished by means of "sticky-back" paper placards, pre-cut to appropriate shape, on which appropriate inscriptions may be applied by hand-printing or typing and then affixed to the console surface at the appropriate switches. Emergency disconnect of the VSS will be accomplished by means of a special switch, located on the control stick column in both crew stations, capable of being actuated by either hand.

Refer to other sections in this document for a more comprehensive description of the secondary flight control systems.

c. Flight Instrumentation

The primary flight instruements will consist of the basic AF Integrated Flight Instrument Group, modified for displays appropriate to the NASA V/STOL aircraft application, which will be centrally located on the main instrument panel. Additional instrumentation for monitoring flight and/or flight control systems will also be located on the main instrument panel. The following will be provided at each crew station, except for those standby instruments noted as being installed in the aft or right crew station only:

Altitude Director Indicator
Horizontal Situation Indicator

Airspeed/Mach/Safe Speed Indicator (Modified)

Altitude/Vertical Velocity Indicator (Modified)

Peripheral Command Indicator for "para-visual" display of angle-of-attack and sideslip

Side-Slip Indicator

Flap Position Indicator

Clock

Primary Hydraulic Pressure Indicator

Utility Hydraulic Pressure Indicator

Standby Magnetic Compass

Standby Attitude Indicator	- aft or right station only
Standby Airspeed Indicator	- aft or right station only
Standby Turn and Slip Indicator	- aft or right station only
Standby Altimeter	- aft or right station only

(1) Attitude Director and Horizontal Situation Indicators

Control of various displays presented by the Attitude Director Indicator (ADI) and Horizontal Situation Indicator (HSI) will be accomplished by a rotary-type switch, located on the main instrument panel, which is commonly referred to as the "Instrument Mode Selector." Information on the Instrument Mode Selector (IMS) switch and the HSI and ADI are presented together in this one section rather than on an individual basis as the operation of these two indicators is directly related to IMS switch position. The possible combinations provided by the five IMS switch positions and the thirteen HSI and ADI indicators lend them to a visual format and, hence, the following charts applicable to each instrument and the IMS switch are presented.

ATTITUDE DIRECTOR INDICATOR OPERATION

Instrument Selector Switch Position			
Indicator	Manual	ILS Localizer and Final	ILS Outbound
Bank Steering	Desired Heading (Heading error and bank) Indicates correct intercept approach to manually set heading marker	When centered indicates correct track (heading and crosswind). Indicates correct intercept approach to a runway course (loc. deviation and bank and crosswind).	Desired heading (heading error and bank). Indicates correct intercept approach to manually set heading marker set on HSI.
Localizer Doppler Alarm Flag	Out of view	Appears if signal strength is weak or unreliable	Out of view
Glide Slope Pointer	Out of view	Indicates glide slope deviation against the displacement index.	Out of view
Pitch Steering Needle	Out of view	Provides lead information to intercept the glide slope.	Out of view
Glide Slope Alarm Flag	Out of view	Appears if signal strength is weak or unreliable	Out of view

HORIZONTAL SITUATION INDICATOR OPERATION

Indicator	Instrument Selector Switch Position		
	Manual	ILS Final-Localizer-Outbound	NAV. (VOR/DME)
Heading Marker (Captains Bars)	← Any desired manually set heading →		No effect
Course Arrow	Arrow becomes fixed and aligned with top of L.L.*	Runway Course Manually Set	No effect
Course Deviation Bar	Aligned with course arrow	Indicates deviation from selected runway course	No effect
Bearing Pointer	Becomes fixed and aligned with top of lubber line	Indicates bearing to VOR station selected	Indicates VOR station bearing
Distance Readout Window	Shuttered	Indicates distance to selected station in nautical miles	Indicates station distance
Mode Lights	Man	ILS and VOR	VOR/DME
To - From Arrow	← Out of View →		Indicates to or from selected radial
Course Readout Window	Indicates Manually set Heading	Indicates runway course (Manually set)	Indicates VOR radial (manually set)

* L.L. - Lubber Line

(2) Compass Control Panel

The compass system control panel located on the right console or center console is used in conjunction with the HSI for basic heading reference during flight. This panel provides the pilot with all necessary controls for using the compass system in either the magnetically slaved or directional gyro modes. Lattitude correction, synchronization and slewing controls are also available on this unit. A null or synchronization indicator in the forward left hand corner of this panel indicates the correspondence between direction gyro and flux valve readings when in the slaved magnetic mode.

(3) Air Speed/Mach Indicator

The Air Speed/Mach Indicator (AMI) is located to the left of the HSI and ADI indicators. This unit will operate in accordance to MIL-SPEC, MIL-A-27329, Amendment I, except for range of operation which will be modified for V/STOL suitability. Basically, the method of presenting the displayed information employs a vertically moving tape with the imprinted scale which is read against a fixed reference line. Command or limit indications are accomplished by translational movement of symbols along the graduated scales. This indicator provides information on the following seven parameters and will cover the ranges indicated for each as follows:

- Aircraft Mach No. - 0.2 to 0.9
- Maximum Safe Mach No. - 0.2 to 0.9
- Indicated Air Speed (-50) to 500 knots
- Safe Speed (in terms of angle-of-attack) -10 to +25°
- Command Mach No. 0.2 to 0.9
- Command Air Speed (-50) to 500 knots
- Acceleration 0 to 7 "g"

(4) Altitude/Vertical Velocity Indicator

The altitude/Vertical Velocity Indicator (AVVI) is located to the right of the HSI and ADI on the main instrument panel and will operate in the manner described in Spec. MIL-A-27328 except as modified

for scale range and radar altimeter readout. Essentially four vertical scales for rate of climb, fine altitude, coarse altitude, and radar altitude respectively are presented on the instrument face. Fine altitude and radar altitude are read against the moving tapes and reference line on the applicable scales. Rate of climb and gross altitude are read against symbol pointer on the fixed scales. Similarity cabin altitude is read by a moving dotted line against the fixed gross altitude scale. The following parameters, and the range associated with each, is presented on this instrument.

* Aircraft pressure altitude - 1000 to + 40,000 ft (sensitive scale) in 10 ft increments.
* Aircraft pressure altitude - (-1000) to +40,000 ft (gross scale) in 1000 ft increments.
* Vertical speed ± 6000 fpm.
o Command Altitude, 0 to 40,000 ft
Δ Radar Altitude, 0 to 1000 ft
* Cabin Altitude 0 to 40,000 ft

Items marked (*) are derived by signals generated within the air data computer while item marked (o) is generated internally within the instrument. Item marked (Δ) is supplied by the doppler velocity sensor. A barometer set knob and barometer readout window are also provided on this unit for correcting pressure altitude.

(5) Peripheral Command Indicator

The peripheral command indicator (PCI) is used for "para-visual" display of angle-of-attack and side-slip. It is located in a position above the basic Integrated Flight Instrument Group where its operation is discernible by direct or peripheral vision.

The purpose of this unit is to display steering commands or other flight data in a relative sense rather than a quantitative value while the pilots attention is focused on another instrument or in the forward field of view. In this application the PCI has been mechanized to display aircraft angles of attack and side slip. Two clear-glass cylinders using a right-hand and left-hand

spiraled pattern combine to form a diamond pattern when viewed in the indicator window. By driving these cylinders in opposite directions, the diamond patterns can be driven towards the left or right depending on the direction of the sensed side slip, and upward or downward depending on the direction of sensed angle of attack. In the presence of side slip and angle-of-attack changes, the resultant PCI indication will be a diagonal motion of the diamond pattern depending on the rate and direction of the sensed parameters.

(6) Side-Slip Indicator

The Side-Slip Indicator is a small instrument servo that positions its graduated tape unit as a function of the electrical transducer signal derived from the aircraft's side-slip sensor. Unlike the para-visual display which indicates side slip sense and angular build-up, this unit provides the pilot with a quantitative readout of this parameter. The moving tape is read against a fixed reference line, and its direction is indicated by the corresponding (Right or Left) display element.

(7) Standby Magnetic Compass

The standby magnetic compass will be located at the center portion of the windshield/canopy bow above the instrument panels. It is a small self-contained magnetic-North-seeking unit provided as an emergency backup and as means for roughly checking the aircrafts primary directional gyro-flux valve heading reference.

(8) Standby Attitude Indicator

The Standby Attitude Indicator, unlike the main ADI which utilizes a remote highly accurate vertical gyro, incorporates a small self-contained two axis gyro for providing pitch and roll attitude information in the event of a failure within the primary attitude system. A pitch trim control is included on this indicator.

(9) Standby Airspeed and Altitude Indicators

The standby airspeed indicator and the standby altimeter both operate from pneumatic pressure inputs directly from the pitot-static tube

and are provided as back-up units to the electrically operated vertical tape instruments. A barometric pressure setting control is included on the standby altimeter.

(10) Standby Turn and Slip Indicator

The standby turn and slip indicator is provided as a back-up unit for the turn and slip display contained in the ADI. The needle is operated from a turn rate gyro. The ball is a self-contained inclinometer.

2. Propulsion, Induction and Vectoring

a. Engine Controls

Basic control of lift-cruise engine power will be accomplished by two levers, one each for the left engine and for the right engine. Control travel will be within limits prescribed by HIAD drawing AD-1. Basic control of power for the six lift engines in Concept "C", or the eight lift engines in Concept "J", will be accomplished by a single master power lever. The lift engine power lever will be capable of being stowed (i.e., pushed down into the quadrant when operation of the lift engines is not required), thereby providing unrestricted access to and/or actuation of the lift-cruise engine power controls. The engine power controls will be located on the left console in Concept "C", and on the left outboard and center consoles in Concept "J", in a configuration as depicted by Figure X-4.

A control will be provided, located in proximity to the engine power levers in the forward or left crew station only, to apply friction for varying the actuating force of the lift-cruise engine power levers.

Starting of engines is accomplished by three push-button-type switches, one each for the left and right lift-cruise engines, and one for the lift engines. The lift engines may be started individually, in groups, or all simultaneously. The engine start switches are located on the left console in Concept "C" and at the lower center of the main instrument panel in Concept "J" in front of the engine power controls.

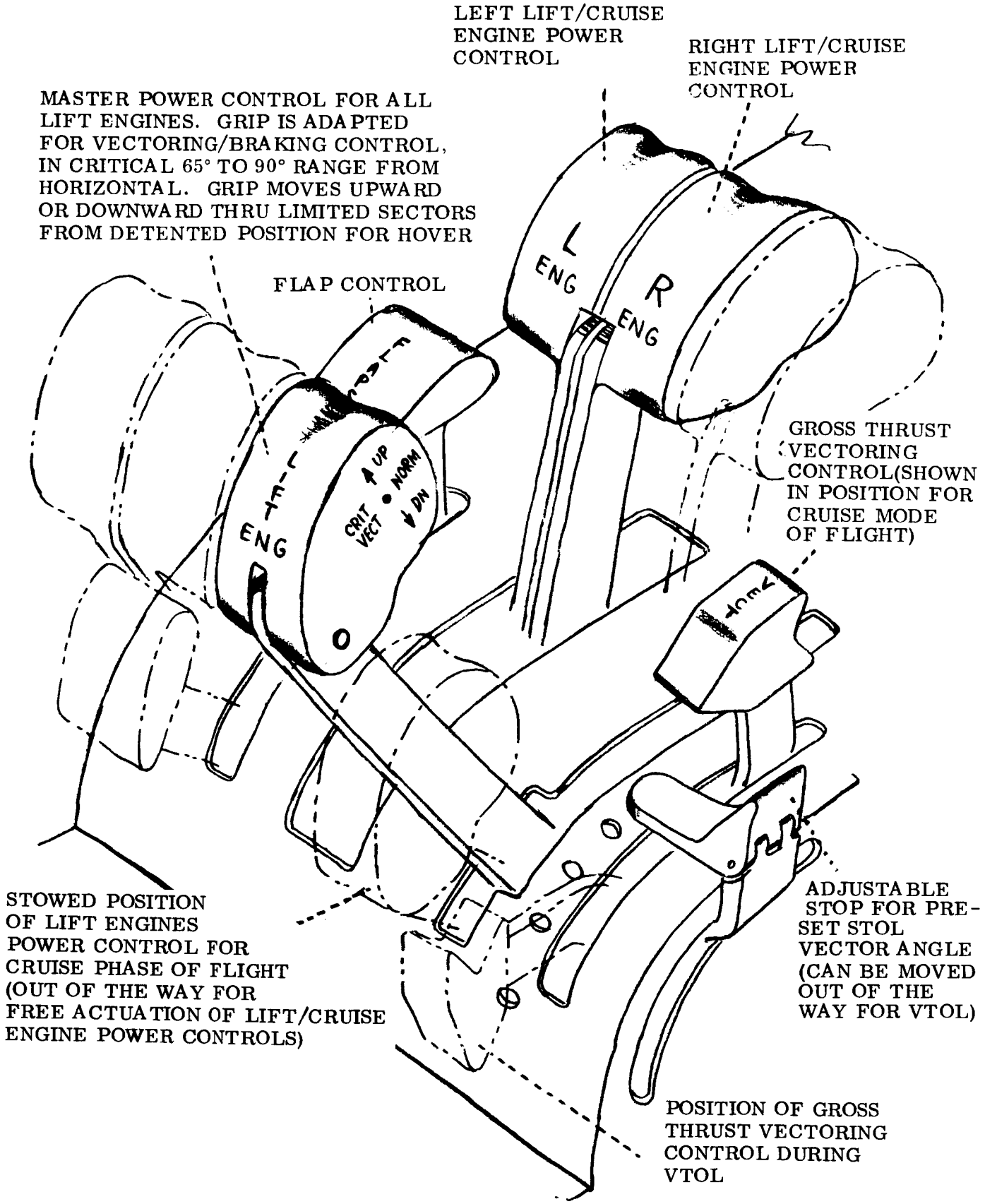


Figure X-4. Engine Vectoring Controls

Controls for selective lift engine operation (i.e., the capability of advancing power for one or more lift engines while other lift engines are maintained at idle) will be located on the left console in Concept "C", and on the center console in Concept "J". These controls consist of six or eight lever-lock and solenoid hold-type toggle switches and six or eight spring-return-to-neutral slide-type switches, one of each to be applicable for each of the lift engines. Operation of these controls is described in Figure X-5.

A switch will be provided for control of the cruise engine exhaust gas diverters and will be located in proximity to the engine starting and engine power controls.

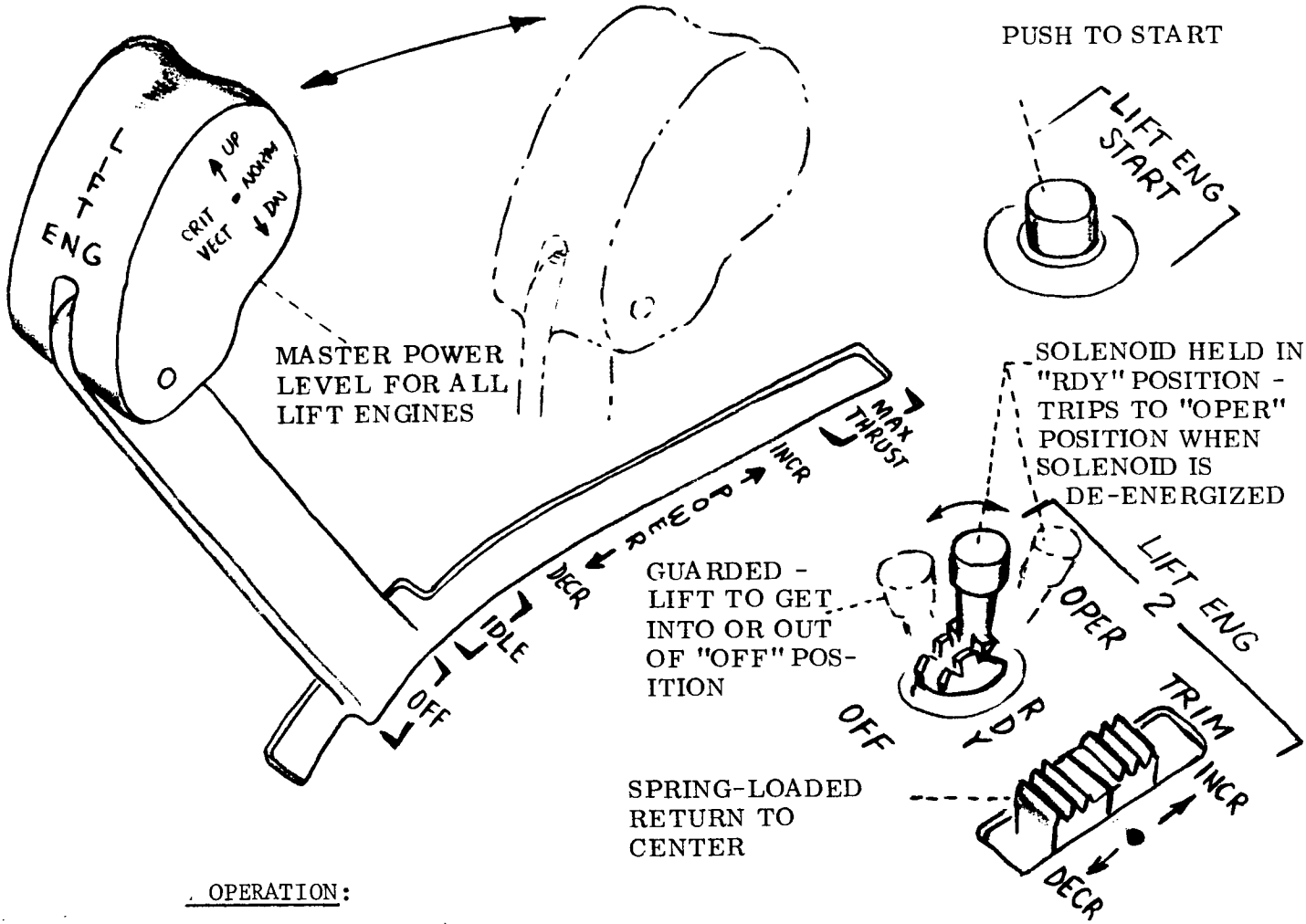
A switch will be provided for control of the lift engine ram air inlet doors and will be located in proximity to the engine starting and engine power controls.

A more comprehensive description of the engine control system and the operation of the diverters and ram air inlet doors is provided in Section IX.

b. Vectoring Controls

Gross thrust vectoring will be accomplished by a lever-type control located just inboard of the engine power controls. The lever will rotate through a sector of 90°. In its forward position, the shape-coded knob will indicate the propulsion force is in an aft-ward direction for CTOL or cruise flight. In its aft position, the knob will indicate the propulsion force is in a downward direction for VTOL or hover. The lever in the forward or left crew station will be capable of controlling movement of the nozzles electro-hydraulically. The lever in the aft or right crew station will be capable of controlling the nozzles hydro-mechanically and of overriding the forward or left vectoring lever.

An adjustable stop for the gross thrust vectoring control will be provided, located adjacent to the control lever, to permit movement of the control lever to pre-determined thrust vector angle for STOL operation.



OPERATION:

1) Move lever from "OFF" to "IDLE" - all fuel control valves are in standby condition.

2) Move #2 Eng Sw. to "RDY" power is applied to open air valve to spin engine - when engine is at proper rotational speed, a sensor will cause closure of a micro switch to permit introduction of fuel.

3) Push "START" switch for ignition, when engine is started, a sensor will cause air valve to shut and also cause removal of power from #2 Eng switch solenoid at which point toggle will automatically move to "OPER" position.

4) In "OPER" position, the #2 Eng trim circuit is activated permitting engine run-up or trim for balance of thrust output (RPM).

5) Move TRIM switch fwd - while holding, Engine will continue to run-up to max power; remove finger, switch returns to center, engine remains at power level reached.

6) Move TRIM switch aft-while holding, engine power will decrease to idle pos. or stop at any power level reached when switch returns to center.

7) Use same procedure for starting any selected engine

8) Use same procedure for gang starting selected engines either in groups or all simultaneously.

9) In "OPER"-all engines will follow power lever movement.

Vectoring/Braking control, for critical 65° to 90° range from horizontal will be accomplished by movement of the grip provided on the master lift engines power lever. The grip will move upward or downward through limited sectors from a detented position for "hover", controlling movement of the nozzles electro-hydraulically.

The configuration of the thrust vectoring controls is depicted in Figure X-4.

A comprehensive description of the thrust vectoring system is provided in Section IX.

c. Fire Extinguishing Controls

The fire extinguishing system is operated by three control handles and an extinguishing agent discharge button located at the top of the main instrument panel. Fire eye sensors monitoring each lift and cruise engine detect the presence of any fire by the attendant increase in IR radiation and immediately causes the built-in warning lamp in the left, center, or right fire extinguishing handle to illuminate depending on the fires location (i.e. left cruise engine, lift engines, and right cruise engine). Pulling the handle involved will immediately shut-off all fuel to the applicable engine(s) and fire the gas generator squib pressuring the extinguishing agent. Depressing the agent discharge button immediately opens the actual extinguishing agent discharge valve supressing the sensed fire. System test can be activated by depressing the fire warning system test button below the right extinguisher pull handle. Correct system operation is indicated by the illumination of all three handle lights.

d. Propulsion System Instrumentation

Propulsion system instrumentation will consist of an integrated Engine Condition Indicator which must be developed for the NASA V/STOL application and a Thrust Vector Indicator. The instruments will be located on the main instrument panel at each crew station in Concept "C"; however, only one centrally located engine condition indicator will be provided in Concept "J".

(1) Engine Condition Indicator

The Engine Condition Indicator provides the pilot with a display of six basic engine parameters associated with the two lift-cruise engines and the six or eight lift engines. To duplicate this same data on an individual basis for each engine would require a prohibitive number of indicators; hence, a special monitoring technique has been adopted.

The indicator will contain six display areas, one for each of the six parameters. Each of the display areas will provide three vertically moving tapes with imprinted scales which are read against a fixed reference line. The two outer scales of each display area will continuously provide information for the condition of the left and right lift-cruise engines. The center scale of each display area will provide information for the condition of each lift engine on a time-shared basis whenever the lift engines are in operation; when not in operation the center scale will appear blank. A lift engine "identity" display is provided for correlation purposes, i.e., to inform the pilot of that lift engine for which the parameters of the center scale are applicable.

The instrument also contains a two-position solenoid-hold toggle switch for selection of either an "auto-scan" or a manually selective engine monitoring mode of operation. Six or eight solenoid-hold integrally lighted push-button-type switches are provided for selection of individual engine condition display.

In the "auto-scan" mode of operation, each lift engine is cyclically monitored for a fixed time interval sufficient to permit pilot assessment of the condition of the engine; i.e., after dwelling on one lift engine, the next lift engine will be monitored and parameters displayed, repetitively completing each cycle. In the manually selective mode of operation, the pilot will depress one of the push-buttons for the lift engine he desires to monitor at which time the center scale display will remain "fixed" for the selected engine until either another engine push button is depressed or the "auto-scan" mode is selected. Once depressed, the solenoid-hold-in feature will retain the switch in the active position, negating the need for the pilot to hold the push-button down. This action

will also automatically release the toggle switch, which is solenoid-held in the "auto-scan" position, causing the system to revert to the manually selective mode of operation. Actuating the toggle switch to the "auto-scan" position will then cause any depressed push-button to be released.

The indicator will also operate in conjunction with the Master Caution signalling system for two lift engine parameters which are considered most critical, i.e., "Lift Engine RPM" and "Lift Engine EGT." In the event any of these parameters are abnormal, the master caution light and the appropriate fault legend light on the caution annunciator panel will both illuminate along with the integral lamp in the push-button switch for the lift engine exhibiting the abnormal condition which immediately isolates the problem area. The pilot need only to depress the push-button switch which has been illuminated to obtain the display of all parameters for the faulty lift engine to assess the situation.

Use of a moving scale read against a fixed reference line is preferred in lieu of a fixed scale/moving reference since it permits use of an extensive scale to enhance readability of quantitative values, not only for flight monitoring but also for ground checkout and maintenance. The use of color-coded tapes may also be employed to simplify flight monitoring by the pilot.

(2) Thrust Vector Indicator

The Thrust Vector Indicator provides a visual display to the pilot of the average thrust vector (i.e., not the position of the nozzles) without diverting his attention from flight duties to look at the position of the gross thrust vectoring control.

3. Fuel System

a. Fuel System Controls

Two toggle-type switches having cover guards, will control the main shutoff valves for the left and right lift-cruise engines. (Fuel shutoff valves for the lift engines are controlled by the selective lift engine operation switches.)

Two lever-lock-type toggle switches will be provided for fuel boost pump operation. One switch will control the two boost pumps in the forward fuel tank and the other will control the two boost pumps in the aft fuel tank.

The fuel system controls will be located on the left console in Concept "C", and on the center console in Concept "J".

Fault signal lights will be provided on the Caution Annunciator panel to indicate low fuel pressure and boost pump failures.

b. Fuel Quantity Indicator

A fuel quantity indicator, displaying contents of the forward and aft tanks and total fuel aboard, will be located on the main instrument panel at each crew station. The indicating system will be of a capacitance type meeting MIL-Spec accuracy requirements.

4. Landing and Deceleration

a. Landing Gear Controls

Normal extension and retraction of the landing gear will be accomplished by a lever type control, having a wheel-shaped knob, located on the left segment of the instrument panel in Concept "C", and in two positions on the main instrument panel in Concept "J".

Emergency override control of the landing gear down-locks is provided by actuation of a momentary contact toggle switch in an upward direction. The switch is located below the landing gear control.

Emergency extension of the landing gear is provided by a control, located at the left side of the pedestal, which is actuated by pulling the handle.

b. Landing Gear Condition Indication

Three electromechanical display elements, one each for the left, right, and nose landing gear, will be provided to indicate "safe" when the gear is "locked" in the up or down position. When the gear is not locked, the

display element will indicate red diagonal stripes. The display elements will be located on the main instrument panel (in proximity to the landing gear control) arranged in a pattern to signify association to the respective gear positions.

The wheel-shaped knob on the landing gear control will illuminate red in color as a warning signal that the landing gear is not locked in position consistant with that selected by the pilot when actuating the control lever. Which of the gear positions that is unsafe can be ascertained by visual reference to the three display elements.

An audio warning signal (beeper) will be introduced in the communications head-set when a landing attempt is made with the "wheels-up."

A toggle switch, located in proximity to the landing gear control, is provided to permit testing of the Landing Gear Unsafe warning light and audio signal, and to permit silencing of the audio signal.

c. Wheel Braking Controls

Normal wheel braking for deceleration after landing or to stop movement of the aircraft on the ground will be accomplished in a conventional manner by depressing the rudder pedals which will apply hydraulic pressure to the wheel brakes.

Emergency acquisition of pressure needed for wheel braking is provided by a control, located at the right side of the pedestal, which is actuated by pulling the handle-after which the rudder pedals must be depressed.

Application of parking brakes is accomplished by means of a control that is centrally located at the base of the pedestal.

d. Nose Wheel Steering

Control of nose wheel steering is accomplished by use of a switch located at the lower left side of the control stick grip.

5. Electrical Power and Lighting

a. Electrical System Power Controls

The electrical system power controls are grouped on a panel located on the forward portion of the right console in Concept "C", and centrally located on the instrument panel in Concept "J".

This panel provides both AC generator and transformer-rectifier switches (DC) associated with the left and right cruise engines respectively. As generator paralleling is accomplished automatically, no separate switch is provided for that purpose. As soon as either cruise engine is up to speed, the appropriate AC generator switch may be placed in the "ON" position. Next the transformer-rectifier switch associated with the generator selected may be activated to supply power to the aircraft's DC buss. These two switches can supply all the aircraft's AC and DC power needs; however, to take advantage of the redundant design provided, both generators and transformer-rectifiers should be used. As soon as the second cruise engine is up to speed, its generator and transformer-rectifier switches can be activated. Paralleling of the AC units is accomplished automatically, as indicated previously, and the pilot is advised when the unit is on the line by the appropriate illuminated generator warning light on the caution annunciator panel going out. In the absence of ground electrical power or during an inflight emergency, the battery switch can be activated for primary buss AC and DC power.

b. Electrical Circuit Protection

Electrical fuses and circuit breakers will be employed for protection against overload conditions. The quantity and sizes of the protective devices will be determined after development of electrical circuitry needed for operation of all installed components and subsystems. Sufficient space has been allocated in the crew stations for the installation of protective devices associated with critical circuits.

c. System Monitoring Take-over Controls

For Concept "C" only, some of the duplicated electrical/electronic control elements in one crew station must be isolated when operation

is accomplished at the other crew station to prevent incompatible signal inputs to the applicable systems. For such cases, a "Take-over" control panel having five solenoid hold-in-type toggle switches is provided whereby either crew member can acquire command of certain systems at which time the control elements at his station are active while the corresponding control elements in the other crew station are electrically deactivated.

The "Take-over" control panel will be located at the forward portion of each right console in Concept "C", readily visible and accessible to the crew member. The orientation of the switches is such that there is inherent indication of which crew station has command of a system, i.e., switches in forward or aftward positions are correlated to forward of aft crew station control. For example, if the aft crew member actuates a "take-over" switch the corresponding switch in the forward crew station will automatically trip to the aftward position, and vice versa.

In the event of malfunction of the control transfer system (relay failure, etc.), all functions will "fail-safe" to the aft crew station for control by the "Safety Pilot."

d. Lighting Controls

For operation of the aircraft at night or other conditions of low ambient light, interior (crew compartment) and exterior lighting systems are installed and will be controlled by switches and/or rheostats in each crew station.

For control of interior (crew compartment) lighting, five rheostats, one toggle switch and one push button switch will be provided, grouped on a panel located on the right console in Concept "C", and on the outboard consoles in Concept "J". Separate rheostats will be used for flight instrument lighting, for non-flight instrument lighting, consoles lighting, general flood lighting, and for high intensity (thunderstorm) lights. The toggle switch will control two illumination levels for signal lights: bright for daytime or dim for night. The push button switch will be used to test the lamps in all signal lights during preflight check.

For exterior navigational or position lights, two rheostats and one toggle switch will be provided, grouped on a panel located on the right console in Concept "C", and on the center console in Concept "J". One rheostat will be used for fuselage lights and the other for wing and tail lights. The toggle switch will control flashing or steady illumination.

Control of landing or taxi light illumination will be accomplished by a toggle switch located in proximity to the landing gear control.

6. Communications and Navigation

a. Communication-Navigation Control Panel

The communications-navigation control panel selected for this application provides all necessary VHF communication and navigation control functions as well as self test functions on a single functionally divided control panel. Specifically it controls the 618M-1 VHF transceiver, 860E-2 DME transceiver, 51RV-1 ILS and VOR receiver and powers the marker beacon receiver. The panel will be located on the left console, behind the engine power controls in Concept "C", and centrally located on the instrument panel in Concept "J".

Operation of the 618M-1 transceiver is provided by the power, frequency selector and volume controls located on the left hand (outboard side) of the control panel. Actual transmission is of course activated only on depressing the microphone button. A transceiver test position is available on the transceiver power switch which provides an audio tone to the pilots headset. Comm. frequency can be varied in 50 KC increments by means of the two knobs on the left side of the panel. A remote Comm. frequency readout is also provided for ready pilot reference, on the main instrument panel in Concept "C" only.

For operation of ILS or VOR the Nav. system power switch must be indexed to the Nav. position. If ILS operation is desired a localizer frequency must be chosen to provide the necessary switching logic between VOR and ILS operation, thus circumventing the need for separate switches for these functions. ILS or VOR operation must be coordinated with the Instrument Selector switch position to assure the proper RADI and HSI displays as applicable to the

function selected. A self test switch for determining the proper operation of all three receiving capabilities associated with the 51RV-1 receiver (VOR, Loc. and Glide Slope) is provided by means of a Nav. test switch located on the panel face. As these test problems must be monitored on the ADI or HSI, the instrument mode selector switch must be properly indexed.

DME operation requires that the Nav. system power switch be indexed to DME or Override. A standby position, in which power is being applied but the transceiver is not operating, is also available. In the DME position the transceiver is locked on to the selected ground station frequency and provides a range read-out on the HSI. Selecting the override position (OVRD) reduces the range of operation to short range DME stations only. DME frequency selection utilizes the same controls previously specified for VOR-ILS operation. A separate volume control is provided on the lower right hand ganged panel knob for any audio signals received during the selected mode of operation. It should be noted that all frequency tuning for Comm. and Nav. operation employs a 2 out of 5 binary method to minimize panel complexity and associated wiring.

b. Audio Control Panel

The audio control panel is basically a modification of a Collins 387C4 panel and solid state audio amplifier mounted on a common chassis designed for console installation. This unit provides both a centralized volume control and selection controls for all audio functions on a subsystem basis. The intercom position provides an audio link to the ground crew only, as hot microphone provisions will be incorporated between both crew stations. The volume control on this panel does not bypass the separate volume controls on the Comm-Nav panel but rather provides a means of altering their effect without changing their basic settings. One exception to this controlling action is the marker beacon function for which a Hi-Lo switch is provided. The panel will be located on the left console, behind the engine power controls in Concept "C", and centrally located on the instrument panel in Concept "J".

c. Air Traffic Control (ATC) Panel

The Air Traffic Control (ATC) Panel controls the operation of its remotely located transponder by providing power, sensitivity and code selection, mode control and self test functions. An altitude reporting switch is also contained on this panel but,since only two ground stations are currently available in the U.S. for this function, the necessary altitude digitizer has not been considered for installation at this time. It should be noted however, that the transponder chosen is compatible with this function should its inclusion be desired at a later date. An identification light built into a depressible button is also provided on the panel face to provide a visual indication to the pilot of transponder operation. Depressing this button, on ground control request, will activate the transponder after the framing time for approximately 14 seconds. Placing the system power switch in the test position initiates a self test problem which checks both receiver and transmitter operation and causes the adjacent test lamp to illuminate for a 10 second interval. An aural tone is also available through the pilots ear phones indicating transponder operation. Two code dials are available for selecting the right and left hand pair of code digits, as desired, which can be read out in the code window centered on the panel face. The panel will be located on the left console behind the engine power controls in Concept "C", and centrally located on the instrument panel in Concept "J".

d. Remote Communication Frequency Indicator

The remote Communication Frequency Indicator provides a digital frequency read out of the Communication Frequency selected by the pilot on the left hand Communication-Navigation control panel in Concept "C" only. By locating this repeater type indicator on the main instrument panel a potential flight safety hazard is eliminated since the display in this location avoids the necessity for diverting the pilots attention from the normal field of view.

e. Microphone Control

Control of radio transmission ("MIC" button) is accomplished by depressing the "thumb" switch at the left side of the control stick grip.

7. Environment

a. Environmental Controls

Control of the cabin air condition system will be accomplished by a rotary switch, a lever-lock toggle switch and a potentiometer. The rotary switch will provide for activating the system and selection of air source (either from the heat exchanger unit or ram air). The potentiometer will govern the temperature level. The toggle switch will permit selection of manual or automatic thermostatic control modes. A switch will be provided for application of electrical power to heating elements in the pitot-static tube to prevent icing. A switch will be provided to control the wind shield and canopy defogging system. A switch will be provided to control the rain removal system. A lever-lock-type switch will be provided for override of the electronic compartment cooling system. These controls will be grouped on a panel located on the right console in Concept "C", and on the center console in Concept "J".

A more comprehensive description of the environmental control system is provided in Section IV.

b. Oxygen Control

A toggle-actuated shut-off valve will be provided, located on the forward portion of the right console in Concept "C", and on the outboard segments of the instrument panel in Concept "J", to control the flow of oxygen from the system to the breathing regulator which is an integral part of the flight helmet. An oxygen supply pressure indicator will be located adjacent to the oxygen shut-off valve.

A more comprehensive description of the oxygen system is provided in Paragraph C.2.

c. Canopy Controls

The canopies will be raised and lowered by electro-mechanical actuators, controlled by a switch located on the left side wall above the left console in each crew station for Concept "C", and on the outboard segments of the instrument panel for Concept "J". Locking devices for the canopies, when closed, will

actuate by a mechanical control located in proximity to the actuating switch. A warning signal light will be provided in each crew station to indicate when either canopy is unlocked.

8. Warning and Caution Signalling

a. Warning Signals

Warning signal lights will be provided to indicate unsafe conditions warranting immediate corrective action. The warning light fixtures each contain two lamps for reliability and the inscribed legent (message) will appear as opaque letters on an illuminated "red" background. The following legend type warning lights will be located on the main instrument panel:

BAIL OUT (For Concept "C" only)
CANOPY UNLOCKED

For Concept "C" only, a bail out warning light will indicate necessity for abandoning the aircraft (ejection) in the event of dire emergency. The light is illuminated when the "Safety" pilot actuates a guarded toggle switch located in the aft crew station only, if voice communication (intercomm) is not possible.

The handles of the three fire extinguishing controls will illuminate red in color for "Fire" warning. The knob of the landing gear control will illuminate red in color for "Gear Unsafe" warning.

An audio "beeper" type signal will be introduced into the pilot's headset if landing attempt is made with "wheels up".

b. Caution Signals

Caution signal lights will be provided to indicate conditions of an impending dangerous nature. The caution signal light fixtures each contain two lamps for reliability and the inscribed legend (message) will appear as illuminated "yellow" letters on an opaque background. A master caution signalling system will be employed, consisting of a Caution Annunciator Panel and a separate master caution signal light.

The Caution Annunciator Panel will contain provisions for use of up-to-24 caution lights for growth potential whenever a fault is sensed within one of the key subsystems, the master caution light, which is in the pilots' primary cone of vision, will illuminate to call attention to the specific fault signal light on the Caution Annunciator Panel. A push button switch is provided at each crew station, conveniently located to permit resetting of the master caution light; however the message on the Caution Annunciator Panel will remain illuminated until the fault is corrected.

The Caution Annunciator Panel will be located on the right segment of the instrument panel in each crew station for Concept "C", and centrally located on the instrument panel for Concept "J". It will include logic circuitry for the caution signalling system. The following legend-type caution lights will be located on the Caution Annunciator Panel:

L. ENG FUEL PRESS LOW
R. ENG FUEL PRESS LOW
LIFT FUEL PRESS LOW
FWD TANK BOOST PUMP
AFT TANK BOOST PUMP
FUEL LEVEL LOW
LIFT ENG EGT
LIFTING ENG RPM
L. GEN FAIL
R. GEN FAIL
L. TRANS-RECT FAIL
R. TRANS-RECT FAIL
ELEC. COMPT O'HEAT
PITCH STAB AUG
ROLL STAB AUG
YAW STAB AUG
PITCH MECH ADV
ROLL MECH ADV
YAW LIMITER

9. Seat and Escape

a. Normal Seat Operation Controls

Adjustment of the seat to maintain proper eye level for crew members of different stature is accomplished by a switch, located at the forward right side of the seat bucket, which controls power to the actuator. A control at the forward left side of the seat is provided for locking and unlocking the shoulder harness inertia reel.

b. Ejection Controls

Ejection of the seat is accomplished by pulling a "D" ring centrally located at the forward edge of the seat, or by raising either or both leg braces and then squeezing either or both exposed triggers. The canopy will be jettisonned automatically, followed by ejection of the seat.

A separate control, located at the aft end of the left console in each crew station for Concept "C", and at the aft end of the center console for Concept "J", is provided to permit jettisonning of the canopy without arming the seat ejection system.

A more comprehensive description of the escape system is provided in Paragraph C.3.

C. CREW ACCOMMODATIONS

1. Air Conditioning System

An air conditioning system will be installed with a heating capacity sufficient to maintain 75°F in the crew compartment at 0°F O.A.T. at pressure altitude up to 25,000 feet. The system will have a cooling capacity sufficient to maintain 85°F in the crew compartment at an O.A.T. of 110°F when one lift-cruise engine is at 48% maximum thrust (86% rpm) or higher. Compressor bleed air from the lift-cruise engines will be used for the air conditioning package. Compressed air from an MA-1A start cart may be used on the ground when the engines are shut-down. Crew compartment air temperature will be automatically controlled. Ram air may be used for cabin ventilation, if desired. The crew compartment will not be pressurized.

A more comprehensive description of the air condition system is provided in Section IV.

2. Breathing Oxygen System

An oxygen system having a high pressure gaseous supply will be installed to provide supplemental breathing oxygen to each crew member. Two high-pressure cylinders of 205 cubic inch capacity each will be employed for storage. These cylinders, when charged to a nominal pressure of 1800 psi, will provide a total of 30.4 cubic feet of free oxygen. Assuming a consumption rate of 27.8 cubic feet of oxygen/hour/man for operation of the vehicle at or near sea level during conduct of the research flight tests, the system will provide 100% oxygen (without dilution) to each crew member for a duration of 34 minutes.

The high pressure supply will be reduced to 70 psi by use of a suitable pressure reducing valve prior to distribution to each crew member's breathing oxygen regulator. A shut-off valve will be provided in the right console of each crew station in Concept "C", and on the outboard segments of the instrument panel in Concept "J", to control flow of oxygen from the source of supply. A pressure gage will be provided adjacent to the shut-off valve at each crew station to indicate the remaining oxygen supply permitting the crew members to

ascertain oxygen duration available, usually accomplished by reference to printed oxygen duration charts. (Such charts which indicate time remaining for various supply pressures at various cabin altitudes in tabular form are normally contained in the vehicular Flight Manual; i.e., the Pilot's Operating Handbook.)

With the shut-off valve in its "OPEN" position, the oxygen will be directed through a survival kit in the seat bucket to the crew member's flight helmet. The flight helmet, designated as Air Force Type HGU-15/P and/or Navy Type AOH-1, contains an integrally mounted demand-type oxygen regulator plus a valve that automatically opens to introduce oxygen into the helmet when the visor is down.

The survival kit is equipped with automatic disconnects to permit separation of the oxygen line from the seat in the event of ejection and to permit separation of the oxygen line from the crew member upon release of the survival kit after such ejection. The survival kit will also contain an emergency oxygen supply for limited breathing duration, either in flight (should there be a malfunction of the main oxygen system) or after ejection (during parachute descent).

The gaseous oxygen cylinders will be provided with self-opening valves permitting direct replacement with fully charged cylinders when supply is depleted; also, a separate filler valve will be provided, accessible from an access door on the exterior of the fuselage, to permit charging the cylinders to full capacity while installed within the airframe.

A schematic diagram of the oxygen system, applicable to Concept "C" and/or Concept "J", is depicted in Figure X-6.

3. Escape System

An anlysis of five escape systems considered representative of current USAF and Navy escape system capability resulted in the selection of the proposed Republic seat configuration. The selection was based on:

- Best current safe escape record in USAF and Navy inventory
- Least modification and re-design and qualification to provide the required capability for the NASA V/STOL aircraft.

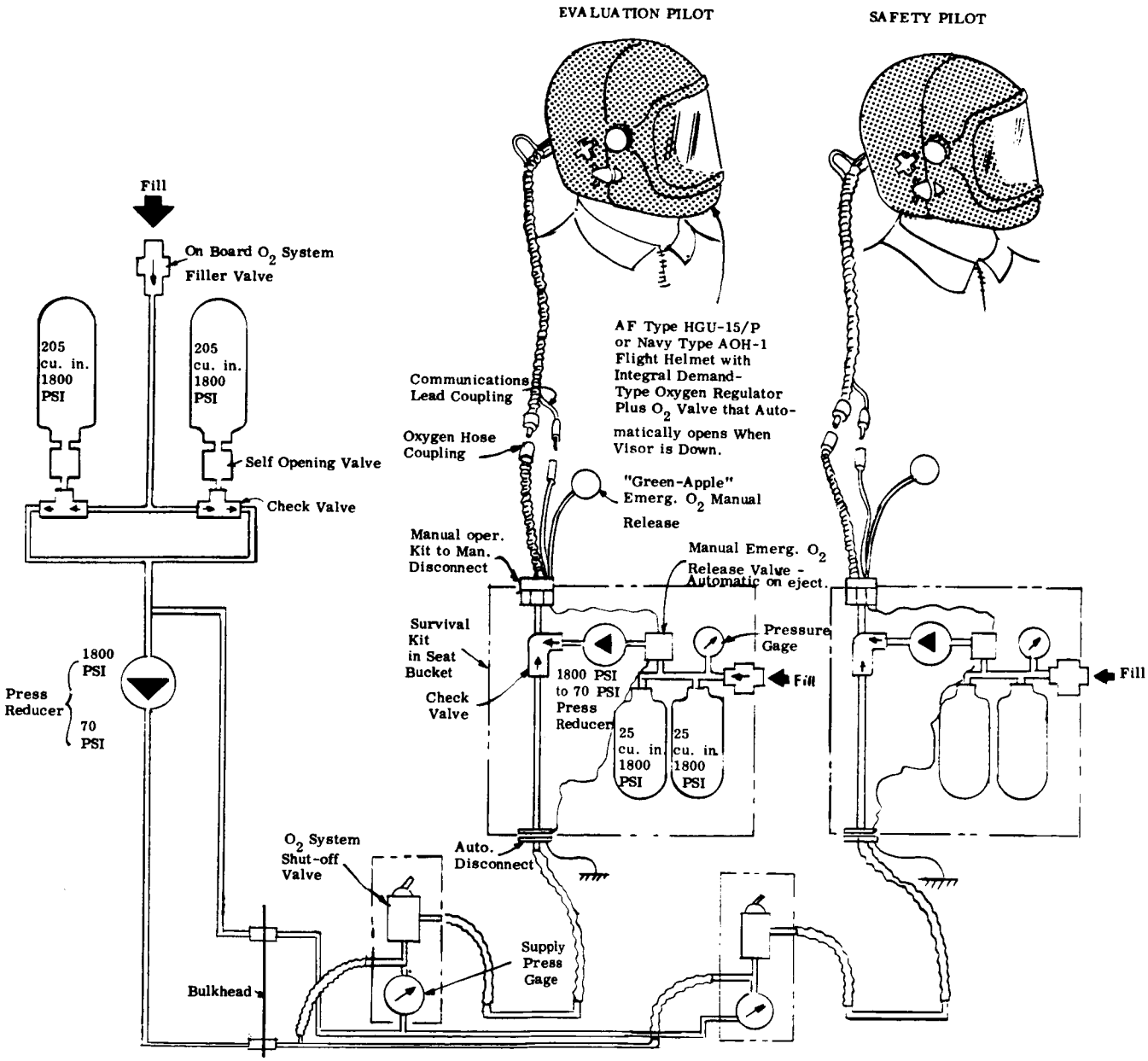


Figure X-6. Oxygen System Schematic Diagram

Some proposed systems appear to have promising characteristics such as recovery capability over 360° aircraft attitude with minimum 200 feet altitude at ejection under inverted condition; however, insufficient validation of their performance precludes their evaluation for the present application.

The proposed Republic escape system has the basic requirement capability of zero-zero performance, high-impulse rocket performance, forced personal parachute deployment, positive and sustained seat and man clearance subsequent to seat/man separation, and automatic deployment of survival kit during the ejection cycle.

A significant factor in the selection of the proposed system is that the configuration is scheduled for testing in July 1967 at Holloman AFB. This testing will serve to confirm the design for the NASA V/STOL program.

Comparison of various ejectable seats is provided in Table X-1 for information purposes.

The crew escape system installation for Concept "C" depicted in Figure X-7 employs a sequenced canopy-seat arrangement for both the front and aft crew stations. Initiation of the ejection cycle by the safety pilot automatically ejects the front seat before aft seat ejection. Initiation of the ejection cycle by the evaluation pilot ejects only the front seat.

The ejection cycle can be initiated in two ways:

- Pulling on the D-ring at the front of the seat
- Raising either or both leg braces and squeezing either or both exposed triggers

The crew escape system shown for Concept "C" features automatic canopy jettison and seat ejection: front cockpit initiation causes jettison of the front canopy and ejection of the front seat; rear cockpit initiation causes (in sequence) front canopy jettison, front seat ejection, rear canopy jettison, rear seat ejection. In addition, it contains automatic torso positioning and restraint, a contoured-lid survival kit with automatic kit deployment and inflation of life raft, a contoured automatic lap belt, a high-impulse rocket motor with integral seat/motor adjustment for minimum C.G. excursion, a forced seat/man ballistic

TABLE X-1

ESCAPE SEAT COMPARISON

	NAA LW-20	Martin Baker MK-9	Douglas ESCAPAC I-C	Douglas ESCAPAC II	Webber A/C "Zero-Zero" Seat	Republic RAD 10500	Republic RAD 20500
Escape Capability	Zero alt: 0-500 KEAS 50' alt: 0-500-650 KEAS	Zero alt: 0-550 KEAS 0-700 KIAS	Zero alt: 0-650 KEAS	Zero alt: 0-650 KEAS	Retro alt. 0-600 KEAS	Zero alt: 0-650 KEAS	Zero alt: 0-650 KEAS
Max. Accel.	9-12G	14G	15G	16G	15.25G	15G	15G
Rate of Rise	150-180G/Sec	180G/Sec	220G/Sec	250G/Sec	175G/Sec	250G/Sec max.	250G/Sec max
Time Event Seat A/C Sep. Pers Chute Infl.	With canopy jet: 0.56 sec 1.8-2.5 sec	With canopy jet: 0.5 sec 4.0 sec	With canopy jet: 0.55 sec 5.5 sec	With canopy jet: 1.9-2.4 sec	With canopy jet: 0.55 sec 5.5 sec	With canopy jet: 0.50 sec 3.30 sec	With canopy jet: 0.50 sec 1.8-2.8 sec
Alt of Trajectory	Zero speed: 125 ft 600 KEAS: 70 ft	Zero speed: 210 ft	Zero speed: 300 ft	Zero speed: 180 ft	Zero speed: 250 ft	Zero speed: 350 ft 650 KEAS: 85 ft	Zero speed: 400 ft 650 KEAS: 85 ft
Rate Recovery Cap. at Zero Alt	5100'/min at zero speed	3000'/min at 150 KIAS	3000'/min at 150 KIAS	5200'/min at zero speed	--	6000'/min at zero speed	6000'/min at zero speed
Dive Angle Recov. Cap. at Zero Alt	Up to 60°	67° nose down 38° nose up	--	--	--	Up to 67°	Up to 67°
Roll Angle Recov.	Up to 60°	Up to 48°	Up to 45°	--	--	Up to 45°	Up to 45°
System Initiation	1 seat "D" ring in front of seat-bucket	1 seat "D" ring in front of seat-bucket	Face curtain & seat "D" ring	Seat "D" ring	Face curtain & Seat "D" ring	Leg braces at side of sear	Seat "D" ring & leg brace
Crewman Restraint	Inertia reel, shoulder restraint	Leg, arm, head & shldr restraint	Inertia reel (shldr restraint can be added)	Power retract inertia reel	Inertia reel; shoulder restraint	Inertia reel, shldr restraint, leg braces *foot stirrups	Ballistic inertia reel, shoulder restraint, leg braces
Stabilization Devices	52" drogue chute	5' drouge chute	Dart system	STAPAC	--	Pitch stabilator plates	Pitch stabilator plates
Seat-Man Sep.	Pers. parachute	Pers parachute	Two inflatable bladders	Two inflatable bladders	Ballistic rotary actuator	Ballistic reel type plus seat retardation chute	Ballistic reel type plus seat retardation chute
Personal Parachute	28' canopy; deployed by a ballistic gun or a drogue chute	24' canopy; deployed by a ballistic gun	28' canopy; deployed by a ballistic gun	28' canopy; deployed by a ballistic gun	26' canopy; deployed by a ballistic gun	26' canopy; deployed by a drogue gun	26' canopy; deployed by a drogue gun
Survival Equip.	Survival Kit	Survival Kit	Survival Kit	Survival Kit	Survival Kit	Survival Kit	Survival Kit
Weight (lbs) Seat Personal Chute Survival Kit	105.7 22.0 31.0 158.7	119.5 22.0 28.5 170.0	107.2 25.0 24.0 156.2	*116.0 25.0 24.0 *165.0 *EST	127.5 22.0 25.0 174.5	101.0 22.0 41.0** 164.0 ** Packed for nominal mission req't over water	104.5 22.0 41.0 167.5

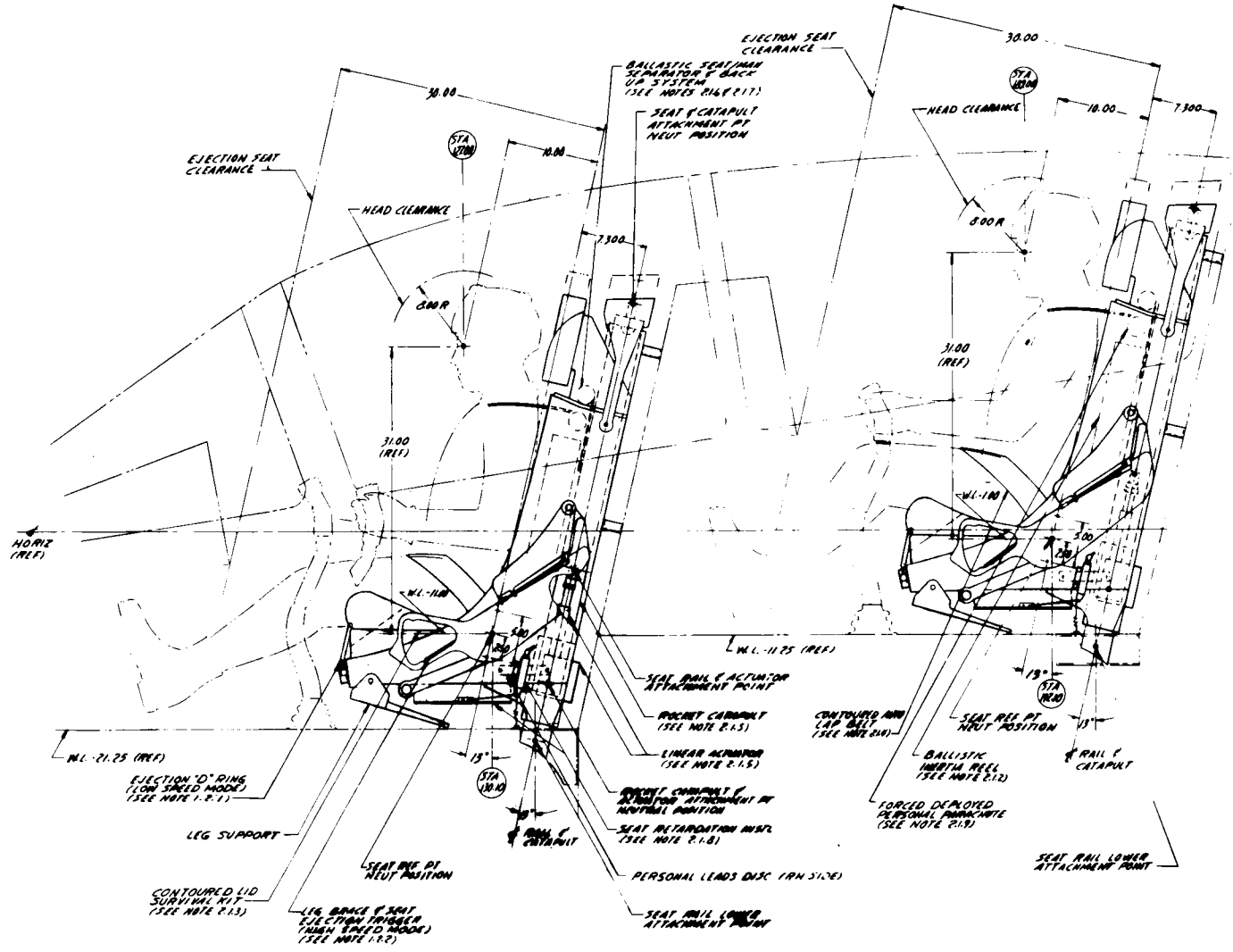


Figure X-7. Escape System - Concept "C"

separator, a seat retardation system to prevent seat/man/chute involvement, and a forced-deployment personal parachute system initiated one second after lap belt release.

The crew escape system installation for Concept "J" depicted in Figure X-8 employs a sequenced canopy-seat arrangement for left and right crew stations. Initiation of the ejection cycle by either the evaluation pilot (L) or the safety pilot (R) automatically ejects both crewmen simultaneously. An automatic sequence first jettisons both canopies, then ejects both seats. The cycle is initiated by pulling on the D-ring at the front of either seat.

The crew escape system shown for Concept "J" features automatic simultaneous canopy jettison and seat ejection for both crew stations initiated by either crewman. In addition, it contains automatic torso positioning and restraint, a contoured-lid survival kit with automatic kit deployment and inflation of life raft, a contoured automatic seat belt, a high-impulse rocket motor with integral seat/motor adjustment for minimum C.G. excursion, a forced seat/man ballistic separator, a backup system for the conventional seat/man separator, a seat retardation system to prevent seat/man/chute involvement, and a forced-deployment personal parachute system initiated one second after lap belt release.

A significant part of the crew escape system shown for Concepts "C" and/or Concept "J" is scheduled for qualification testing in July 1967 at Holloman AFB, New Mexico. It is part of ECP action requested of Republic Aviation by USAF/SMAMA for optimization of the F-105 escape system to realize zero-zero escape capability.

4. Relief Provisions

Each crew station will be provided with a device for relief purposes. The pilot relief provisions will consist of a cylindrical container of sufficient volume to receive and retain 500 cc of liquid. The top cover of the container will be equipped with a conical-horn-type receiver to funnel the liquid into the container. A simple valve, controlled by a lever at the outside of the receiver, is included to prevent spillage and emission of obnoxious odors.

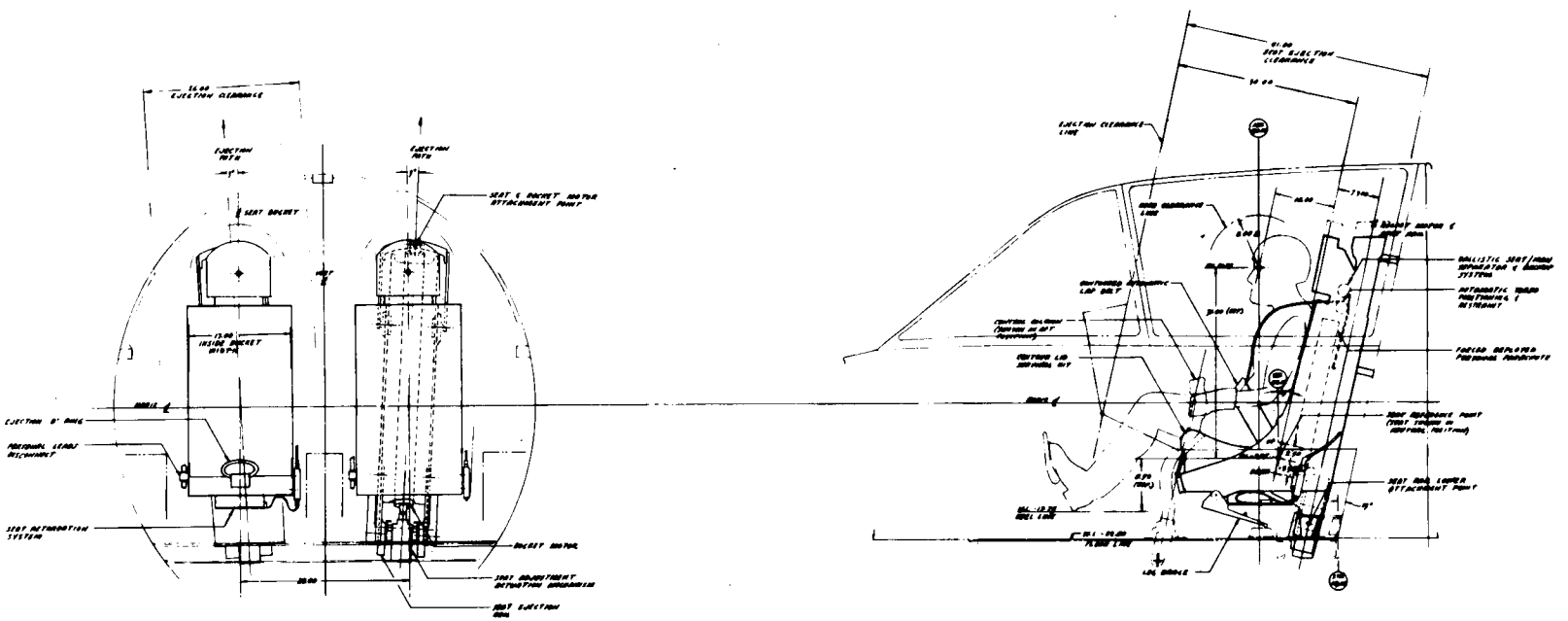


Figure X-8. Escape System - Concept "J"

The relief container will be installed at the aft end of the left console for Concept "C", and the aft end of the outboard consoles for Concept "J", in a depressed well having a bayonet-type lock for retention purposes.

5. Stowage Provisions

Each crew station will be provided with a case of sufficient size for stowage of Pilot's Operating Handbook, Radio Facility Charts, Maps, and other appropriate flight data documents. For Concept "C", the data case will be installed at the aft end of the right console in each crew station. For Concept "J", only one data case will be provided, located at the aft end of the outboard console in the right crew station. The data case will be of such configuration as to enable insertion and removal of stored material with ease.

Compartments for stowage of spare lamps and spare electrical fuses are included in the design of the pedestal (below main instrument panel) at each crew station.

6. Blind Flight Simulation Provisions

It must be acknowledged that any head-type device for obscuring the "Evaluation Pilots" external vision to simulate "blind flying" (IFR) conditions would also impair the "Safety Pilots" external field of view creating a hazard to flight safety.

Cursory investigative effect for a means to safely simulate "blind-flying" conditions during conduct of research flight tests has been conducted, but not to an extent that would permit evaluation of several possible techniques from an engineering and economic view point.

There is one practical method to satisfy this requirement, i.e., by obscuring the forward or left crew member's external vision by a light-liltering device immediately in front of his eyes, in conjunction with tinted glass in the windshield and canopy. The visor of the forward or left crew members flight helmet can be fabricated in the proper color to permit visual reference to controls and displays within the crew compartment but preclude external vision through the windshield and canopy. (It is to be noted that except for a slight

change in real world coloration, the aft or right crew member's external vision is otherwise unaffected). If sophistication is desired, the flight helmet can be modified to have the visor raised and lowered by electro-mechanical means, in which case the visor can be spring-loaded in a raised position and solenoid-held in a lower position in front of the eyes. The solenoid can be energized by power applied through actuation of a switch in the aft or right crew station. Simulation of total IFR conditions or even momentary "blind" conditions, such as passing through cloud banks, can thus be controlled by the Safety Pilot without interrupting the Evaluation Pilot's task.

It is felt that additional study effort is required before attempting recommendation of "blind-flying" provisions for this vehicle.

7. Noise Protection

Structural design techniques and use of acoustic insulation at the inner walls of the crew compartment will serve to attenuate noise, generated by operation of the engines to levels of human tolerance. To further attenuate noise to a more comfortable level permitting more intelligible voice communications, the AF type HGU-15/P or Navy Type AOH-1 flight helmets are recommended for use by the pilots.

The Navy has conducted extensive service evaluation of these helmets, the results of which indicate enthusiastic acceptance by pilots for several reasons; the most noteworthy reason having direct application to the NASA V/STOL test aircraft is its effectiveness in attenuation of ambient noise!

8. Cabin Lighting

Crew compartment lighting is provided for operation of the aircraft at night or under conditions of low ambient light. "White" lighting will be used based upon current Air Force requirements for military aircraft.

The instruments are integrally lighted (i.e., lamps within the instrument case) which negates the need for separate lighting fixtures at each indication. Trans-illuminated plastic lighting plates having imbedded "grain-of-wheat" type lamps and wiring will be used for control panels and other instructional placards

containing information used during night flights. The Variable Stability System (VSS) control panel will not be provided with a trans-illuminated placard because applicable markings at the function select switches will differ depending on the programmed test flight. Separate lighting fixtures will be strategically located to provide flood-lighting illumination at areas which cannot otherwise be lighted.

The light levels of adjacent instruments and control panels, while being within tolerances prescribed by applicable specifications, differ greatly. To overcome this condition, a panel containing a grouping of up-to-24 potentiometers (one each for individual instruments or control panels) will be provided for "trimming" brightness levels. Once adjusted to obtain a balance of illumination levels throughout each crew station, the brightness may then be varied using the applicable rheostats on the interior lighting control panels. For Concept "C", the interior light level balancing panels will be located at the aft end of the right console in each crew station. For Concept "J", these panels will be located on a semi-vertical extension of the center console at the aft bulkhead, between the left and right crew stations.

A moveable utility light, normally stowed at the aft end of the right console in Concept "C", and on the outboard consoles in Concept "J", is provided for use as warranted.

SECTION XI

QUALITY ASSURANCE AND RELIABILITY

A. QUALITY ASSURANCE PHILOSOPHY

1. Organization

The proposed program envisions the fabrication of two test aircraft or modification of existing aircraft. The program is to be conducted on an austere basis, utilizing a limited experimental shop approach for fabrication but with approved standard engineering release procedures.

Operationally, the Contractor's Quality Assurance Manager assigned to this program would receive direction from the NASA V/STOL Program Manager, but with Quality Assurance policy direction from the Contractor's Quality Assurance and Reliability Manager.

As much as feasible, all functions, activities and operations are to be conducted in a single integrated area, permitting maximum on-the-spot liaison between responsible personnel of all the appropriate disciplines. Overall control of quality is enhanced utilizing this mode of operation as each function tends to monitor the others when working in this close relationship.

The Quality Assurance personnel and the scope of their activities will be kept to a minimum commensurate with the stated approach, but a level of effort will be maintained sufficient to guarantee that the Customer's interest in quality matters will be well represented and that the end products will conform completely with his requirements. The Contractor will have in existence a Quality Assurance and Reliability organization with extensive experience in all phases of aero space activity.

2. Procurement

The assurance of supplier capability before and during the tenure of contract is basic to the quality assurance system. Before subcontracts are

issued, each potential supplier must be evaluated to determine the ability of his personnel, equipment, and procedures to perform adequately the quality assurance function for the parts, assemblies, or tools to be manufactured by him. In addition, after contract award, the supplier's operations must be surveyed on periodic visits to check procedures, to spot-check product quality by actual inspections, and to initiate any necessary corrective action. In selecting sources, consideration will be given to the nature of the supplies to be furnished, quality evidence furnished by the supplier, and the quality history of products previously supplied by him.

Subcontracts issued for AVE supplies and supporting data will be reviewed prior to reliease to ensure that all applicable specifications, plus any specific quality assurance requirements and to specifications. The level of inspection performed will be adjusted to suit the article involved and the quality history of previous shipments of the same article by the same supplier. Statistical sampling techniques, unless contractually prohibited, may be used.

To obtain a current history of each supplier's quality and to provide a basis for taking corrective action at the proper point when required, a record will be maintained for each part, component, subassembly, or shipment of raw materials from each vendor or subcontractor. This record will indicate the number of lots received, the quantities of each item rejected or accepted at receiving inspection, the sampling plan used (if any) and, for completeness, the number of rejections of the item attributed to each supplier.

B. RELIABILITY INVESTIGATIONS

As the conceptual V/STOL aircraft is required as a research tool for the sole purpose of analyzing aerodynamic and operating characteristics it is indicated that the aircraft requires an adequate but not rigorous reliability effort. Achievement of a sufficiently high aircraft reliability is assured, and inclusion of design parameters emphasizing safety, life, and effectiveness characteristics has been considered.

1. Reliability Block Diagram and Mathematical Model

The V/STOL aircraft block diagram shown in Figure XI-1 is based upon the assumption that all subsystems are needed for a successful mission. All components were placed in series because the mission will be considered a failure if any subsystem fails in whole or part.

Development of the mathematical model was based upon the block diagram. The diagram and model are similar for both Concepts C and J. As a series configuration was selected for all parts, the considerable redundancy designed into the system for safety purposes acts to lower the aircraft's reliability. This is so because the probability of any single failure, with consequent mission abort, is increased.

The mathematical model is:

$$R = \prod_{i=1}^{n} e^{-\lambda_i t_i} = R_1 \times R_2 \times R_3 \times \ldots\ldots R_n$$

R = probability of not having any failure during the mission,

λ_i = failure rate of a subsystem,

t_i = operating time of a subsystem,

n = number of subsystems in the aircraft.

2. Prediction

The reliability prediction is presented in Table XI-1. Failure rates are based upon manufacturer's information, FARADA, and extrapolation of data from existing aircraft such as the F-105. Results indicate that an aircraft system reliability of .97 for Concept C can be achieved through utilization of presently attainable components and techniques. This value can be reached without performing special reliability tests or undergoing developmental programs which produce advancements in the state-of-the-art.

TABLE XI-1

RELIABILITY PREDICTION

Subsystem	Time (Hr.)	Failure Rate, λ (10^{-6} Failures/Hr.)	Reliability
1. Airframe	0.283	833.33	0.99976389
2. Landing Gears	0.150	31,643.20	0.99525352
3. Lift Engines (6 Engines)	0.200	1,500.00	0.99820000
4. Lift/Cruise Engines (2 Engines)	0.283	1,100.00	0.99937666
5. Diverter Value (2 Valves)	0.283	250.00	0.99985834
6. Vectoring Nozzle (8 Nozzles)	0.200	181.82	0.99970912
7. Fuel	0.283	2,061.86	0.99941581
8. Throttle Controls	0.283	857.60	0.99975701
9. Escape System (2 Seats)	0.333	1,183.10	0.99921126
10. Left Hydraulic System	0.283	7,954.18	0.99776320
11. Right Hydraulic System	0.283	7,954.18	0.99776320
12. Left Engine Alternator	0.283	5,072.65	0.99856275
13. Right Engine Alternator	0.283	5,072.65	0.99856275
14. Mechanical Flight Control	0.283	3,676.20	0.99895841
15. S.A.S. and Fly-by-Wire	0.283	1,543.70	0.99956262
16. V.S.S.	0.283	5,000.00	0.99858334
17. Environmental System	0.283	6,530.80	0.99814961
18. Navigation	0.283	1,389.70	0.99960625
19. Communication	0.283	2,222.30	0.99937035
20. Cockpit Instruments (2 Pilots)	0.283	9,000.00	0.99490000
	TOTAL RELIABILITY =		0.97229433

It is felt that a mission accomplishment of 97 times out of 100 attempts is sufficient for an experimental aircraft of this nature where failure to complete the scheduled phase has minimal military impact. However, should a higher success ratio be desired by the NASA, a reliability apportionment among the aircraft components would be performed and, with the application of more stringent requirements, an over-all aircraft reliability of .99 or higher would easily be within reach.

3. Trade-Off Studies

One of the reliability trade-studies performed to aid the V/STOL aircraft design concerned engine operation.

Equations were established to determine the probability of no engine failures, one-or-less engine failures and two-or-less engine failures. These are:

Mission	Unreliability	
	Symbol	Applicable Equation
Probability of exactly no engine failures	U_o	$U_o = 1-R^n$
Probability of 1 or less failures	U_1	$U_1 = 1-(R^n+nR^{n-1}Q)$
Probability of 2 or less failures	U_2	$U_2 = 1-\left[R^n+nR^{n-1}Q+\frac{n(n-1)R^{n-2}Q^2}{2!}\right]$

Utilizing these equations, a family of curves was generated to show the relationship between the number of lift engines and an unsuccessful flight per 100,000 takeoff or landing attempts. The two curves plotted in Figure XI-2 represent the probability of no engine failures, and one-or-less engine failures.

The Concept C design incorporates six lift engines and will function with one engine failure. Thus the number of unsuccessful flights per 100,000 attempts is .09, representing a probability of success of .9999991.

4. Failure Mode and Effect Analysis

A Failure Mode and Effect Analysis (FMEA) was conducted for Concept C. An FMEA is a "how can it fail" type of analysis used to evaluate the effects of failure modes as they affect performance and safety. A unit is assumed to fail in each possible mode and the remainder of the system is examined to determine the over-all effect. Thus, defects of the aircraft design concept are brought forward for consideration during design evaluation.

The mission profile considered consists of a vertical takeoff, transition to conventional flight and back to a vertical landing. The following twelve subsystems were investigated:

1. Airframe	5. Flight Controls	9. Throttle Control
2. Landing Gear	6. Fuel	10. Propulsion
3. Communication	7. Hydraulic	11. Environmental
4. Navigation	8. Electrical	12. Cockpit Instruments

At the level of study undertaken, an FMEA for Concept J would be similar to that for Concept C. Airframe, landing gear, communication, hydraulic, environmental, instruments, navigation and flight controls subsystems are different, thus requiring some FMEA modification. Fuel, electrical, throttle control, and propulsion subsystems are similar.

A summary of the FMEA study resulted in the following quantities of possible failure mode effects which have not yet been resolved through design adjustment. However, no major or unique problem area exists.

Failure Mode Effects	Quantities
Safety	6
Either safety or data accomplishment	12
Data accomplishment	97
Either data accomplishment or nuisance	4
Nuisance	1

The six possible safety failures occurred in two areas:

- Flight control
- Pilots' oxygen supply

The twelve failures of either safety or data accomplishment occurred in six areas:

- Landing gear
- Propulsion
- Hot gas reingestion
- Throttle control
- Hydraulic actuators
- Electrical contactors

The high number of data accomplishment failures stem from the purpose of the aircraft, which is to gather performance data. Thus a completely satisfactory operating mission is required at all times. Any failure which could not allow all desired data to be acquired was therefore classified as a failure of data accomplishment, even though a test run could have been completed with much derived information. In addition, the FMEA was conducted primarily on an assembly level so that the hypothesized failure caused entire subassemblies to be inoperative rather than only individual components.

5. Design Reviews

Following are some of the areas which were evaluated in conjunction with design and various other engineering disciplines.

- Flight control surfaces, mechanical linkages and landing gear. These items incorporate sufficient safety factors since they were designed for a minimum of 300 hours whereas only 100 aircraft flying hours are anticipated.

- Oxygen Supply. The critical need for oxygen exists during vertical takeoff and landing when the cockpit is contaminated with reingested gases. Two separate oxygen supply tanks are being considered rather than one, so that loss of an oxygen supply will not affect both pilots.

- Hydraulic System. Only a leak in the barrier of a dual actuator will cause the loss of both hydraulic systems, and this is an extremely remote occurrence.
- Even with the loss of both systems, many hydraulic control subsystems and parts have mechanical/electrical backup devices which can override the hydraulic actuators.
- Electrical System. To short out the electrical system, a second failure must occur at the same time that the contactor fails in a closed position. This is an extremely remote occurrence.
- Hot gas reingestion. This was one of the main areas affecting engine aircraft design. Engineering has utilized various techniques and designs to prevent the build-up of hot gas fountains at the engine's intakes. These will reduce potential engine degradation or failure.
- Fuel System. The present system, of two independent pumping power sources, hydraulic and electrical, was selected over a system of only electrical powered pumps because of the inherent safety of flight. No single failure point exists which will cause the loss of all fuel pumps.

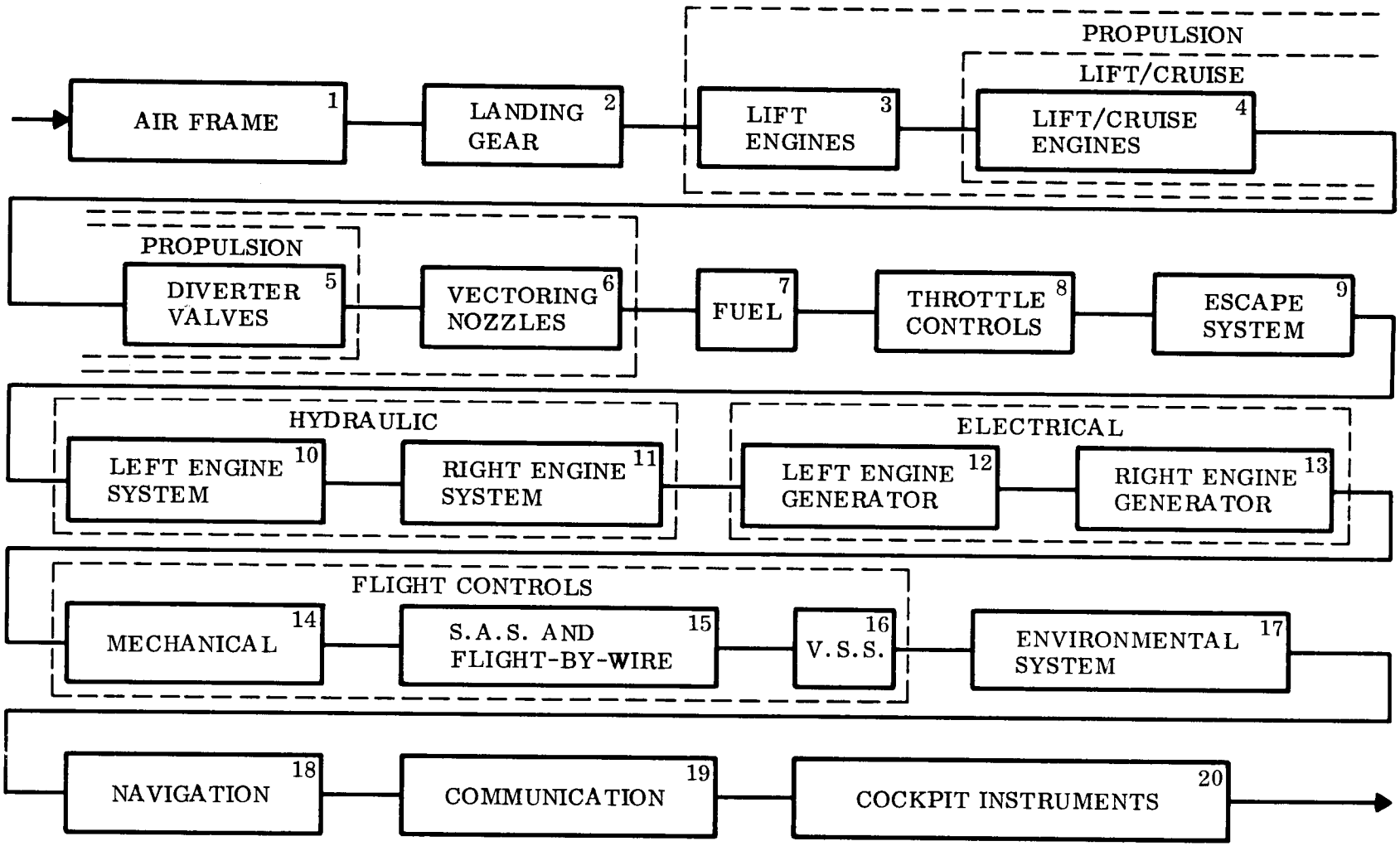


Figure XI-1. Reliability Block Diagram NASA V/STOL

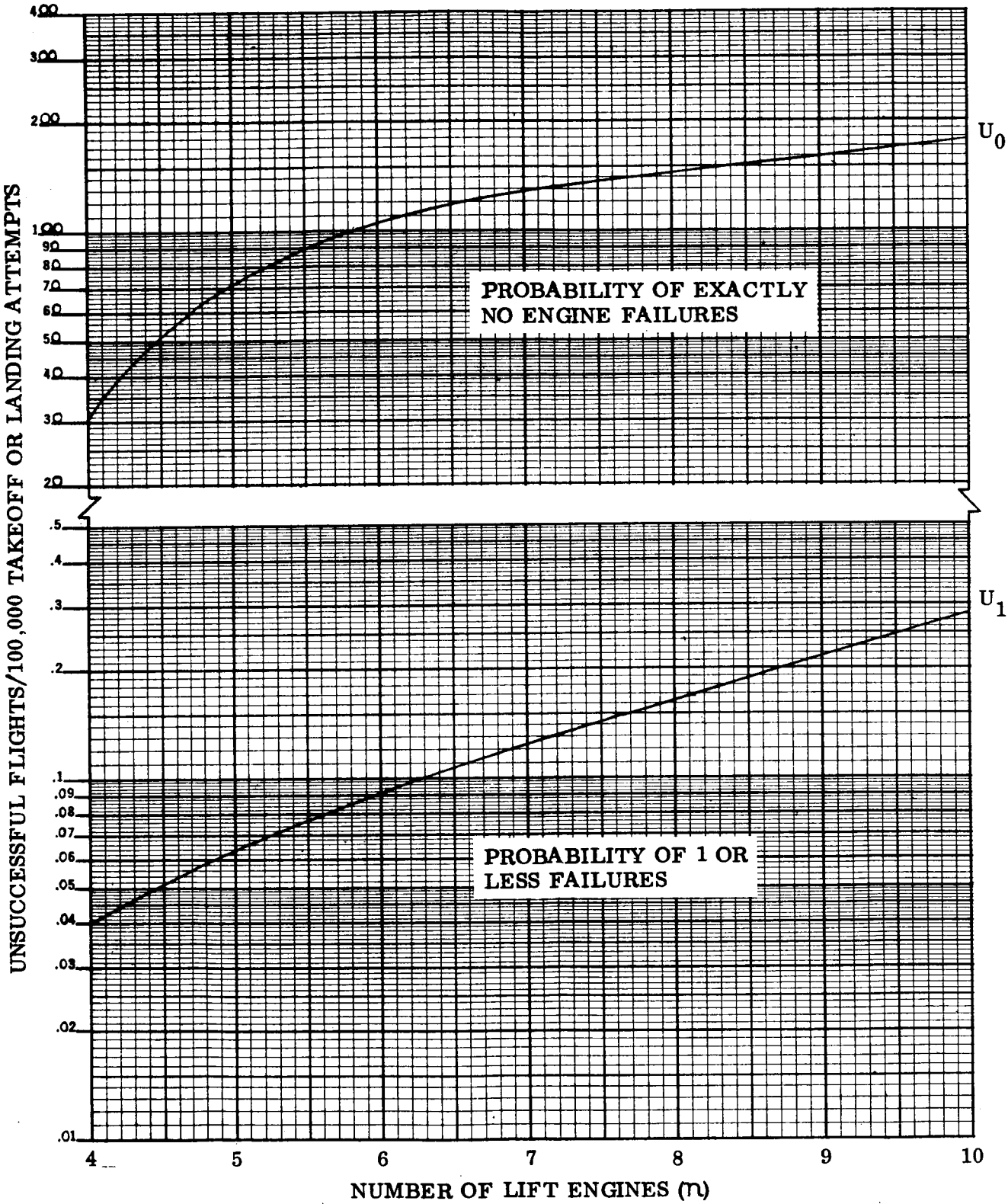


Figure XI-2. Engine Trade-Off Study