NASA CR-66421

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

by

G. Rosenthal
J. Chung, E. Schiller
J. Wright, and E. Vogel

Distribution of this report is provided in the interest of information exchange. Responsibility for the contents resides in the author or organization that prepared it.

Prepared under Contract No. NAS 1-6778 by
FAIRCHILD HILLER CORPORATION
Republic Aviation Division
Farmingdale, New York 11735

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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USE IN BOTH BOOKS
PART I PART II

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."

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SECTION I

SUMMARY

This report provides a preliminary design description of a conceptual new V/STOL Jet Operations Research Airplane. Included also are the supporting analyses and data to demonstrate compliance of this design with the NASA technical requirements. Supplemental data are also included to describe a modified existing aircraft concept (derived from the North American Sabreliner) which meets a lesser set of requirements.

The document was prepared as a final report in compliance with the NASA Statement of Work (L-7151) pertaining to Contract No. NAS 1-6778 "V/STOL Jet Operations Research Aircraft Design Study."

Section II of the report presents the basic description and dimensional data of the new conceptual aircraft (Concept J) which contains two lift-cruise and six lift turbojet engines of the General Electric YJ85-GE-19 type. The modified existing aircraft (Concept J) has two lift-cruise and eight lift engines of the same type. The format is in compliance with that for standard aircraft characteristics presentations.

The estimated conventional and subaerodynamic performance of both aircraft are summarized in the following section. Included in the section are geometry and basic aerodynamic characteristics. This is followed by a summary of propulsion and thermodynamic analytical design data and system design data in Section IV. Included in the latter is also the reaction control system and environmental control system performance.

Final weight and balance estimates are summarized in Section V.

Section VI provides a summary of stability and control in hovering and vertical flight, transition and conventional flight, a survey of the flight dynamics analyses of the stability augmentation system, variable stability system,

and modeling of an operational V/STOL aircraft. Included also are time histories of normal and emergency subaerodynamic flight maneuvers.

The preliminary structural, design and analysis, acoustical, flutter and design criteria investigations performed are summarized in Section VII. This section includes a materials list, structural and component diagrams as well as detailed structural analyses.

A survey of the avionics equipment, antennas, the electrical power distribution system schematic, and loads profile is presented in Section VII.

Section IX contains a description and schematics of the flight controls system and the hydraulic system.

Section X consists of a summary of cockpit arrangement and instrumentation rationale, crew subsystems, and indicates final recommended crew station and escape system layout and corresponding visibility available.

Section XI presents a brief discussion of systems availability during the proposed research program, quality assurance considerations, and reliability calculations for the new aircraft concept.

SECTION II

AIRCRAFT DESCRIPTION AND SPECIFICATION

A. DESCRIPTION

1. Background

The aircraft mission is to gather engineering data on the performance and stability and control characteristics needed by fighter-type jet V/STOL configurations in the hovering and transition speed range. This information will be used to define the levels of control power and automatic stabilization required, the necessary landing aids and pilot displays, the piloting problems, the operating procedures, and the interplay of these factors for the take-off and landing approach of jet V/STOL aircraft in instrument weather.

2. Configuration Approach

In configuring the research aircraft, emphasis is placed on those characteristics appropriate to the low speed mode for supersonic jet V/STOL fighter designs that might be developed in the next few years. It has the mixed propulsion system of a V/STOL fighter design, but does not have cruise engines sized for transonic or supersonic performance. Afterburners are not included.

Provisions are provided for the in-service installations of two additional lift engines. This design enables a study of the in flight operational characteristics of a pure lift engine configuration (one not using the deflected thrust of the cruise engines for any part of the lift). The weight of these extra engines are considered part of the research payload and are not to be installed during other flights. The space provided for these two engine installations is normally used to house the diverter ducts from the cruise engines in the lift mode.

The shape of the lower fuselage is designed to minimize the base loss caused by the low pressures on surfaces adjacent to the exit nozzles. The wings are placed as high as feasible to reduce the horizontal areas subjected to high velocity circulatory air masses during hover and transition. The exit nozzles are

shielded by exit area doors that serve to minimize ground-induced suckdown, jet exhaust mixing, jet-induced lift loss, to reduce temperatures on structural members and tires, and to reduce nose-up pitching moments in transition.

Control during hovering and transition flight, when normal flight control surfaces are ineffective, is obtained from a reaction control system utilizing bleed air from the compressor of the engines in normal operation. The moment arms used by the reaction jets are maximized to obtain the greatest control power.

3. Configuration Comparison

A comparison is presented in Table II-1 of the proposed NASA V/STOL jet operations research aircraft with two studies for a V/STOL fighter aircraft.

The proposed NASA V/STOL aircraft are Concept C, a new airplane design and, Concept J, a modified North American stretched Sabreliner. The V/STOL fighter aircraft are composites obtained from various studies.

TABLE II-1. COMPARISON OF TYPICAL V/STOL FIGHTER GEOMETRY WITH NASA V/STOL

		V/STOL FIGHTER		NASA V/STOL	
		Study 1	Study 2	"C"	"J"
COMPARISON FACTORS					
Wing Span/Fus. Length		0.665	0.86	0.728	0.883
Mass Distribution					
$\sqrt{I_{xx}/Wb^2}$		0.10	0.11	0.10	0.10
$\sqrt{I_{yy}/WL^2}$		0.18	0.20	0.19	0.16
$\sqrt{I_{zz}/WLb}$		0.23	0.24	0.23	0.19
WING ASSEMBLY					
Theo. Area	ft^2	330	430	204	350
Span	ft	44.49	51.0	32.0	45.3
Aspect Ratio	-	6.0	6.0	5.0	5.9

TABLE II-1. COMPARISON OF TYPICAL V/STOL FIGHTER GEOMETRY WITH NASA V/STOL (Cont'd)

		V/STOL FIGHTER		NASA V/STOL	
		Study 1	Study 2	"C"	"J"
Taper Ratio	-	0.40	0.35	0.30	0.32
Sweep	deg				
LE	deg	25	25	25	31.5
TE	deg	10.25	8.25	2	14
0.25c	deg	21.55	21.11	20	28
Dihedral	deg	-2.0	3.0	-4	+3
Incidence	deg	0.0	0.0	0.0	0.0
M.A.C., $\bar{c}$	ft	7.87	8.97	7.0	8.4
HORIZONTAL TAIL					
Theo. Area	ft^2	123.78	117.0	66	81
Span	ft	24.80	23.75	15	18
Aspect Ratio	-	2.82	2.20	3.4	4.0
Taper Ratio	-	0.19	0.35	.43	.30
Sweep c/4	deg	43.5	50.42	25	30
Dihedral	deg	0.0	-	0	0
Incidence	deg	0.0	-	0	0
M.A.C.	ft			4.25	5.0
Tail Arm, ℓ_{HT}/c	-	1.997	2.238	2.05	2.17
Area Ratio, S_{HT}/S_W	-	0.3751	0.2721	.324	.232
Tail Vol_{TH}, $\frac{S_{HT}\,\ell_{HT}}{S_W\,\bar{c}}$	-	0.7890	0.6090	.663	.502
VERTICAL TAIL					
Theo. Area	ft^2	90.42	91.0	48	64
Span	ft	10.96	10.005	9.0	10.6
Aspect Ratio	-	1.19	1.10	1.7	1.75
Taper Ratio	-	0.34	0.25	.33	.23
Sweep c/4	deg	44.50	49.53	29	30
M.A.C.	ft	9.940	9.883	5.75	6.83

TABLE II-1. COMPARISON OF TYPICAL V/STOL FIGHTER GEOMETRY WITH NASA V/STOL (Cont'd)

	V/STOL FIGHTER		NASA V/STOL	
	Study 1	Study 2	"C"	"J"
Tail Arm, ℓ_{VT}/b_w	0.3547	0.4085	.47	.37
Area Ratio, $S_{VT_{TH}}/S_w$	0.2740	0.2116	.235	.183
Tail Vol_{TH}, $\frac{S_{VT}\,\ell_{VT}}{S_w\,b_w}$	0.0972	0.0864	0.110	.066
LATERAL DIRECTIONAL DERIVATIVES				
C_{n_β} / degree	0.0029	0.0028	.0018	.0016
C_{ℓ_β} / degree	-0.0019	-0.0016	-.00035	-.00050
$C_{n_\beta}/C_{\ell_\beta}$	-1.525	-1.750	-5.15	-3.20
Neutral Point, $NP/\bar{c}$	0.265	0.300	0.39	0.38
Static Margin, dC_m/dC_L	0.065	-0.10	-.09	.08
Lift Slope, C_{L_α}/RAD	5.05	5.50	4.70	4.58
SKIN FRICTION DRAG AT S.L., M=0.8				
Wetted Area, S_{wet} ft^2	2154	2443	1164	1660
S_{ref} ft^2	330	430	204	350
Area Ratio, S_{wet}/S_{ref}	6.50	5.7	5.7	4.75
Fuselage Length ft	66.9	59.2	44	51.3
C_{D_o}	0.0200	0.0169	0.0188	0.0150
$C_{D_o} S_{REF}$	6.60	7.27	3.85	5.26
C_{f_e}	0.0031	0.0030	0.0033	0.0032

TABLE II-1. COMPARISON OF TYPICAL V/STOL FIGHTER GEOMETRY WITH NASA V/STOL (Cont'd)

VTO WEIGHT AND FUEL		V/STOL FIGHTER		NASA V/STOL	
		Study 1	Study 2	"C"	"J"
Design VTO Wgt	lb	44700	39650	15300	19757
Wgt Empty	lb	29260	27750	10528	14491
VTO Fuel	lb	13315	9900	3440	3886
Max. Int. Fuel	lb	14211	15700	3710	4210
VTO Fuel/Wgt	-	0.298	0.250	0.225	0.197
VTO Fuel/Max. Int Fuel		0.940	0.630	0.927	0.923
1.3 VTO Wgt		58000	51500	19890	25684

B. SPECIFICATIONS

The performance specifications for Concept C are summarized in Figure II-1 through II-10. Concept J is summarized in Figure II-11 through II-20.

The Characteristics Summary Chart presented in Figure II-1 and the Standard Aircraft Characteristics Charts are presented in Figure II-2 through II-10 that summarize the conventional flight characteristics of new aircraft, Concept C. Similarly, for the modified aircraft, Concept J, the Characteristics Summary Chart is presented on Figure II-11 and the Standard Aircraft Characteristics Charts are presented in Figures II-12 through II-20.

The Characteristics Summary Chart presents a general three-view of the aircraft, its dimensions, status, power plant, and special features. On the second page of this chart is a basic research mission profile, conventional performance charts, load capacity, weights, and notes on the basic data used to define the performance characteristics.

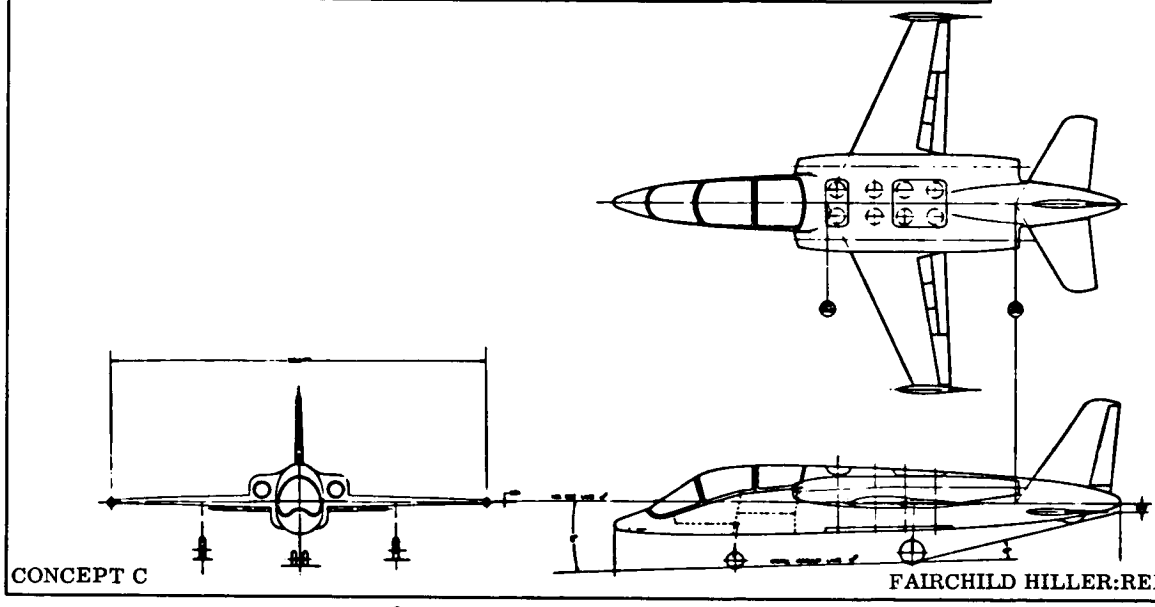
The Standard Aircraft Characteristic chart consists of nine pages, the first being a cover sheet. The second page presents the general arrangement, tank and cargo space, and the interior arrangements. The third page presents the mission and its principal characteristics. The loading and performance on a typical ferry

mission is presented on the fourth page. Standard performance charts and notes are presented on the next two pages. Supplemental charts applicable to the configuration are on the next three pages.

Substantiation data is presented in Section III - Performance Data.

Characteristics Summary

V/STOL Research Aircraft ..... Concept C



Wing area 204 ft^2

Span 320 ft

Length 42.9 ft

Height 13.5 ft

STATUS

PROPOSAL

NAVY EQUIVALENT: NONE

CONCEPT C

POWER PLANT

(8) YJ85-GE-19 Turbojet

Engine Ratings

Sea Level Static	Thrust (lb)	RPM	Min
Max.	3015	16700	5
Mil.	2930	16500	30
Norm.	2780	16350	Cont.

FEATURES

Crew: 2 (tandem)

Thrust Vectoring: Spherical nozzles

V/STOL Capability: Lift engines, jet reaction controls, 30 min. oxygen supply

Maneuvering Stabilator

Tricycle Landing Gear, Wide Track

Ejection Seats: zero altitude - zero velocity to limits of flight envelope

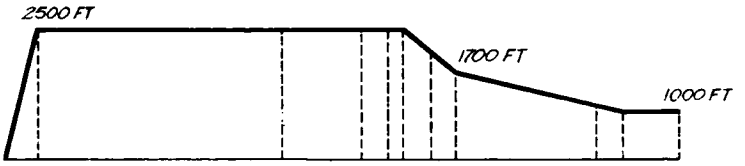
Fuel Capacity: 570 gal (max. internal)

Figure II-1 (1/2)

Characteristics Summary Basic Mission

CONCEPT C

IFR RESEARCH FLIGHT PATTERN



PERFORMANCE

RANGE	SPEED	CLIMB
540 nautical miles with 800 lb payload at 400 knots average in 1.35 hours	MAX 540 knots at 25,000 ft alt., mil. power	5850 fpm sea level, take-off weight 15570 lb normal power

CEILING	TAKE-OFF		LOAD
36000 ft 100 fpm, take-off weight 15570 lb normal power	Ground Run 2500 ft no assist	0 ft jet lift assist	Fuel: 570 gal Protected 100%
	Over 50 ft Height 3920 ft no assist	0 ft jet lift assist	

WEIGHTS	STALLING SPEED
Empty: 10,528 lb Take-off: 15,570 lb VTO Lift-off: 15,300 lb Limited by landing gear load factor	145 knots flaps up, spoiler up, VTO take-off weight 15300 lb 134 knots flaps up, VTO take-off weight 15300 lb 109 knots flaps down, VTO take-off weight 15300 lb

NOTES

FOF..ULA: RESEARCH MISSION

Vertical lift-off with military power on lift and lift-cruise engines, acceleration at constant altitude with military power to aerodynamic flight speed, and shut down lift engines at conversion speed. Climb on course with military power to circuit cruise altitude, cruise at cruise altitude at optimum cruise speed. Decelerate to conversion speed, start up lift engines, continue deceleration to hover over landing area, and a vertical let down. Range-free allowances include 1 minute at hover power, all engines operating, for starting and engine warm up, plus 5 percent of initial fuel for reserve.

GENERAL DATA

(a) Lift engines operated at 5 percent bleed in trimmed flights below conversion speed.

(b) Cruise is performed in level flights, increasing altitude by 5000 ft increments when the cruising ceiling increases accordingly. All cruising climbs at military power.

(c) Estimated Data

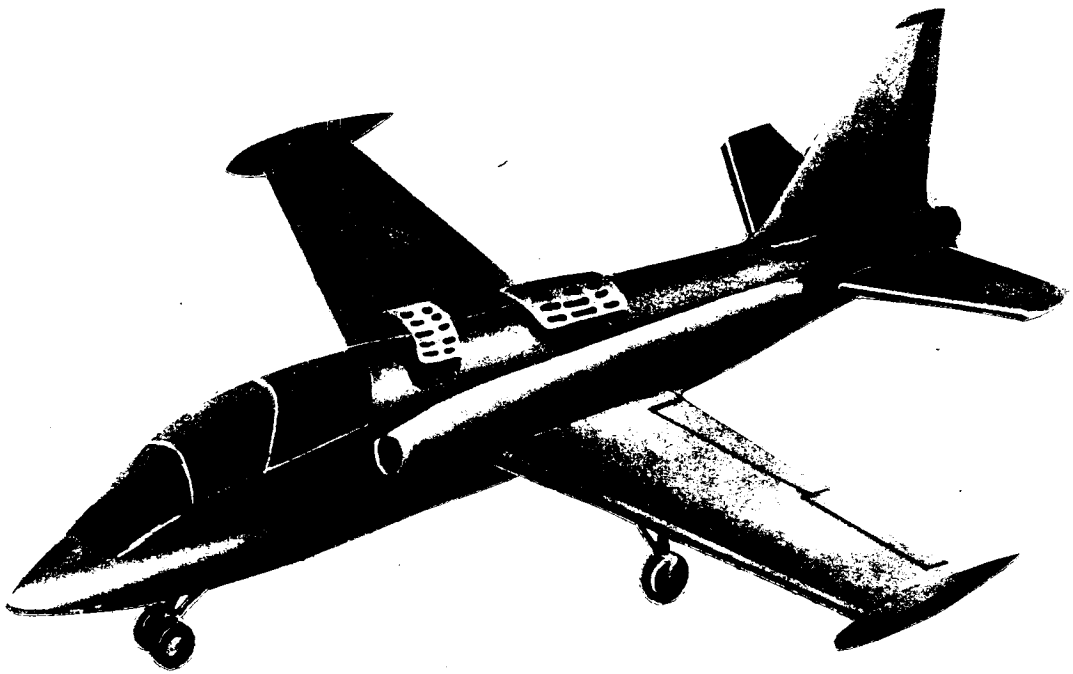
(d) Fuel density: 6.5 lbs/gal (JP-4)

(e) Military Power: 3015 lb at 16,700 rpm,
Normal Power: 2780 lb at 16,350 rpm.

REVISION BASIS

Initial Issue

Figure II-1 (2/2)



Standard Aircraft Characteristics

NASA
JET V/STOL
RESEARCH
AIRPLANE

CONCEPT C

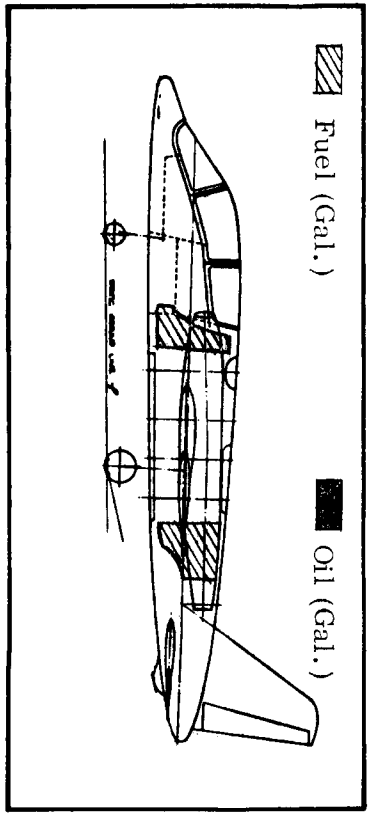
FAIRCHILD HILLER: REPUBLIC

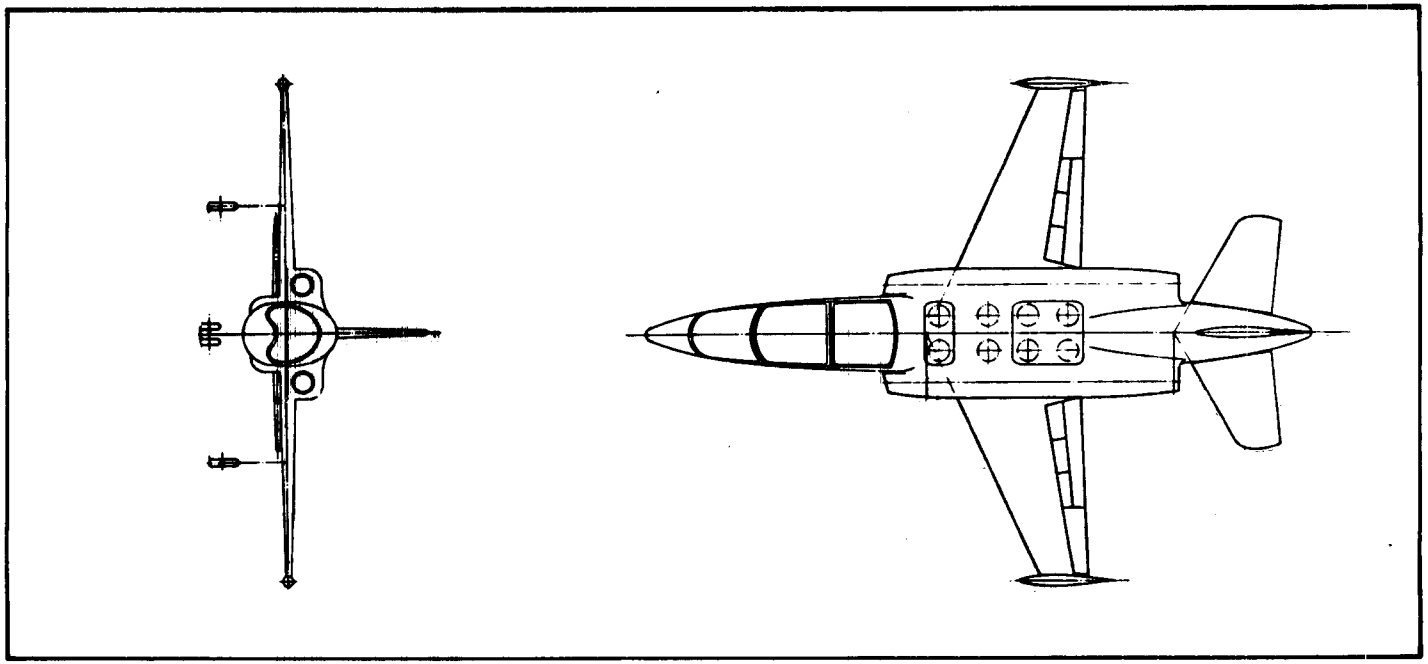
EIGHT
YJ85-GE-19
ENGINES

6/67

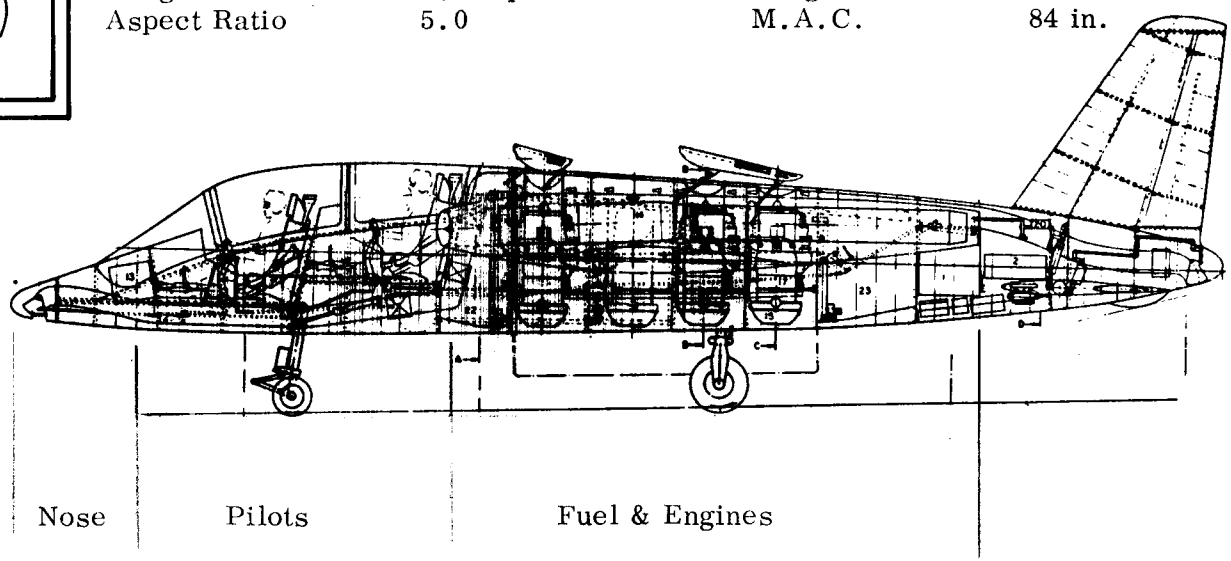
Figure II-2

Concept C
(Eight YJ85-GE-19 Engines)





Wing Area	204 sq. ft.	Wing Section	NACA 64A212-218
Aspect Ratio	5.0	M.A.C.	84 in.



9 June 1967

CONCEPT C
(Eight YJ85-GE-19 Engines)

Figure II-3

POWER PLANT	Lift	Lift/Cruise
Number	6	2
Model	YJ85-GE-19	YJ85-GE-19
Mfr.	Gen Elec	Gen Elec
Eng Spec No	E 1129	E 1129
Type	Axial	Axial
Length	40.5	40.5
Diameter	20.3"	20.3"
Weight (Dry)	387 lb	387 lb
Pivoting Sphere Nozzle	6	2
Diverter Valve	None	2

ENGINE RATINGS			
S.L. Static	LB	RPM	Min
Max.	3015	16700	5
Mil	2950	16500	30
Norm	2780	16350	Cont.
Wing			
Span	32.0'		
Incidence	0		
Dihedral	-4°		
L.E. Sweep Back	25°		
Length	42.9'		
Height	13.5'		
Tread	16.5'		

MISSION AND DESCRIPTION

Navy Equivalent: None Mfr's Model: Concept C

The principal mission of this aircraft is to serve as a jet V/STOL research aircraft to define the levels of control power & automatic stabilization required for a supersonic jet V/STOL fighter aircraft that might be developed in the future.

Concept C is a tandem two place, jet V/STOL airplane characterized by swept-back wings and empennage, low horizontal stabilator, & lift-cruise engines located above the wing-fuselage juncture. Six lift engines are located in the fuselage.

This airplane incorporates a fast operating zero-altitude-zero-velocity escape system, automatic seat ejection at low real decision time flight conditions, variable stability system for simulation of proposal aircraft characteristics, stability augmentation system, and a closed air system during vertical flight near the ground.

DEVELOPMENT	
NASA Statement of Work L-7151	13 Jun 66
Contract No NAS 1-6778	22 Nov 65
Prelim-Conceptual Design Study Report	6 Apr 67

WEIGHTS		
Loading	Lb	L. F.
Empty	10,528	-
Design Gross	15,300	3.75
V.T.O.	15,300	3.75
Landing	15,300	3.03
Ramp	15,570	-

FUEL		
Location	No Tank	Gal
Fuselage	2	570
Grade		JP-4
Specification		MIL-F-5624

ELECTRONICS	
VHF Communication	61BM-1A
Air Traffic Control	621A-3
VOR-ILS Receiver	51RV-1
Marker Beacon	51Z-4
Distance Measuring Equip	860-E-2
Audio Control Unit	387C-4
Gyro Compass	C704130
Doppler Sensor	AN/APN-163

Figure II-4

LOADING AND PERFORMANCE - TYPICAL MISSION			
CONDITION		FERRY MISSION	
Take-off weight	lb	15,570	
Fuel: Grade JP-4 (6.5 lb/gal)	lb	3,710	
Payload: 2 crew + 800 lb	lb	1,200	
Wing loading	psf	76.3	
Stall speed - power-off	knots	109.5	
Take-off ground run at S. L. 80°F - calm	ft	2,550	(1) (2)
Take-off to clear 50 ft - calm	ft	3,930	(1) (2)
Maximum speed/altitude	knots/ft	540/25,000	(2)
		535/25,000	(3)
		580/S. L.	(2)
		561/S. L.	(3)
Rate of climb at S. L.	fpm	6,090	(2)
Rate of climb at S. L. (one engine out)	fpm	830	(2)
Time: S. L. to 20,000 ft	min	3.8	(2)
Time: S. L. to 25,000 ft	min	5.2	(2)
Service Ceiling (100 fpm)	ft	38,000	(2)
Service Ceiling (one engine out - 100 fpm)	ft	19,500	(2)
Ferry range	n. mi.	541	
Avg. cruising speed	knots	390	
Cruise altitude	ft	25,000	
Total mission time	hr	1.38	
Landing Weight	lb	12,420	
Fuel: Grade JP-4 (6.5 lb/gal)	lb	557	
Stall speed - power-off	knots	97.5	
Stall speed - with approach pwr	knots	117	
Ground roll at S. L. 80°F - calm	ft	4150	
Total distance from 50 ft - calm	ft	6410	

Notes:

(1) Take-off Thrust
(2) Military Power
(3) Normal Power

Figure II-5

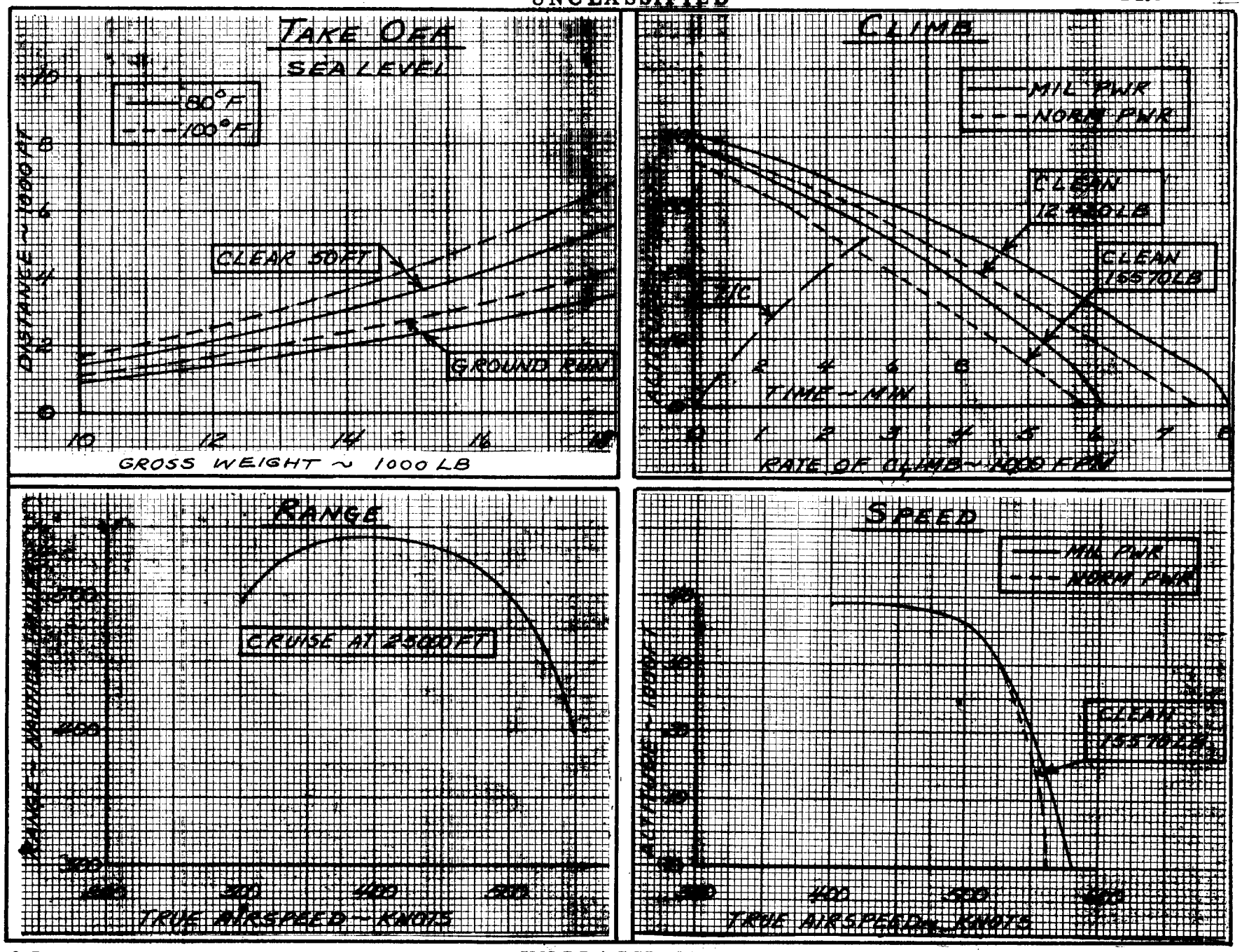


Figure II-6

CONCEPT C
(Eight YJ85-GE-19 Engines)

NOTES

FORMULA: FERRY MISSION

Take-off with military power, climb on course with military power to cruise altitude, cruise at cruise altitude at optimum cruise speed. Range-free allowances include 5 minutes at normal power at sea level for starting engines and take-off, plus 20 minutes loiter at speeds for maximum endurance at sea level, plus 5 percent of initial fuel load for reserve.

FORMULA: RESEARCH MISSION

Vertical lift off with military power on lift and lift-cruise engines, acceleration at constant altitude with military power to aerodynamic flight speed, and shut down lift engines at conversion speed. Climb on course with military power to circuit cruise altitude, cruise at cruise altitude at optimum cruise speed. Decelerate to conversion speed, start-up lift engines, continue deceleration to hover over landing area, and a vertical let-down. Range-free allowances include 1 minute at hover power, all engines operating, for starting and engine warm-up, plus 5 percent of initial fuel load for reserve.

GENERAL DATA

(a) Lift engines operated at 5 percent bleed in trimmed flight below conversion speed.

(b) Cruise is performed in level flights, increasing altitude by 5000 ft increments when the cruising ceiling increases accordingly. All cruising climbs are at military power.

(c) Estimated data

(d) Fuel density: 6.5 lb/gal (JP-4)

(e) Military power: 3015 lb at 16,700 rpm,
Normal power: 2780 lb at 16,350 rpm.

REVISION BASIS

Initial Issue

Figure II-7

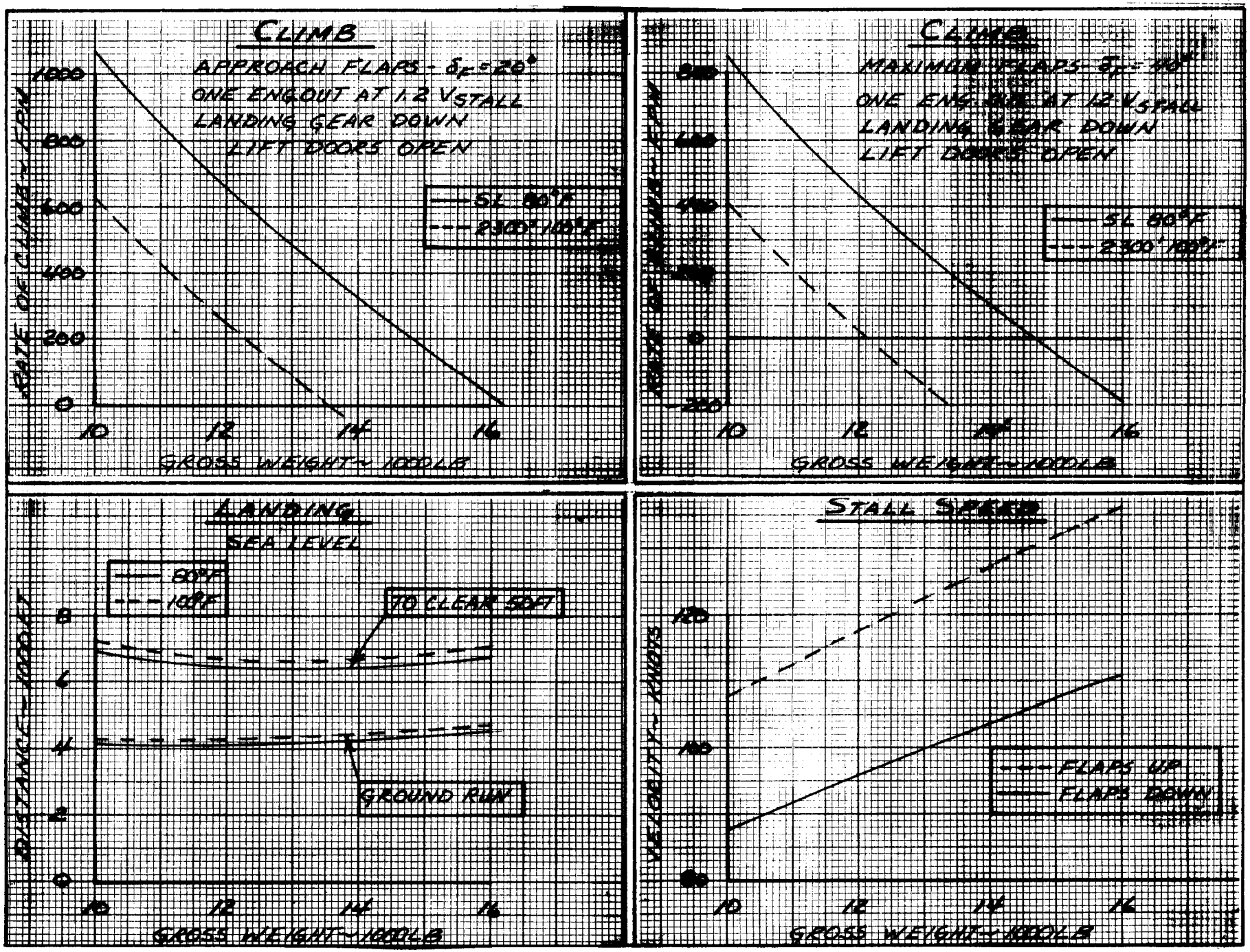
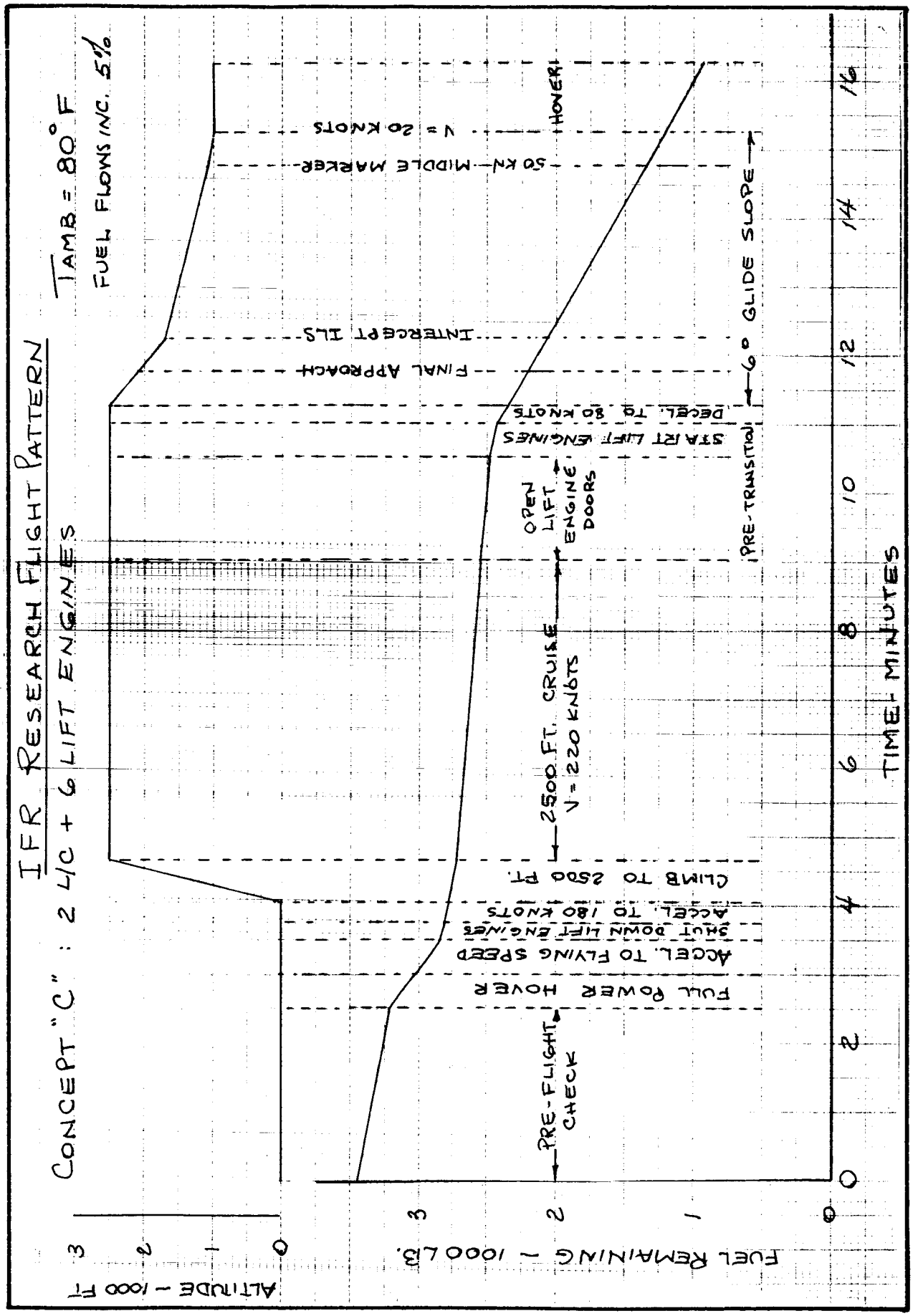


Figure II-8

(Eight YJ85-GE-19 Engines)



CONCEPT C
(Eight YJ85-GE-19 Engines)

Figure II-9

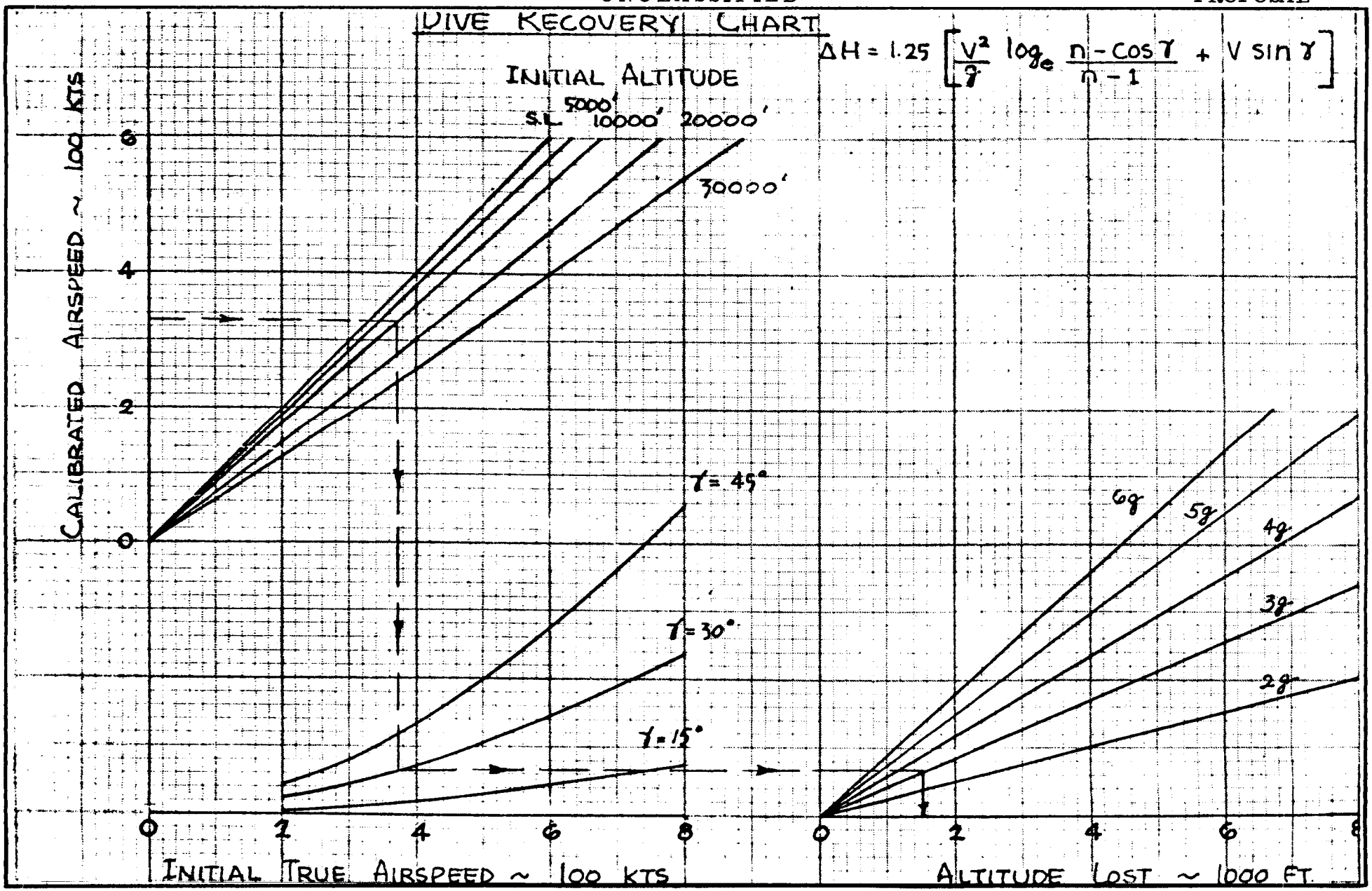
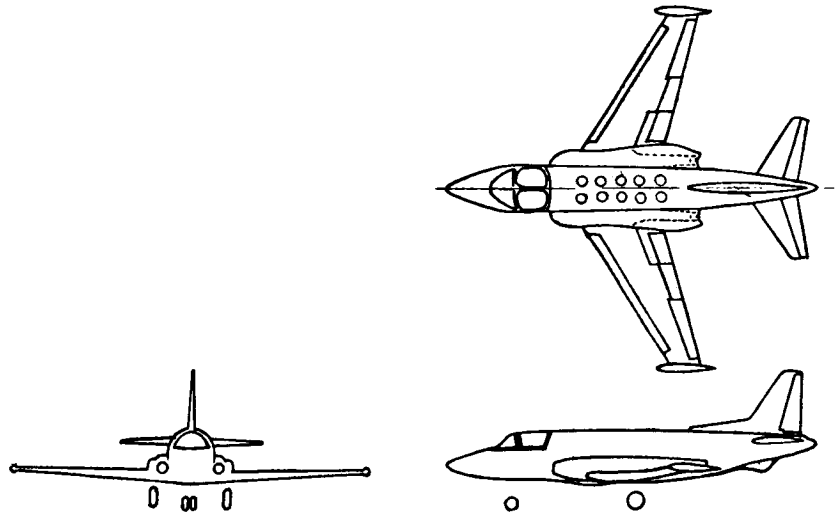

DIVE RECOVERY CHART
$\Delta H = 1.25 \left[\frac{V^2}{g} \log_e \frac{n - \cos\gamma}{n - 1} + V \sin\gamma \right]$
INITIAL ALTITUDE
S.L.
5000'
10000'
20000'
30000'
γ = 45°
γ = 30°
γ = 15°
6g
5g
4g
3g
2g
CALIBRATED AIRSPEED ~ 100 KTS
0
2
4
6
8
INITIAL TRUE AIRSPEED ~ 100 KTS
ALTITUDE LOST ~ 1000 FT

Figure II-10

Characteristics Summary

V/STOL Research Aircraft ..... Concept J



CONCEPT J

FAIRCHILD HILLER:REPUBLIC

Wing area 350 ft^2

Span 44 ft

Length 46.9 ft

Height 15.4 ft

STATUS

PROPOSAL

NAVY EQUIVALENT: NONE

CONCEPT J

POWER PLANT

(8) YJ85-GE-19 Turbojet

Engine Ratings

Sea Level Static	Thrust (lb)	RPM	Min.
Max.	3015	16700	5
Mil.	2930	16500	30
Norm.	2780	16350	Cont.

FEATURES

Crew: 2 (side by side)

Thrust Vectoring: Spherical nozzles

V/STOL Capability: Lift engines, jet reaction controls, 30 min. oxygen supply

Ejection Seats: Zero altitude - zero velocity to limits of flight envelope

Fuel Capacity: 600 gal (max. internal)

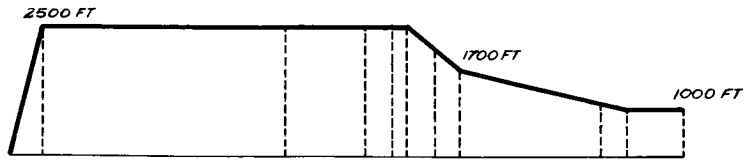
Figure II-11 (1/2)

CONCEPT J
(ten YJ85-GE-19 engines)

Characteristics Summary Basic Mission

CONCEPT J

IFR RESEARCH FLIGHT PATTERN



PERFORMANCE

RANGE	SPEED	CLIMB
435 nautical miles with 800 lb payload at 400 knots average in 1.09 hours	MAX 500 knots at 25,000 ft alt., mil. power	3700 fpm sea level, take-off weight 20081 lb normal power

CEILING	TAKE-OFF		LOAD
32000 ft 100 fpm, take-off weight 20081 lb normal power	Ground Run 2700 ft no assist	0 ft jet lift assist	Fuel: 600 gal Protected 100%
	Over 50 ft Height 4400 ft no assist	0 ft jet lift assist	

WEIGHTS	STALLING SPEED
Empty Take-off: 20081 lb VTO Lift-off: 19757 lb Limited by landing gear load factor	109 knots flaps up, VTO take-off weight 19757 lb 102 knots flaps down, VTO take=off weight 19757 lb

NOTES

FORMULA: RESEARCH MISSION

Vertical lift-off with military power on lift and lift-cruise engines, acceleration at constant altitude with military power to aerodynamic flight speed, and shut down lift engines at conversion speed. Climb on course with military power to circuit cruise altitude, cruise at cruise altitude at optimum cruise speed. Decelerate to conversion speed, start up lift engines, continue deceleration to hover over landing area, and a vertical let down. Range-free allowances include 1 minute at hover power, all engines operating, for starting and engine warm up, plus 5 percent of initial fuel for reserve.

GENERAL DATA

(a) Lift engines operated at 5 percent bleed in trimmed flights below conversion speed.

(b) Cruise is performed in level flights, increasing altitude by 5000 ft increments when the cruising ceiling increases accordingly. All cruising climbs at military power.

(c) Estimated Data

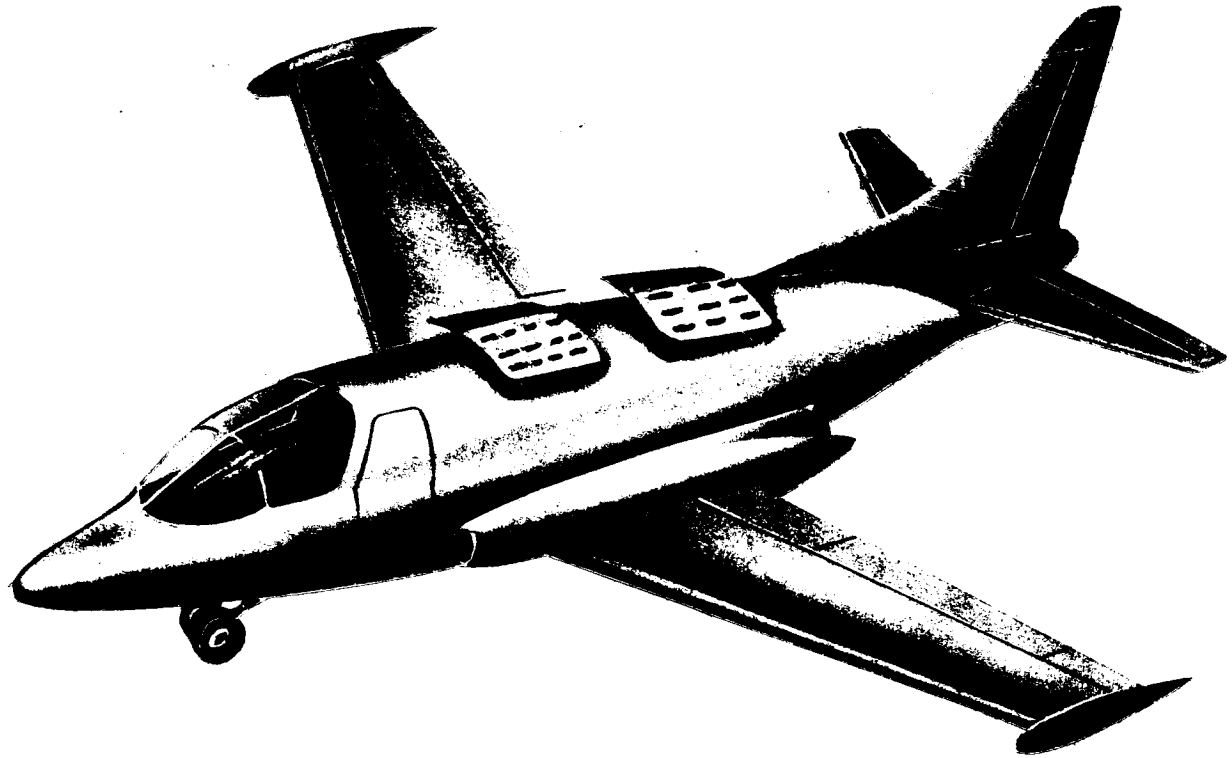
(d) Fuel density: 6.5 lbs/gal (JP-4)

(e) Military Power: 3015 lb at 16,700 rpm,
Normal Power: 2780 lb at 16,350 rpm.

REVISION BASIS

Initial Issue

Figure II-11 (2/2)



Standard Aircraft Characteristics

NASA
JET V/STOL
RESEARCH
AIRPLANE

CONCEPT J

FAIRCHILD HILLER: REPUBLIC

TEN
YJ85-GE-19
ENGINES

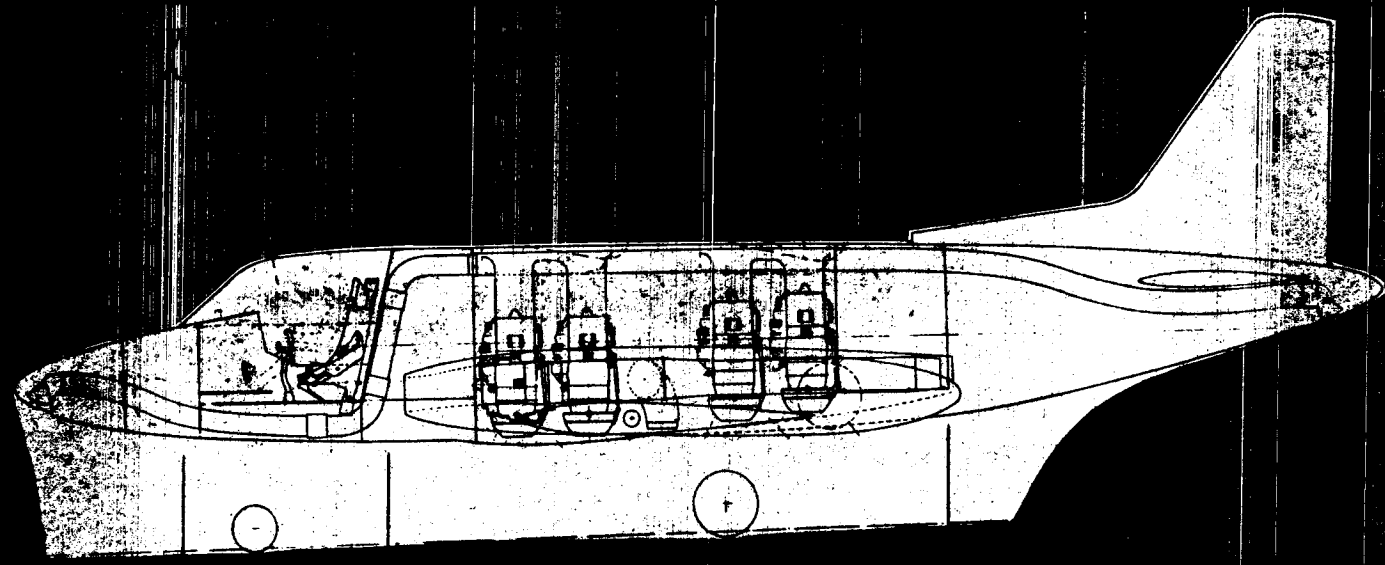
UNCLASSIFIED

Figure II-12

Concept J
(Ten YJ85-GE-19 Engines)

UNCLASSIFIED

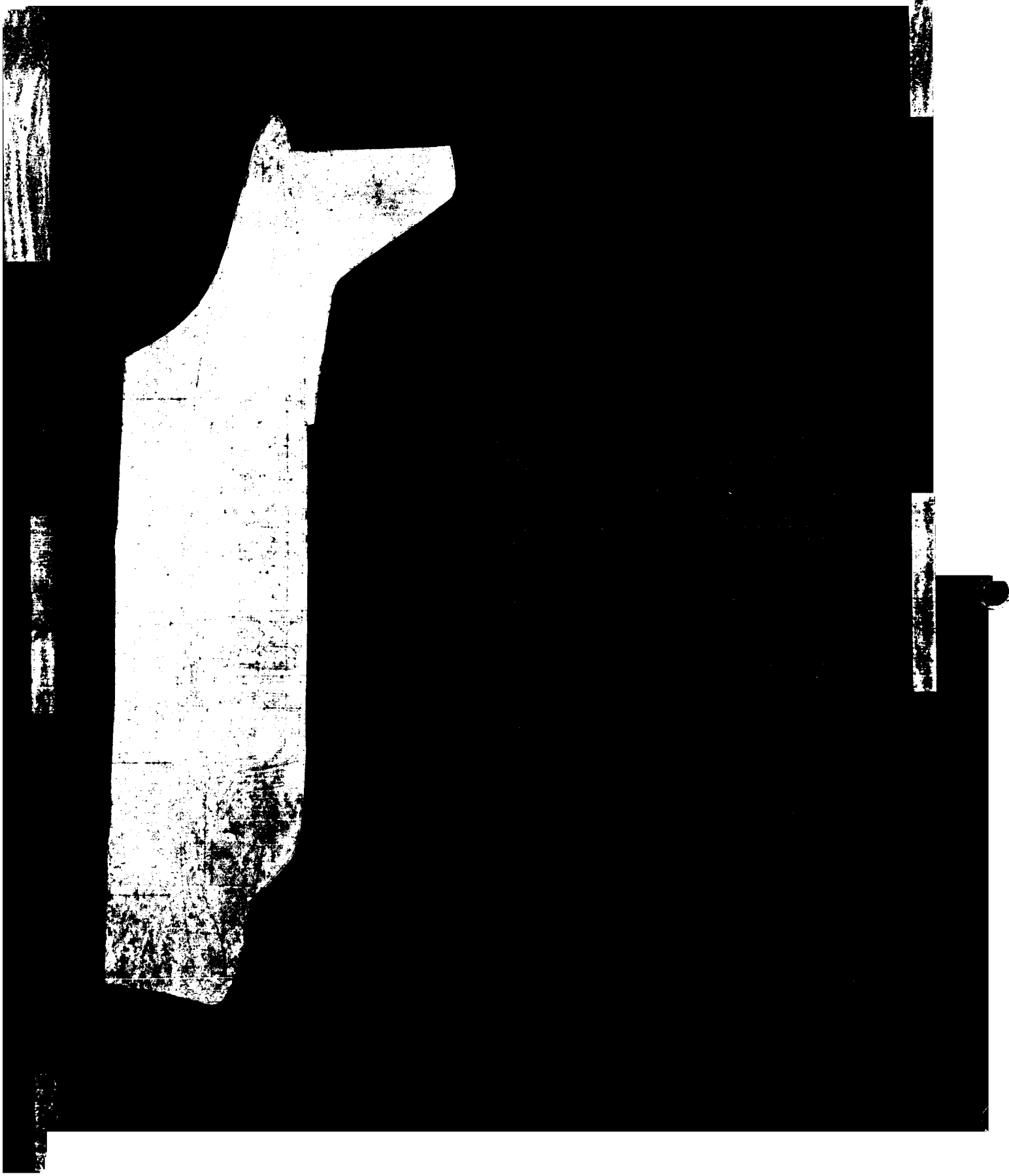
PROPOSAL



6 June 1967

UNCLASSIFIED

CONCEPT J
(Ten YJ85-GE-19 engines)



POWER PLANT

	Lift	Lift/Cruise
Number	8	2
Model	YJ85-GE-19	YJ85-GE-19
Mfr.	Gen Elec	Gen Elec
Eng Spec No	E1129	E1129
Type	Axial	Axial
Length	40.5"	40.5"
Diameter	20.3"	20.3"
Weight (Dry)	387 lb	387 lb
Pivoting Sphere Nozzle	8	2
Diverter Valve	None	2

ENGINE RATINGS

S.L. Static	LB	RPM	Min
Max.	3015	16700	5
Mil	2950	16500	30
Norm	2780	16350	Cont.

DIMENSIONS

Wing	
Span	44.64
Incidence	
Dihedral	3°
L.E. Sweep Back	31.5°
Length	46.9
Height	15.36
Tread	9.2

MISSION AND DESCRIPTION

Navy Equivalent: None **Mfr's Model:** Concept J

The principal mission of this aircraft is to serve as a jet V/STOL research aircraft to define the levels of control power & automatic stabilization required for a supersonic jet V/STOL fighter aircraft that might be developed in the future.

Concept J is a North American series 60 Sabreliner modified as a side-by-side 2 place Jet V/STOL airplane characterized by swept back wings and empennage, low horizontal tail. Lift-Cruise Engines are located above the wing-fuselage juncture. Eight Lift engines are located in the fuselage.

This airplane incorporates a fast operating zero-altitude-zero velocity escape system, automatic seat ejection at low real decision time flight conditions, variable stability system for simulation of proposal aircraft characteristics, stability augmentation system and a closed air system during vertical flight near the ground.

DEVELOPMENT

NASA Statement of Work L-7151	13 Jun 66
Contract No NAS 1-6778	22 Nov 65
Prelim-Conceptual Design Study Report	6 Apr 67

WEIGHTS

Loading	Lb	L. F.
Empty	14490	-
Design Gross	19750	3.0
V.T.O.	19750	3.0
Landing (VTO)	19750	2.03
Ramp	20080	

FUEL

Location	No Tank	Gal
Fuselage	2	600
Grade		JP-4
Specification		MIL-F-5624

ELECTRONICS

VHF Communication	61BM-1A
Air Traffic Control	621A-3
VOR-ILS Receiver	51RV-1
Marker Beacon	51Z-4
Distance Measuring Equip	860E-2
Audio Control Unit	387C-4
Gyro Compass	C704130
Doppler Sensor	AN/APN-163

LOADING AND PERFORMANCE - TYPICAL MISSION			
CONDITION		FERRY MISSION	
Take-off weight	lb	19,830	
Fuel: Grade JP-4 (6.5 lb/gal)	lb	4300	
Payload: 2 crew + 800 lb	lb		
Wing loading	lb/ft^2	56.6	
Stall speed - power off	knots	102.0	
Take-off ground run at S.L. 80°F - calm	ft	2710	(1)(2)
Take-off to clear 50 ft 80°F - calm	ft	4350	(1)(2)
Maximum speed/altitude	knots/ft	505/25,000	(2)
		495/25,000	(3)
		525/S.L.	(2)
		492/S.L.	(3)
Rate of climb at S.L.	fpm	4300	(2)
Rate of climb at S.L. (one engine out)	fpm	1050	(2)
Time: S.L. to 20,000 ft	min	6.8	(2)
Time: S.L. to 25,000 ft	min	10.	(2)
Service ceiling (100 fpm)	ft	32,000	(2)
Service ceiling (one engine out - 100 fpm)	ft	15,800	(2)
Ferry range	n. mi.	435	
Avg. cruising speed	knots	400	
Cruise altitude	ft	25,000	
Total mission time	hr	1.23	
Landing Weight	lb	16,497	
Fuel: Grade JP-4(6.5 lb/gal)	lb	967	
Stall speed - power-off	knots	93	
Stall speed - with approach pwr	knots	111	
Ground roll at S.L. 80°F - calm	ft	3680	
Total distance from 50 ft - calm S.L. - 80°F - calm	ft	5900	

Notes:

(1) Take-off thrust
(2) Military power
(3) Normal power

Figure II-15

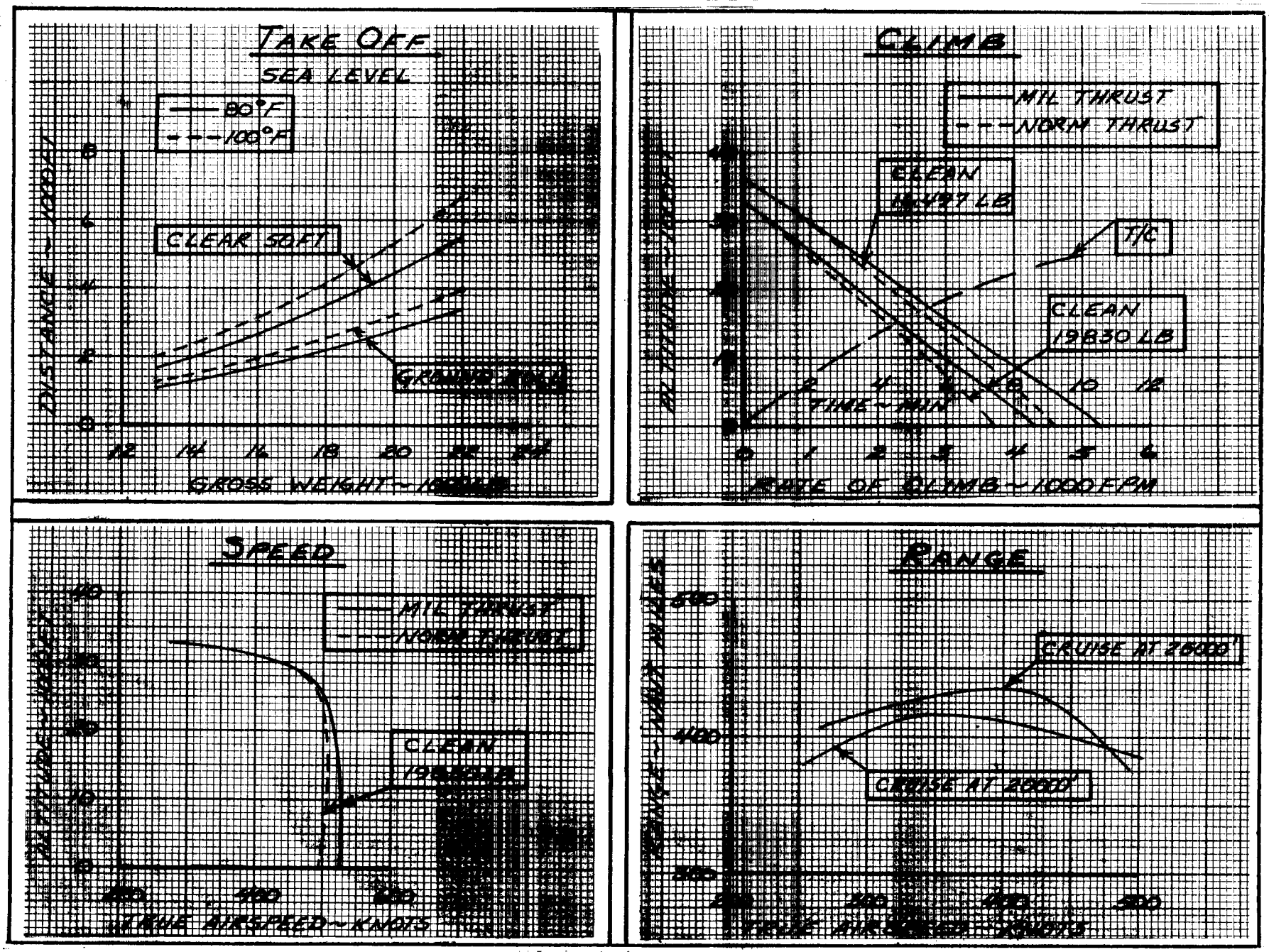


Figure II-16

CONCEPT J
(Ten YJ85-GE-19 engines)

NOTES

FORMULA: FERRY MISSION

Take-off with military power, climb on course with military power to cruise altitude, cruise at cruise altitude at optimum cruise speed. Range-free allowances include 5 minutes at normal power at sea level for starting engines and take-off, plus 20 minutes loiter at speeds for maximum endurance at sea level, plus 5 percent of initial fuel load for reserve.

FORMULA: RESEARCH MISSION

Vertical lift off with military power on lift and lift-cruise engines, acceleration at constant altitude with military power to aerodynamic flight speed, and shut down lift engines at conversion speed. Climb on course with military power to circuit cruise altitude, cruise at cruise altitude at optimum cruise speed. Decelerate to conversion speed, start-up lift engines, continue deceleration to hover over landing area, and a vertical let-down. Range-free allowances include 1 minute at hover power, all engines operating, for starting and engine warm-up, plus 5 percent of initial fuel load for reserve.

GENERAL DATA

(a) Lift engines operated at 5 percent bleed in trimmed flight below conversion speed.

(b) Cruise is performed in level flights, increasing altitude by 5000 ft increments when the cruising ceiling increases accordingly. All cruising climbs are at military power.

(c) Estimated data

(d) Fuel density: 6.5 lb/gal (JP-4)

(e) Military power: 3015 lb at 16,700 rpm,
Normal power: 2780 lb at 16,350 rpm.

REVISION BASIS

Initial Issue

Figure II-17

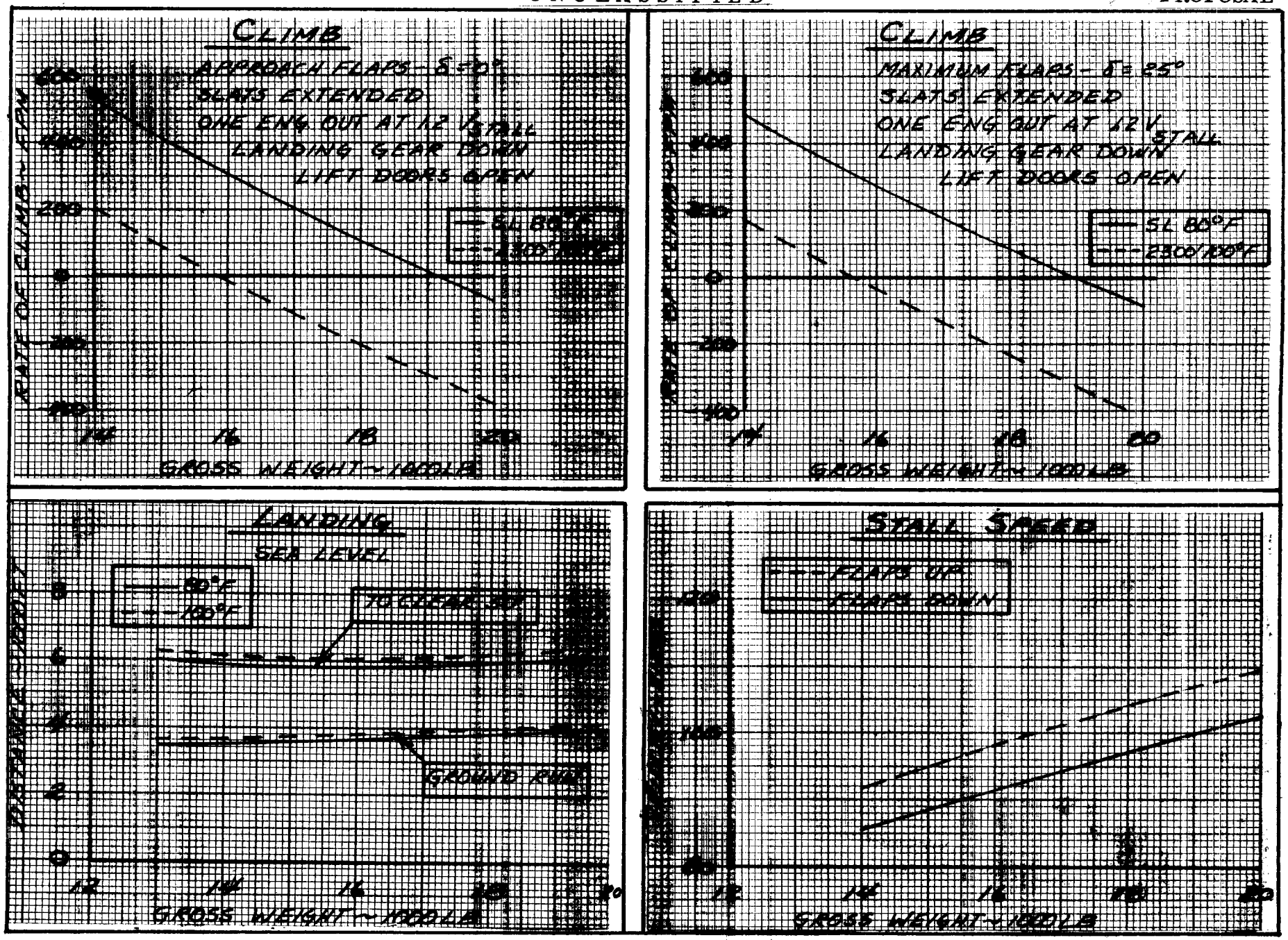
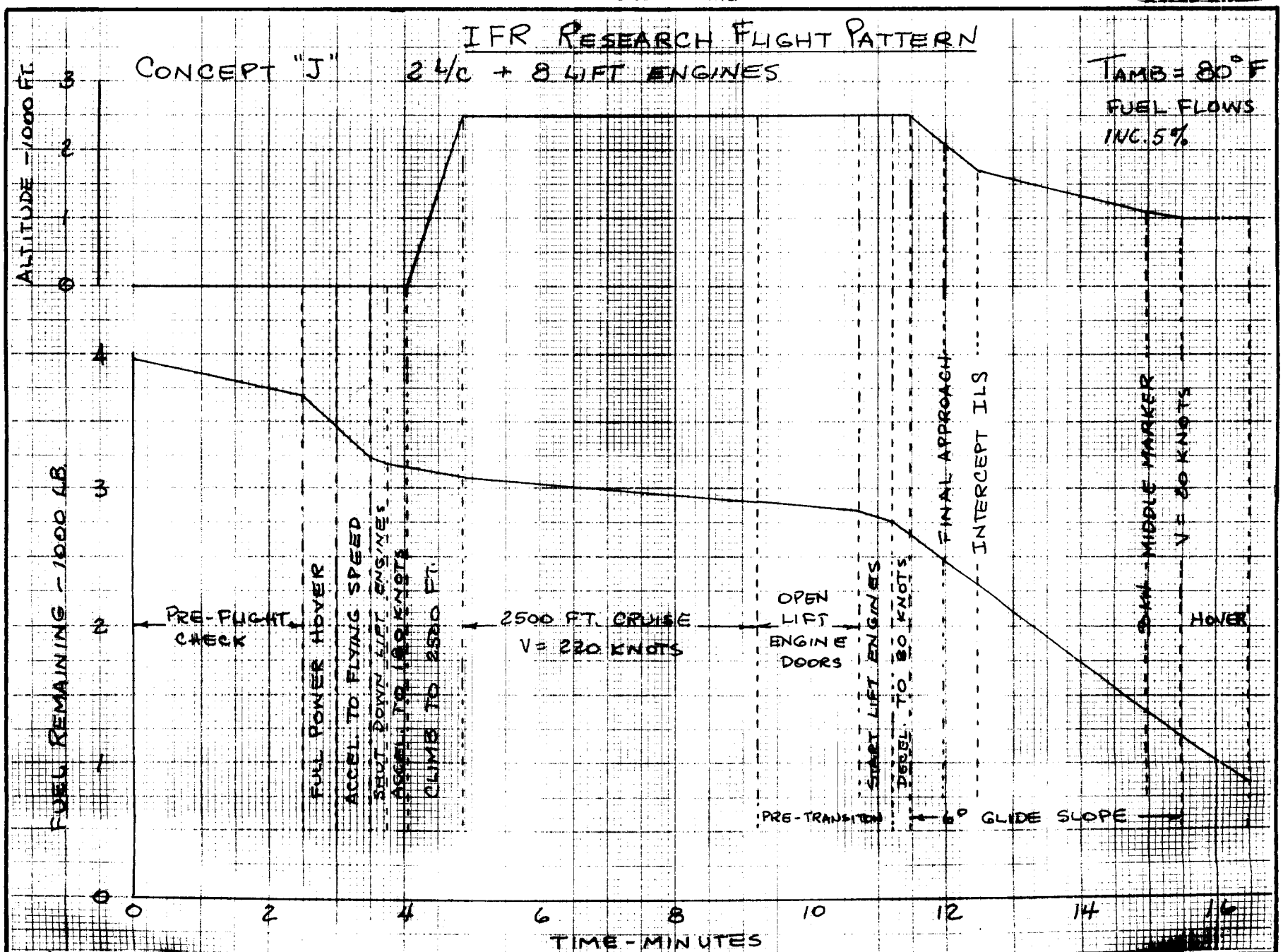

CLIMB
APPROACH FLAPS – δ = 10°
SLATS EXTENDED
ONE ENG OUT AT 1.2 V STALL
LANDING GEAR DOWN
LIFT DOORS OPEN
SL 80°F
2300/100°F
RATE OF CLIMB ~ FPM
GROSS WEIGHT ~ 1000 LB
CLIMB
MAXIMUM FLAPS – δ = 25°
SLATS EXTENDED
ONE ENG OUT AT 1.2 V STALL
LANDING GEAR DOWN
LIFT DOORS OPEN
SL 80°F
2300/100°F
GROSS WEIGHT ~ 1000 LB
LANDING
SEA LEVEL
80°F
100°F
TO CLEAR 50'
GROUND RUN
DISTANCE ~ STANDARD FT
GROSS WEIGHT ~ 1000 LB
STALL SPEED
FLAPS UP
FLAPS DOWN
GROSS WEIGHT ~ 1000 LB

Figure II-18


IFR RESEARCH FLIGHT PATTERN
CONCEPT "J"
2 4/c + 8 LIFT ENGINES
TAMB = 80° F
FUEL FLOWS INC. 5%
ALTITUDE - 1000 FT.
FUEL REMAINING - 1000 LB.
TIME - MINUTES
PRE-FLIGHT CHECK
FULL POWER HOVER
ACCEL. TO FLYING SPEED
SHUT DOWN LIFT ENGINES
ASCEND TO 180 KNOTS
CLIMB TO 2500 FT.
2500 FT. CRUISE
V = 220 KNOTS
OPEN LIFT ENGINE DOORS
START LIFT ENGINES
DECEL. TO 80 KNOTS
FINAL APPROACH
INTERCEPT ILS
PRE-TRANSITION
6° GLIDE SLOPE
MIDDLE MARKER
V = 80 KNOTS
HOVER

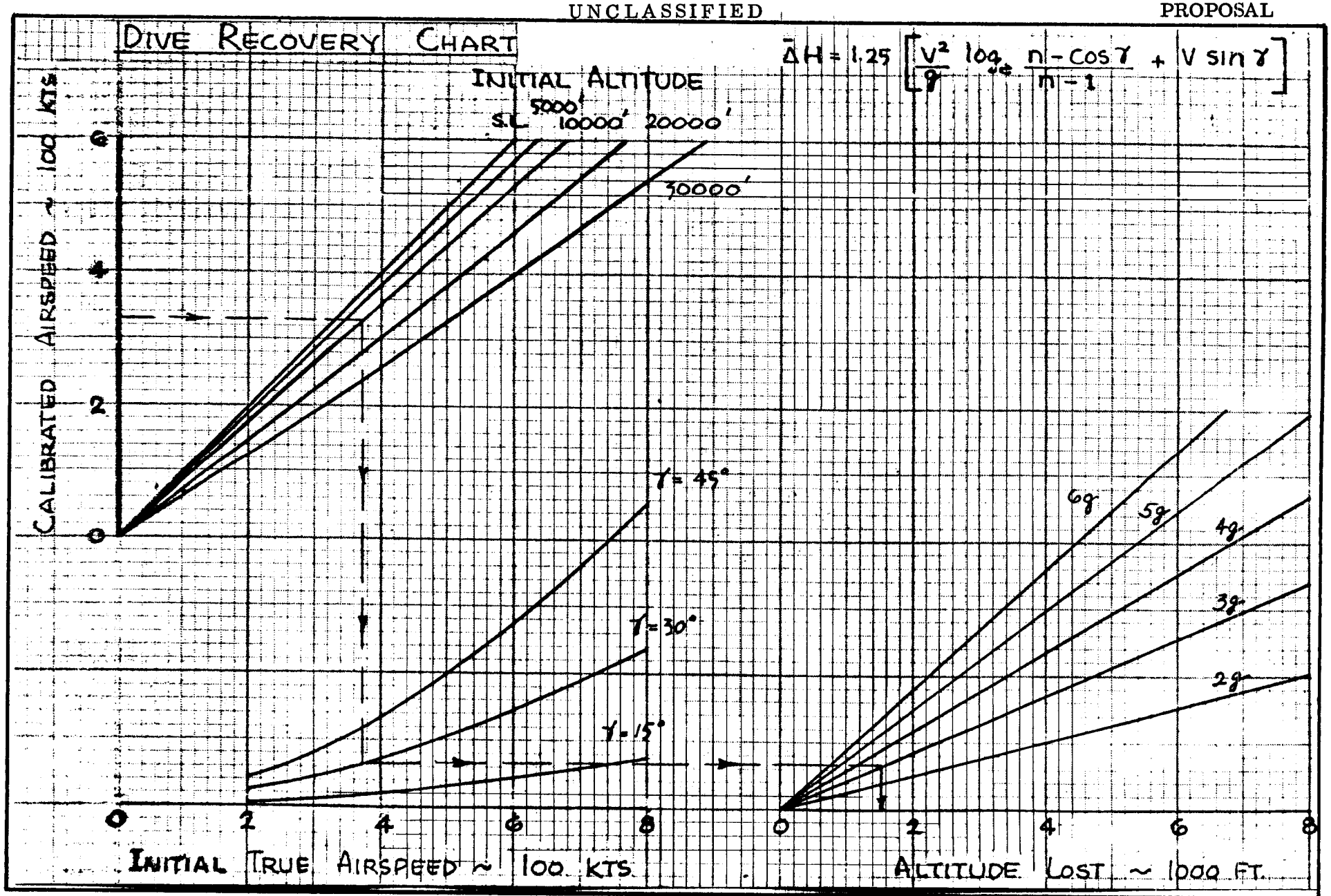

DIVE RECOVERY CHART
$\bar{\Delta H} = 1.25 \left[\frac{V^2}{g} \log_e \frac{n - \cos\gamma}{n-1} + V \sin\gamma \right]$
INITIAL ALTITUDE
S.L.
5000'
10000'
20000'
30000'
CALIBRATED AIRSPEED ~ 100 KTS
0
2
4
6
$\gamma = 45°$
$\gamma = 30°$
$\gamma = 15°$
6g
5g
4g
3g
2g
0
2
4
6
8
INITIAL TRUE AIRSPEED ~ 100 KTS.
ALTITUDE LOST ~ 1000 FT.

Figure II-20

SECTION III

PERFORMANCE DATA

A. INTRODUCTION

The performance calculations presented in this section are used to substantiate the specifications presented in Section II. The procedures are in conformance with normal engineering practice and the estimates are in accordance with the most recent and available aerodynamic test data.

There are no stated minimum conventional flight performance requirements such as for speed, climb, cruise, conventional take-off and landings, or ferry range. The vehicle is designed for a fighter airplane (Class III) at altitudes up to 25,000 feet at 450-knot speed or limit Mach number of 0.8. Normal fighter aircraft minimums are complied with.

This is the final preliminary design report on the proposed NASA V/STOL Jet Operations Research Airplane Concept C, a new design.

This aircraft has a high research utility and has a more maneuverability and safety than the Lockheed XV-4B. Data from previous V/STOL aircraft have been used in designing this aircraft, including the Ryan XV-5A, the Hawker-Siddely P-1127, and the Short SC-1.

A second aircraft, Concept J, which is a modification of the North American Sabreliner is also presented for alternative consideration. This aircraft has less maneuverability and limited growth potential.

B. TABULATED DATA

1. Aircraft Dimensional Data

General and aerodynamic geometry data used in the analysis are presented here for Concepts C and J.

		CONCEPT C	CONCEPT J
WING ASSEMBLY			
Theo. Area	ft^2	204	350
Span	ft	32.0	45.3
M.A.C., $\bar{c}$	ft	7.0	8.4
Aspect Ratio	-	5.0	5.9
Taper Ratio	-	.30	.32
Sweep	Deg.		
L.E.		25	31.5
T.E.		2	14
$0.25\bar{c}$		20	28
Dihedral	Deg.	-4	3
Incidence	Deg.	0	0
FLAPS			
Type	-	Single Slotted Fowler	Slotted
Area	ft^2	25.5	34.00
Deflection	Deg.	40	25
Percent Wing Chord	%	30	25
AILERONS			
Area	ft^2	10	22
Deflection	Deg.	±20	±15.5
Percent Wing Chord	%	30	.25
SPOILERS			
Area	ft^2	8	-
Deflection	Deg.	60	-
Percent Wing Chord	%	15	-

		CONCEPT C	CONCEPT J
SLATS			
Area	ft^2	-	36.34
HORIZONTAL TAIL			
Theo. Area	ft^2	66	81
Span	ft	15	18
M.A.C.	ft	4.25	5.0
Aspect Ratio	-	3.4	4.0
Taper Ratio	-	.43	.30
Sweep $0.25\bar{c}$	Deg.	25	30
Dihedral	Deg.	0°	0
Incidence	Deg.	0	0
Tail Arm, $\ell_{HT}/\bar{c}_{TH}$	-	2.05	2.17
Area Ratio, $S_{HT_{TH}}/S_{W_{TH}}$	-	.324	.232
Tail Volume, $\frac{S_{HT_{TH}}\ \ell_{HT}}{S_{W_{TH}}\ \bar{c}_{TH}}$	-	.663	.502
Deflection	Deg.	+20, -25	-1 to -9
ELEVATOR			
Area	ft^2	--	21
c_r/c_t	%	--	35
Deflection	Deg.	--	22 up, 11 down
VERTICAL TAIL			
Theo. Area	ft^2	48	64
Span	ft	9.0	10.6
M.A.C.	ft	5.75	6.83
Aspect Ratio	-	1.70	1.75
Taper Ratio	-	.33	.23

		CONCEPT C	CONCEPT J
VERTICAL TAIL (cont'd)			
Sweep $0.25\bar{c}$	Deg.	29	30
Tail Arm, ℓ_{VT}/b_W	-	.47	.37
Area Ratio, S_{VT}/S_W	-	.235	.183
Tail Volume, $\frac{S_{VT}\,\ell_{VT}}{S_W\,b_W}$	-	0.110	0.066
RUDDER			
Area	ft^2	10	16
c_r/c_t	%	.30	.375
Deflection	Deg.	±25	±25

2. Weights

The weights used in the performance calculations are presented in Table III-1 and comply with the definitions given in this section.

TABLE III-1 TABLE OF WEIGHTS (pounds)

	Concept C	Concept J
Empty Weight	11860	14491
Design VTO Weight	15300	19757
Basic Weight	11060	15071
Maximum Take-Off Weight	15570	20081
Maximum Landing Weight	15570	19830
Landing Weight	12420	16497

a. Empty Weight

The airplane weight empty does not include crew, fuel, oil, cargo, and disposable or special equipment.

b. Design VTO Weight

This is the weight at which specified flight structural design requirements are met in vertical lift-off and let-down and in conventional flight load factors.

c. Basic Weight

The basic weight is the operating weight empty which is the empty weight plus trapped fuel and oil and all fixed equipment for normal operation.

d. Maximum Take-Off Weight

The maximum take-off weight is the greatest weight for take-off.

The maximum (overload) take-off weight is determined with the aircraft fully loaded with fuel and cargo to capacity for which space and tankage is normally provided without exceeding the bearing capacity of the floor or supporting structure. The aircraft and its components meets the minimum load factor for each phase of the operation, including taxi and ground handling. The conventional take-off ground run at sea level and with standard atmospheric conditions does not exceed 8,000 ft. The maximum rate-of-climb under the same atmospheric conditions with normal power is above 500 ft/min. The service ceiling at military power with one engine inoperative is above sea level. Throughout the flight, the center-of-gravity remain within the limits for satisfactory flight and ground handling.

The maximum (normal) take-off weight does not exceed the maximum (overload) take-off weight. The military power service ceiling is above sea level on a hot day (100°F) with one engine inoperative, gear up, flaps set for take-off. The distance over a 50-foot height on the same day does not exceed 10,000 feet with a rate-of-climb above 300 ft/min. The airplane can cruise at

airspeeds for long range at 5,000 feet pressure altitude on the same day without exceeding 85 percent of normal power.

The maximum (VTO) take-off weight is the greatest weight for take-off established by the design requirements for maximum hover endurance plus overload fuel corresponding to one minute at hovering power. The ferry mission starts with this gross weight.

e. Landing Weights

The maximum landing weight is the greatest weight established for landing by design requirements.

The landing weight includes the fuel reserve remaining at the end of flight for a gas turbine general purpose fighter aircraft. The fuel allowance for reserve and landing is the sum of 5 percent of the initial fuel plus 20 minutes at speeds for maximum endurance at sea level. All engines used in landing approach shall be operating.

f. Operational Weight Limitations

The weight limitations are structural considerations based on the design load factors specified by NASA. Further details are given in Sections V and VII.

C. AERODYNAMIC DATA

1. Drag

a. Methods and Analysis

The variation of aircraft minimum drag coefficient with Mach number was determined by computing the drag of principal components, including interference, with reference to available test data, and taking the sum of the parts. This method was only applied at one flight condition (M = .6, 36,000 feet).

In order to determine the drag variation throughout the complete flight regime, this value was then adjusted by the method which is outlined in the section on compressibility corrections (Section III.C.1.d).

A typical analysis is presented based on Concept J.

b. Basic Data

(1) Fuselage Drag for Concept J

The fuselage drag is the summation of the pressure drag due to the forebody and the afterbody and the friction drag of the external surfaces. A typical analysis is presented using Concept J as a reference.

(a) Pressure Drag

(Reference 1). The forebody drag was determined from the data of S. F. Hoerner (Reference 1). The pressure drag of the normal visibility wind screen (vee) was .0012 based upon reference area (350 sq. ft.). The pressure coefficient over the forebody, excluding canopy, as obtained from S. F. Hoerner (3-12),* was .01. Applying this value to the proper area it was determined that the fuselage forebody with the wind screen contributes a pressure drag increment of .0006, based upon 350 sq. ft.

The afterbody pressure drag coefficient obtained from S. F. Hoerner (15-5), applied over an area on the fuselage representing approximately half the boat tail, was .03. Applying this value to the respective area on Concept J gave the increment in pressure drag contributed by the fuselage boat tail as .0011 based on 350 sq. ft.

Total fuselage pressure drag as obtained from the addition of all the contributions, calculated in the foregoing sections, amounts to .0029 based on reference area.

(b) Friction Drag

Skin friction drag on the fuselage at M = 0.6 and 36,000 feet in turbulent air is computed on the basis of D.R. Report 1415 BuAer and NACA TN 3391. The length is 85 percent of the fuselage length. The total fuselage wetted area is on the order of 800 square feet. The specific values are given in the configuration drag analysis.

(2) Engine Nacelle Drag

The drag analysis for the nacelles on Concept J are presented. The uninstalled drag of engine nacelles with a fineness ratio of

* ( ) Refers to Section Number

approximately 5 to 1, at mass flow ratios over .3 is given in S. F. Hoerner (9-9) as .04 based on nacelle frontal area. Adjusting this value for the part of the nacelle buried in the fuselage and wing, which is approximately 20%, gave the uninstalled drag of the nacelle as .0017 based on the airplane reference area of 350 sq. ft. S. F. Hoerner (8-10) gives the following expression for the interference drag for a wing attached to a wall, based on wing thickness:

$$(C_D)_t = .75\ (t/c) - \frac{.0003}{(t/c)^2}$$

where (t/c) is the thickness ratio of the wing root. This expression was applied, taking thickness ratio as that measured along a longitudinal cut where the nacelle attached to the fuselage of Concept J. With (t/c) = .127 and t = 16 inches, it was determined that the drag increment of nacelle-fuselage interference was .0006. Adding this amount to the uninstalled nacelle drag gave installed drag of two nacelles of .0023.

(3) Miscellaneous

(a) Lifting Surface Interference

Wing-fuselage interference, as determined in exactly the same manner described previously in the section dealing with nacelle drag, was calculated to be .0003 for the Concept J with a wing root thickness ratio of .11 and wing root thickness equal to 11 inches.

Empennage-fuselage interference was calculated by a method outlined in S. F. Hoerner (8-12) which gives the drag increment due to mutual interference between a fuselage and an ordinary low horizontal tail-single vertical tail arrangement, equal to .3 for six surface juncture corners, based on empennage thickness ratio of .08. Converting this value to reference area gave .0001 as the empennage-fuselage interference drag increment.

(b) Wing Tip Pods

The drag increment wing tip pods for tip reaction controls of the type included in Concept J is .05 based on the maximum cross sectional area of the pod, according to the data presented in S. F. Hoerner (13-17).

Converting this value to drag based on reference area resulted in the drag increment, due to both wing tip pods with a maximum diameter of 8 inches, equal to .0001.

(c) Protuberances

It was estimated that protuberances incorporated in Concept J would present a source of drag which would be equivalent to that of a 6 sq. in. flat plate perpendicular to the stream flow (S. F. Hoerner). Applying a value for C_{D_π} equal to 1.17 to this area it was estimated that the drag of protuberances amounted to .0008.

(d) Vents

It was estimated that each vent and port would present an area of .02 sq. ft. to the stream. Assuming ten such vents are incorporated in the subject design, and that the ΔC_{D_π} for such a drag source is .5 (S. F. Hoerner) (9-16), it was estimated that the drag contribution of vents was .0003 based on reference area.

(4) Configuration Drag for Concept J

The configuration drag is the summation of the pressure drag and friction drag of each component.

(a) Wing and Empennage

The total drag of both wing and empennage on Concept J was determined by assuming two dimensional flow over all lifting surfaces. This condition provides that the minimum drag of these surfaces based on their respective exposed areas is equal to the value of minimum section drag as obtained from Abbot and Von Doenhoff for 64 series airfoils. The magnitude of this value ranges between .0055-.0063 for sections with thickness ratios between .06 and .11, and, basing these values on the reference area of 350 sq. ft. gave the total drag of the wing as .0054, and that of the empennage as .0021, with 300 sq. ft. of exposed wing area and 118 sq. ft. exposed empennage area. The skin friction included in these values is slightly conservative, but no correction was made for differences in full scale and wind tunnel test Reynolds number.

(b) Skin Friction Drag

Skin friction drag was calculated for Concept J at the reference flight condition of M = .6, 36,000 ft. by applying the data of D.R. Report No. 1415 BuAer, and NACA TN 3391, assuming lengths pertinent to each component of the wetted area breakdown as presented in the following tabulation:

Component	Wetted Area sq. ft.	Pertinent Length ft.	Length Source
Wing	595	7.9	Exposed $\bar{C}$
Horizontal Tail	127	4.6	Exposed $\bar{C}$
Vertical Tail	109	6.5	Exposed $\bar{C}$
Fuselage	796	38.0	85% Fuselage Length

(c) Total Minimum Drag

The total minimum drag coefficient for Concept J at the reference flight condition of M = .6, 36,000 ft. was determined by adding all the components in the following manner. Fuselage skin friction includes 10% roughness reflected by the length used to calculate Reynolds number (85% actual fuselage length).

Component	ΔC_D Configuration 1 VEE CANOPY
Fuselage Pressure	.0029
Fuselage Skin Friction	.0052
Fuselage Total	.0081
Wing Total	.0054
Empennage Total	.0021
Interference Lifting Surfaces	.0004
Gaps and Protuberances	.0011
Nacelles and Interference	.0023
Wing Tip Pods	.0001
	.0195

c. Corrections for Altitude and Compressibility

(1) Drag Variation with Altitude

Drag varies with altitude principally as a function of skin friction. The computed values of drag due to skin friction, $C_{D_{SF}}$, were increased 5 percent to correct the flat plate values for three dimensional effects. At the reference flight condition, M = 0.6, 36,000 ft., an average length is computed.

$$L_{AV} = R/\ (R/ft.),\ ft.$$

where (R/ft.) is 1.39×10^6 at M = 0.6, 36,000 ft. and R is the Reynold's number corresponding to complete configuration, M = 0, skin friction coefficient at 36,000 ft., $(C_f)_{M=0}$. As mentioned previously, the usual component breakdown procedure was applied in order to calculate total skin friction drag at M = .6. Reversing the procedure by dividing this value by the ratio of compressible to incompressible skin friction obtained in NACA TN 3391 assuming zero heat transfer in the boundary layer, gives aircraft $(C_f)_{M=0}$.

The computations for L_{AV} were accomplished with the following information:

Configuration	No. 1
$C_{D_{SF}}$, Skin Friction, M = .6, 36,089	.0116
$(C_f)_{M=0}$	.00269
R (Reference 4)	16.8×10^6
L_{AV}	12 ft.

Multiplying the L_{AV} = 12 ft. by (R/ft.) gave the aircraft Reynolds number for any flight condition. $(C_f)_{M=0}$ was then obtained from D.R. Report No. 1415 BuAer and, after correcting for compressibility with the data of NACA TN 3391 the skin friction based on reference area was obtained by dividing by the wetted area ratio, S_{WET}/S_{REF}, of 4.65 on Concept J (Figure III-1).

(2) Drag Variation with Mach Number

The drag variation is estimated by the form factor in the subsonic speed range and the drag divergence Mach number in the transonic range.

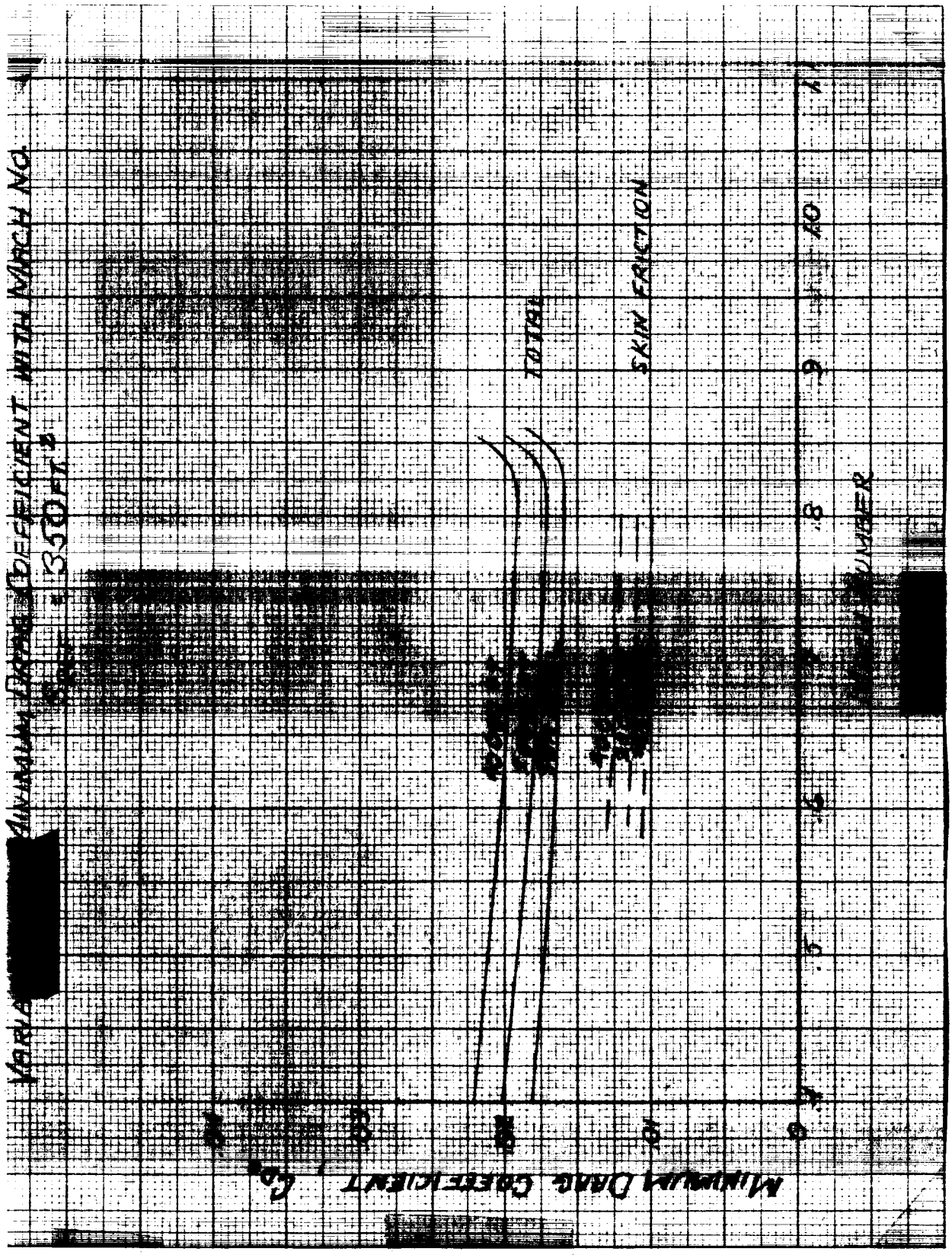

MINIMUM DRAG COEFFICIENT WITH MACH NO.
S = 350 FT²
TOTAL
SKIN FRICTION
MACH NUMBER
.4
.5
.6
.7
.8
.9
1.0
1.1
MINIMUM DRAG COEFFICIENT, C_{D_0}

Figure III-1

(a) Subsonic

The factor, used on the skin friction drag to obtain an estimated total minimum drag variation with Mach number in the subsonic speed range, is called the form factor, F. For concept J, the form factor has a value of 1.60.

$$F = \left\{ \frac{\text{total minimum drag coefficient}}{\text{skin friction drag coefficient}} \right\}_{M,\ h}$$

(b) Transonic

The minimum drag variation deviates sharply near transonic speeds due to the rapid build-up of pressure drag. The calculated drag variation is corrected with reference to the drag divergence Mach number. This drag divergence Mach number is defined as that Mach number at which the variation of drag coefficient with Mach number exceeds 0.1.

$$M_D = \text{Mach number for } \frac{dC_D}{dM} = 0.1$$

Using the method outlined in North American Report ADL-52-2, the divergence Mach number for Concept J is 0.867.

(3) Induced Drag Variation with Mach Number

The induced drag factor due to lift as given in Section III.C.2.b is corrected for Mach number effects.

The variation of induced drag factor, k, with Mach number was obtained applying an empirical correction for Mach number illustrated in the following equations:

$$\beta_R = \sqrt{1 - M_R^2}$$

$$C = k_{(M_R)} \ (\beta_R A)^{0.9} / \beta_R$$

$$\beta = \sqrt{1 - M^2}$$

$$k_M = \beta C / (\beta A)^{0.9}$$

where

k_{M_R} = induced drag at reference Mach number

M_R = Mach number (referenced)

M = Mach number

A = aspect ratio

k_M = induced drag at Mach number

d. Drag Comparisons

In order to establish confidence in the drag estimation, the total drag of Concept J at M = .6, 36,000 ft. (based on reference area) has been included with the test data obtained from contemporary aircraft and plotted in Figure III-2. . Curves of constant equivalent aircraft drag, (C_{f_e}), have been faired through these data points and are an indication of aircraft cleanliness. C_{f_e} is C_{D_o} at M = .6, 36,000 ft., based on aircraft wetted area. Concept J carries a value of .00416 at C_{D_o} = .0198 with $2S_{REF}/S_{WET}$ equal to .423.

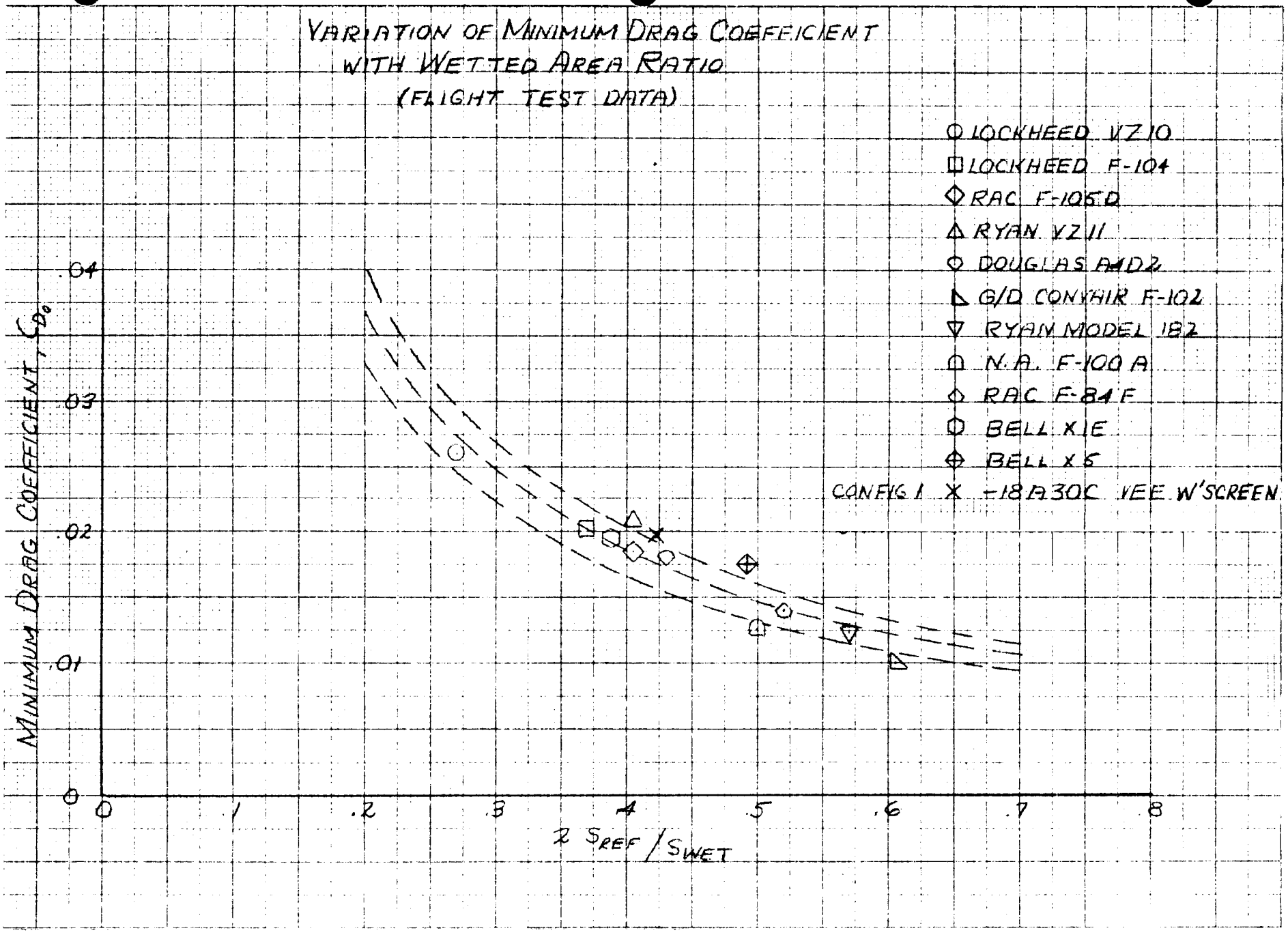

VARIATION OF MINIMUM DRAG COEFFICIENT
WITH WETTED AREA RATIO
(FLIGHT TEST DATA)
LOCKHEED VZ10
LOCKHEED F-104
RAC F-105D
RYAN VZ11
DOUGLAS A4D2
G/D CONVAIR F-102
RYAN MODEL 182
N.A. F-100A
RAC F-84F
BELL X1E
BELL X5
CONFIG 1 X F-18A30C VEE W'SCREEN
MINIMUM DRAG COEFFICIENT, C_{D_0}
.04
.03
.02
.01
0
0
.1
.2
.3
.4
.5
.6
.7
.8
$\approx S_{REF}/S_{WET}$

Figure III-2

2. Lift

a. Methods and Analysis

The wing-body lift curve has been derived using the component build-up shown on Figure III-3. The basic wing parameters are presented in this section. This analysis includes auxiliary lift control devices.

The wing for the proposal airplane (P) represents, as well as possible, with one design, the characteristic variation of load factor with angle-of-attack and attitude near stall speed as would be appropriate to a supersonic jet V/STOL fighter configuration (F) that may be developed for the U.S. Military Services in the next few years.

$$\left(\frac{n}{\alpha}\right)_F = \left\{\frac{L/W}{\alpha}\right\}_F = \left\{\frac{C_{L_\alpha}\, q_s}{(W/S)}\right\}_F = \left\{\frac{C_{L_\alpha}\, q_s}{(W/S)}\right\}_P$$

where n = load factor

α = angle-of-attack

L = aerodynamic lift

W = airplane gross weight

C_{L_α} = lift curve slope

q_s = dynamic pressure at stall speed

S = reference wing area

W/S = wing loading

The stall speed (100 knots) for the proposed vehicle is much lower than the stall speed for a jet fighter (170 knots); therefore, the lift curve for the proposed vehicle would have a higher slope or the wing loading would have to be lower. The wing loading is set by the specified stall speed and the maximum lift coefficient ($C_{L_{max}}$).

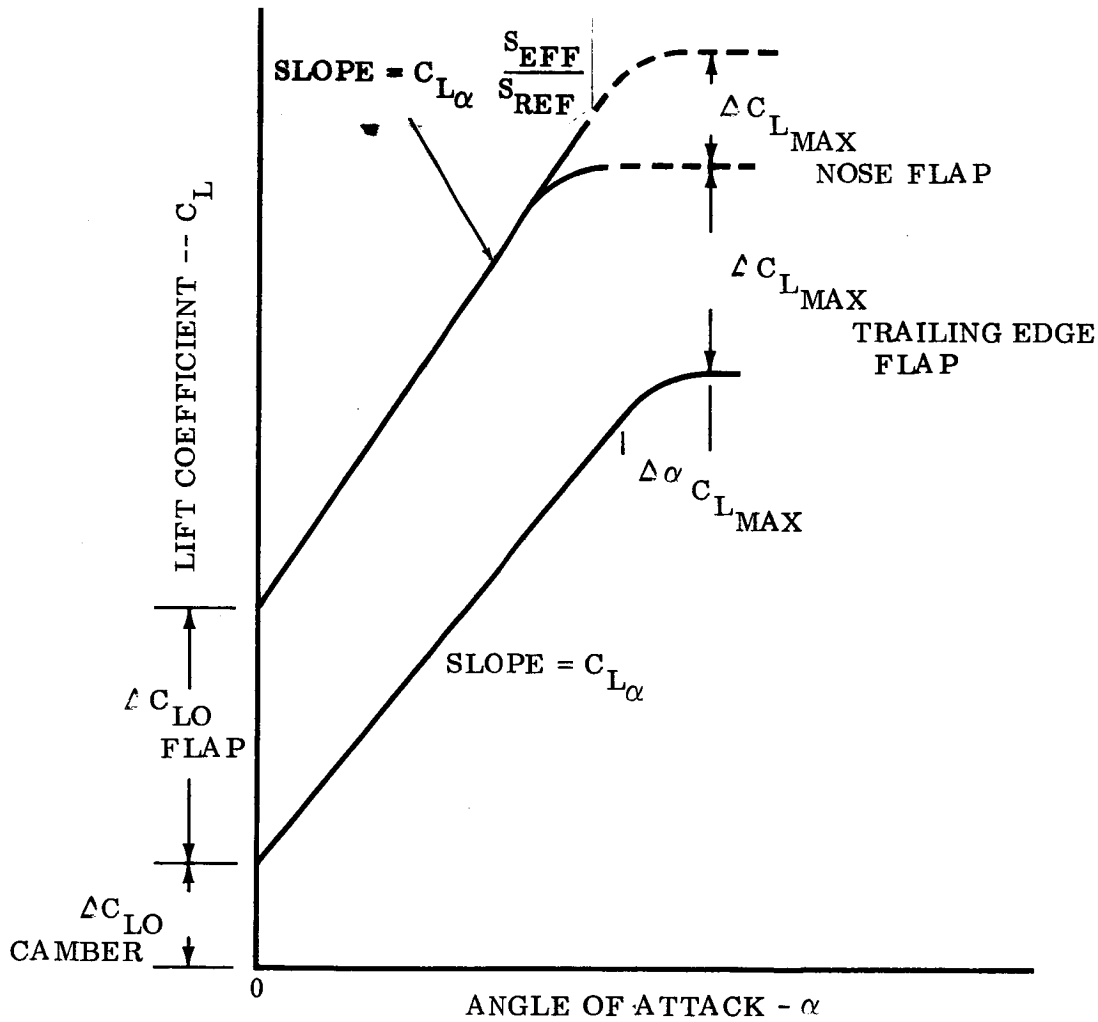


Figure III-3. Lift Curve Component Build-Up

$$(W/S) = C_{L_{max}} q_s$$

or $$\left(\frac{n}{\alpha}\right)_F = \left\{\frac{C_{L_\alpha}}{C_{L_{max}}}\right\}_F = \left\{\frac{C_{L_\alpha}}{C_{L_{max}}}\right\}_P$$

Due to the lift curve curvature in the stall region, the useful maneuvering range may generally be considered as all angles-of-attack below the ground angle-of-attack for take-off and landing.

$$C_{L_{Take\ Off}} = 0.9\ C_{L_{max}} \quad \text{(MIL-C-5011A paragraph 3.2.5 (b))}$$

$$C_{L_{Landing}} = C_L \text{ at } 1.10\ V_s \quad \text{(MIL-C-5011A paragraph 3.2.6 (a))}$$

Using the take-off lift coefficient as the simulation parameter in the linear lift range, it is possible to solve for attitude simulation near stall speed.

$$\left\{\frac{1}{\alpha_{To}}\right\}_F = \left\{\frac{C_{L_\alpha}}{.9\ C_{L_{max}} - C_{L_o}}\right\}_F = \left\{\frac{C_{L_\alpha}}{.9\ C_{L_{max}} - C_{L_o}}\right\}_P$$

where

C_{L_o} = lift coefficient at zero angle-of-attack

α_{To} = take-off angle-of-attack or ground angle at take-off

or

$$\left\{C_{L_\alpha}\right\}_P = \left\{C_{L_{max}}\right\}_P \left\{\frac{n}{\alpha}\right\}_F$$

and $$\left\{C_{L_\alpha}\right\}_P = \left\{.9\ C_{L_{max}} - C_{L_o}\right\}_P / \left\{\alpha_{To}\right\}_F$$

This solution of the problem is in the selection of a combination of maximum lift-coefficient ($C_{L_{max}}$), and wing area (S) to satisfy the specified stall speed dynamic pressure (q_s) and the take-off ground angle of the supersonic jet V/STOL fighter configuration $(\alpha_{To})_F$ being simulated is to be obtained with a compatible lift curve slope (C_{L_α}).

The maximum lift coefficient is designed on a research operational optimum combination of airfoil section, wing planform, and flaps for gentle stall characteristics. The ground angle-of-attack is designed on a research operational optimum combination of aspect ratio and sweep for low roll inertia. These parameters were perturbated by material considerations, including the landing gear installation, the reaction jet control system for roll control, and the hover flight control power requirements.

b. Basic Data

(1) Airfoil Section

A fixed wing section in the transonic and supersonic flight regime has a small leading edge radius and an aft maximum thickness station (50 percent chord) to reduce pressure drag and transonic pressure pulsations. This causes a low maximum lift coefficient ($C_{L_{max}}$) for the basic wing as a result of early boundary layer flow separation. The increase in the induced drag factor (k) is due to a lower wing efficiency factor (e).

For variable sweep configurations, a thicker wing section is used in the unswept position. This thickness ratio decreases as the wing is swept as a result of the change in wing geometry.

On the proposal airplane, a more forward location for the maximum thickness was selected to reduce the abrupt stalls inherent in thin airfoils. The gentler stall characteristics of the 64AOXX series airfoil was selected. A wing thickness of 12 percent kept constant to the outboard flap station. From the outboard flap station to the wing tip, the thickness was increased linearly to 18 percent to provide volume for the reaction jet control ducts, to prevent wing tip stall, and to provide a small degree of added directional stability (C_{n_β}).

Camber was incorporated in the basic airfoil to provide a simple construction at the maximum lift coefficient using a single-slotted Fowler flap.

(2) Wing Planform

A relationship is determined graphically for the variation of theoretical aspect ratio (AR_{th}) with wing sweep ($\Lambda_{C/2}$) which will provide a specified lift curve slope ($C_{L_{\alpha_P}}$) at the estimated geometry ratios (taper ratio (λ) and fuselage width to wing span ratio (2 r/b)). Figure III-4 presents the graph used in this analysis.

The maximum lift coefficient for a selected aspect ratio and wing sweep is determined from Figure III-5 for the flaps up configuration, tail off.

(3) Trailing Edge Flap

The variation of maximum lift coefficient at optimum trailing edge flap deflection is presented in Figure III-6 as a function of the theoretical aspect ratio and wing sweep. The effect of a 2 percent camber is included in this presentation. This graph shows that a single-slotted flap with 50 percent Fowler action compares favorably with a double-slotted flap in its lift capacity. Wing sweep causes a measurable decrease in the maximum lift coefficient.

(a) Lift Increment Due to Flap Deflection at Zero Angle-of-Attack

The incremental lift coefficient due to flap deflection at zero angle-of-attack is graphically determined with the aid of Figure III-7 The geometric parameters considered include:

b_i	inboard flap chord station
b_o	outboard flap chord station
λ	taper ratio
b	total wing span
AR_{th}	theoretical aspect ratio
$\Lambda_{c/2}$	sweep of the half-chord
C_f/C	flap chord to wing chord ratio

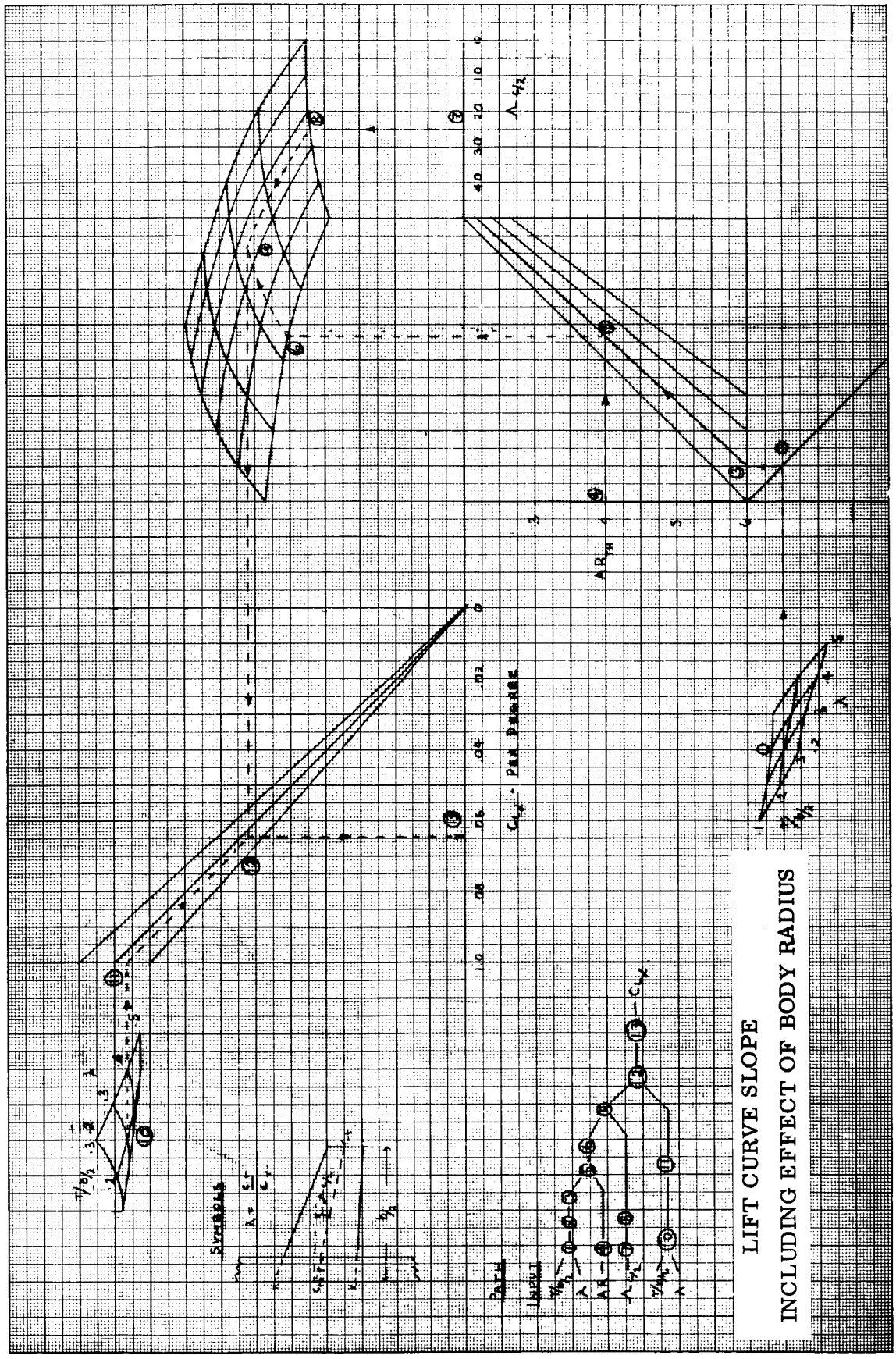


Figure III-4

Wing Planform Effects

$C_{L\ max}$ = 1.3 (Including 2% Camber L. E. Droop)

$\lambda = .3 \quad b_o/(b/2) = .7 \cdot \frac{C_4}{C} = .3 \quad b_1/(b/2)$

AR	4	5	6
	.3	.278	.244

No Ground Effects

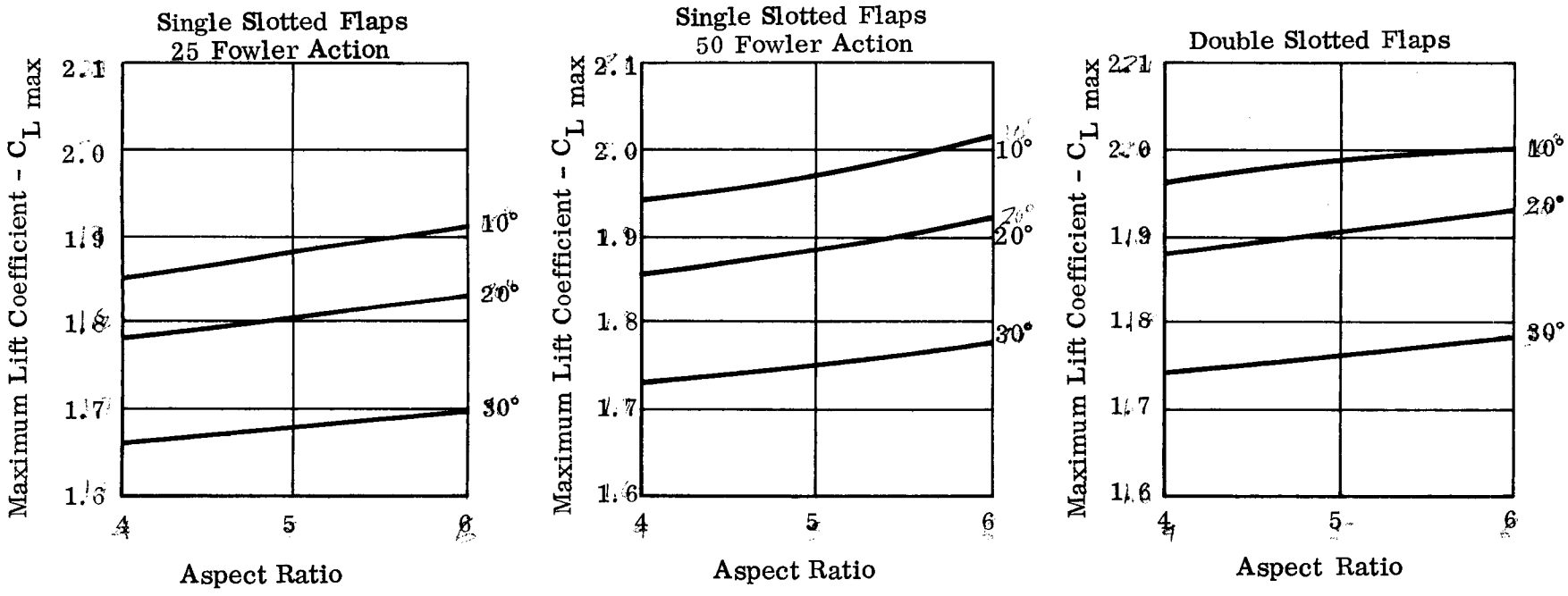


Figure III-6

INCREMENTAL LIFT DUE TO FLAP DEFLECTION AT ZERO ANGLE OF ATTACK

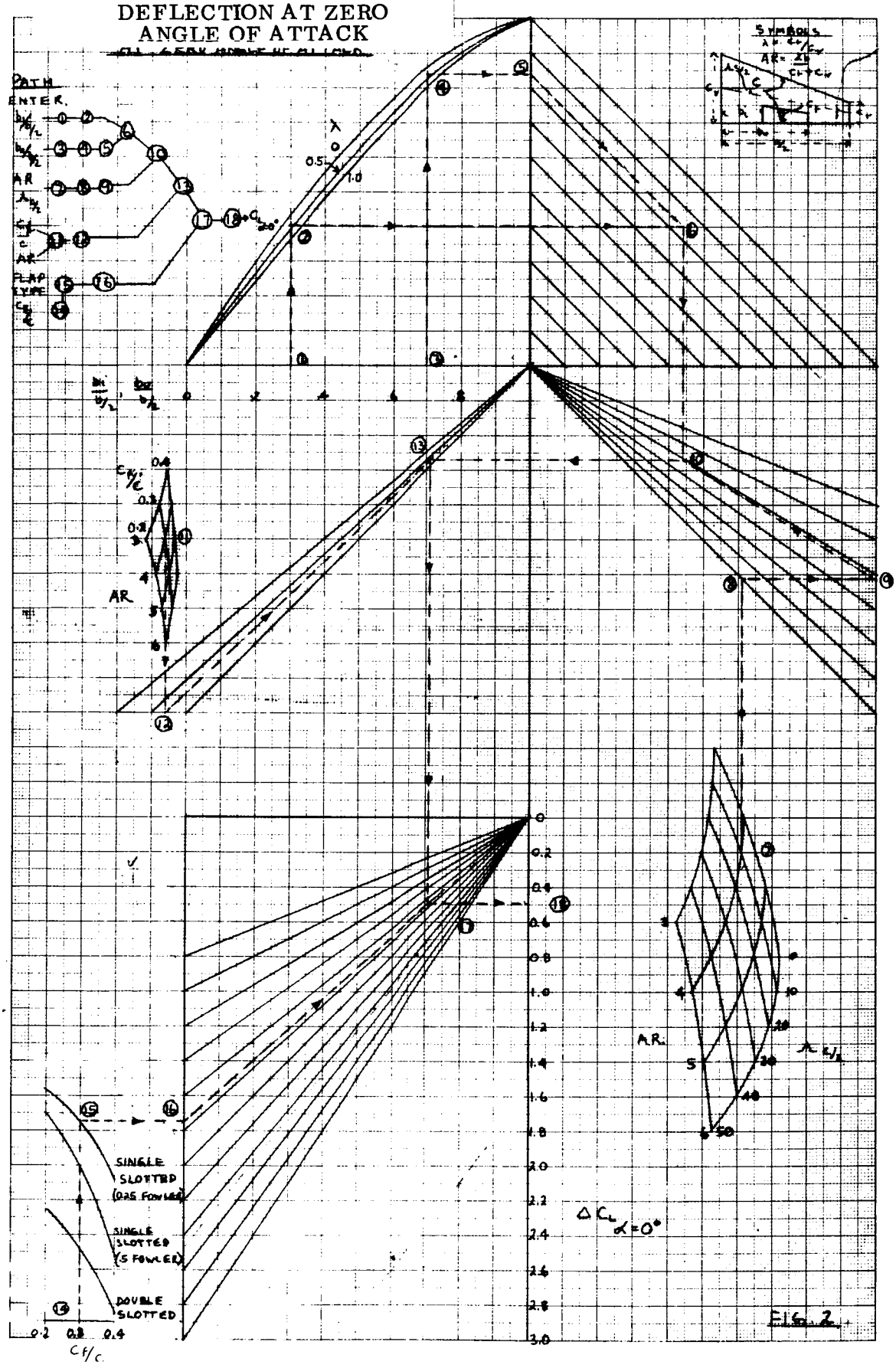


Figure III-7

The incremental lift coefficient due to flap deflection at zero angle of attack is computed along a question mark path starting clockwise on Figure III-7. The Optimum two-dimensional increment in lift coefficient is factored in for flap chord ratio of 0,2 to 0,4 at the lower left. These factors combine to provide the optimum incremental lift coefficient due to flap deflection at zero angle of attack (ΔC_{L_o}).

The incremental lift coefficient at other flap deflections ($\Delta C_{L_{\alpha = 0°,\ \delta = \delta_1}}$) at zero angle of attack may be obtained by a ratio (K_F) relative to the two-dimensional values.

ΔC_{L_1}	=	Two-dimensional lift coefficient increment at desired flap deflection and zero angle-of-attack
ΔC_{L_2}	=	Two-dimensional lift coefficient increment at optimum flap deflection and zero angle-of-attack
K_F	=	$\Delta C_{L_1} / \Delta C_{L_2}$
$\Delta C_{L_{\alpha = 0°,\ \delta = \delta_1}}$	=	$L_F \Delta C_{L_o}$

Values of ΔC_{L_1} and ΔC_{L_2} may be obtained from many sources, two of which are References 2 and 3.

(b) Maximum Lift Increment Due to Flap Deflection

The basic empirical methods of estimating $\Delta C_{L_{max}}$ due to trailing edge flap deflection for unswept wings given in References 4 and 5 have been modified and are presented in chart form on Figures III-8 and III-9. together with a correlation of the estimate with available data on swept configurations. These data are correlated in Reference 6.

The computation path begins at the top left of the chart with the input for taper ratio, inboard and outboard flap span stations. Following the indicated path, flap chord ratio, wing thickness chord ratio, flap deflection, and the wing mid-chord sweep angle are successively accounted for in arriving at the increment in maximum lift coefficient.

These data are correlated in Figure 6.

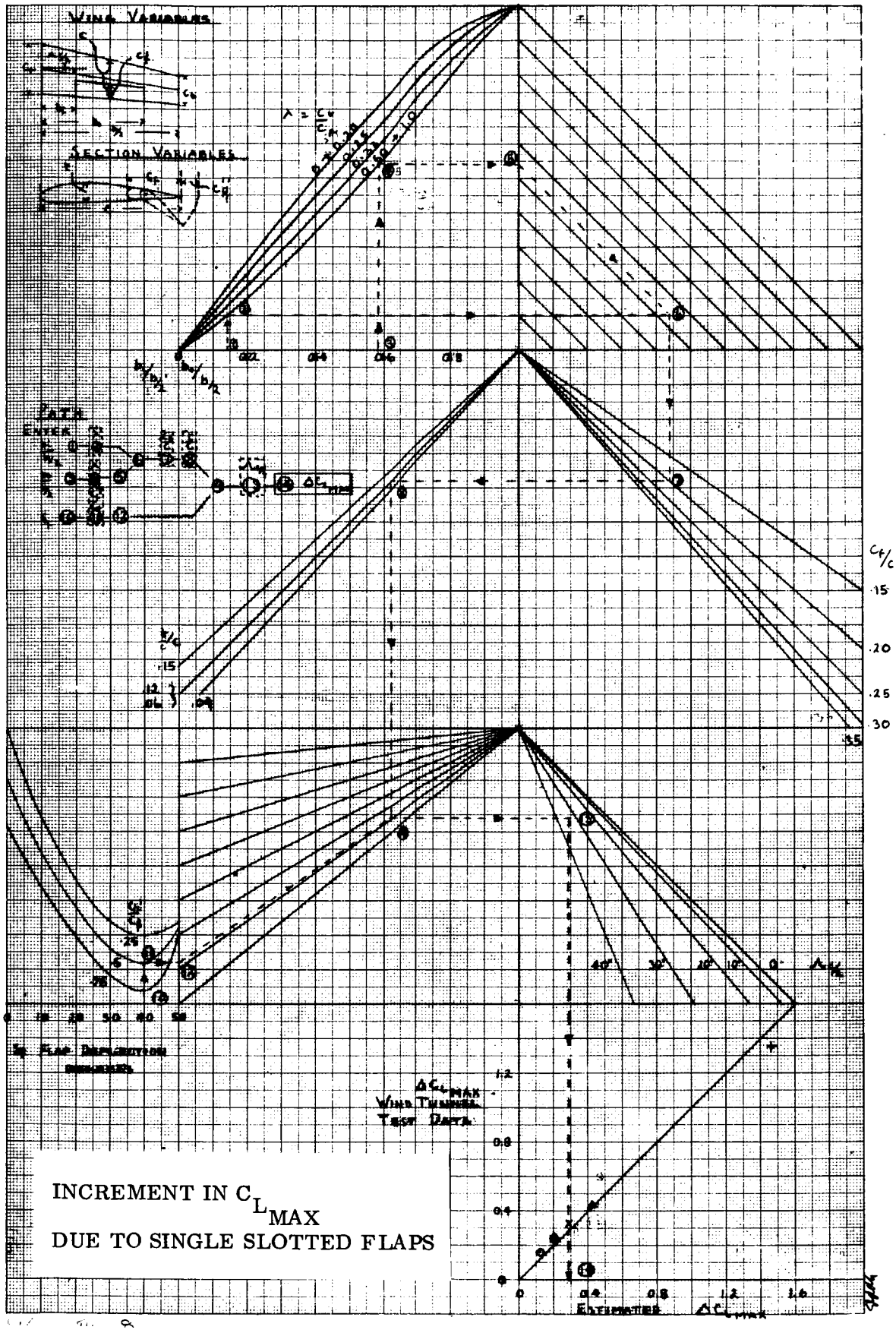

WING VARIABLES
SECTION VARIABLES
PATH ENTER
INCREMENT IN $C_{L_{MAX}}$
DUE TO SINGLE SLOTTED FLAPS
$\Delta C_{L_{MAX}}$ WIND TUNNEL TEST DATA
ESTIMATED $\Delta C_{L_{MAX}}$

Figure III-8

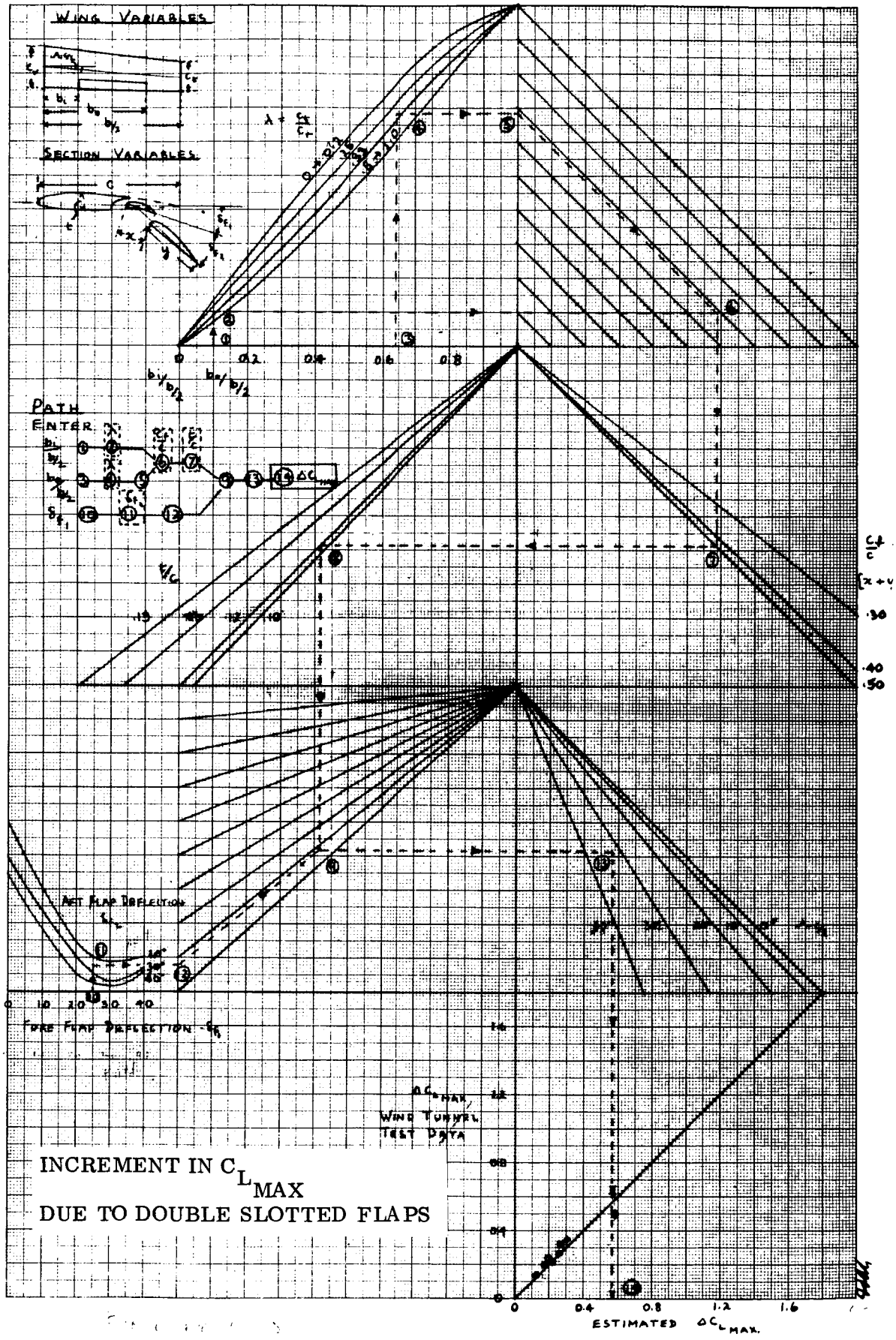

WING VARIABLES
SECTION VARIABLES
PATH
ENTER
AFT FLAP DEFLECTION
FORE FLAP DEFLECTION
WIND TUNNEL TEST DATA
INCREMENT IN $C_{L_{MAX}}$
DUE TO DOUBLE SLOTTED FLAPS
ESTIMATED $\Delta C_{L_{MAX}}$

Figure III-9

Fowler action is included in the term $\frac{C'f}{C}$ for the single slotted flap chart, and in $\frac{Cf}{C}$ for the double slotted flap chart. It can be seen that $\Delta C_{L_{max}}$ may be estimated to within 0.1, the only assumption being that the flap slot is near optimum and that aspect ratio effects are small.

(c) Lift Curve Slope Increment Due to Flap Deflection

The change in the lift curve slope for a full span slotted trailing edge flap is obtained from Figure III-10. The value is corrected for the total trailing edge flap span and the flap deflection.

$$\Delta C_{L_\alpha \text{Full Span}} = \Delta C_{L_\alpha (t/c)} + \Delta C_{L_\alpha (AR)} + \Delta C_{L_\alpha (\Lambda)}$$

$$\Delta C_{L_\alpha (\delta_F)} = \Delta C_{L_\alpha \text{Full Span}} \left(\frac{b_F}{b}\right)^{(1.0 - 0.6\,(b_F/b))} \left(\frac{1 - \cos \delta_F}{1 - \cos 50°}\right)$$

$$C_{L_\alpha} = C_{L_{\alpha (AR = 4.0)}} + \Delta C_{L_\alpha (\delta_F)}$$

where

b_F = span from left outboard flap chord to right outboard flap chord.

δ_F = trailing edge flap deflection

$C_{L_{\alpha (AR = 4.0)}}$ = Reference lift curve slope (AR = 4.0, t/c = 0.06) is listed below:

$\Lambda_{c/4}$, Deg	C_{L_α}, Deg^{-1}
20	0.0585
30	0.0562
40	0.0532

(4) Leading Edge Slats

A deflection of 30 degrees gives the maximum increment in lift for a leading edge slat. Aspect ratio and wing thickness ratio have

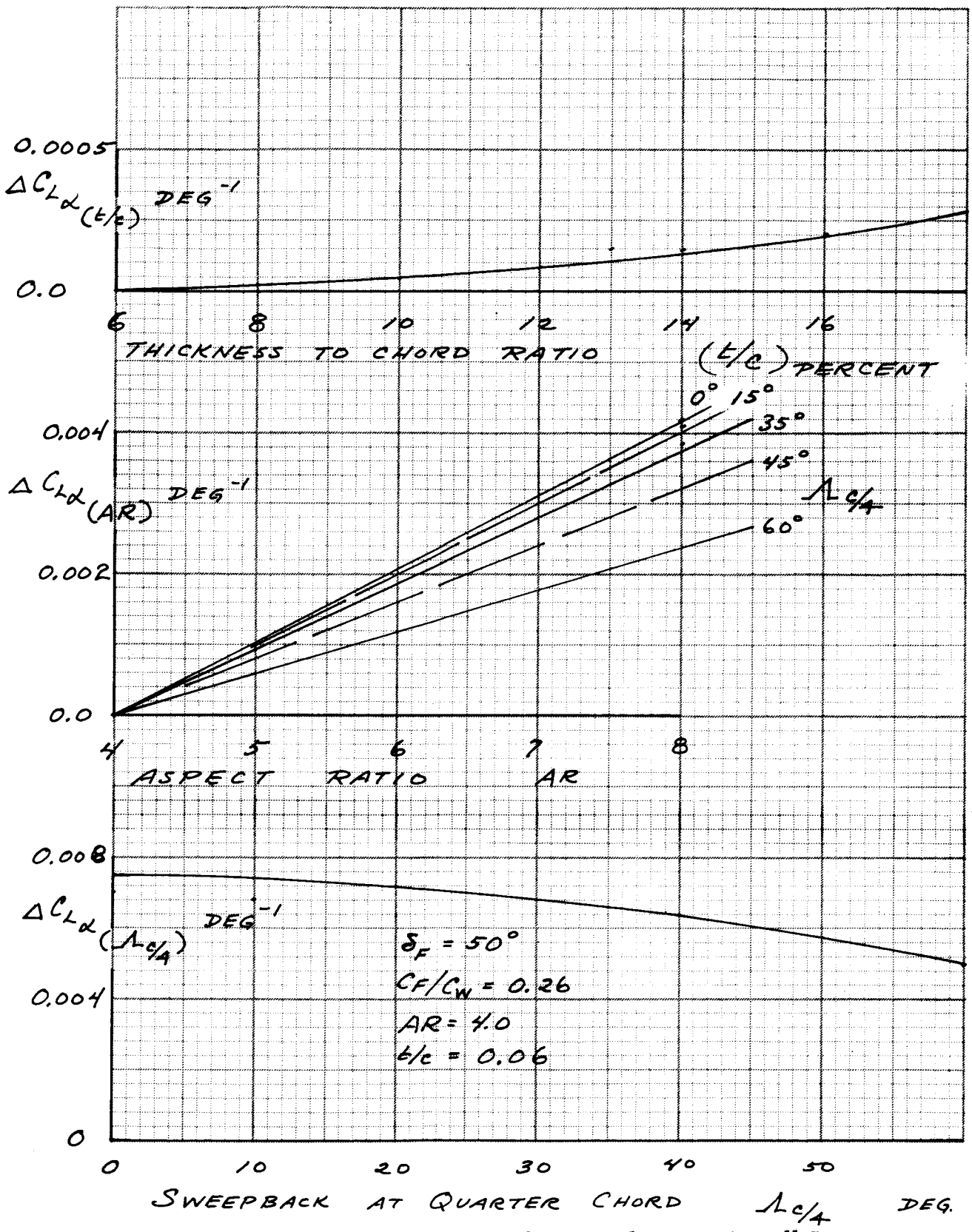


Figure III-10. Incremental Change in Lift Curve Slope Due to Full Span Slotted Trailing Edge Flaps Deflected 50 Degrees

negligible influence on this value. The lift curve slope increment is affected by the sweep angle of the wing quarter chord locus, wing aspect ratio, wing thickness ratio, and the ratio of slat span to wing span.

The incremental lift due to slats at zero angle-of-attack is zero. The increment increase in the maximum lift coefficient is obtained from Figure III-II based on the sweep angle of the wing quarter chord and the slat span.

The change in the lift curve slope for a full span slat is 0.55 the change in the lift curve slope due to the trailing edge flaps. This value is reduced by the 1.4 power of the ratio of slat span to wing span.

$$\Delta C_{L_{\alpha_{Slat}}} = (0.55\, \Delta C_{L_{\alpha_{Flap}}}) \left(\frac{2 b_s}{b}\right)^{1.4}$$

where:

b_s = span from inboard slat chord to outboard slat chord

The preceding analysis is also presented in Reference 7. The method has given reasonable correlation with data.

(5) Nose Droop

The effect of nose droop was evaluated in Reference 6 for a number of swept wing configurations. As no simple correlations were apparent, the nose droop was assumed to correspond to a given percentage camber in the wing. This corresponded reasonably well with the tabulated data for the trailing edge flap angles and aspect ratios under study.

(6) Lift Spoilers

Negative lift increments due to spoiler action may be estimated using Reference 8; or by regarding the spoiler as a split flap and using the data of Reference 4. Using either approach a spoiler extending from the fuselage side to the span station .6 b/2, located at the 70% chord line and with a 15% chord,

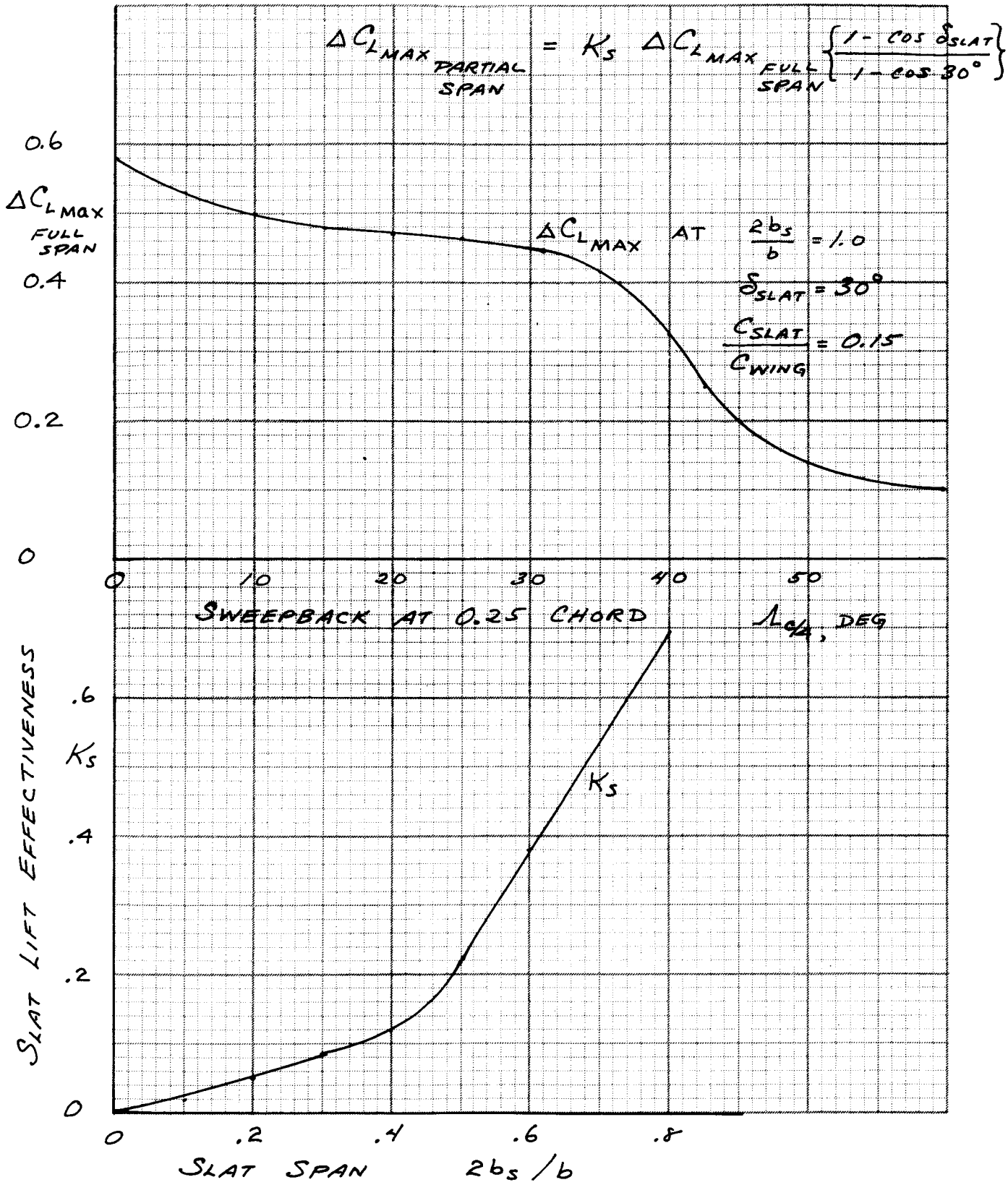


Figure III-11. Maximum Lift Increment Due to Leading Edge Slats

when deflected 60° up would provide a lift loss of $\Delta C_L = .3$ on Concept C. A spoiler of this size would require to be segmented and perforated over the outer edge of the spoiler to prevent undesirable buffet.

(7) Lift-Drag Polar

(a) Induced Drag Due to Lift

The coefficient of drag due to lift is expressed by

$$C_{D_i} = K \left\{ C_L - C_{L_b} \right\}^2 + J \left\{ C_L - C_{L_b} \right\}^4$$

where:

$K = \dfrac{1}{\pi \, AR \, e}$

$J = J \left\{ K, t/c \right\}$ (see Figure III-7)

e = airplane efficiency factor

Examination of the airplane efficiency factor, e, shows that the majority of flight test data fall within ±6% of the 0.80 level. Therefore, it is concluded that a value of 0.80 for the airplane efficiency factor will give the drag due to lift within ±6%.

Efficiency factors were also obtained for a sizable number of low-thickness-ratio wing-body configurations tested in wind tunnels. Figure III-12 is a plot of the values versus a leading-edge-radius Reynolds number parameter Ω defined as

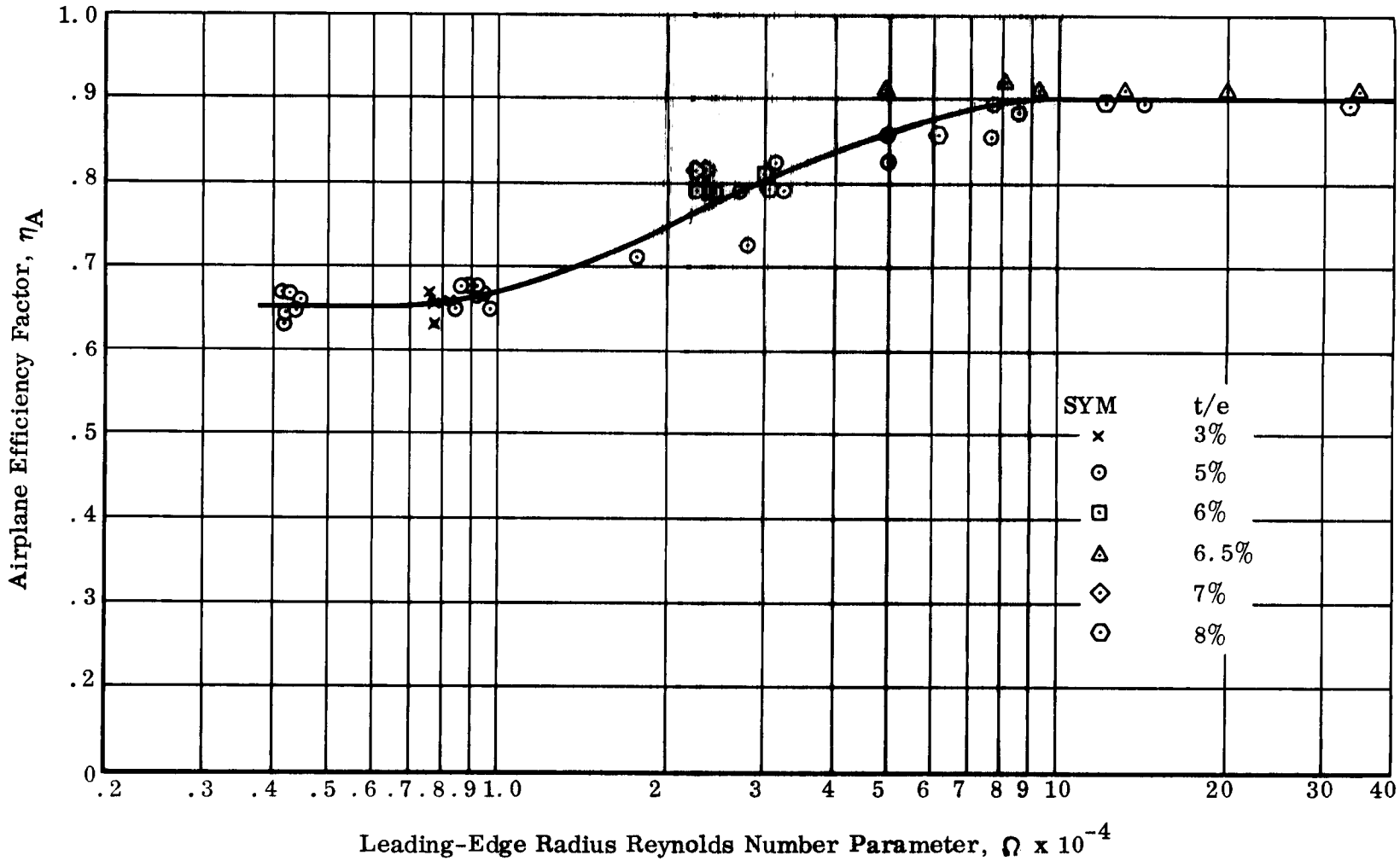
$$\Omega = R_e^* \left\{ 1 + \frac{R_e^*}{45,000} \left(1 - M^2 \right) \right\}$$

where:

R_e^* = Reynolds number based on leading-edge radius

For nearly all rounded-leading-edge wings, regardless of planform or thickness ratio, the leading-edge-radius Reynolds-number parameter, Ω, for full-scale aircraft has a value greater than 10^5. Consequently (Figure III-12), a value of $e = 0.8$ was chosen for all airplane configurations using rounded-leading-edge airfoil sections.

Efficiency Factor Versus Leading-Edge Radius
Reynolds Number Parameter



$$\Omega = R_{e\,(L.E.\ Radius)}^{*} \times \left[1 + \frac{R_e}{45000} (1 - M^2) \right]$$

Figure III-12

The last quantity of the equation to be evaluated is J. By analyzing a large amount of wind-tunnel data it has been determined that the data for the various configurations may be divided into two classes identified by thickness ratio. These two classes of configurations are (1) those that have "thin" airfoil sections, i.e., $t/c \leq 7\%$, and (2) those that have "thick" airfoil sections, i.e., $t/c \geq 8.5\%$. The results of the analysis also indicate that for the thin-wing cases J = K, and for the thick-wing cases J = O.

C_{L_b} has been determined from an analysis of NASA data and is presented in minor subheading (c) of this section.

(b) Minimum Drag Increment Due to Flap Deflection

The minimum drag of the airplane in the flaps up condition is given in Section III. C. 1. c. Flap deflection will increase the airplane profile drag. An analysis of data for both single and double slotted flaps is given in Reference 6. The data is presented on Figures III-13 and -14 as a function of flap span and the flap deflection angle. The data shows good correlation and collapses to a single curve when presented as a function of $\frac{\Delta C_D}{1-\cos\delta_F}$. Comparison of Figures III-13 and -14 shows that profile drag associated with double slotted flaps is 10% greater than that of single slotted flaps.

(c) Lift Offset Increment Due to Flaps

The lift coefficient is offset and not zero at the minimum drag coefficient for all asymmetric airfoil profiles as a result of camber, flap deflections, nose droop, a spoiler actuation. The location of this offset is assumed to agree with a parabolic lift-drag polar and is defined as C_{L_b}. This is the value used in the induced drag formula.

C_{L_b} has been determined from an analysis of NASA data and is presented on Figure III-15.

(d) Comparison with NASA Data, Aspect Ratio 10, Double Slotted Flap

In the re-evaluation of the lift-drag polar analysis, the procedure was applied to an unknown airfoil whose basic data for the flaps undeflected was known. This wing data was supplied by NASA Langley.

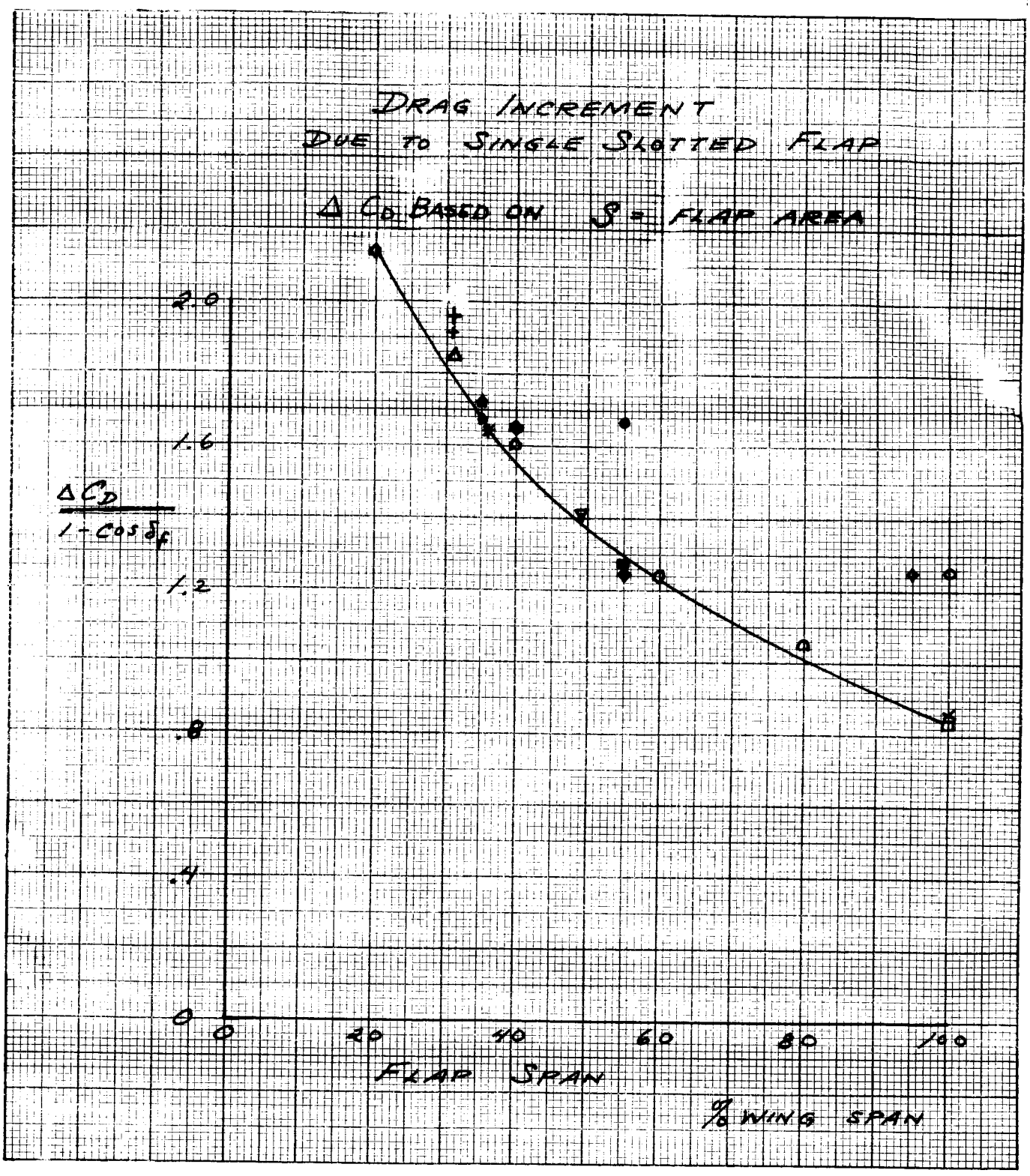


Figure III-13

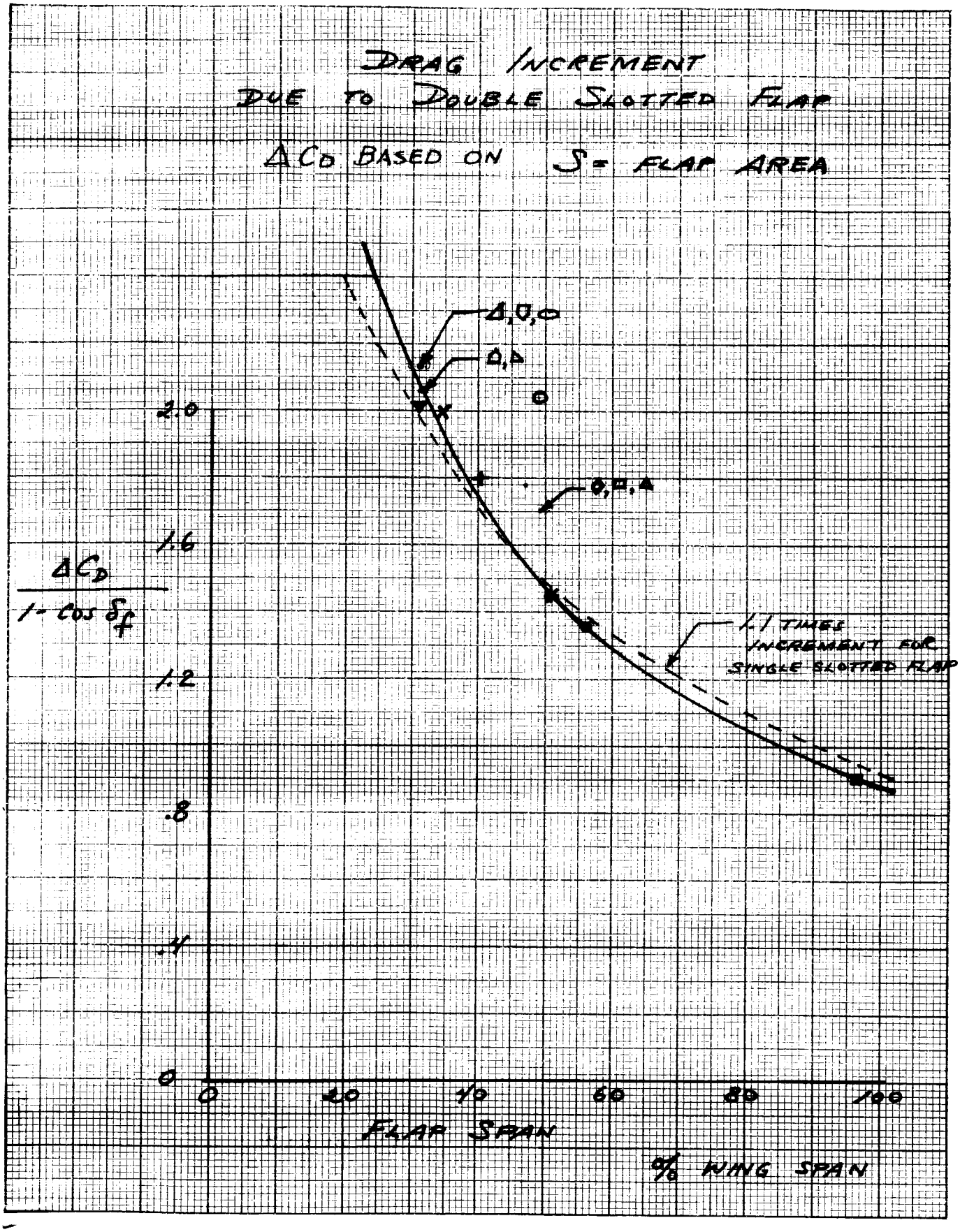


Figure III-14

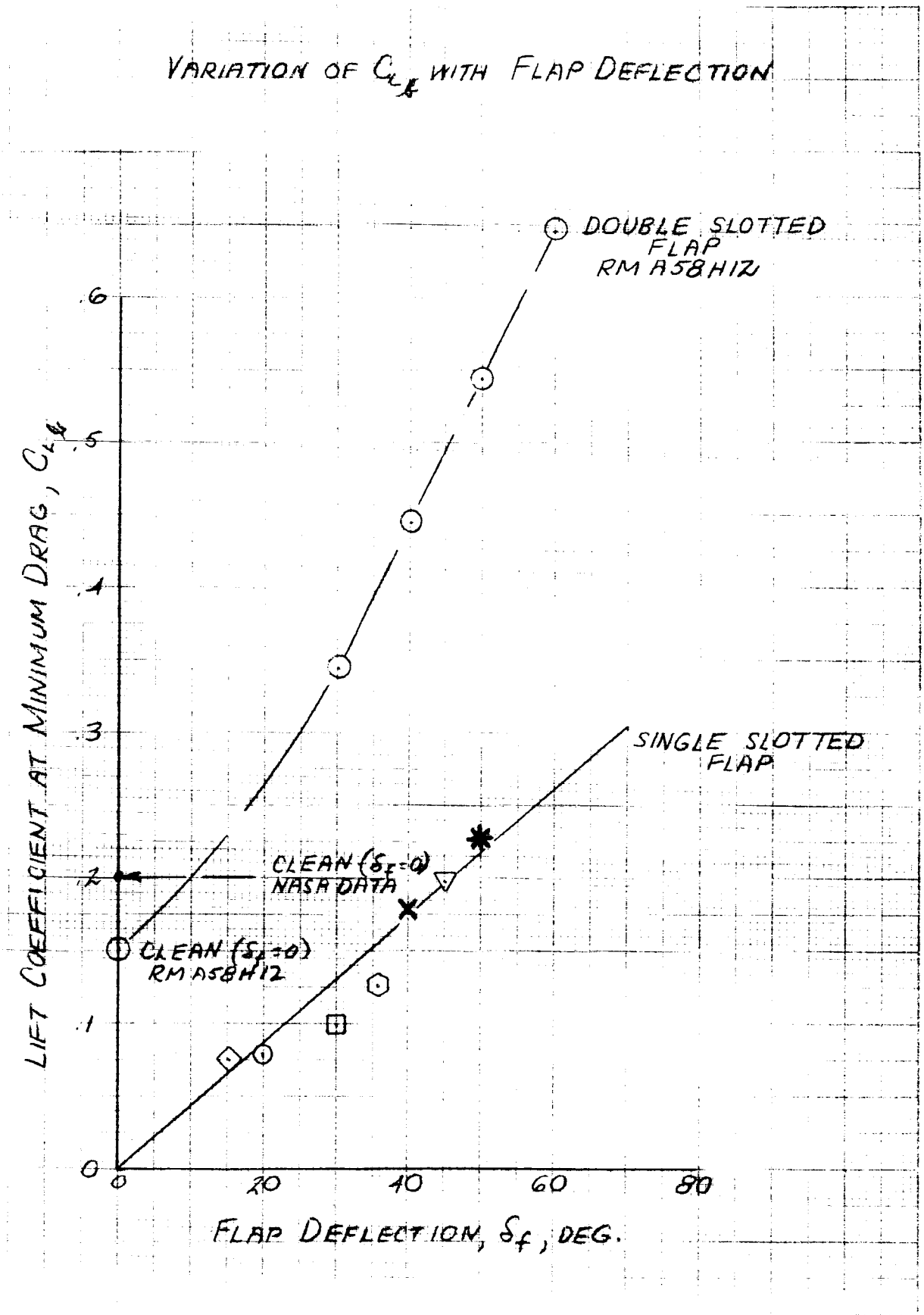


Figure III-15

The wing has an aspect ratio of 10, with double slotted flaps over the full span including an arrangement under the body. The body was assumed to have a split flap with a body width to wing span of 0.10. This data is reproduced on Figure III-16.

The variation of lift coefficient offset (C_{L_b}) as determined for a double slotted flap was obtained from NACA RM A58H12 and presented in Figure III-15 . The increment in minimum drag was determined from Figure III-17 . The ratio of wing area to flap area for the split flap is 30.0 and for the double slotted flap, this is 3.3.

The resulting values used in the lift-drag polar are presented.

TABLE III-2. NASA DOUBLE SLOTTED FLAP PARAMETERS

Flap Deflection (degree)	0	30	40	50
C_{L_b}	0.150	0.345	0.445	0.545
$\Delta C_{D_{min}}$ due to flaps	0	0.0536	0.0936	0.1428
$C_{D_{min}}$	0.0185	0.0721	0.1121	0.1613

The above values used with a wing efficiency of 0.80 for a thick cambered airfoil will provide data that are within 5 percent of the experimental data. This error is less near the optimum angle-of-attack and increases as the angle-of-attack produces separation.

c. Effects of Compressibility

(1) Maximum Lift Coefficient

The lift coefficient is an integrated value normal to the flight path of the pressures over the upper and lower surfaces of the vehicle and ratioed to the free air dynamic pressure. This process does not consider the negative differential pressure limit at vacuum. This process also does not consider the compressibility of air. To compensate for these two phenomena, the maximum lift coefficient decreases with Mach number with reference to the vacuum line and increases with Mach number with reference to compressibility. The resulting values are presented on Figure III-18.

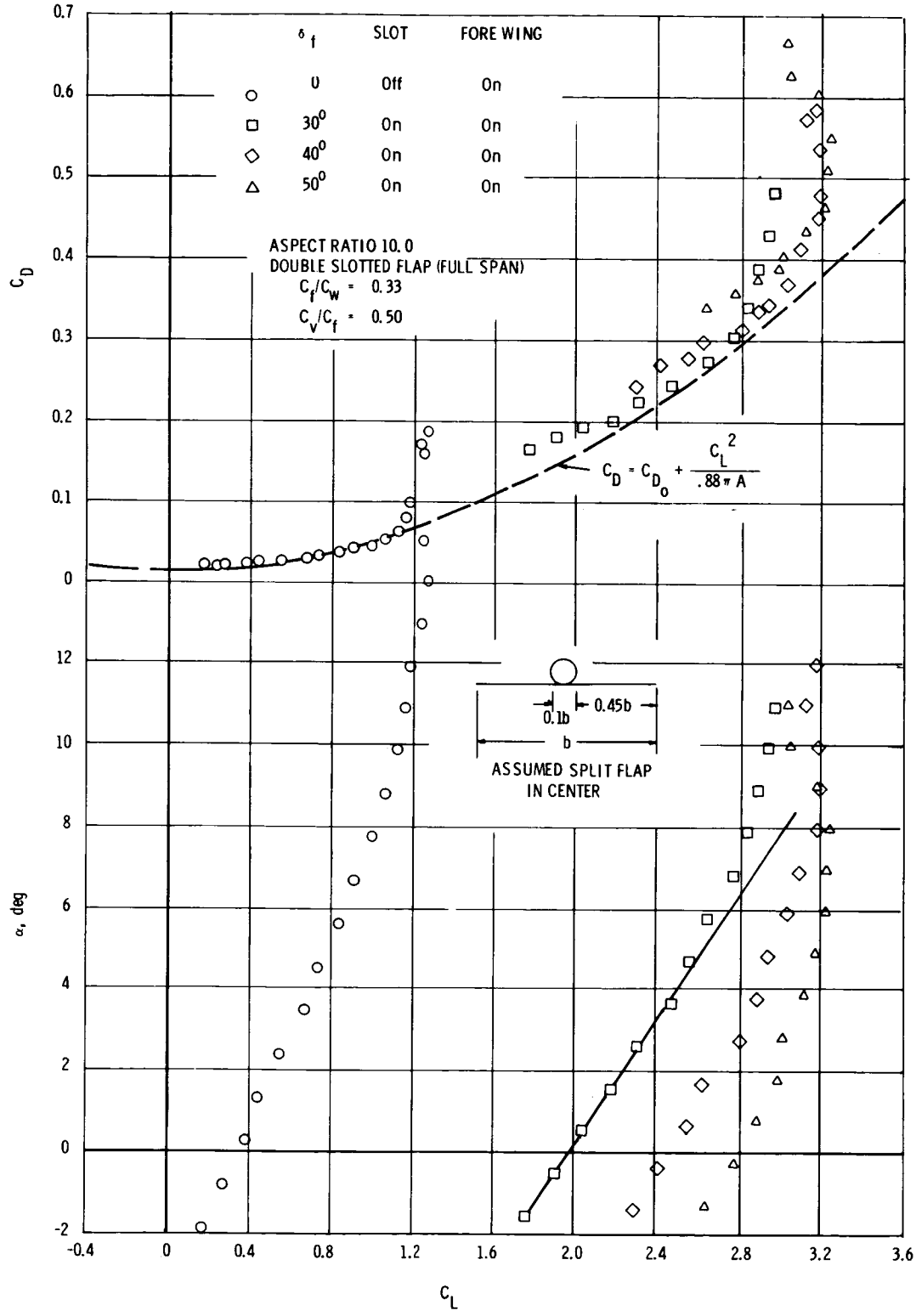


Figure III-16

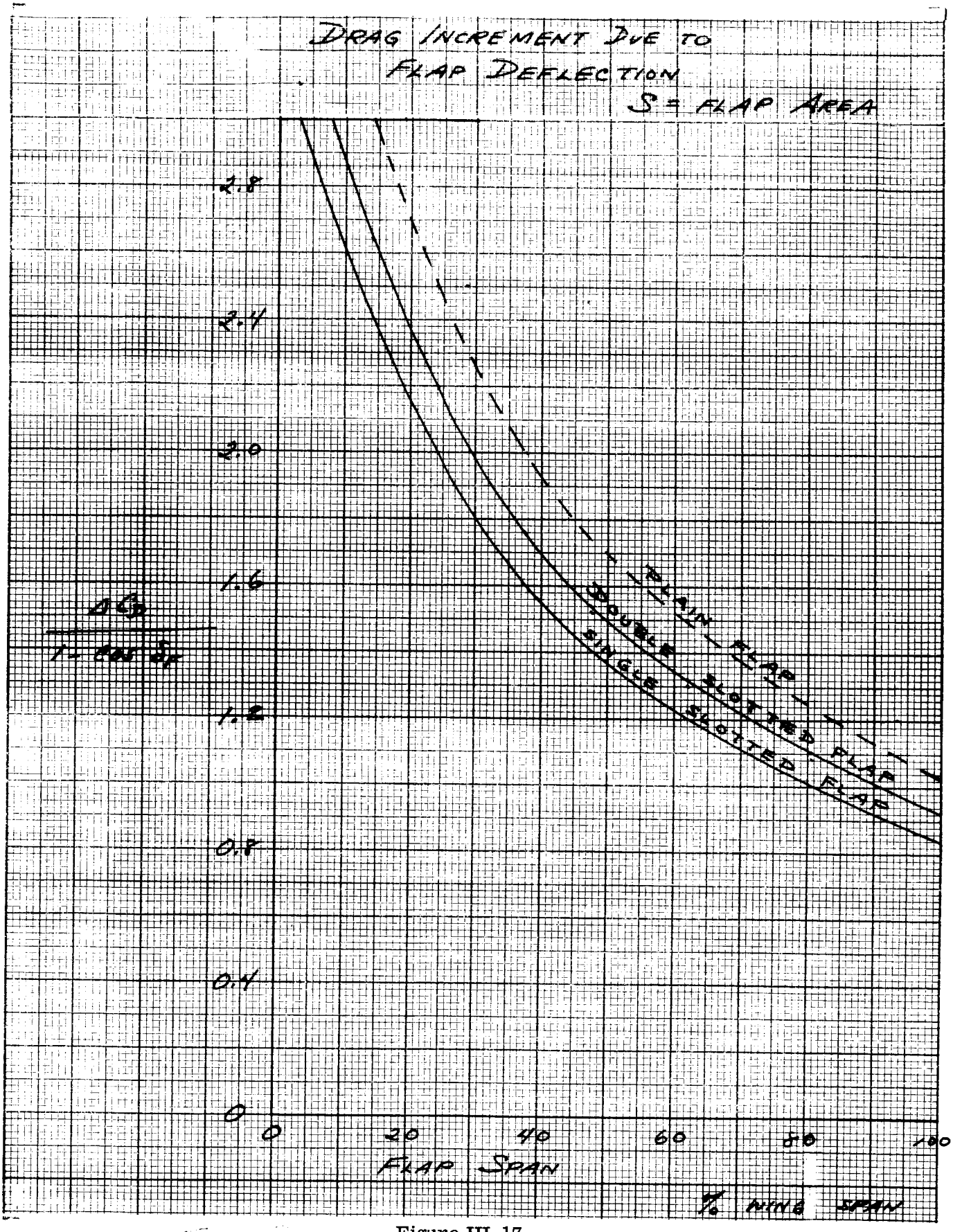


Figure III-17

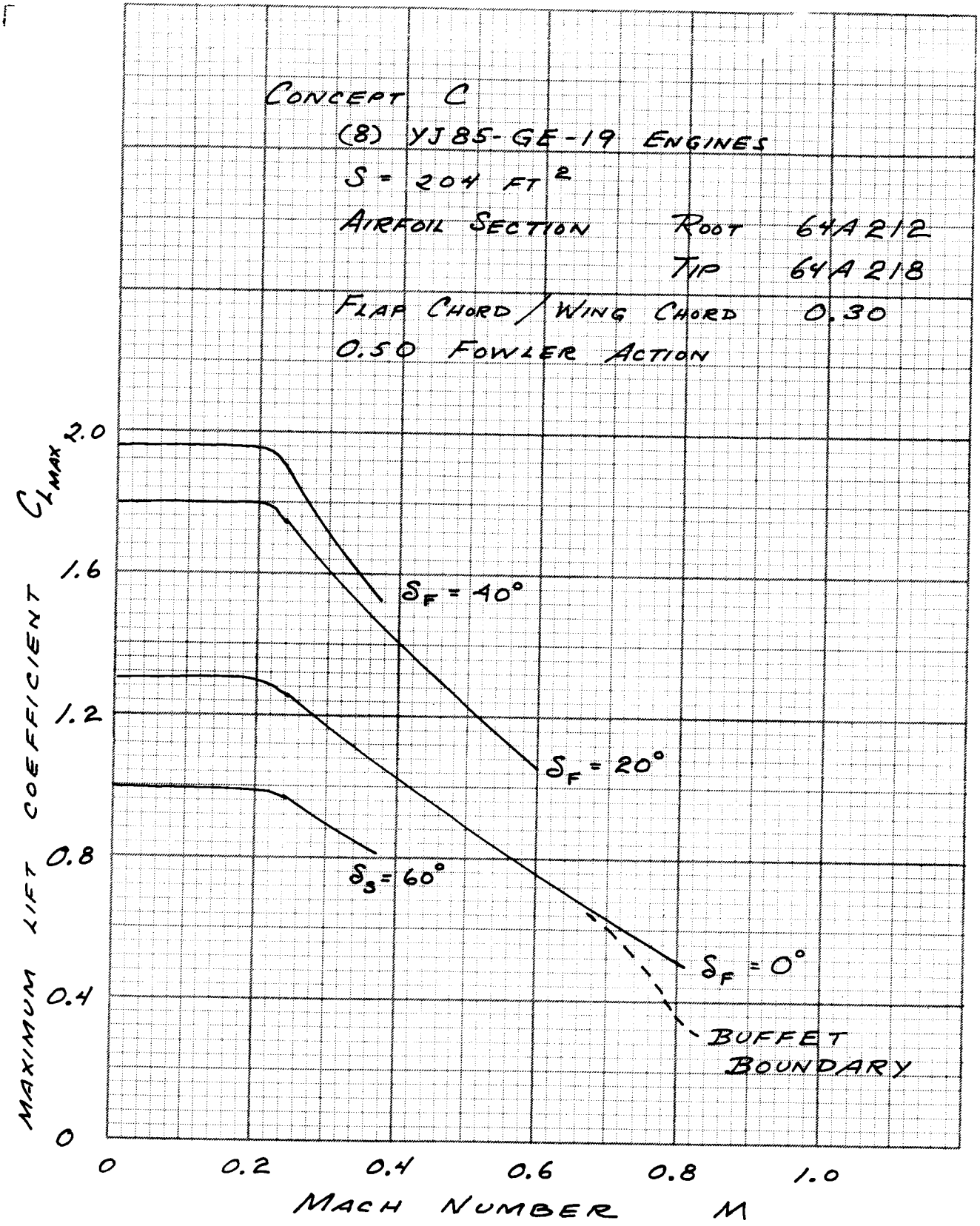


Figure III-18. Effect of Mach Number on Maximum Lift Coefficient

(2) Lift Curve Slope

In incompressible flow, the lift curve slope in two-dimensional flow has been tacitly assumed as 2π per radian with slight variations due to the trailing edge angle of the airfoil section. For wings with aspect ratios greater than three, an empirical formula is applied which is sufficient for engineering requirements.

$$C_{L_\alpha} \simeq 2\pi \frac{1}{p + (2/AR)}$$

where

P	=	semiperimeter of the wing
b	=	wing span
S	=	wing area
AR	=	b^2/S
p	=	P/b

Low aspect ratio wing theory is obtained from the analysis of a long flat triangular wing traveling point-foremost.

$$C_{L_\alpha} = 2\pi \frac{b^2/4}{S}$$

In an elastic compressible fluid, the incompressible streamline pattern is modified by the expansion of the volume of fluid passing through low pressure regions and by the compression occurring in the high pressure regions. The expansion of the stream tube in the low pressure region forces the streamlines outward with a net curvature increase and a greater disturbance normal to the airfoil section. The effect of compressibility is thus an increase in the horizontal perturbation velocities by the Prandtl-Glauert correction factor, $1/\sqrt{1-M^2}$. As the stream Mach number approaches one, the correction factor approaches infinity because the perturbation velocity is no longer small relative to the free stream velocity. The Prandtl-Glauert formula will show increasing departures from reality at some Mach number below unity, the exact value of which depends on the thickness or angle-of-attack of the airfoil.

$$C_{L\alpha(M)} = C_{L\alpha} \frac{1}{\sqrt{1-M^2}}$$

In the transonic region, the lift curve slope peak is empirically approximated by a value obtained by low aspect ratio wing theory.

$$C_{L\alpha(max)} = 2\pi \frac{b\ 2/4}{S} \quad \text{in transonic speed range, all wings}$$

The variation of the lift curve slope with Mach number is presented in Figure III-19.

3. References

1. Fluid-Dynamic Drag, S. F. Hoerner, Midland Park, N.Y., 1958.
2. Theory of Wind Sections, Ira H. Abbott, Albert E. Von Doenhoff, Dover Publications, 1959.
3. A Correlation of Two Dimensional Data on Lift Coefficient Available with Blowing, Suction, Slotted and Plain-Flap High-Lift Devices, J.M. Riebe, NACA RML55D29a, 1955.
4. Unpublished Data: Notes on High Lift Devices from Fairchild Hiller Transport Division.
5. Aerodynamic Characteristics of Flaps - A.D. Young, ARC Tech. Report No. R&M 2622.
6. Part II - Technical Data Substantiation Report - V/STOL Jet Operations Research Airplane Design Study, Report No. FHR 3391-3I, 7 April 1967.
7. Aircraft Preliminary Design Methods Used in the Weapons Systems Analysis Division, Bureau Navweps, Report No. R5-62-13 dated June 1962.
8. USAF Stability and Control Datcom.

D. POWER REQUIRED

1. Method

The thrust required is equal to the drag in level conventional flight. At the cruise altitude, past experience was used to reduce the number of speeds selected in obtaining the maximum range. The power required is graphically presented in Figures III-20 to III-27 for Concept C and III-28 to III-35 for Concept J.

2. Graphical Presentation

a. Probable Weight

The probable weight was studied in 3000 to 4000 pound increments from approximately zero fuel to full internal fuel.

b. Probable Altitudes

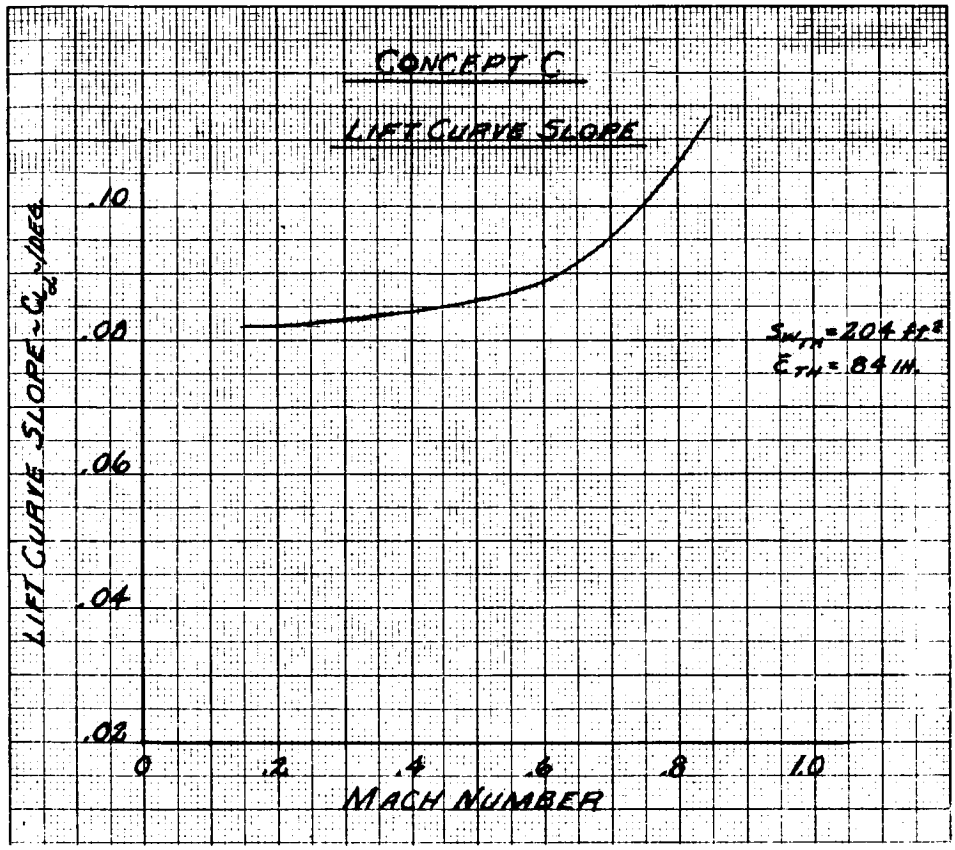
Altitudes from sea level to 36000 ft. were studied in 5000 ft. increments.

c. Speeds from Maximum Endurance to Maximum Speed

To bracket the speeds from maximum endurance to maximum speed, all anticipated speeds were checked from Mach number 0.20 to 0.90 below 25,000 ft. and 0.40 to 0.90 above 25,000 ft. altitude.

E. POWER AVAILABLE

The power available was interpolated from the engine data presented in Section IV for the particular speed, altitude, and temperature.



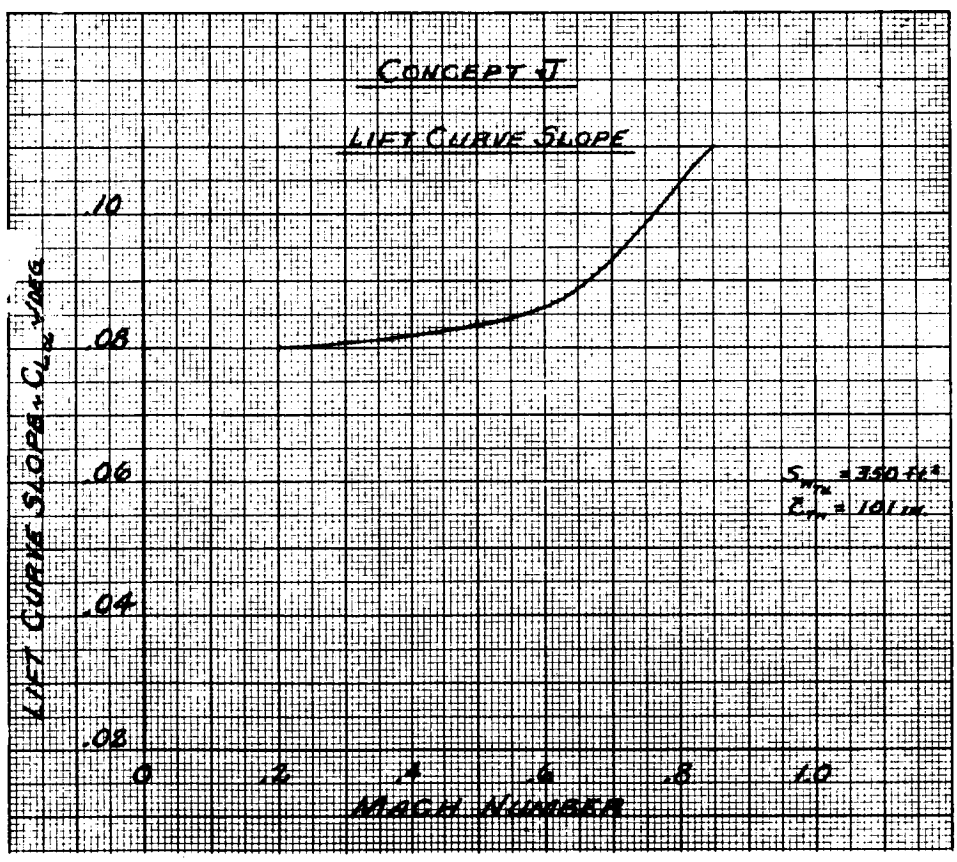


Figure III-19

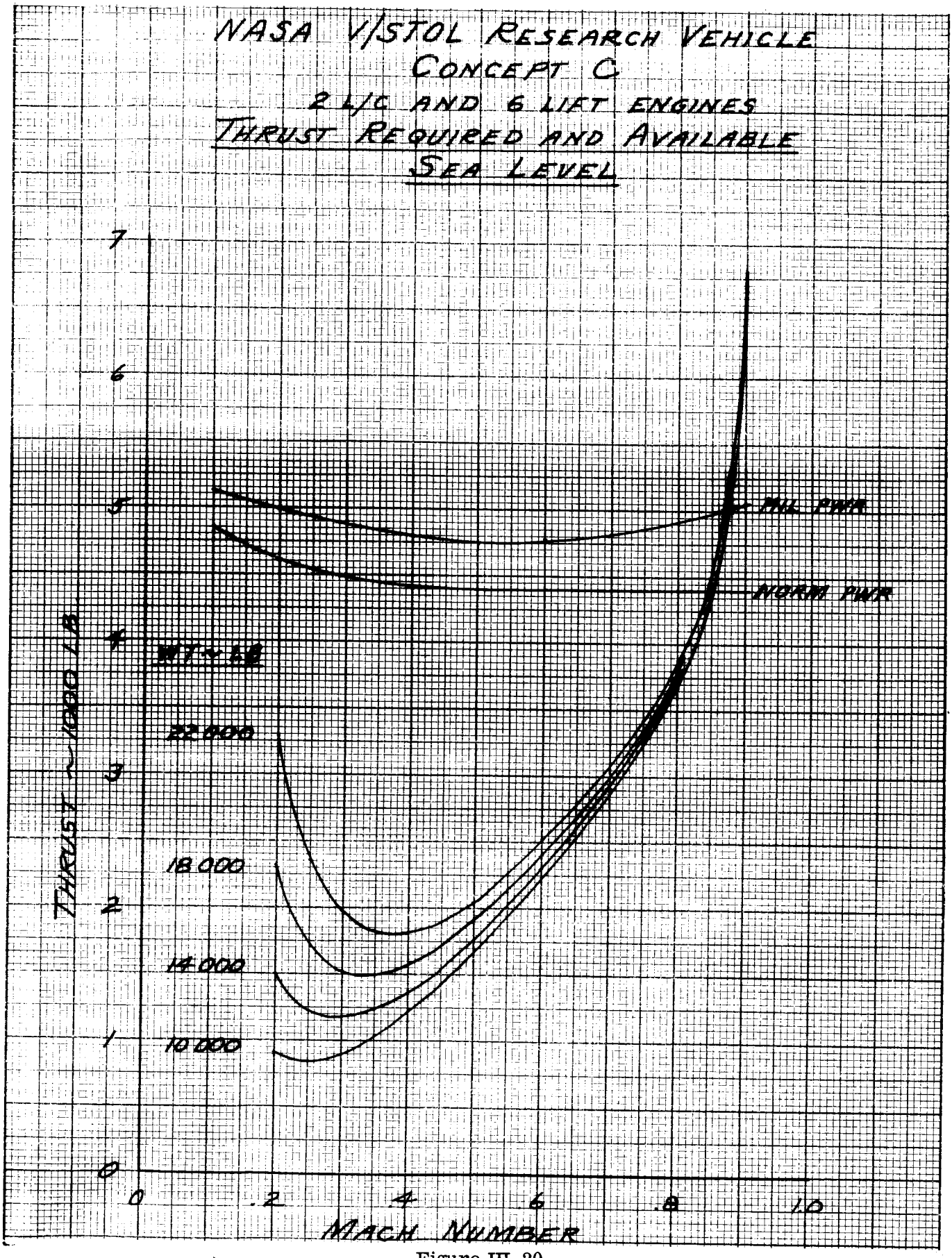


Figure III-20

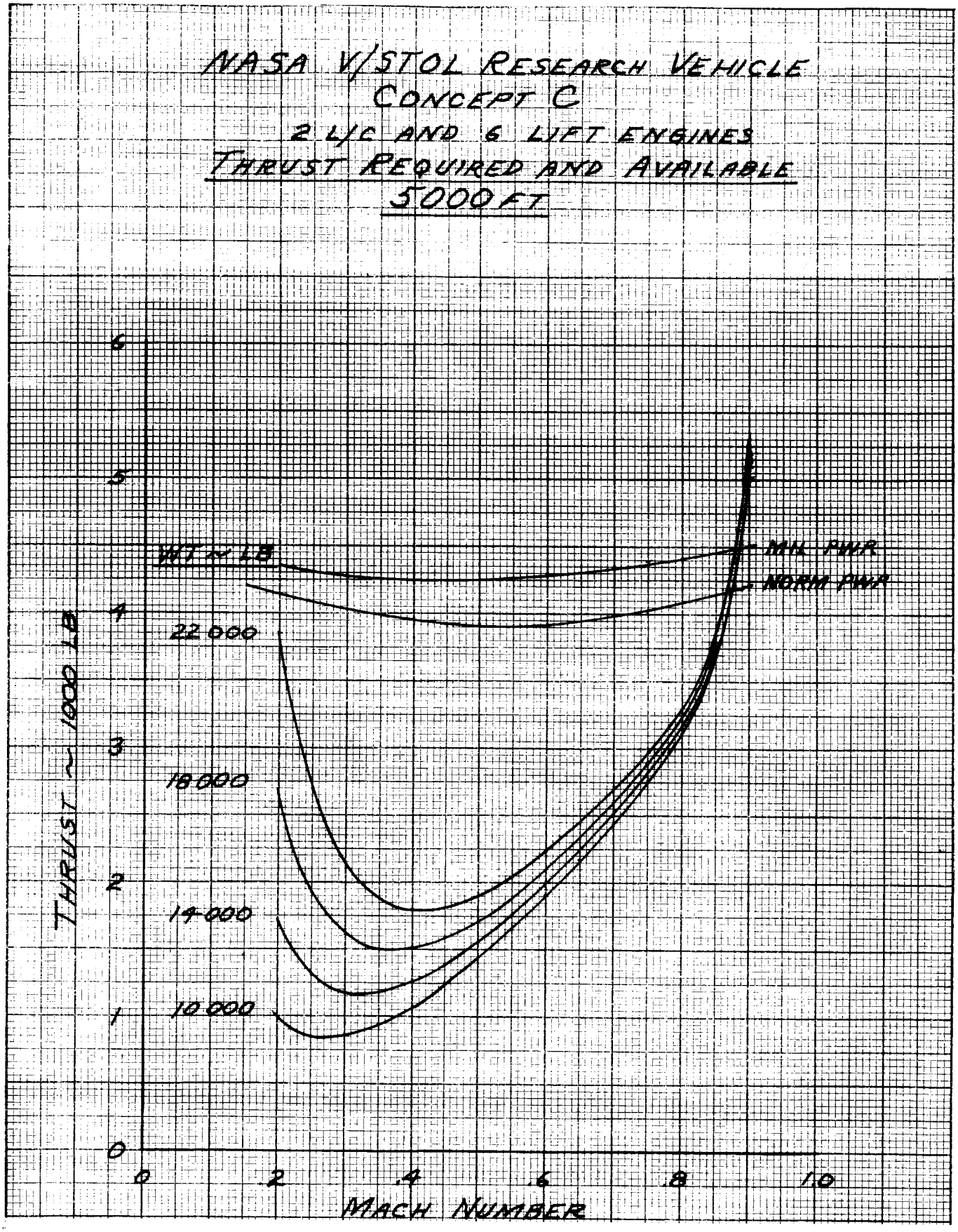


Figure III-21

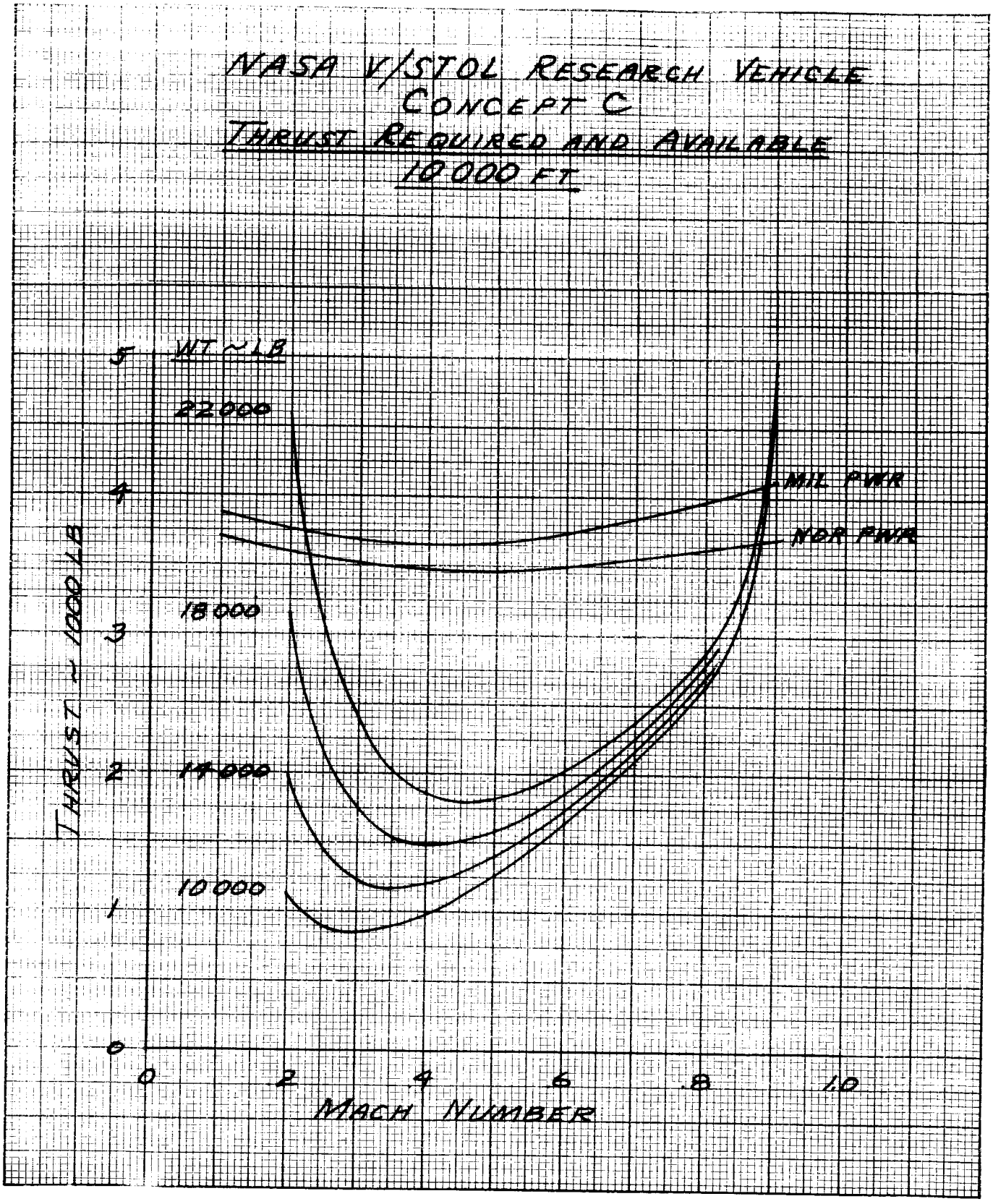


Figure III-22

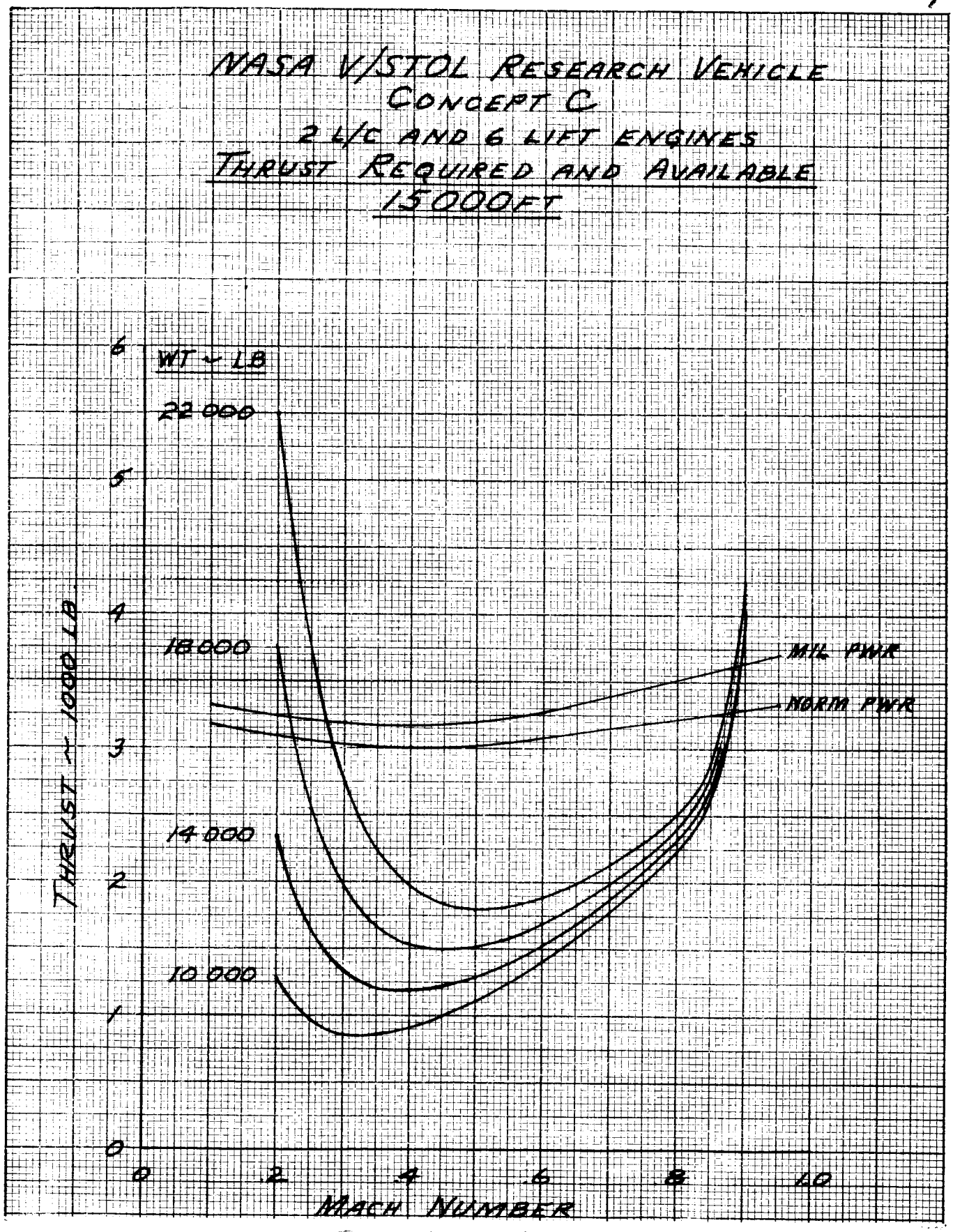
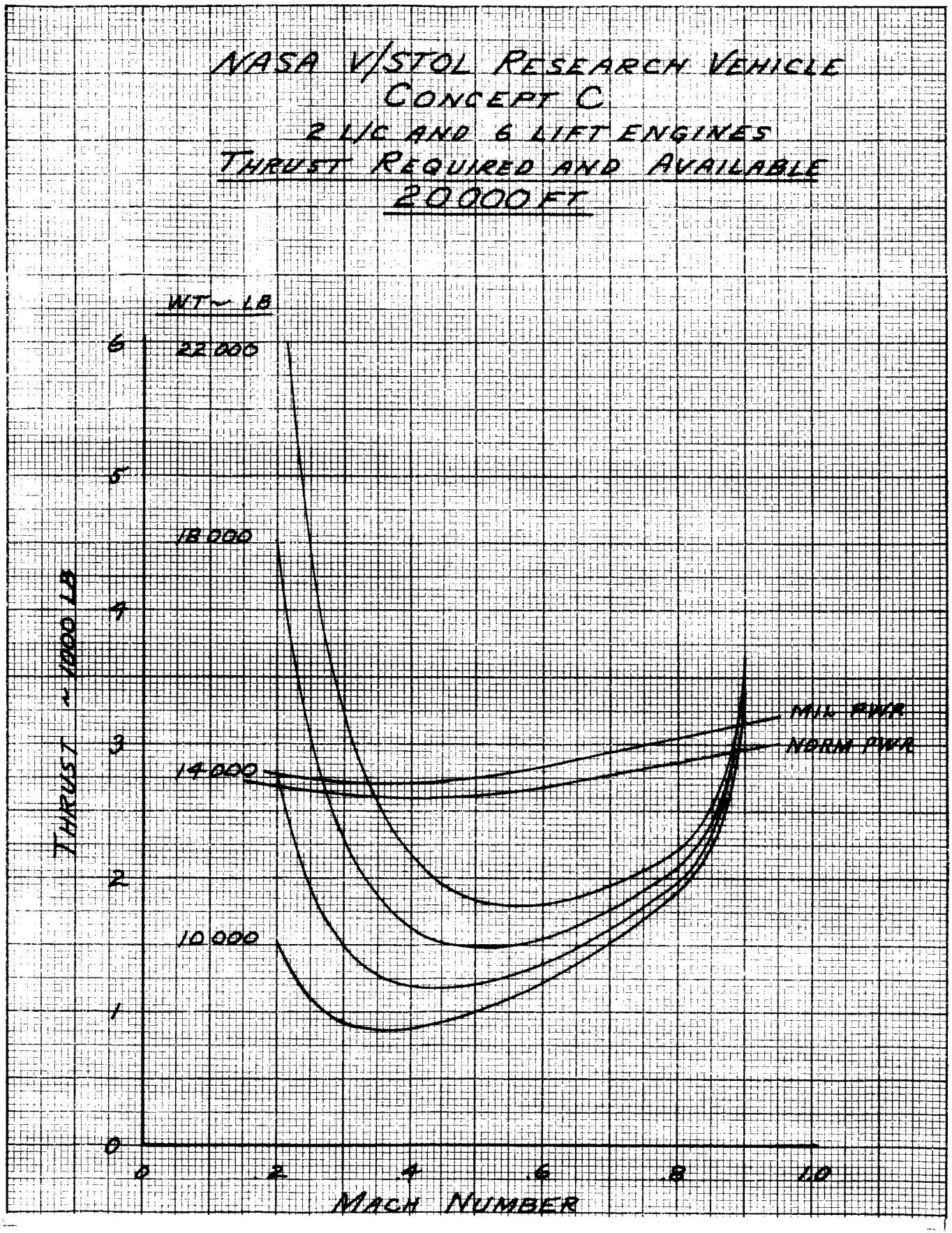


Figure III-23



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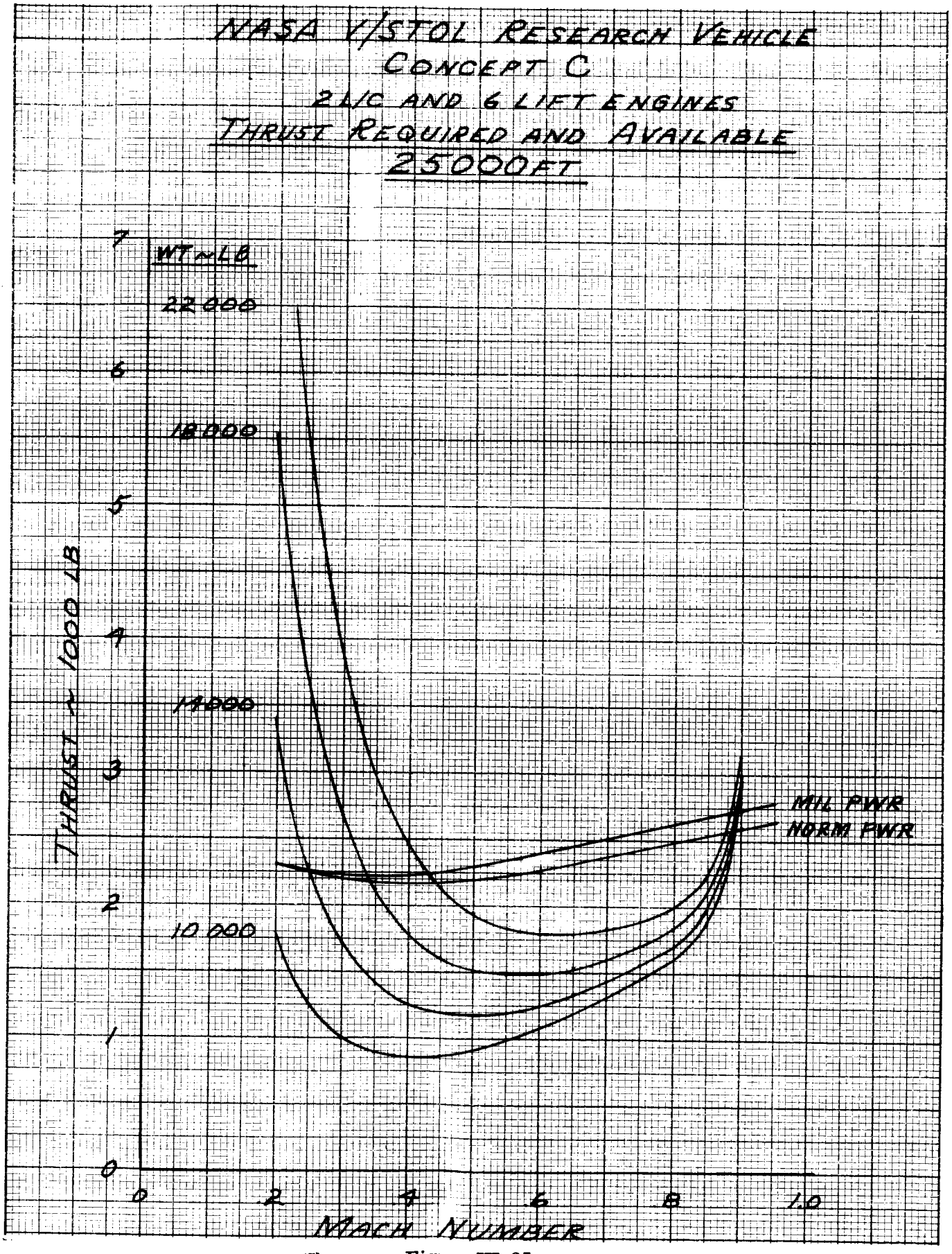


Figure III-25

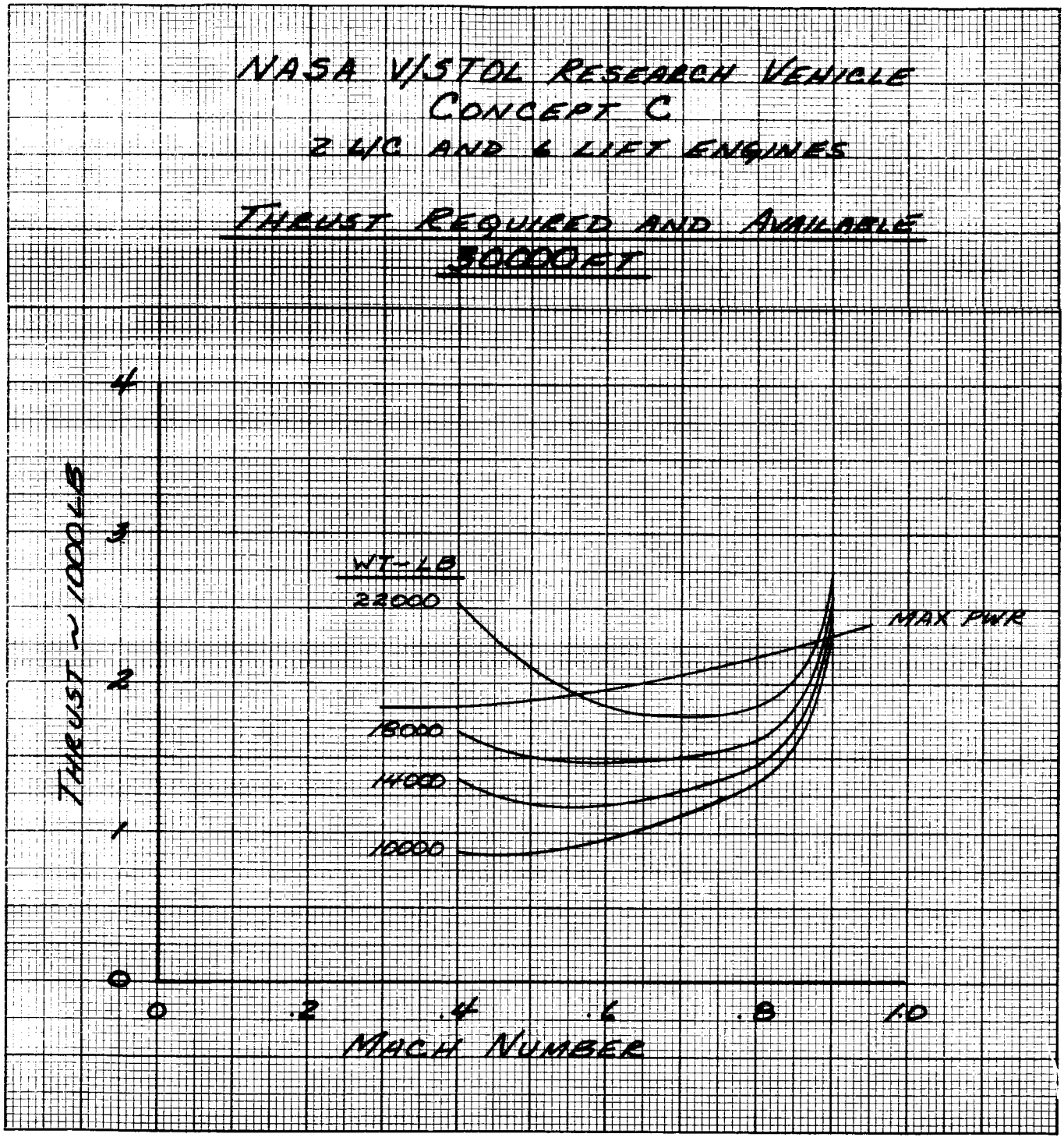


Figure III-26

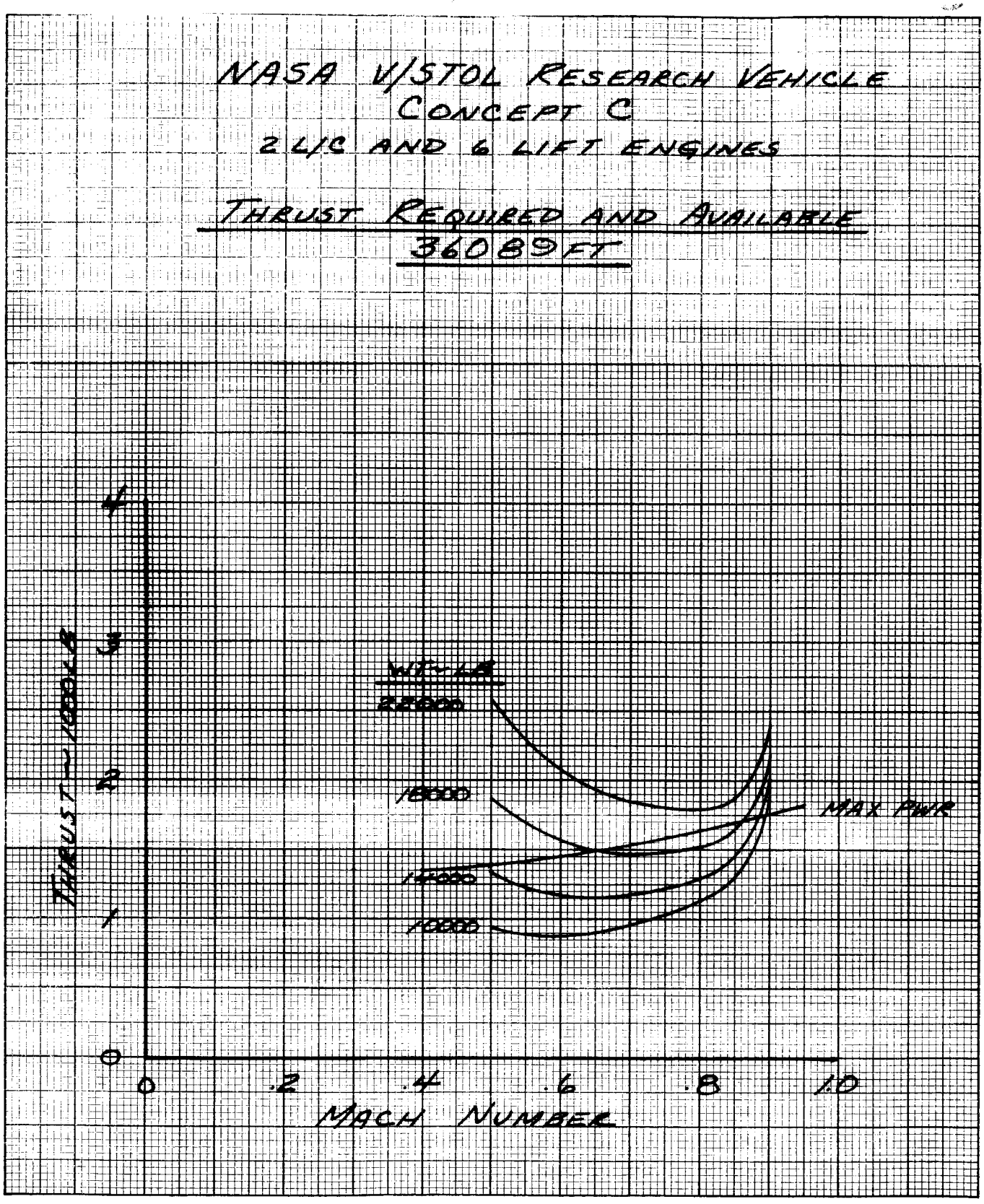


Figure III-27

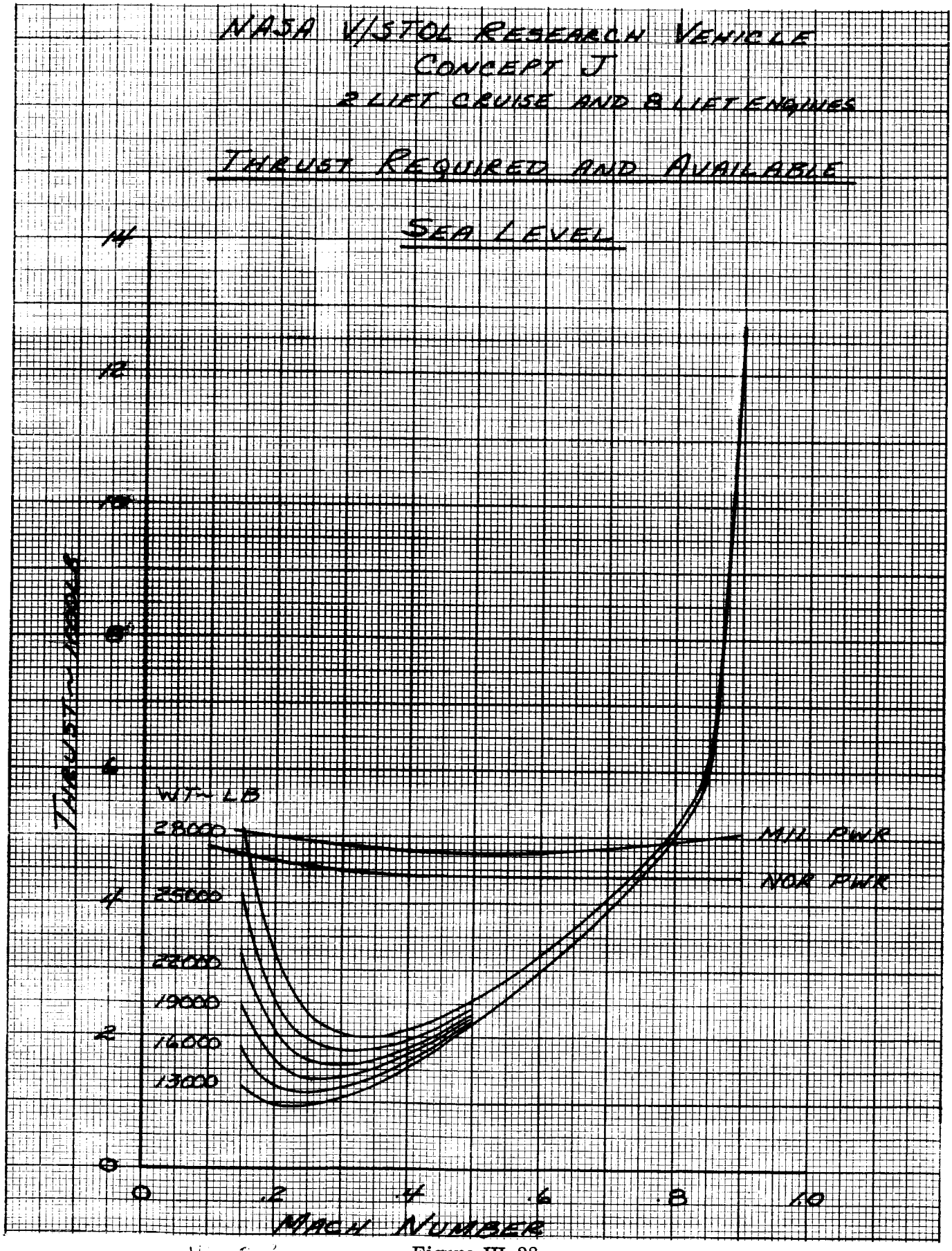


Figure III-28

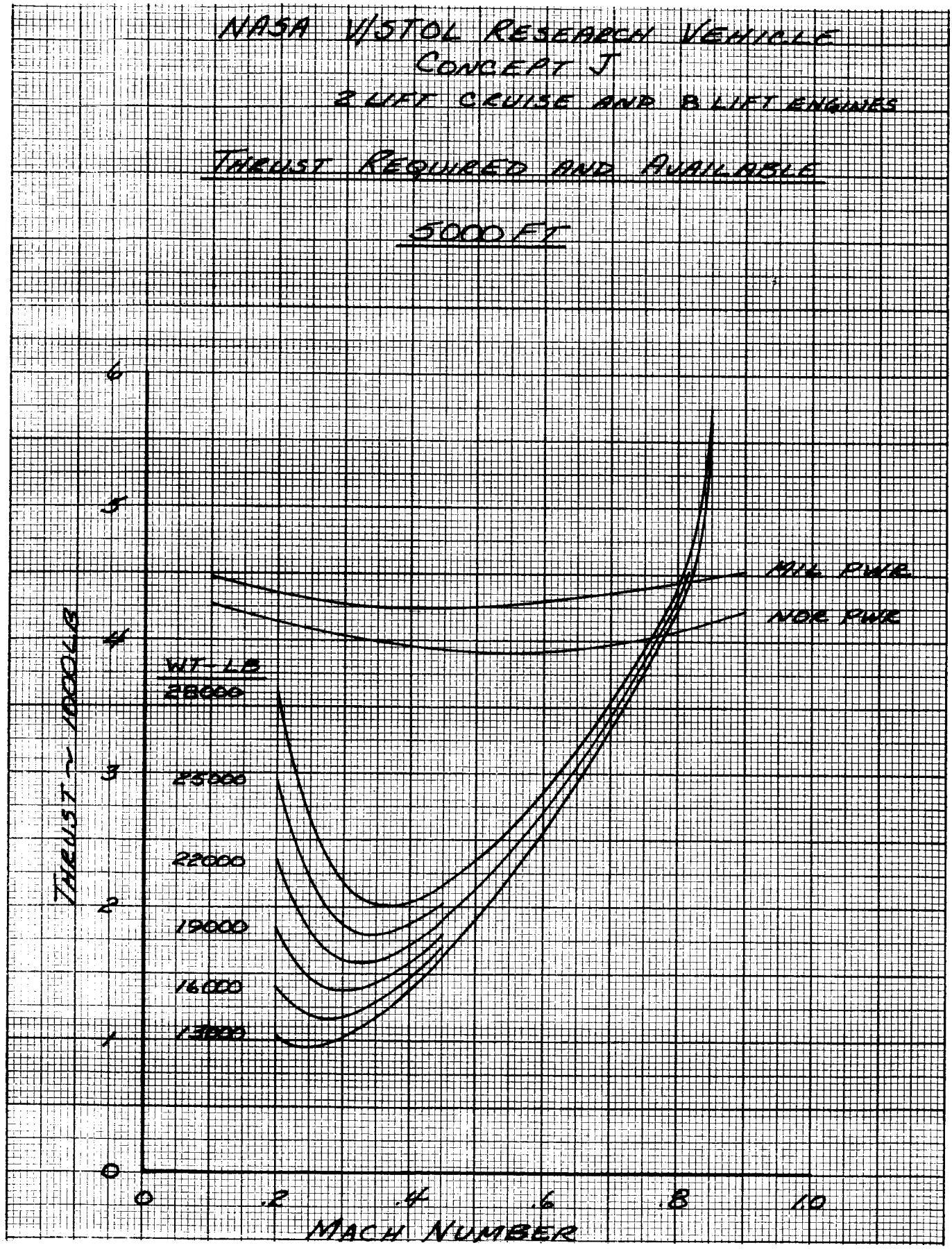


Figure III-29

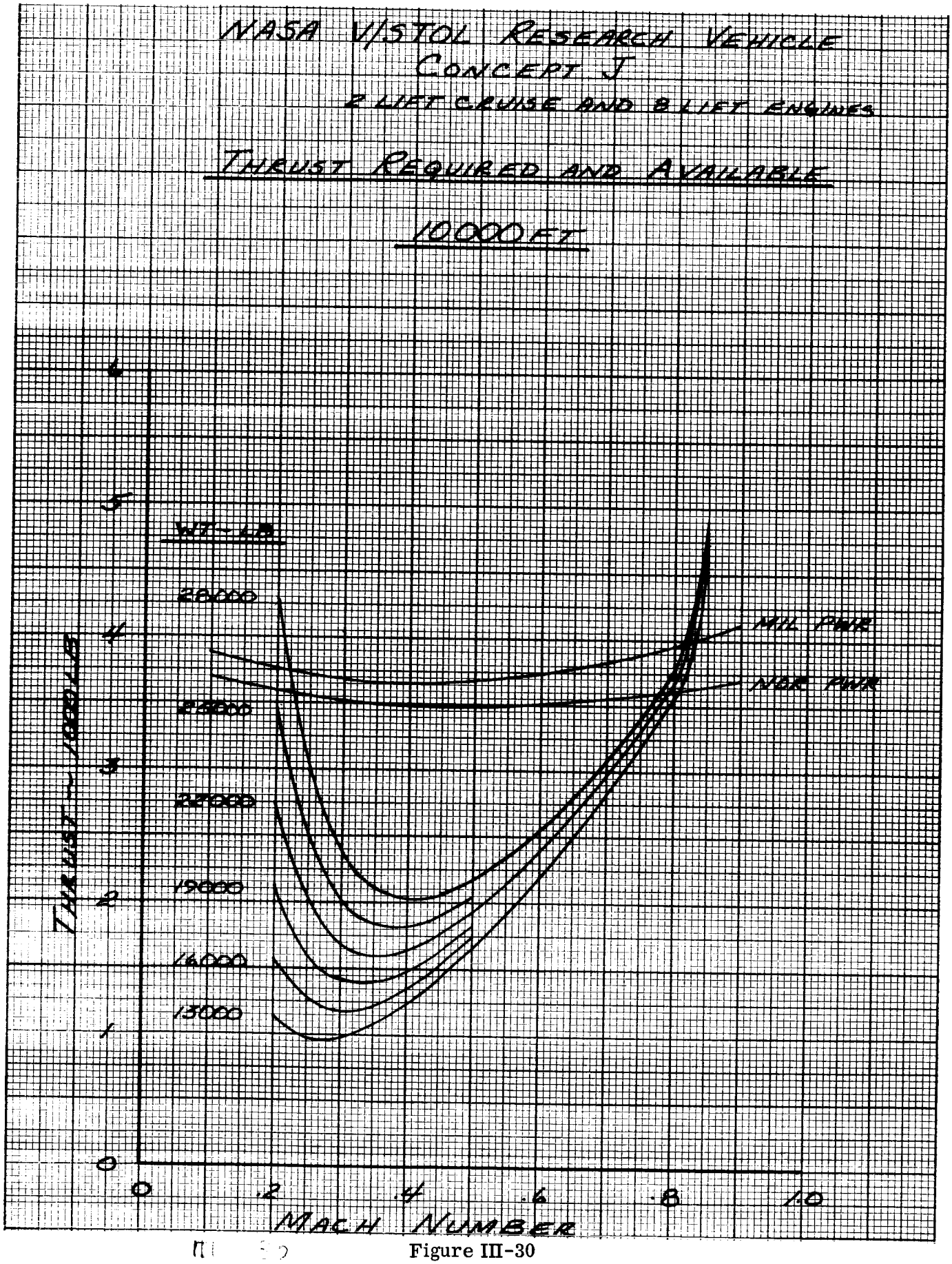


Figure III-30

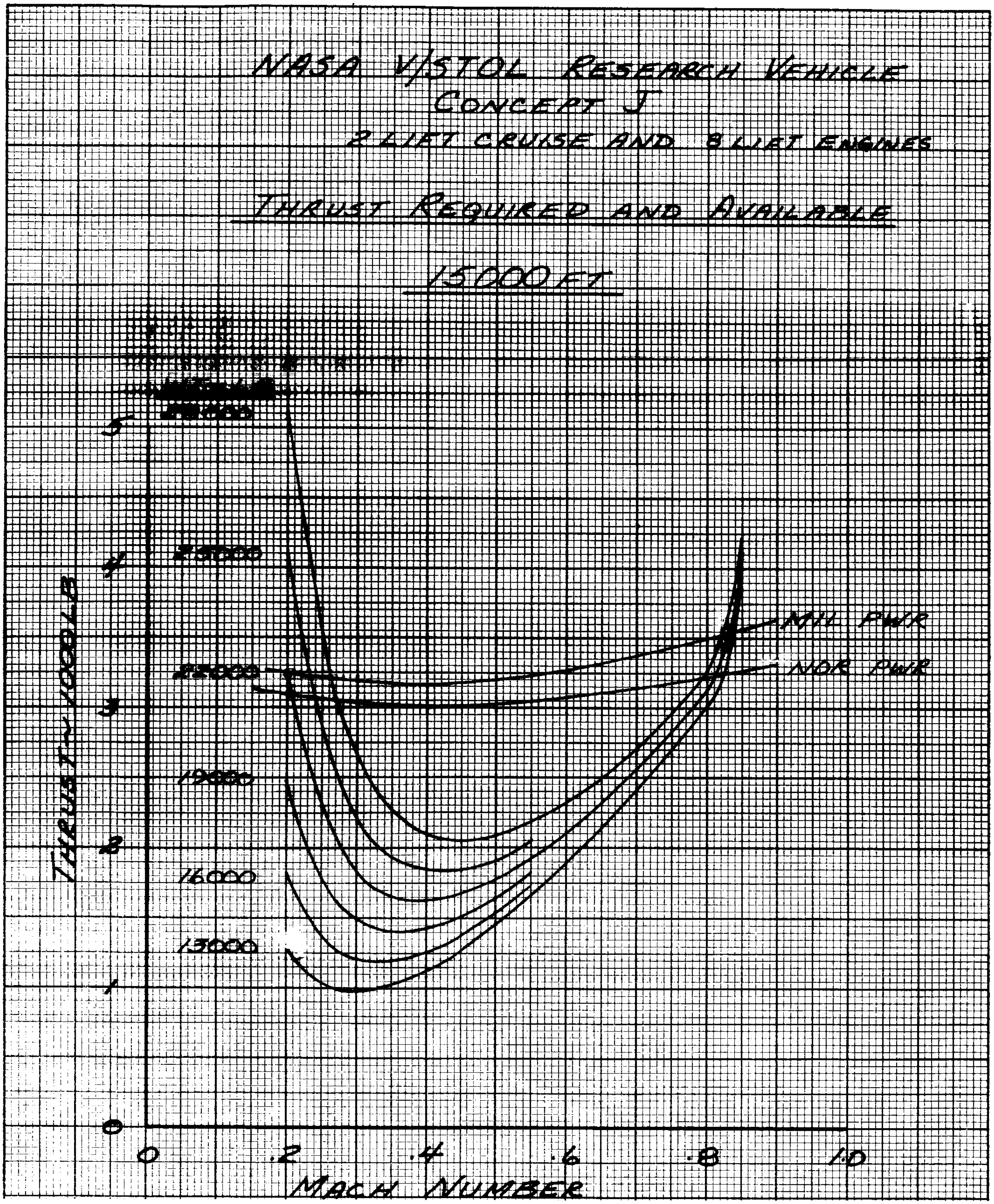


Figure III-31

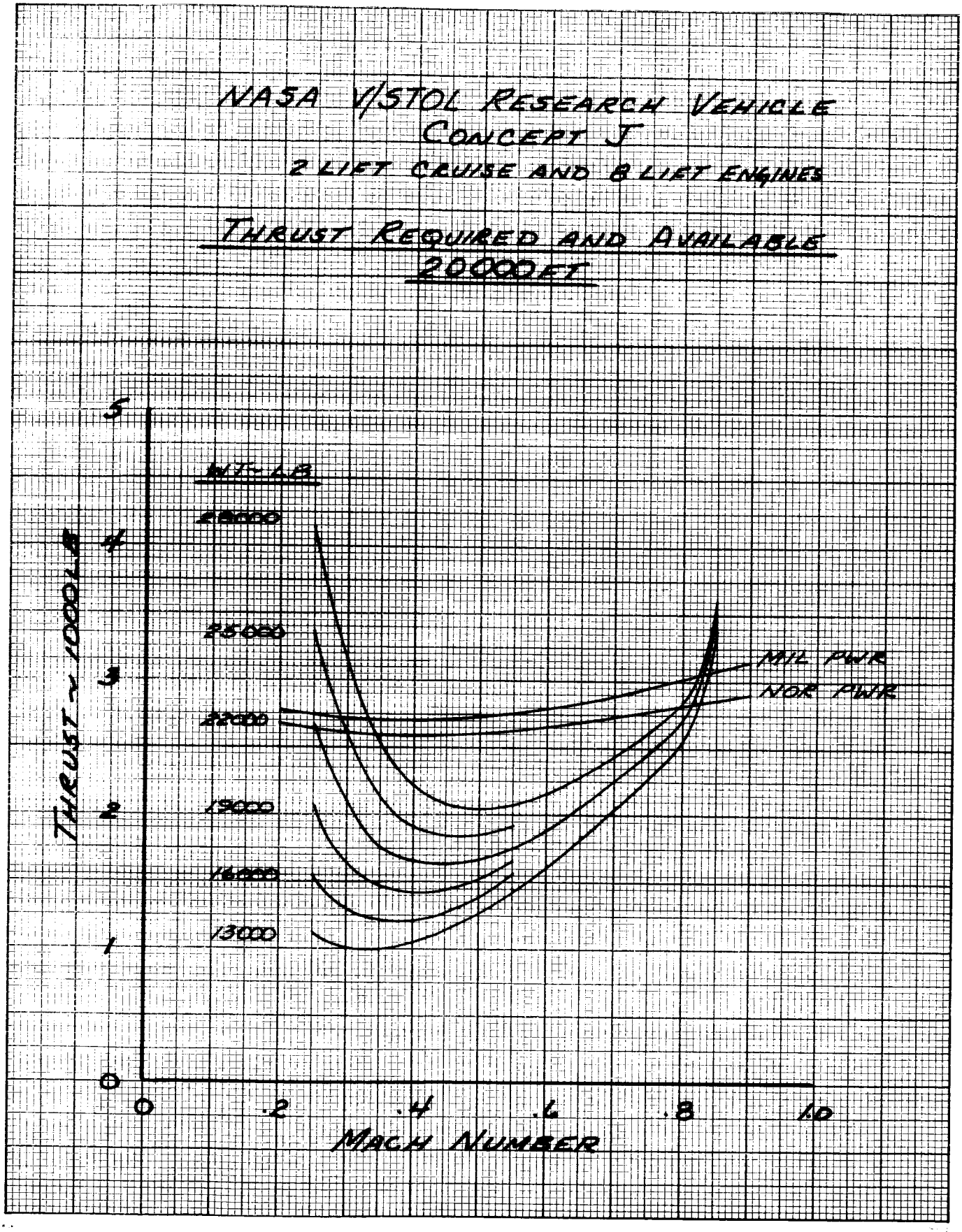


Figure III-32

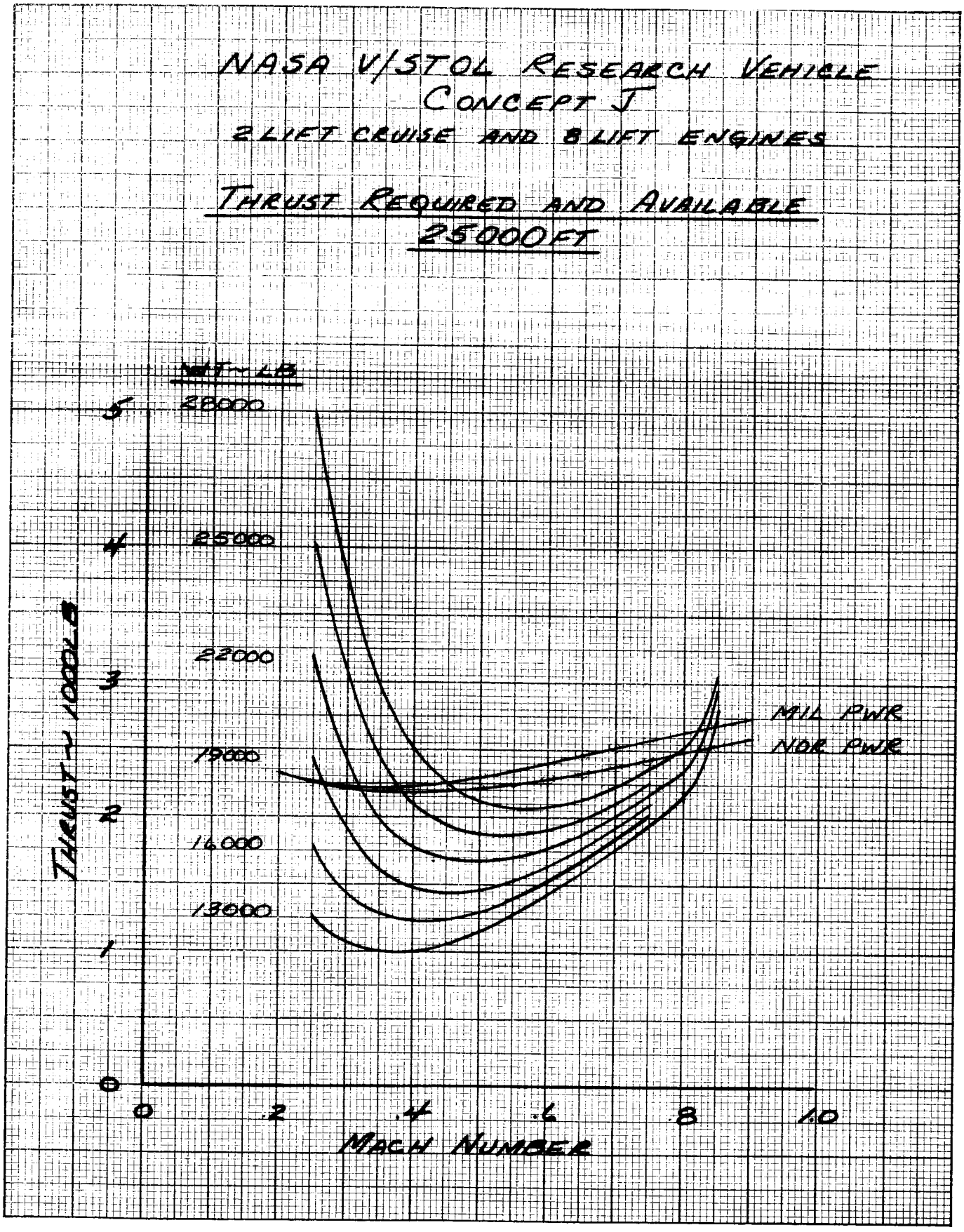


Figure III-33

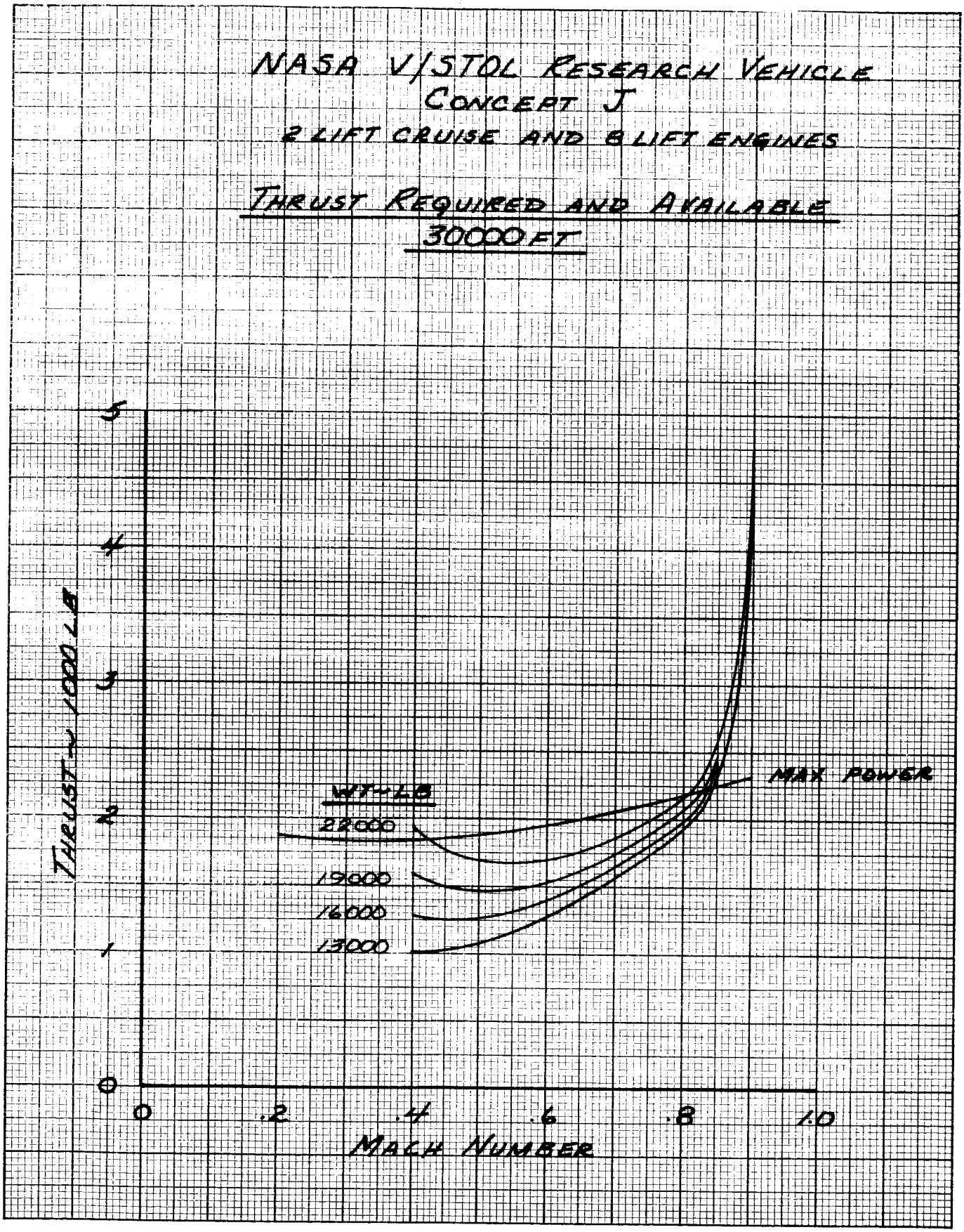


Figure III-34

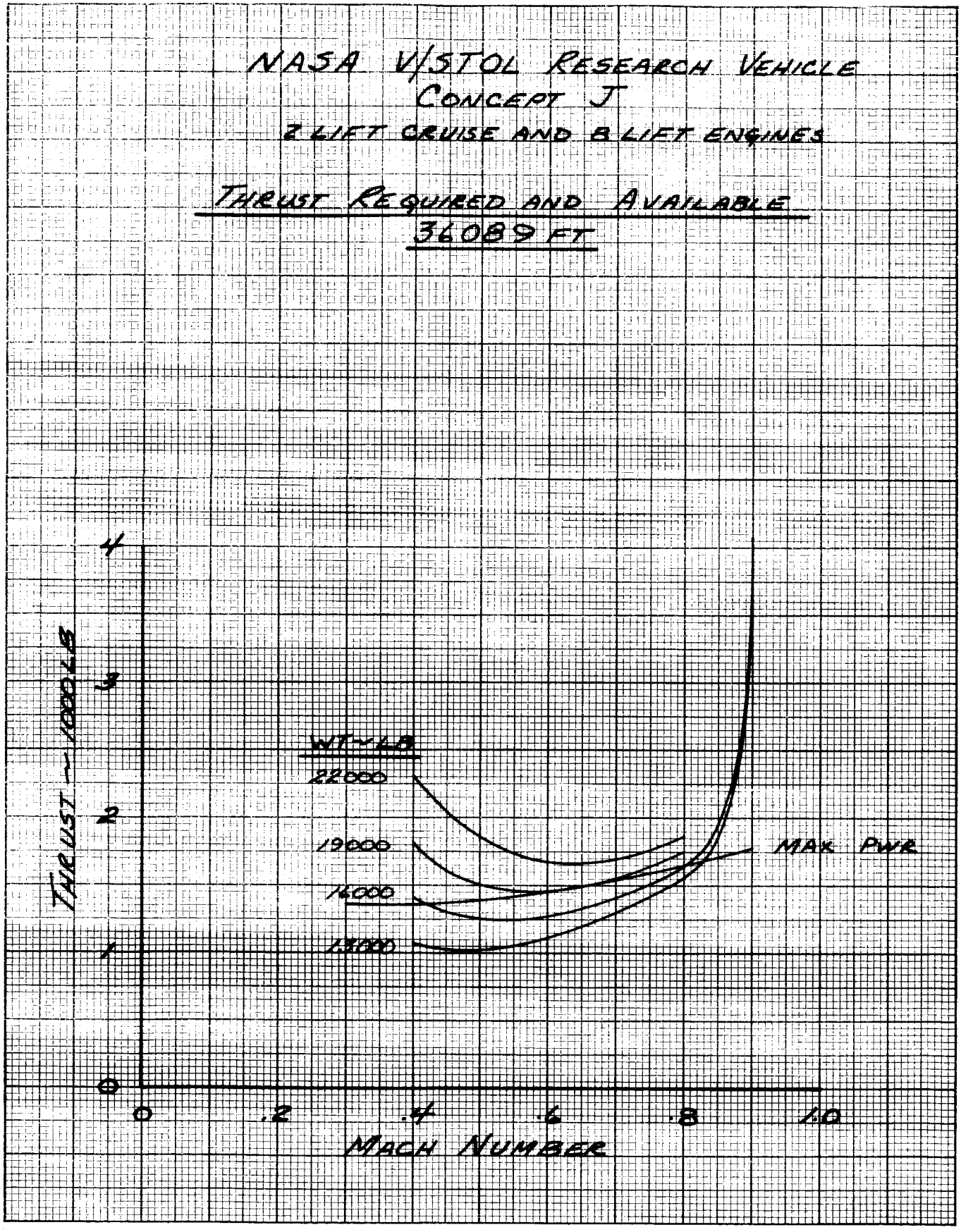


Figure III-35

F. PERFORMANCE DATA: CONVENTIONAL FLIGHT

The performance data presented in this section includes fuel flow, take-off, rate-of-climb, maximum and cruise speeds, ceilings, landings, cruise control, and mission analysis.

1. Fuel Flow Data

The fuel flow is a function of the thrust, speed, and altitude. This is obtained from Section IV - Propulsion Data.

2. Take-Off Calculation

a. Method and Sample Calculation

The take-off distances for a land-based Navy aircraft are those normally obtainable in service operation at sea level at 80°F and 100°F and on hard surfaced runways. The take-off lift coefficient is not greater than the maximum attainable as limited by ground angle or 90 percent of maximum lift coefficient, not utilizing ground effect. Distances to clear a 50-foot height are based on speeds that are at least 120 percent of the power on stall speeds for the take-off considerations. Normally distances are for hard surface runways having a rolling coefficient of friction of 0.025. No additional assistance is included.

The take-off flight path to clear a 50 ft. height is determined for the airplane in the following configuration:

Approach Flaps Deflected	20 degrees
Landing Gear Down	

Sample calculations are given in Tables III-3 and III-4 and a brief review of the calculation procedure is presented.

b. Ground Run

The distance traversed on the ground in accelerating from the static condition to the lift off speed is established using Hartman's single point method of NACA TN 557. Since the variation of the excess thrust with the speed square approaches a straight line, the acceleration at 71% of the lift off speed (or 50% V_U^2) represents the average acceleration for the ground run. The lift off

TABLE III-3

TAKE-OFF CALCULATION

MODEL CONCEPT C

ENGINE 2 YJ85-GE-19 L/C

THRUST MIL WING AREA 204 FT² CONFIG. $\delta_F = 20°$

PR. ALT. S.L. α_U _____ C_{LU} 1.62 $C_{L_{MAX.}}$ 1.80 σ .961 μ .025 TEMP. 80°F

AIR DISTANCE	1	GROSS WEIGHT		LB	15570			
	2	$V_U = \left(\frac{295W}{C_{LU}\sigma S}\right)^{1/2}$	V_U^2	KNOTS	14462			
			V_U		120.2			
	3	$V_{50} = 1.14 \times V_U$	V_{50}^2	KNOTS	18769			
			V_{50}		137			
	4	C_{DU}			.236			
	5	$D_U = C_{DU}\frac{V_U^2\sigma S}{295}$		LB.	2268			
	6	T_U		LB.	4810			
	7	$F_U = T_U - D_U$		LB.	2542			
	8	$C_{L50} = C_{LU}/(1.14)^2$			1.25			
	9	C_{D50}			.15			
	10	$D_{50} = C_{D50}\frac{V_{50}^2\sigma S}{295}$		LB.	1871			
	11	T_{50}		LB.	4760			
	12	$F_{50} = T_{50} - D_{50}$		LB.	2889			
GROUND ROLL	13	$.707\ V_U$		KNOTS	85			
	14	$qS = \frac{(.707V_U)^2\sigma S}{295}$		LB.	9532			
	15	$C_{L_{GR.}}$			.72			
	16	$C_{D_{GR.}}$			.08			
	17	$T_R = (C_{D_{GR.}} - \mu C_{L_{GR.}})qS + \mu W$		LB.	980			
	18	T_A at (13)		LB.	4900			
	19	$F_{GR.} = T_A - T_R$		LB.	3920			
$X_{AIR} = \frac{W}{F_U + F_{50}}\left[.0887(V_{50}^2 - V_U^2) + 100\right]$					1381			
$X_{GR} = \frac{.0443\ W V_U^2}{F_{GR}}$					2546			
$X_{TOT.} = X_{AIR} + X_{GR}$					3927			

RAD-199

TABLE III-4

TAKE-OFF CALCULATION

MODEL CONCEPT J

ENGINE 2 YJ85-GE-19 L/C

THRUST MIL WING AREA 350 CONFIG. CLEAN $\delta_F = 0°$

PR. ALT. S.L. α_U _____ C_{LU} 1.31 $C_{LMAX.}$ 1.455 σ .961 μ .025 TEMP. 80°F

AIR DISTANCE	1	GROSS WEIGHT		LB	19830			
	2	$V_U = \left(\frac{295W}{C_{LU}\sigma S}\right)^{1/2}$	V_U^2	KNOTS	13276			
			V_U		115.2			
	3	$V_{50} = 1.14 \times V_U$	V_{50}^2	KNOTS	17240			
			V_{50}		131.3			
	4	C_{DU}			.148			
	5	$D_U = C_{DU}\frac{V_U^2 \sigma S}{295}$		LB.	2240			
	6	T_U		LB.	4820			
	7	$F_U = T_U - D_U$		LB.	2580			
	8	$C_{L50} = C_{LU}/(1.14)^2$			1.01			
	9	C_{D50}			.096			
	10	$D_{50} = C_{D50}\frac{V_{50}^2 \sigma S}{295}$		LB.	1887			
	11	T_{50}		LB.	4780			
	12	$F_{50} = T_{50} - D_{50}$		LB.	2893			
GROUND ROLL	13	$.707\ V_U$		KNOTS	81.4			
	14	$qS = \frac{(.707V_U)^2 \sigma S}{295}$		LB.	7555			
	15	$C_{L_{GR.}}$			.14			
	16	$C_{D_{GR.}}$			.02			
	17	$T_R = (C_{D_{GR.}} - \mu C_{L_{GR.}})qS + \mu W$		LB.	620			
	18	T_A at (13)		LB.	4920			
	19	$F_{GR.} = T_A - T_R$		LB.	4300			
$X_{AIR} = \frac{W}{F_U + F_{50}}\left[.0887(V_{50}^2 - V_U^2) + 100\right]$					1638			
$X_{GR} = \frac{.0443\ W V_U^2}{F_{GR}}$					2712			
$X_{TOT.} = X_{AIR} + X_{GR}$					4350			

RAD-199

speed is based on 90 percent of maximum lift coefficient, not utilizing ground effects.

$$V_S = \sqrt{\frac{295\,(W - T_U \sin \alpha_U)}{C_{L_{max}}\, S\sigma}}$$

$$V_U = \sqrt{\frac{295(W - T_U \sin \alpha_U)}{C_{L_{U_{wing}}}\, S\sigma}} = V_S/\sqrt{0.9}$$

where

$$C_{L_U} = C_{L_{U_{wing}}} + \frac{T_u \sin \alpha_U}{q_U\, S}$$

where:

σ	=	Ambient density ratio
V_U	=	Lift off speed - knots
W	=	Weight - lb
T_U	=	Thrust at lift off speed - lb
α_U	=	Angle of attack at lift off - degrees
$C_{L_{U_{wing}}}$	=	Lift off lift coefficient = $.9\, C_{L_{max}}$
S	=	Wing area = 204 sq ft
q	=	Dynamic pressure - ± lb/sq ft = $\frac{V^2\sigma}{295}$
V_S	=	Power on stall speed - knots

To obtain the average acceleration for the ground run, the net accelerating force is determined at $.707\, V_U$ with the airplane in the three point attitude by the following expression:

$$F_{gr} = T_{gr} - (C_{D_{gr}} - \mu C_{L_{gr}})\, q\, S - \mu W$$

where:

F_{gr}	=	Average net ground run accelerating force - lb
T_{gr}	=	Thrust available at $.707\, V_U$ - lb

$C_{D_{gr}}$ = Drag coefficient in the ground attitude

$C_{L_{gr}}$ = Lift coefficient at ground angle of attack

μ = Rolling coefficient of friction = .025

The reduction in induced drag coefficient due to proximity to the ground is neglected and $C_{D_{gr}}$ used in the computations for take-off is that which would be obtained away from the ground at the same angle of attack.

The ground roll distance traversed in accelerating from static conditions to V_U is obtained from the following expression:

$$X_{gr} = \frac{.0443W\ V_U^2}{F_{gr}}$$

c. Air Distance

The equation for the determination of the air distance to a 50 ft height from lift off is derived based on speeds that are at least 120 percent of the power on stall speed.

$$V_{50} = 1.2\ V_S = 1.14\ V_U$$

where:

Σ Forces = Sum of significant forces along the flight path

X_{air} = Air distance from lift off to 50 ft height - ft

g = Acceleration of gravity = 32 ft/sec^2

W = Weight - lb

V_{50} = Speed at 50 ft height - knots

V_U = Lift off speed - knots

T_{50} = Thrust available at V_{50} - lb

D_{50} = Thrust required at V_{50} - lb = $C_{D50}\ q_{50}\ S$

T_U = Thrust available at V_U - lb

D_U = Thrust required at V_U - lb = $C_{DU}\ q_U\ S$

F_{50} = Excess thrust at 50 ft height - lb = $T_{50} - D_{50}$

F_U = Excess thrust at lift off - lb = $T_U - D_U$

$C_{L_{50}}$ = Lift Coefficient at 50 ft height = $C_{L_U}/(1.14)^2$

$C_{D_{50}}$ = C_D at $C_{L_{50}}$

Work done on aircraft = change in kinetic energy + change in potential energy

$$\Sigma \text{ Forces } X_{air} = \left(\frac{1}{2}\ \frac{W}{g}\ V_{50}^{2}\right) - \left(\frac{1}{2}\ \frac{W}{g}\ V_U^{2}\right) + 50\ W$$

$$\Sigma \text{ Forces } X_{air} = \frac{W}{2g}\ (V_{50}^{2} - V_U^{2}) + 50\ W$$

$$X_{air} = \frac{\frac{W}{2g}\ (V_{50}^{2} - V_U^{2}) + 50\ W}{\left\{\frac{(T_{50} - D_{50}) + (T_U - D_U)}{2}\right\}}$$

$$X_{air} = \frac{W}{F_U + F_{50}}\ \left[.089\ (V_{50}^{2} - V_U^{2}) + 100\right]$$

Take off distance as a function of gross weight at SL - 80°F and 100°F and 2300 ft - 80°F and 100°F are presented on Figures III-36 and III-37.

d. Graphical Presentation

The take off distances as a function of gross weight are presented on Figure III-36. The solid line represents a sea level, 80°F day. The dashed line represents a sea level, 100°F hot day. Both the ground run and the total distance to a 50 ft height are shown.

In no case does the ground run exceed 8,000 ft. The service ceiling (rate-of-climb = 100 fpm) at military power, gears up, flaps up, and with one engine inoperative is not less than sea level as shown in the Standard Aircraft Characteristics Charts in Section II.

Similar graphs are also presented for 2300 ft altitude in Figure III-37.

3. Rate of Climb

a. Method and Sample Calculation

The rate-of-climb at the maximum take-off weight and landing weight are presented as a function of altitude at military and normal power in the

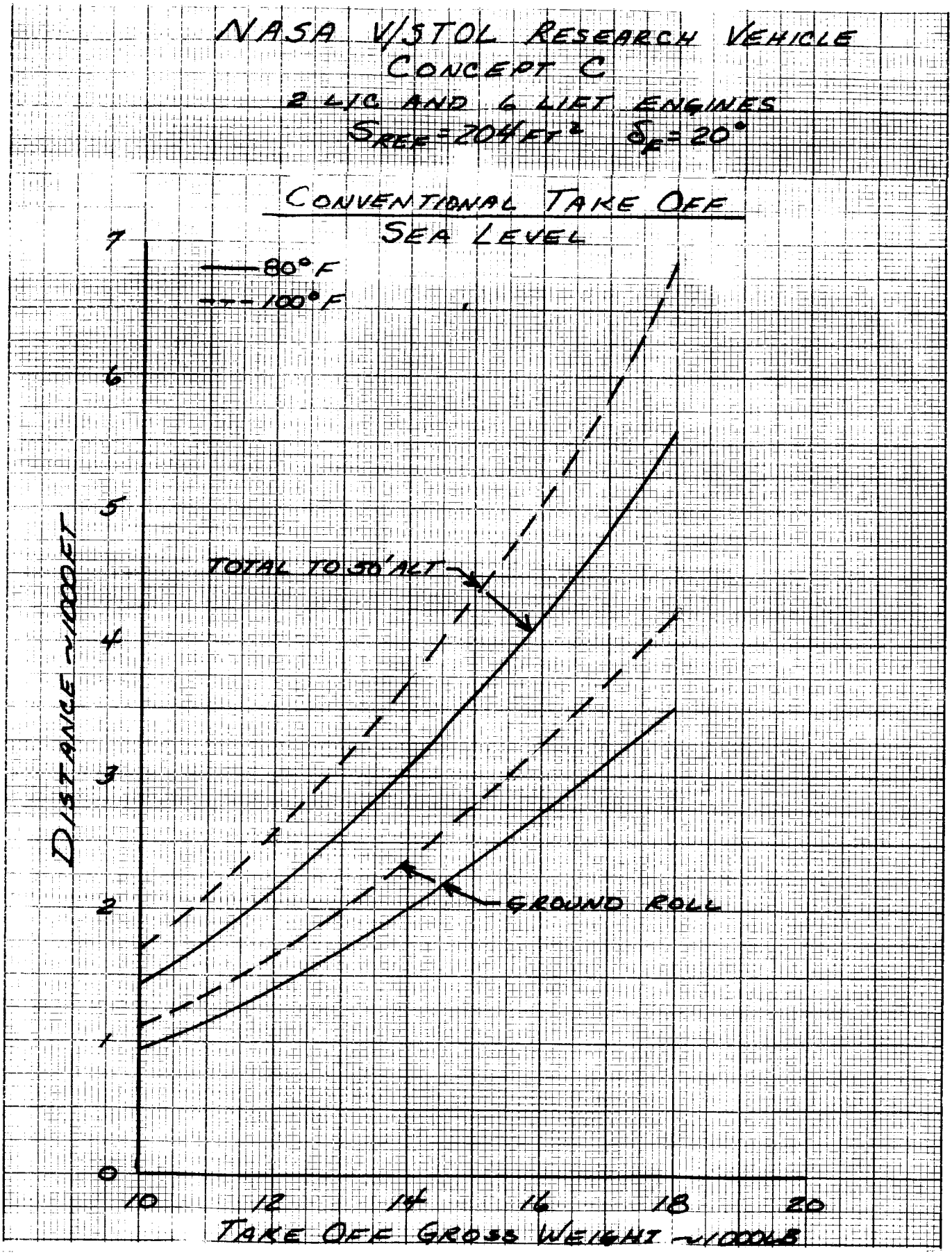
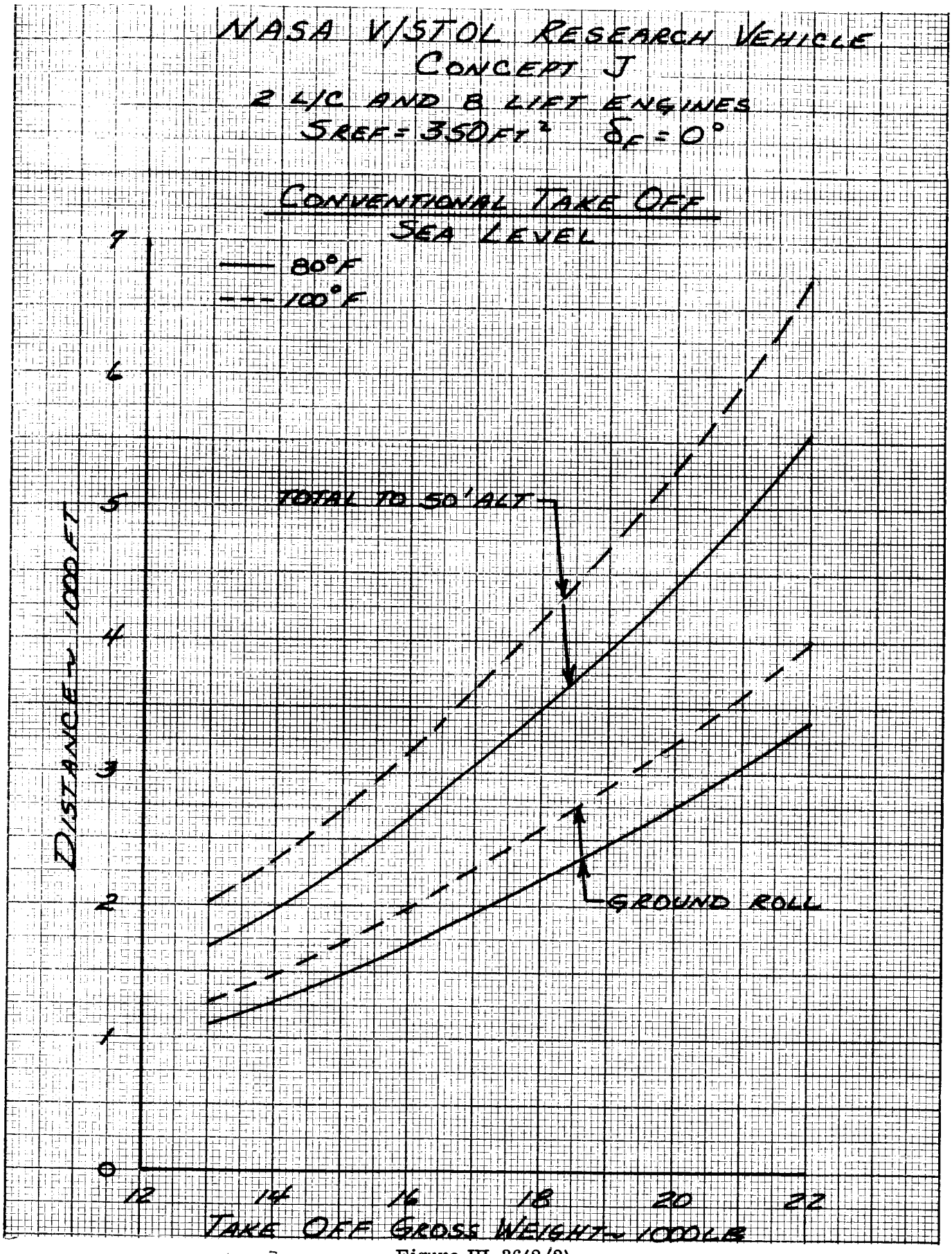


Figure III-36(1/2)



III-36 2/2 Figure III-36(2/2)

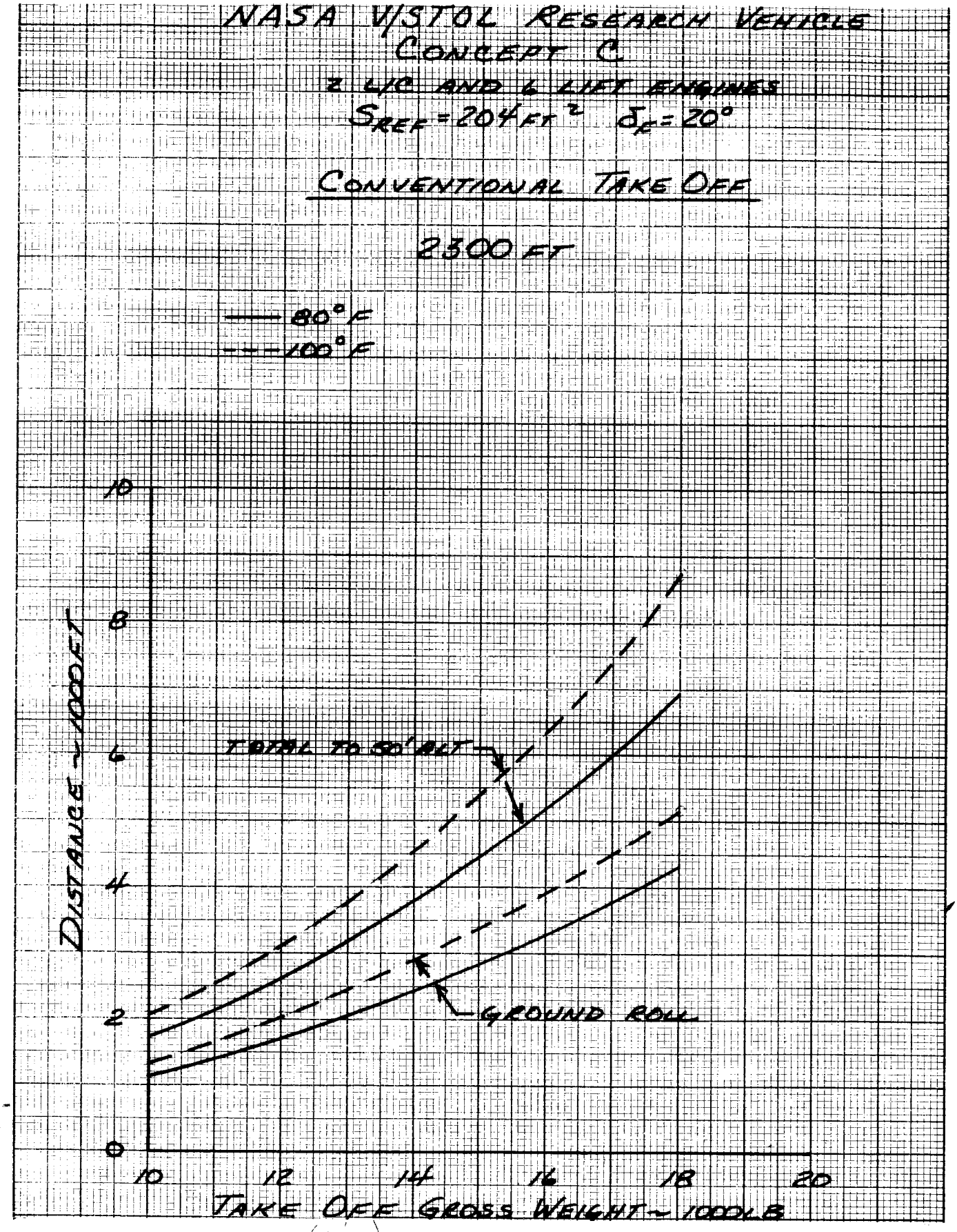
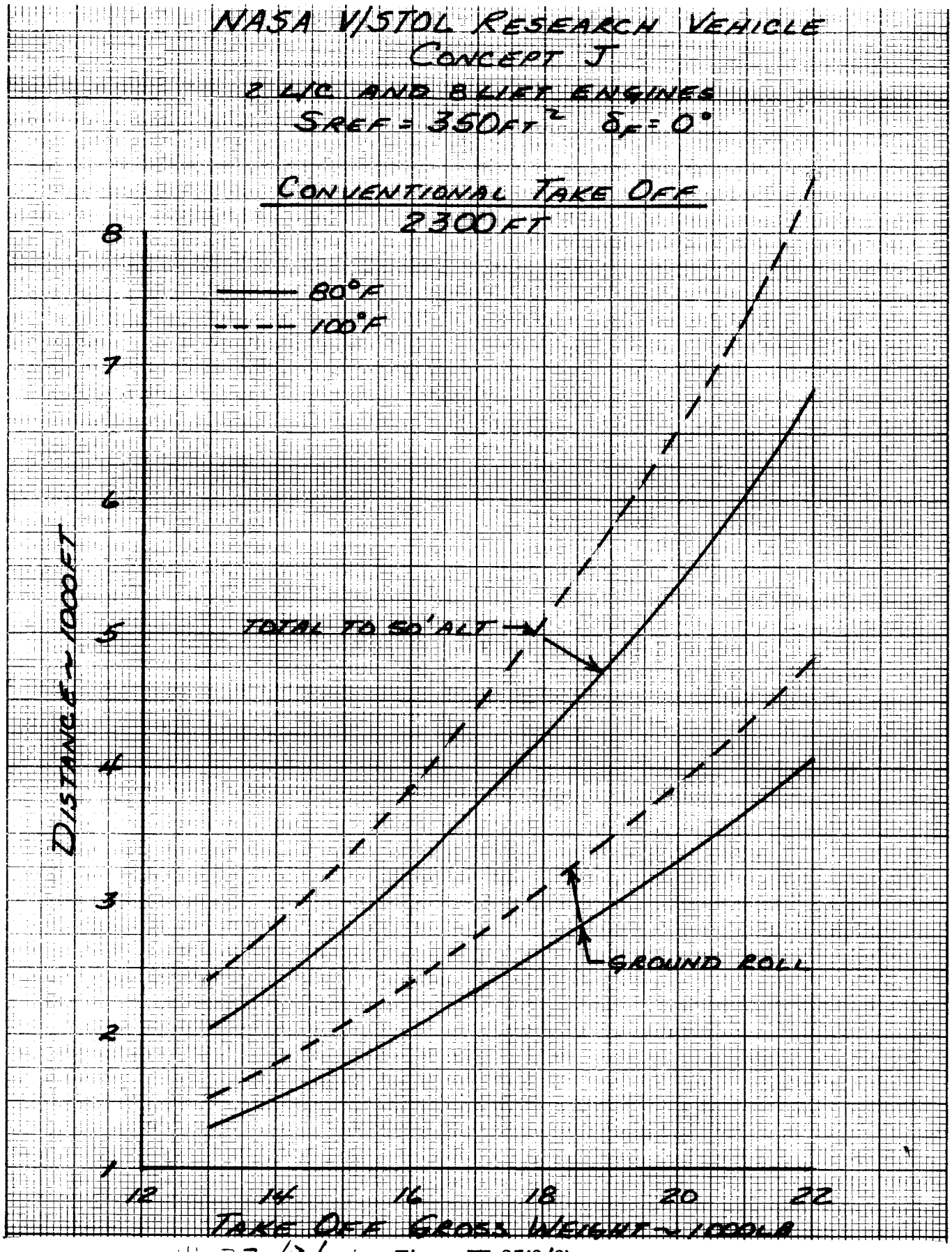


Figure III-37(1/2)



III-37 (2/2) Figure III-37(2/2)

Standard Aircraft Characteristics Chart in Section II. The rates of climb were corrected for changes in kinetic energy due to forward acceleration. A time to climb curve conforming to this performance data was also included.

Sample calculations are presented in Table III-5 for the basic rate-of-climb variation with speed and altitude. A second calculation, as used in a climb program and includes the kinetic energy correction term, is presented in Table III-6. The values in columns 1, 3, and 8 of Table III-6 were obtained from the data of Table III-5. Fuel Flow was obtained from Section IV.

b. Rate of Climb Data

(1) General Climb Data

The rate of climb as a function of excess power was determined in Table III-9 for various airspeeds and altitudes from sea level to 25000 ft in 5000 ft increments.

$$R/C = \frac{101.3\ (T-D)\ V}{W}$$

where:

R/C	=	Rate of climb - ft/min
T-D	=	Excess thrust - lb
V	=	True airspeed - kn
W	=	Weight - lb
T	=	Engine thrust - lb
D	=	Drag - lb
101.3	=	$\frac{60\ \text{sec}}{1\ \text{min}} \times \frac{1.68\ \text{ft/sec}}{1\ \text{knot}}$

A plot of this rate-of-climb as a function of true airspeed will show maxima where the change in the rate-of-climb is zero with a small change in airspeed. In a climb program, the reduction in true airspeed with forward velocity is used to increase the rate of climb by an exchange between kinetic energy and potential energy.

MODEL CONCEPT C

RATE OF CLIMB

Configuration CLEAN

Power Setting MILITARY

Gross Weight 14000 LB

$R/C = \frac{(101.3)\ V\ T_{ex}}{G.W.}$

ALT. SEA LEVEL						ALT. 15000 FT.				
V	D	T	T_{ex}	R/C	MACH	V	D	T	T_{ex}	R/C
	Fig.	Fig.	T-D		NO	KN	Fig.	Fig.	T-D	
KN	Lb.	Lb.	Lb.	FT/MIN		KN	Lb.	Lb.	Lb.	FT/MIN
430	2608	4800	2192	6814	.65	407	1687	3307	1620	4769
463	2911	4830	1919	6425	.70	438	1862	3362	1500	4755
496	3298	4870	1572	5642	.75	469	2072	3427	1355	4603
ALT. 5000 FT.						ALT. 20000 FT.				
422	2242	4329	2087	6376	.65	399	1486	2862	1376	3970
455	2500	4363	1863	6125	.70	430	1624	2916	1292	4018
487	2818	4399	1581	5573	.75	460	1789	2977	1188	3957
ALT. 10000 FT.						ALT. 25000 FT.				
415	1937	3783	1846	5537	.65	391	1335	2448	1113	3150
446	2153	3836	1683	5436	.70	421	1435	2498	1063	3236
478	2413	3898	1485	5141	.75	451	1560	2548	988	3224

Table III-5 Sheet 1

MODEL CONCEPT J

RATE OF CLIMB

Configuration CLEAN
Power Setting MILITARY
Gross Weight 19000 LB

$$R/C = \frac{(101.3)\ V\ T_{ex}}{G.W.}$$

ALT. SEA LEVEL						ALT. 15000 FT.				
V	D	T	T_{ex}	R/C	MACH	V	D	T	T_{ex}	R/C
	Fig.	Fig.	T-D				Fig.	Fig.	T-D	
	Lb.	Lb.	Lb.	FT/MIN			Lb.	Lb.	Lb.	FT/MIN
264.5	1727	4800	3073	4332	.40	250.4	1426	3160	1734	2314
297.6	2008	4779	2771	4394	.45	281.7	1517	3168	1651	2479
330.7	2341	4769	2428	4279	.50	313.0	1657	3187	1530	2552

ALT. 5000 FT.						ALT. 20000 FT.				
259.8	1585	4220	2635	3648	.40	245.6	1416	2680	1264	1655
292.3	1800	4222	2422	3773	.45	276.3	1444	2694	1250	1840
324.8	2064	4236	2172	3759	.50	307.0	1526	2720	1194	1954

ALT. 10000 FT.						ALT. 25000 FT.				
255.2	1485	3660	2175	2959	.40	240.6	1459	2220	761	976
287.1	1636	3664	2028	3102	.45	270.7	1422	2241	818	1181
319.0	1837	3679	1841	3130	.50	300.8	1446	2277	832	1333

Table III-5 Sheet 2

CLIMB TO ALTITUDE
INCLUDING KINETIC ENERGY CORRECTION

MODEL CONCEPT C M = .70 POWER MILITARY

CONFIGURATION CLEAN WEIGHT AT START OF CLIMB 15000 LB

1	2	3	4	5	6	7	8	9	10
ALT.	Δ ALT.	V_{climb}	ΔV_{climb}	V_{climb} AVER.	$\frac{\Delta V_{climb}}{\Delta h}$	$\frac{\Delta C}{C'}$	C' R/C Uncorr.	C' AVER.	C_{KE} AVER.
	(1) − (1)↓	Fig.	(3) − (3)↓	$\frac{(3)+(3)\downarrow}{2}$	$\frac{(4)}{(2)}$	0.0886x (5) x (6)	Fig. Read vs. (17)	$\frac{(8)+(8)\downarrow}{2}$	$\frac{(9)}{1+(7)}$
Feet	Feet	Knots	Knots	Knots	Kts/ Ft.		Ft./Min.	Ft./Min	Ft./Min.
0		463					6030		
	5000		−8	459	−.0016	−.06426		5800	6205
5000		455					5570		
	5000		−8	451	−.0016	−.06314		5370	5745
10000		447					5170		
	5000		−9	442.5	−.0018	−.07080		4755	5085
15000		438					4340		
	5000		−8	434	−.0016	−.06076		4100	4385
20000		430					3860		
	5000		−9	425.5	−.0018	−.06808		3400	3640
25000		421					2940		

11	12	13	14	15	16	17	18	19	20
Δt	Total Time	Fuel Flow	Fuel Flow Avg.	ΔFuel Used	Total Fuel Used	Assumed G.W.	Actual G.W.	ΔHoriz. Dist.	Total Horiz. Dist.
$\frac{(2)}{(10)}$	(11) + (12)↓	5% CONS Fig.	$\frac{(13)+(13)\downarrow}{2}$	$\frac{(11) \times (14)}{60}$	5% CONS. (15) + (16)↓		↑(18) − (15)	See ΔH Below	(19) + (20)↓
Min.	Min.	Lb/Hr.	Lb/Hr.	Lbs.	Lbs.	Lbs.	Lbs.	N.Miles	N.Miles
	0	6857			0		15000		0
.806			6442	86				6.2	
	.806	6027			86		14914		6.2
.870			5638	82				6.5	
	1.676	5250			168		14832		12.7
.983			4888	82				7.3	
	2.659	4325			250		14750		20.0
1.140			4172	79				8.2	
	3.799	3820			329		14671		28.2
1.374			3535	81				9.7	
	5.173	3250			410		14590		37.9

$$\Delta H = \frac{(5) \times (11)}{60}$$

Table III-6 Sheet 1

CLIMB TO ALTITUDE
INCLUDING KINETIC ENERGY CORRECTION

MODEL CONCEPT J POWER MILITARY M=.46

CONFIGURATION CLEAN WEIGHT AT START OF CLIMB 19000 LB

1	2	3	4	5	6	7	8	9	10
ALT.	Δ ALT.	V_{climb}	ΔV_{climb}	V_{climb} AVER.	$\frac{\Delta V_{climb}}{\Delta h}$	$\frac{\Delta C}{C'}$	C' R/C Uncorr.	C' AVER.	C_{KE} AVER.
	(1)-(1)↓	Fig.	(3)-(3)↓	$\frac{(3)+(3)↓}{2}$	$\frac{(4)}{(2)}$	0.0886x (5)x(6)	Fig. Read vs. (17)	$\frac{(8)+(8)↓}{2}$	$\frac{(9)}{1+(7)}$
Feet	Feet	Knots	Knots	Knots	Kts/ Ft.		Ft./Min.	Ft./Min	Ft./Min.
0		304					4379		
	5000		-5	301.5	-.001	-.02713		4082	4207
5000		299					3790		
	5000		-6	296	-.0012	-.03256		3463	3565
10000		293					3136		
	5000		-5	290.5	-.001	-.02614		2836	2920
15000		288					2535		
	5000		-6	285	-.0012	-.03135		2228	2296
20000		282					1921		
	5000		-5	279.5	-.001	-.02515		1608	1655
25000		277					1294		

11	12	13	14	15	16	17	18	19	20
Δt	Total Time	Fuel Flow	Fuel Flow Ave.	ΔFuel Used	Total Fuel Used	Assumed G.W.	Actual G.W.	ΔHoriz. Dist.	Total Horiz. Dist.
$\frac{(2)}{(10)}$	(11)+(12)↓	Fig.	$\frac{(13)+(13)↓}{2}$	$\frac{(11) \times (14)}{60}$	(15)+(16)↓		(18)-(15)	See ΔH Below	(19)+(20)↓
Min.	Min.	Lb/Hr.	Lb/Hr.	Lbs.	Lbs.	Lbs.	Lbs.	N.Miles	N.Miles
	0	5979			0		19000		0
1.19			5583	111				5.98	
	1.19	5188			111		18889		5.98
1.40			4830	113				6.95	
	2.59	4473			224		18776		12.93
1.71			4172	119				8.35	
	4.30	3871			343		18657		21.28
2.18			3579	130				10.44	
	6.48	3287			473		18527		31.72
3.02			3004	151				14.20	
	9.50	2722			624		18376		45.92

$$\Delta H = \frac{(5)(11)}{60}$$

Table III-6 Sheet 2

$$W\,\Delta h + \frac{W}{g}\,\frac{V^2}{2} = \text{constant}$$

$$\frac{\Delta R/C}{R/C} = \frac{-1}{g}\,V\,\frac{dV}{dh} = \frac{-1}{32.16}\left(\frac{6080}{3600}\right)^2 V\,\frac{dV}{dh}$$

$$= 0.0886\;V\,\frac{dV}{dh}$$

where: V = true airspeed - knot

dV/dh = rate of change of airspeed with altitude - knot/feet

This correction factor is similar to the Phillips' presentation, Jour. of Aero. Sciences, March 1946. A form of this correction factor is incorporated in Table III-6. The maximum ratio of climb which includes this factor as a function of gross weight and altitude is presented in Figure III-38 at military power and in Figure III-39 at normal power.

(2) Time to Climb

The take-off time is that normally obtainable in service operation at sea level with NACA standard atmosphere and no wind. The time to climb calculation is included in Table III-6. This is a simple summation of the time required to climb incremental heights of 5000 ft. This is shown on the top of Figure III-40 for a military power climb.

(3) Distance Covered in Climb

The distance covered in a climb is presented in Table III-6 This is a simple summation of the horizontal distance covered during each segment of the climb program. For the shallow climb profiles used, this is the product of the true airspeed times the time in each segment. This is shown on the bottom of Figure III-40 for a military power climb.

(4) Fuel Consumed vs Altitude

The fuel consumed during the climb was also obtained by the arithmetic sum of the fuel consumed in each segment of the climb, Table III-6 The fuel flow is present in Section IV. The fuel used as a function of weight and altitude is presented in Figure III-41.

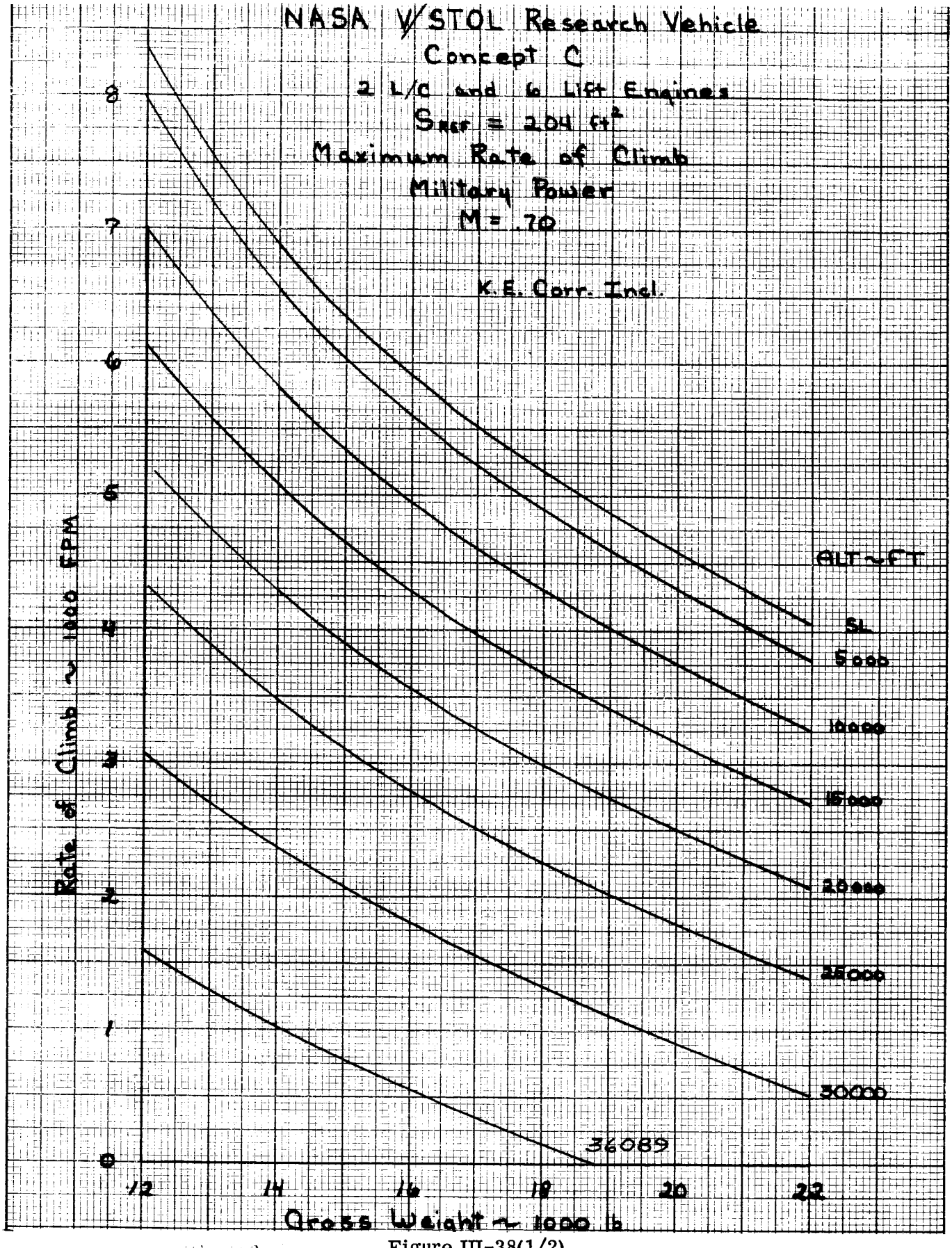


Figure III-38(1/2)

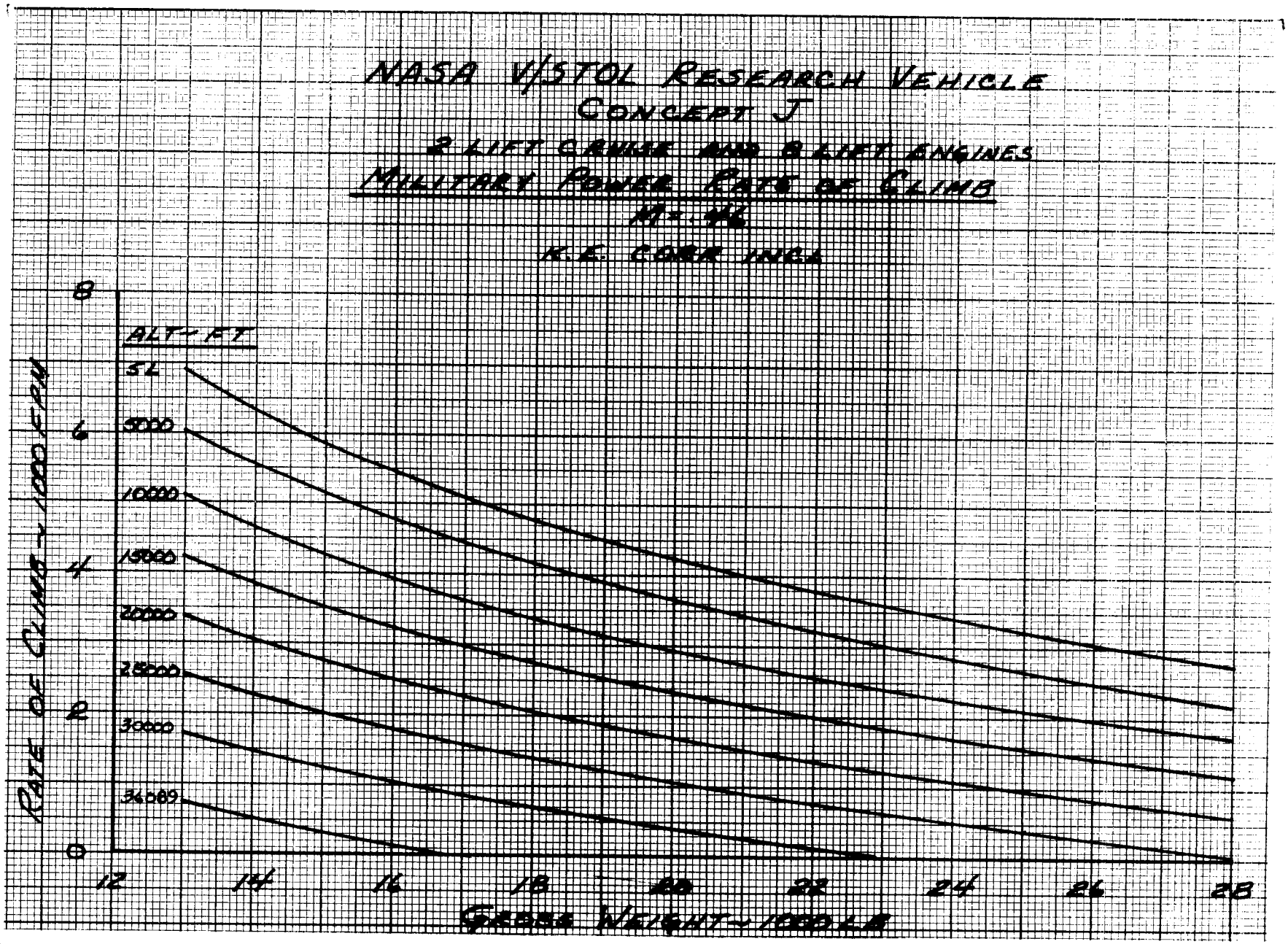


Figure III-38(2/2)

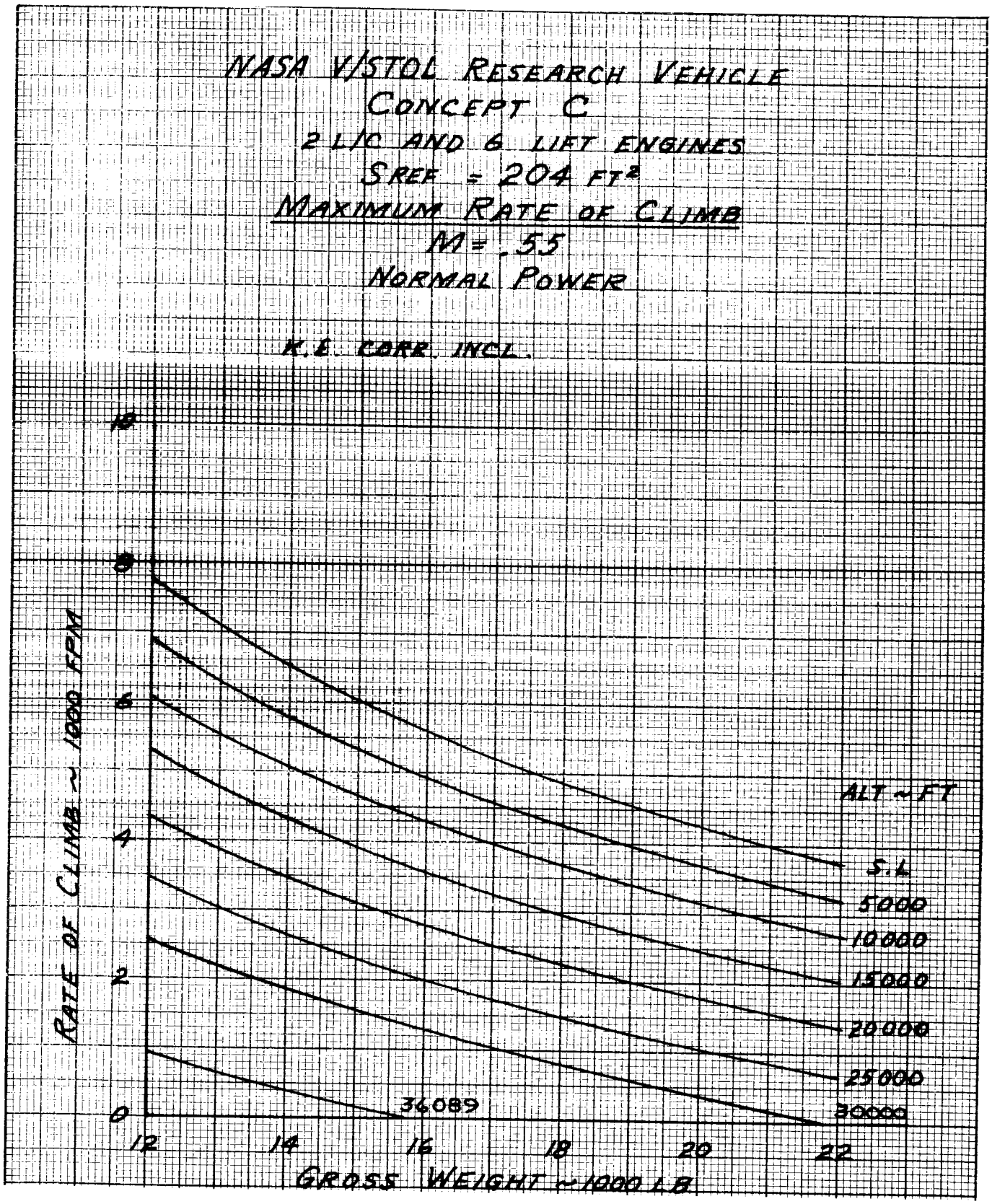


Figure III-39(1/2)

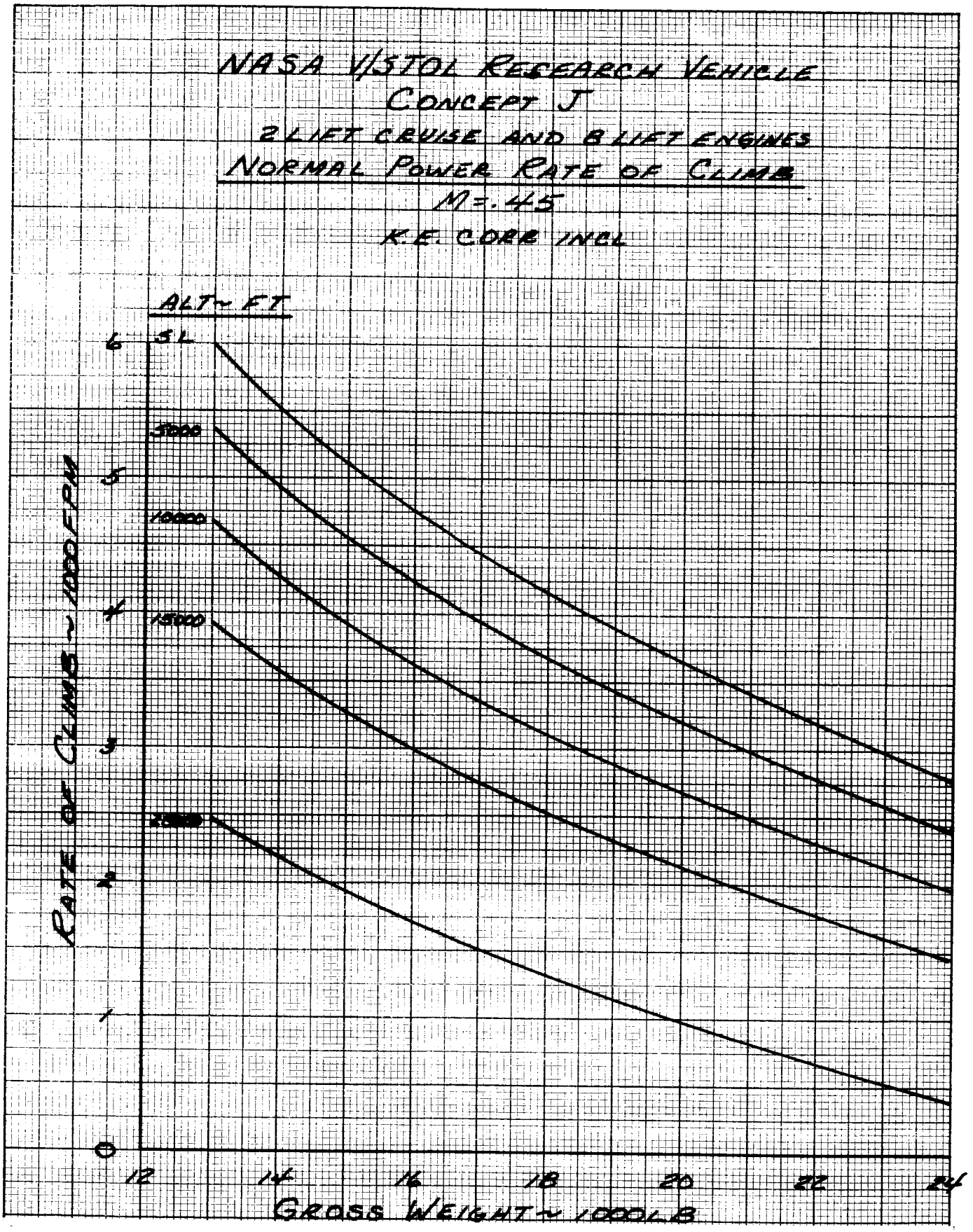


Figure III-39(2/2)

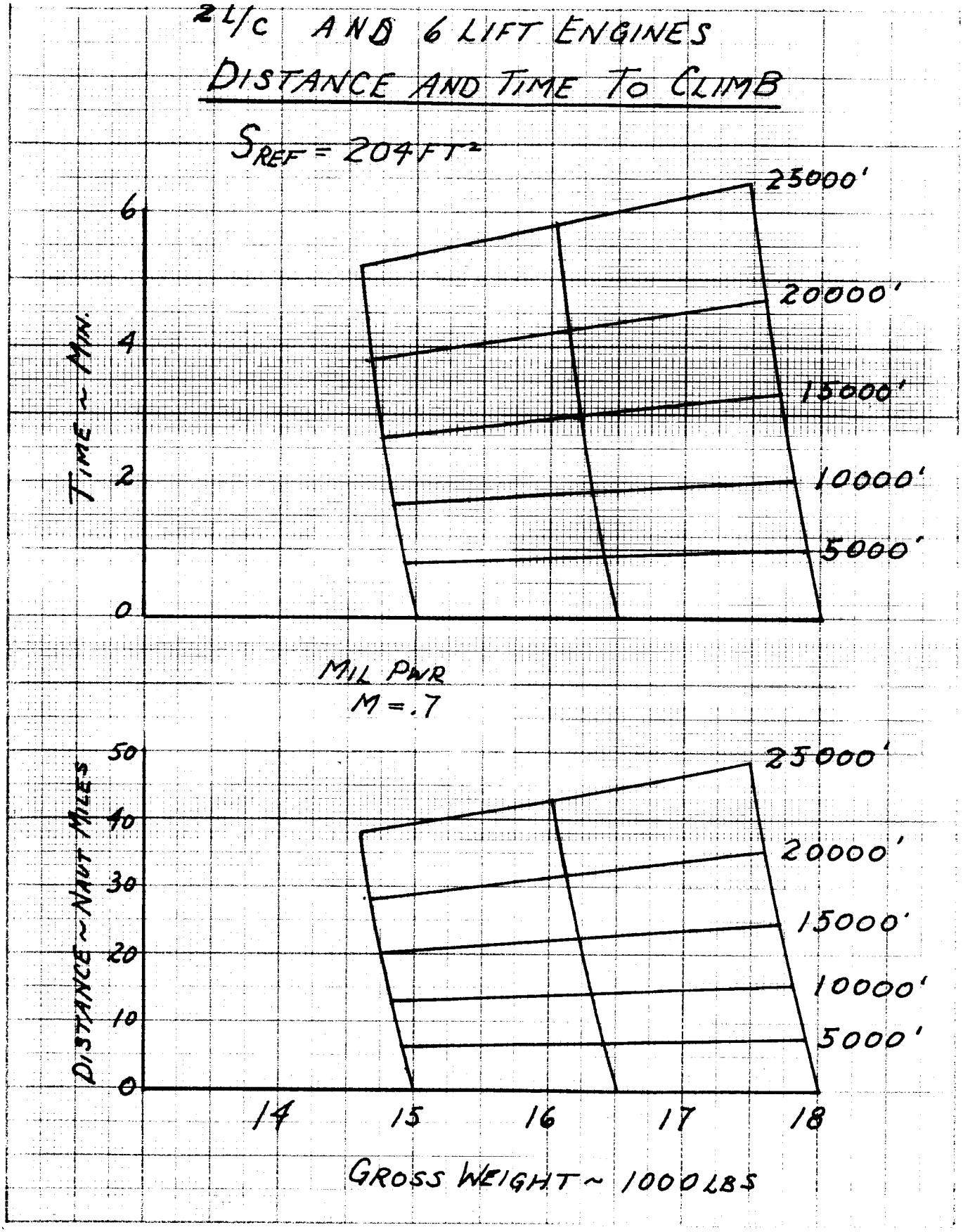


Figure III-40(1/2)

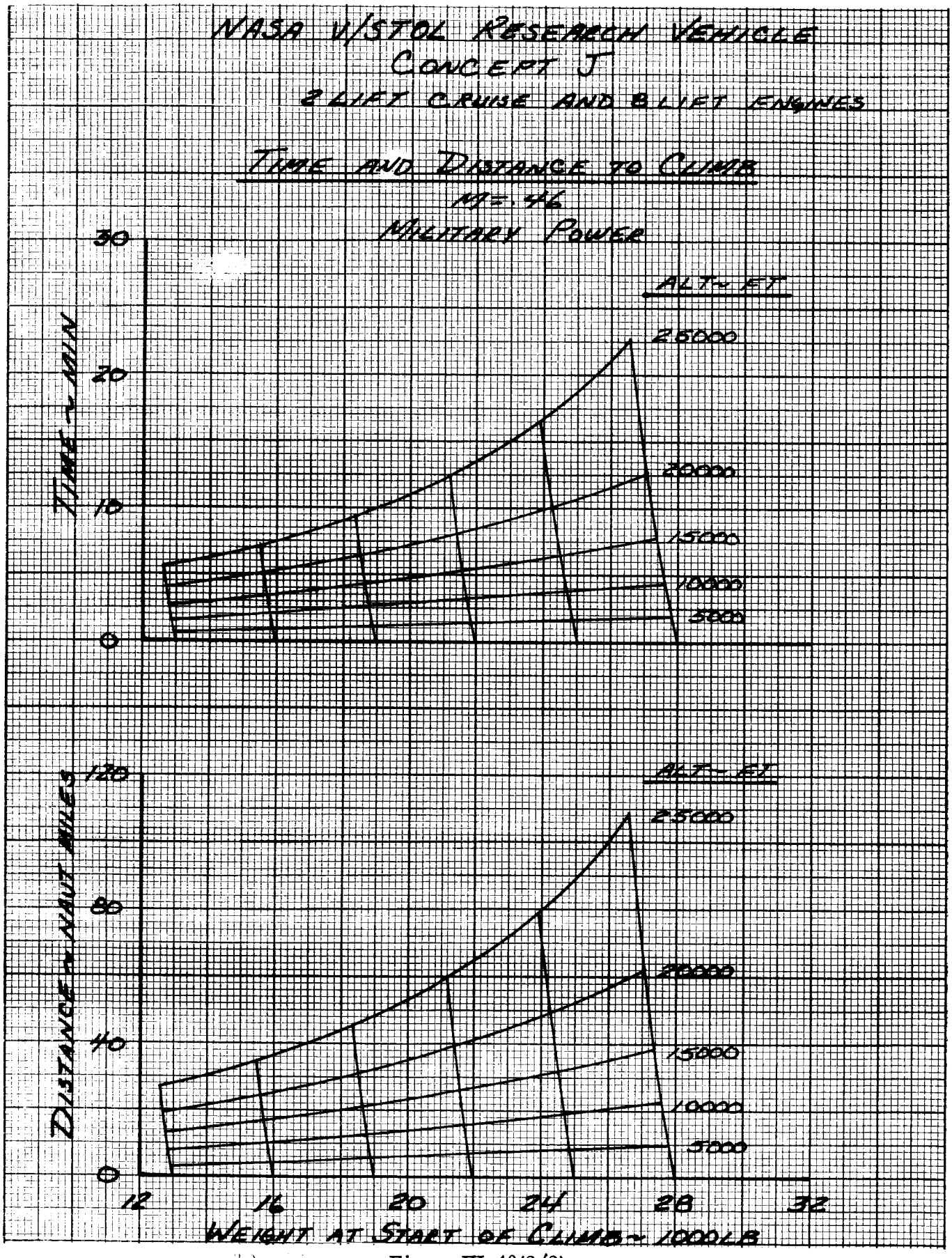


Figure III-40(2/2)

NASA V/STOL RESEARCH VEHICLE

CONCEPT C

2 L/C AND 6 LIFT ENGINES

FUEL TO CLIMB

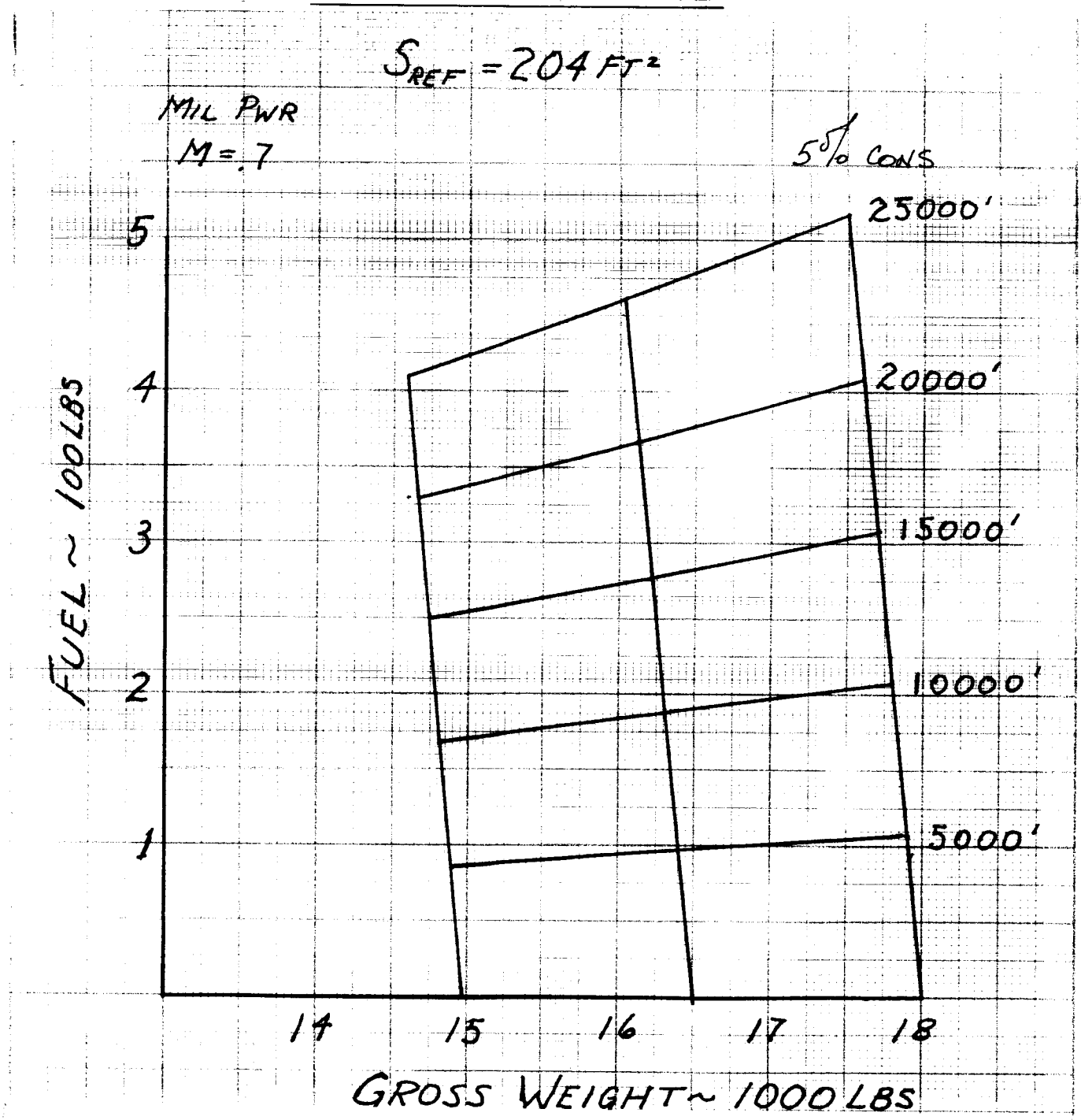


Figure III-41(1/2)

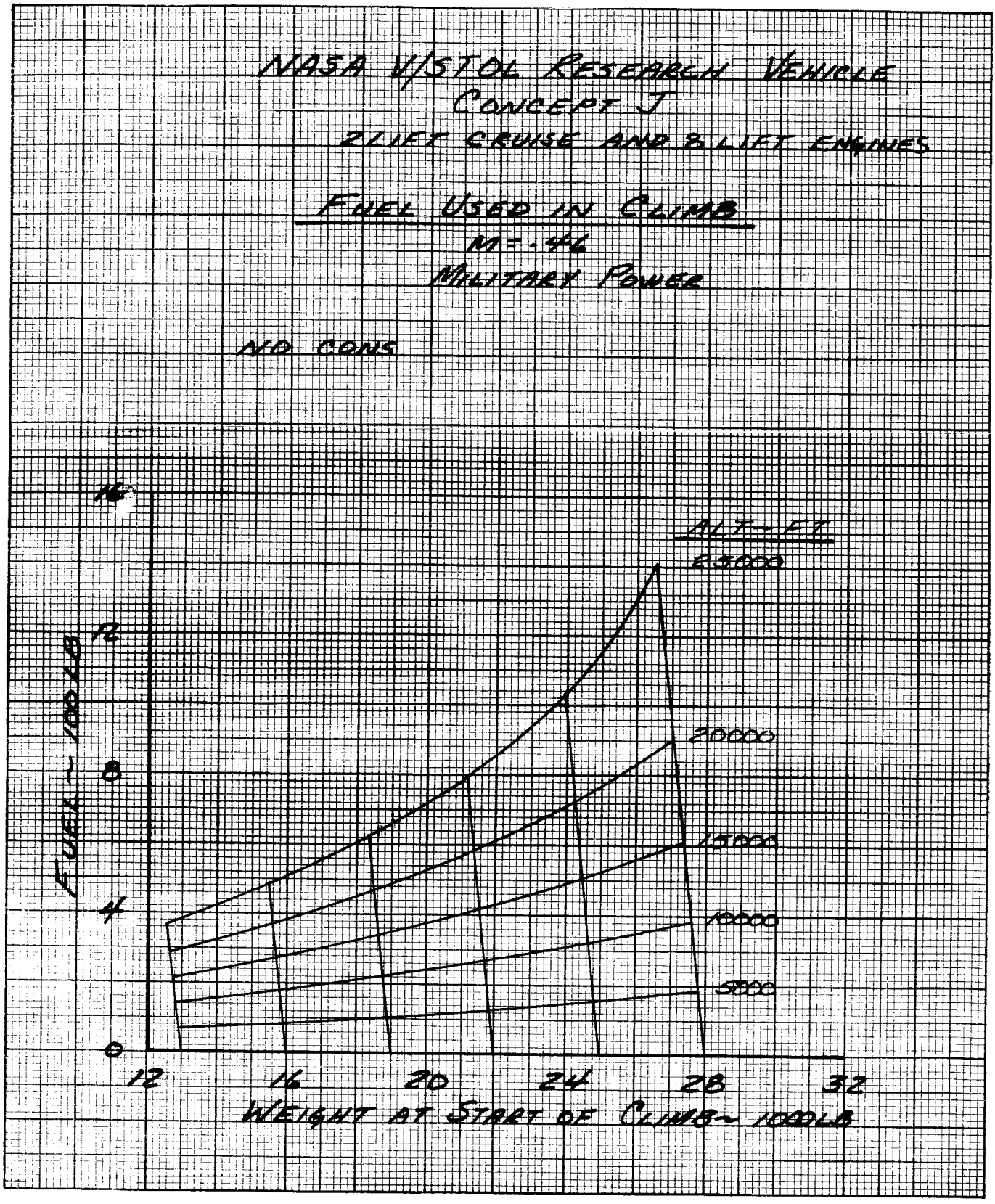


Figure III-41(2/2)

c. One Engine Out Rate of Climb Curves

Single engine inoperative performance at military power is presented on Figures III-42 as a function of gross weight and altitude. The rate-of-climb is also presented as a function of altitude for the maximum take-off gross weight in Figure III-43.

d. Emergency Climb

The emergency climb which may occur at the end of conversion to aerodynamic flight after a vertical lift-off and acceleration is specified by NASA.

The rate-of-climb for one engine out is presented as a function of gross weight up to maximum weight using maximum power on operating engines at approach speed. The approach speed is 1.2 times the power-off stall speed. The applicable landing configuration is with approach flap setting (less than full flap), with the lift engine doors open, lift engine windmilling, and with the landing gear down. The condition using full flaps is also presented.

These conditions are presented as supplemental charts in the Standard Aircraft Characteristics Charts in Section II.

4. Speeds

The level flight speed-altitude profile is presented in this section.

a. Maximum Speed vs Altitude

The maximum level flight speed is presented as a function of altitude at military and normal power and maximum take-off gross weight in the Standard Aircraft Characteristics Chart in Section II. A typical thrust required calculation is presented in Table III-7. In Sections III.D and III.E are presented the power required and power available. The intercept of these two curves provide the variation of maximum speed with gross weight and altitude. These are shown on Figures III-20 through III-35.

b. Speed for Maximum Endurance

The airspeed for maximum endurance corresponds to the speed for minimum fuel flow attainable at momentary weight and altitude except as limited

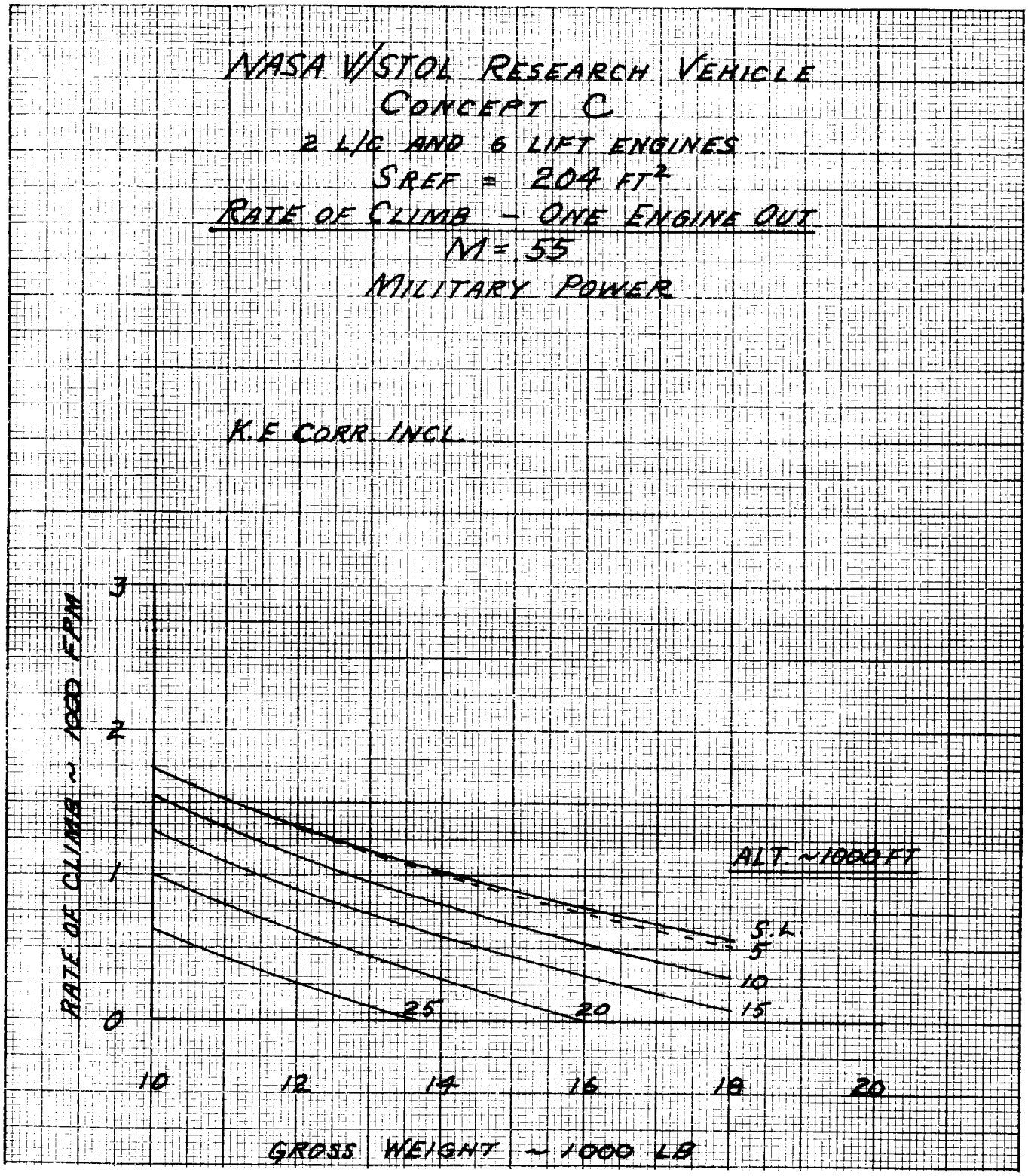


Figure III-42(1/2)

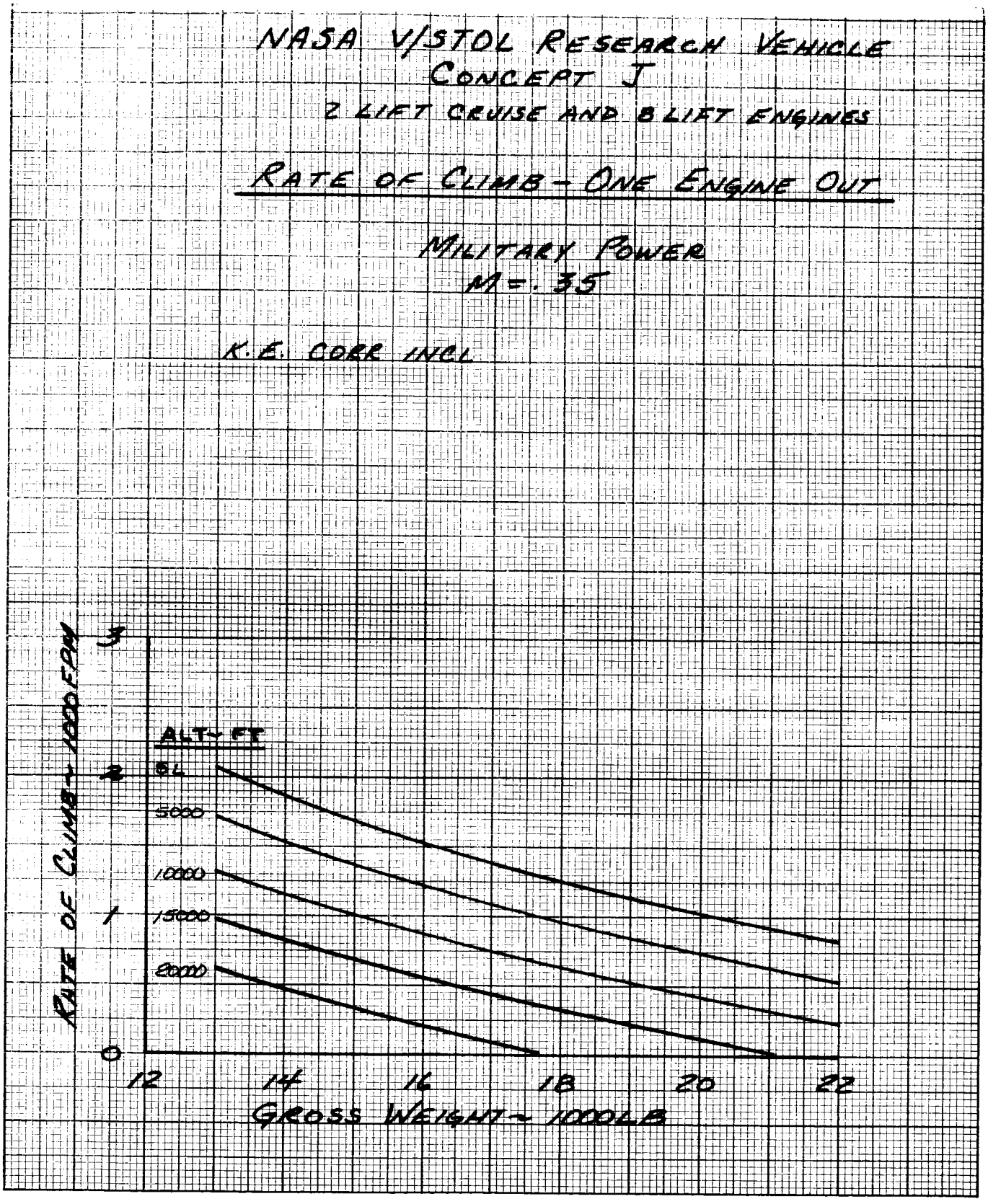


Figure III-42(2/2)

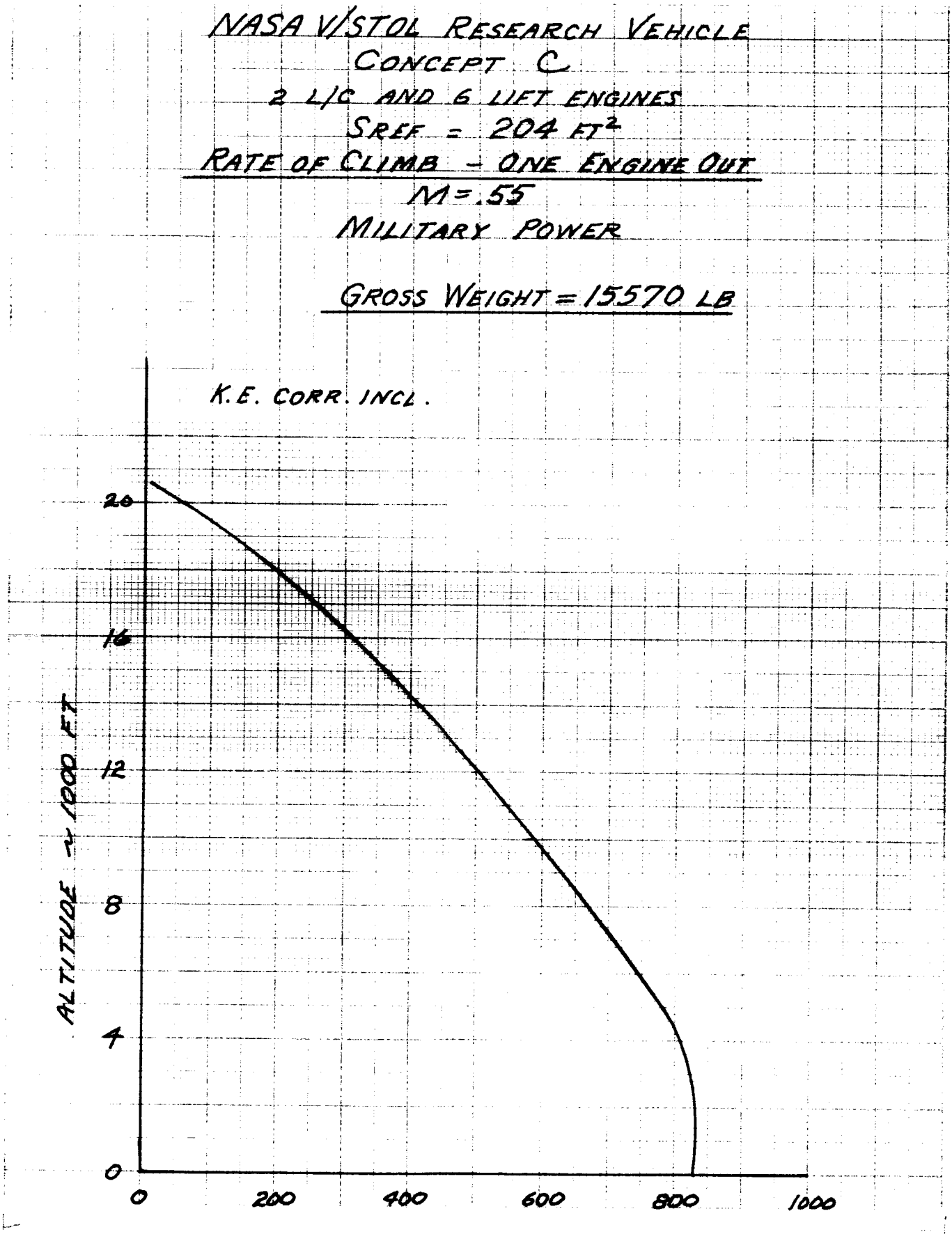


Figure III-43(1/2)

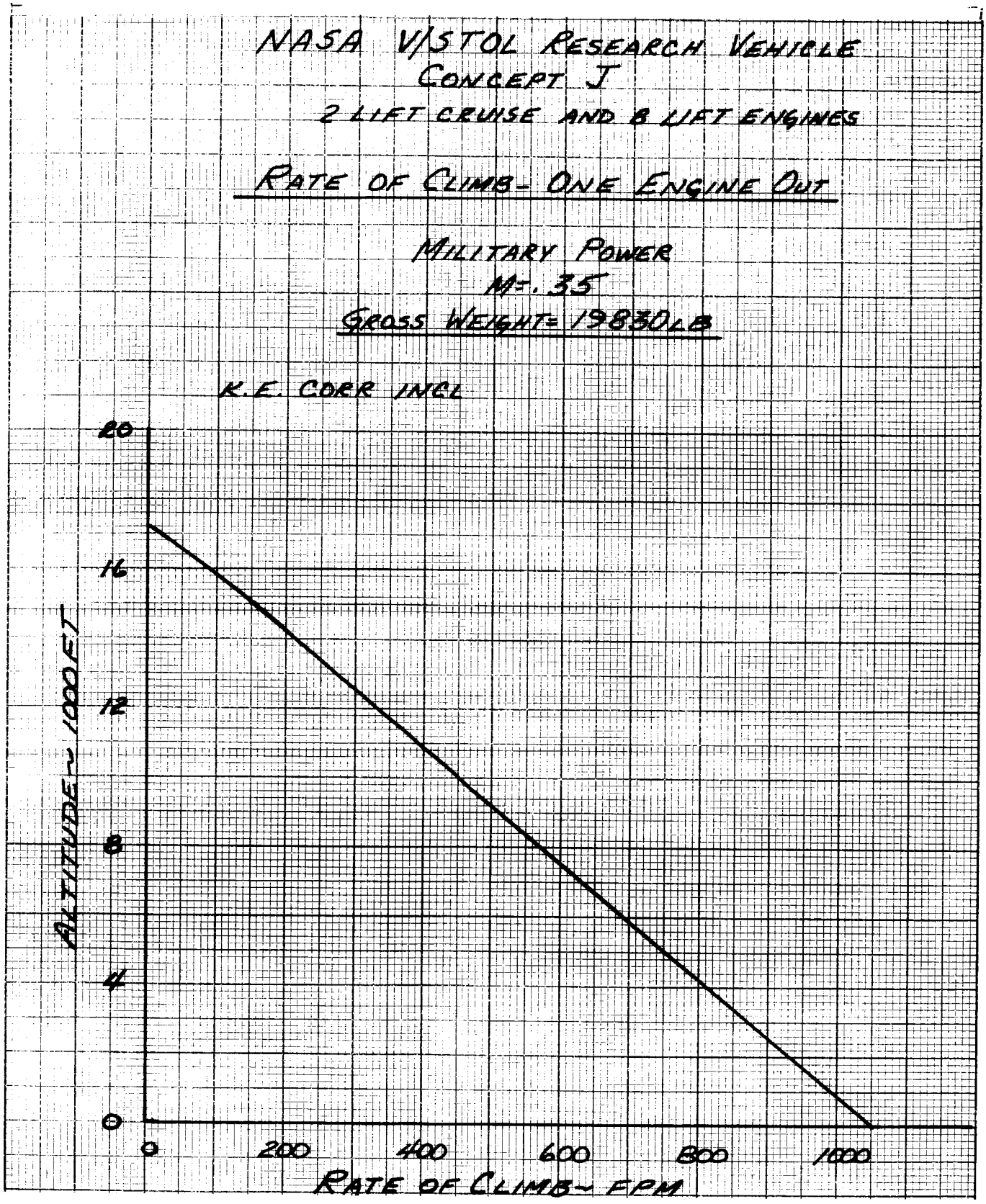


Figure III-43(2/2)

THRUST REQUIRED
CONCEPT J

Configuration CLEAN — Altitude = 25000 Ft.

Weight = 19000 Lbs. — V_a = 601.8 Knots — σ = .4481 — S = 350 Sq. Ft.

1	2	3	4	5	6	7	8	9	10
True Airspeed	qS	C_L	Mach Number	C_{D_o} Clean	K	ΔC_{D_i}	ΔC_D	C_D Total	Thrust Req.
		G.W. / (2)	(1) / Va	Fig.	Fig.	Fig.	Fig.	(5) + (6) +(7) + (8)	(2)x(9)
Knots	Lbs.					(6)x $(3)^2$			Lbs.
271	38851	.488	.45	.0191	.073	.0174		.0365	1422
301	47957	.395	.50	.0187	.073	.0114		.0301	1446
331	58102	.326	.55	.0183	.073	.0078		.0260	1516
361	69302	.274	.60	.0179	.073	.0055		.0234	1619
391	81202	.234	.65	.0174	.073	.004		.0214	1742
421	94153	.202	.70	.0170	.073	.003		.0199	1878

Table III-7

by handling characteristics of the aircraft. Since the speed for maximum endurance for a jet airplane occurs at approximately the lift coefficient for maximum L/D, the determination of this item is accomplished with a minimum of calculations. The thrust required is determined at several true airspeeds bracketing the speed at the lift coefficient of maximum L/D for several gross weights. At these speeds and the corresponding thrust, minimum fuel flow curves are established. These values are shown on Figure III-44.

c. Stall Speed vs Weight

The power off stall speed as a function of gross weight is presented for two flap configurations. For the flaps full down, the data is presented along the solid line. For the flaps up, the data is presented along the dashed line. This data is shown in the Supplemental Performance Charts included in the Standard Aircraft Characteristics in Section II - Aircraft Description and Specification.

Concept C has a higher stall speed, not shown, when spoilers are used to spoil lift and thus achieve a higher stall speed.

5. Ceilings

a. Service and Cruise Ceiling

The ceilings are defined by rate-of-climb, weight, and power setting. The ceiling definitions do not change if a single engine is inoperative.

CEILING DEFINITIONS

Ceiling	Rate-of-Climb	Weight	Power
Service Ceiling	100 ft/min	stated	normal
Cruise Ceiling	300 ft/min	momentary	normal

The ceilings listed below are for the maximum take-off gross weight.

V/STOL AIRCRAFT CEILINGS - NORMAL OPERATION

		Concept C	Concept J
Gross Weight	lb	15570	19830
Service Ceiling	ft	36000	32000
Cruise Ceiling	ft	35000	30000

Figure III-45 presents data for various gross weights for Concept C.

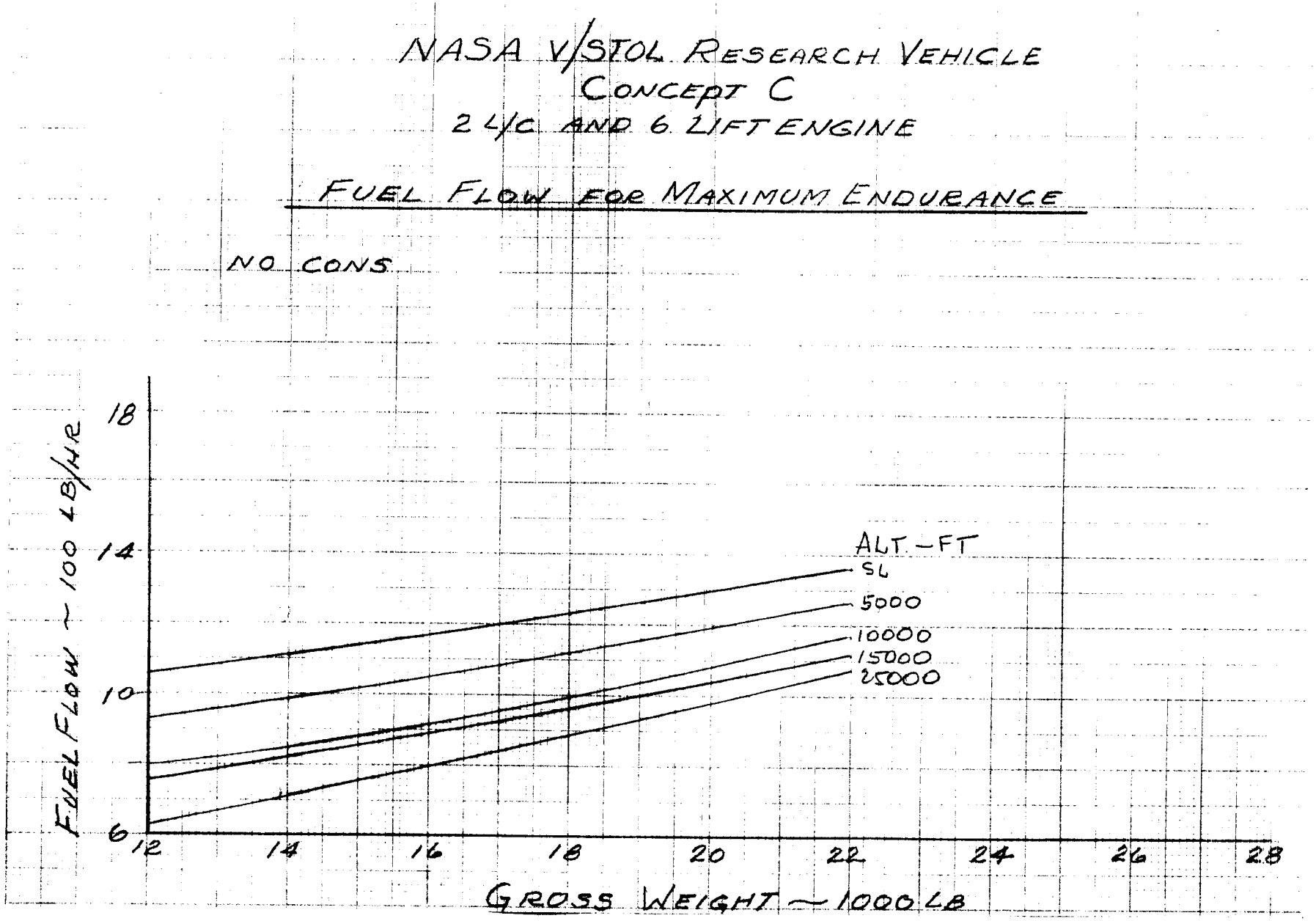


Figure III-44(1/2)

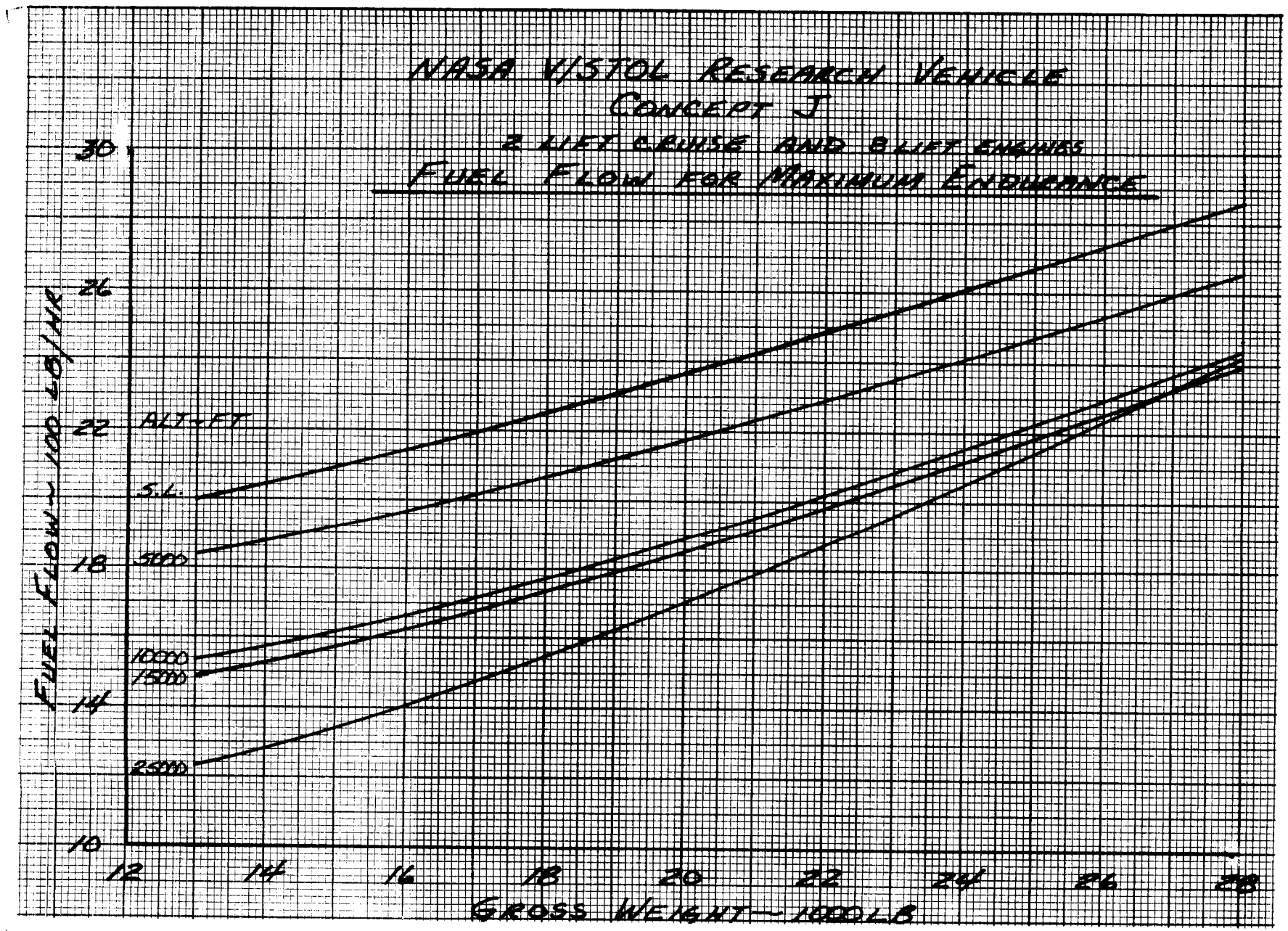


Figure III-44(2/2)

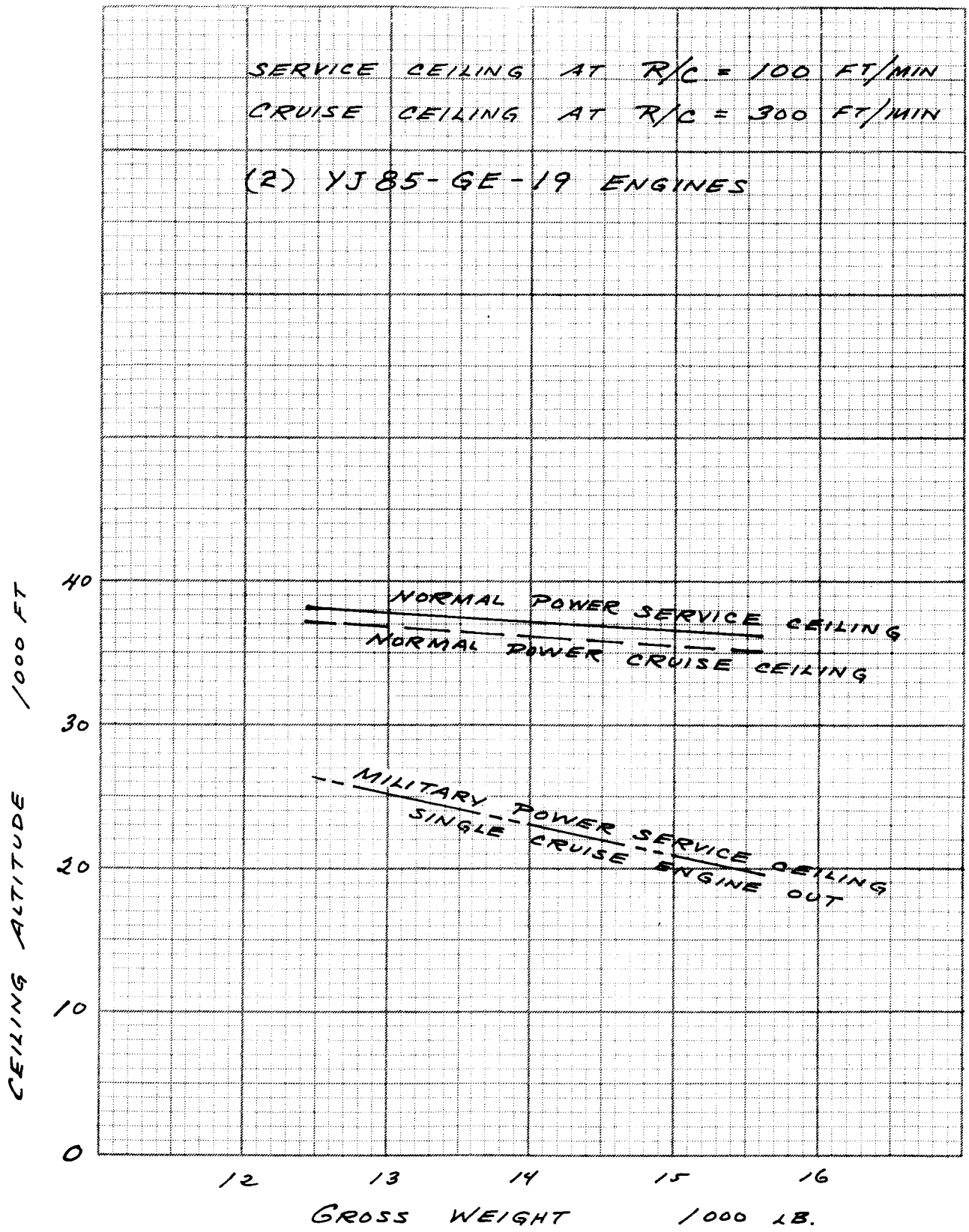


Figure III-45. Concept C Ceiling Altitudes

b. One Engine Out: Service Ceiling

The ceilings listed below are for the single engine inoperative, the remaining engine is operated at military power.

V/STOL AIRCRAFT CEILINGS
MILITARY POWER, SINGLE ENGINE OUT

		Concept C	Concept J
Gross Weight	lb	15300	19500
Service Ceiling, Clean	ft	20000	16000

Figure III-45 presents data for various gross weights for Concept C, clean.

c. Sea Level Service Ceiling, Vertical Power Approach

The maximum gross weight in the vertical power approach configuration was investigated for safety of flight with a sea level service ceiling, 80°F.

GROSS WEIGHT FOR SEA LEVEL SERVICE CEILING - 80°F
SINGLE ENGINE OUT, APPROACH FLAPS, 1.2 STALL SPEED,
LANDING GEAR DOWN, AND LIFT ENGINE DOORS OPEN

		Concept C	Concept J
Flap Setting	degrees	20.0	0.0
Max. Gross Weight	lb	15500	18100

6. Landings

a. Method and Sample Calculation

Landing distances are presented for operation at standard sea level on hard surface runways with no wind. The landing lift coefficient utilizing ground effect is based on the maximum angle-of-attack attainable with the main landing gear oleo strut positioned for the static position, provided the landing lift coefficient shall not exceed that for 110 percent of power-off stall speed for the landing configuration. Distances to clear a 50-foot height are based on speeds that are at least 120 percent of the power-off stall speed for the landing configuration. Normally, distances for a hard-surface runway have a braking coefficient of friction of 0.30.

The distance required for landing over a 50 ft height is calculated as a function of weight for the airplane with flaps deflected 40° and the landing gear down. To represent realistic flight conditions the landing distance is computed with a maximum skinking speed during the glide of 20 fps and at idling thrust on the ground. Landing distances are shown for landing speeds equivalent to 1.10 V_{stall}.

An outline of the formulae used in establishing landing distances as demonstrated in Table III-8 follows.

b. Basic Data

(1) Air Distance

The air distance is assumed to consist of the distance necessary to dissipate the potential energy existing at a height of 50 ft, S_{a_1}, and the kinetic energy change from the speed at 50 ft to the speed at touchdown, S_{a_2}.

$$FS_{a_1} = 50\ W$$

$$S_{a_2} = \frac{50\ W}{F_{ge}} \quad \text{where } F_{ge} = (D-T) \text{ for } R/C = 20 \text{ fps}$$

$$FS_{a_2} = \frac{W}{2g}\left(V_{50}^2 - V_{TD}^2\right)$$

$$S_{a_2} = \frac{\frac{W}{2g}\left(V_{50}^2 - V_{TD}^2\right)}{F}$$

where $$F = \frac{(D_{50} - T_{50\ idle}) + (D_{TD} - T_{TD\ idle})}{2} = \frac{F_{50} + F_{TD}}{2}$$

$$X_{air} = S_{a_1} + S_{a_2}$$

and

$$X_{air} = \frac{50\ W}{F_{ge}} + \frac{.0886W\left(V_{50}^2 - V_{TD}^2\right)}{F_{50} + F_{TD}}$$

(2) Ground Braking Distance

$$X_{gr} = \frac{.0443W\ V_{TD}^2}{D_{gr} - \mu\ W}\ \ell n\ \frac{D_{gr} - T_{avg}}{\mu\ W - T_{avg}}$$

LANDING CALCULATION
MODEL CONCEPT C
ENGINE 2 YJ85-GE-19 L/C

WING AREA 204 ft² TEMP 80°F PR ALT S.L CONFIG $\delta_F = 40°$

$C_{L_{MAX}}$ 1.96 $C_{L_{TD}}$ 1.619 $C_{L_{50}} = C_{L_{TD}}/(1.09)^2$ 1.363 $\Delta C_{D_{CH}}$ —

α_{TD} — σ .961 μ .30 R/S_{GL} 20 FPS

1	GROSS WEIGHT		LB	12400			
2	$V_{50} = \left(\frac{295W}{C_{L_{50}}\sigma S}\right)^{1/2}$	V_{50_L}	KNOTS	13690			
		V_{50_L}		117			
3	$V_{TD} = \left(\frac{295W}{C_{L_{TD}}\sigma S}\right)^{1/2}$	V_{TD_L}	KNOTS	11525			
		V_{TD}		107.3			
4	$C_{D_{50}}$			.191			
5	$D_{50} = C_{D_{50}} \frac{V_{50}^2 \sigma S}{295^2}$		LB	1738			
6	$C_{D_{TD}} = C_{D_{POLAR}} - \Delta C_{D_{G.E.}}$			.243			
7	$qS @ V_{TD} = \frac{V_{TD}^2 \sigma S}{295^2}$		LB	7659			
8	$D_{TD} = C_{D_{TD}} qS$		LB	1861			
9	T_{50} IDLE		LB	1130			
10	T_{TD} IDLE		LB	1120			
11	$F_{GL} = R/S\ G.W./V_{50}(1.689)$		LB	1255			
12	$F_{50} = D_{50} - T_{50}$		LB	608			
13	$F_{TD} = D_{TD} - T_{TD}$		LB	741			
14	$C_{D_{TD}} + \Delta C_{D_{CHUTE}}$			.243			
15	$D_{GR} = (C_{D_{TD}} + \Delta C_{D_{CH}}) qS$		LB	1861			
16	T_0 IDLE @ $V=0$		LB	1040			
17	T_{AVG} IDLE $= \frac{T_{TD} + T_0}{2}$		LB	1080			
	$X_{AIR} = \frac{50W}{F_{GL}} + \frac{.0886W(V_{50}^2 - V_{TD}^2)}{F_{50} + F_{TD}}$		FT	2258			
	$X_{GR} = \frac{.0443WV_{TD}^2}{D_{GR} - \mu W} \ln \frac{D_{GR} - T_{AVG}}{\mu W - T_{AVG}}$		FT	4152			
	$X_{TOT} = X_{AIR} + X_{TOT}$		FT	6410			

Table III-8 Sheet 1

LANDING CALCULATION

MODEL Concept J

ENGINE _____

WING AREA 350 ft^2 TEMP 80°F PR ALT S.L. CONFIG $\delta_F = 25°$

$C_{L_{MAX}}$ 1.68 $C_{L_{TD}}$ 1.388 $C_{L_{50}} = C_{L_{TD}}/(1.09)^2$ 1.168 $\Delta C_{D_{CH}}$ —

α_{TD} _____ σ .961 μ .3 R/S_{GL} 20 FPS

1	GROSS WEIGHT		LB	16497			
2	$V_{50} = \left(\frac{295W}{C_{L_{50}}\sigma S}\right)^{1/2}$	V_{50}^2 / V_{50}	KNOTS	12387 / 111.3			
3	$V_{TD} = \left(\frac{295W}{C_{L_{TD}}\sigma S}\right)^{1/2}$	V_{TD}^2 / V_{TD}	KNOTS	10424 / 102.1			
4	$C_{D_{50}}$			.132			
5	$D_{50} = C_{D_{50}} \frac{V_{50}^2 \sigma S}{295}$		LB	1864			
6	$C_{D_{TD}} = C_{D_{POLAR}} - \Delta C_{D_{G.E.}}$			.170			
7	$qS @ V_{TD} = \frac{V_{TD}^2 \sigma S}{295}$		LB	11884			
8	$D_{TD} = C_{D_{TD}} qS$		LB	2020			
9	T_{50} IDLE		LB	1120			
10	T_{TD} IDLE		LB	1120			
11	$F_{GL} = R/S\ G.W./V_{50}(1.689)$		LB	1755			
12	$F_{50} = D_{50} - T_{50}$		LB	744			
13	$F_{TD} = D_{TD} - T_{TD}$		LB	900			
14	$C_{D_{TD}} + \Delta C_{D_{CHUTE}}$			.170			
15	$D_{GR} = (C_{D_{TD}} + \Delta C_{D_{CH}}) qS$		LB	2020			
16	T_0 IDLE @ $V = 0$		LB	1040			
17	T_{AVG} IDLE $= \frac{T_{TD} + T_0}{2}$		LB	1080			
	$X_{AIR} = \frac{50W}{F_{GL}} + \frac{.0886W(V_{50}^2 - V_{TD}^2)}{F_{50} + F_{TD}}$		FT	2216			
	$X_{GR} = \frac{.0443WV_{TD}^2}{D_{GR} - \mu W} \ln \frac{D_{GR} - T_{AVG}}{\mu W - T_{AVG}}$		FT	3680			
	$X_{TOT} = X_{AIR} + X_{TOT}$		FT	5896			

8

Table III-8 Sheet 2

where:

X_{gr} = Braking distance - ft

W = Weight - lb

V_{TD} = Touchdown speed $= \sqrt{\frac{295W}{C_L S_\sigma}} = 1.10\ V_{stall}$ - kn

σ = Ambient density ratio

D_{gr} = Thrust required at V_{TD} - lb

μ = Braking coefficient of friction = .30 (hard-surface runway)

$T_{avg} = \frac{T_{TD} + T_o}{2}$ where: T_{TD} is thrust at V_{TD} - lb

and T_o is thrust at V_o - lb

The derivation of this formula is shown below.

Derivation:

Summation of decelerating forces $= F = M(-a) = \frac{W}{g} \frac{dV}{dt}$

Since: $dx = V\,dt$

Then: $F = -\frac{W}{g} V \frac{dV}{dx}$

and: $X = -\frac{W}{g} \int_{V_1}^{V_2} \frac{Vdv}{F}$ since F can be expressed as $D-T+\mu\ (W-L)$

Then: $X = -\frac{W}{g} \int_{V_1}^{V_2} \frac{Vdv}{(\mu\ W-T) + (C_{D\mu} - \mu\ C_L)\ q\ S}$

Let: $a = (\mu\ W-T)$ and $b = (C_D - \mu\ C_L)\ \frac{1}{2}\ \rho S$

Then: $X = -\frac{W}{g} \int_{V_1}^{V_2} \frac{Vdv}{a + bV^2}$

If: V_1 = Initial velocity

V_2 = Final velocity; $V_2 = 0$

For constant touchdown attitude:

$$C_D = C_{D_{TD}}$$

$$C_L = C_{L_{TD}}$$

and $$C_{L_q} S = W$$

$$X = \frac{.0443W\ V_{TD}^{\ 2}}{D_{gr} - \mu W} \log_e \frac{D_{TD} - T_{avg}}{\mu W - T_{avg}}$$

c. Distance vs Gross Weight

46 Landing distance as a function of gross weight is presented on Figures III-46 and III-47 for SL and 2300 feet respectively.

7. Cruise Control

a. Method and Sample Calculation

The airspeed for long-range operation is the greater of the two speeds at which 99 percent of the maximum miles per pound of fuel are attainable at the momentary weight and altitude. The cruising ceiling is that altitude at which rate-of-climb is 300 ft/min at normal power at momentary weight. The average cruise speed is the total cruise distance divided by time for cruise but does not include the distance and time in the climb.

The fuel consumption data, regardless of source, is increased by 5 percent for all engine power conditions as a service tolerance to allow for practicable operations. Corrections and allowances are made for powerplant installation losses and detailed in Section IV. Nautical miles per pound of fuel for continuous operation (up to 100 percent maximum continuous thrust) are obtained from the following relationship.

$$\text{N Mi/lb} = \frac{V}{T \times S.F.C.} = \frac{V}{w_f}$$

where:

V = True airspeed - kn

w_f = Fuel flow - lb/hr

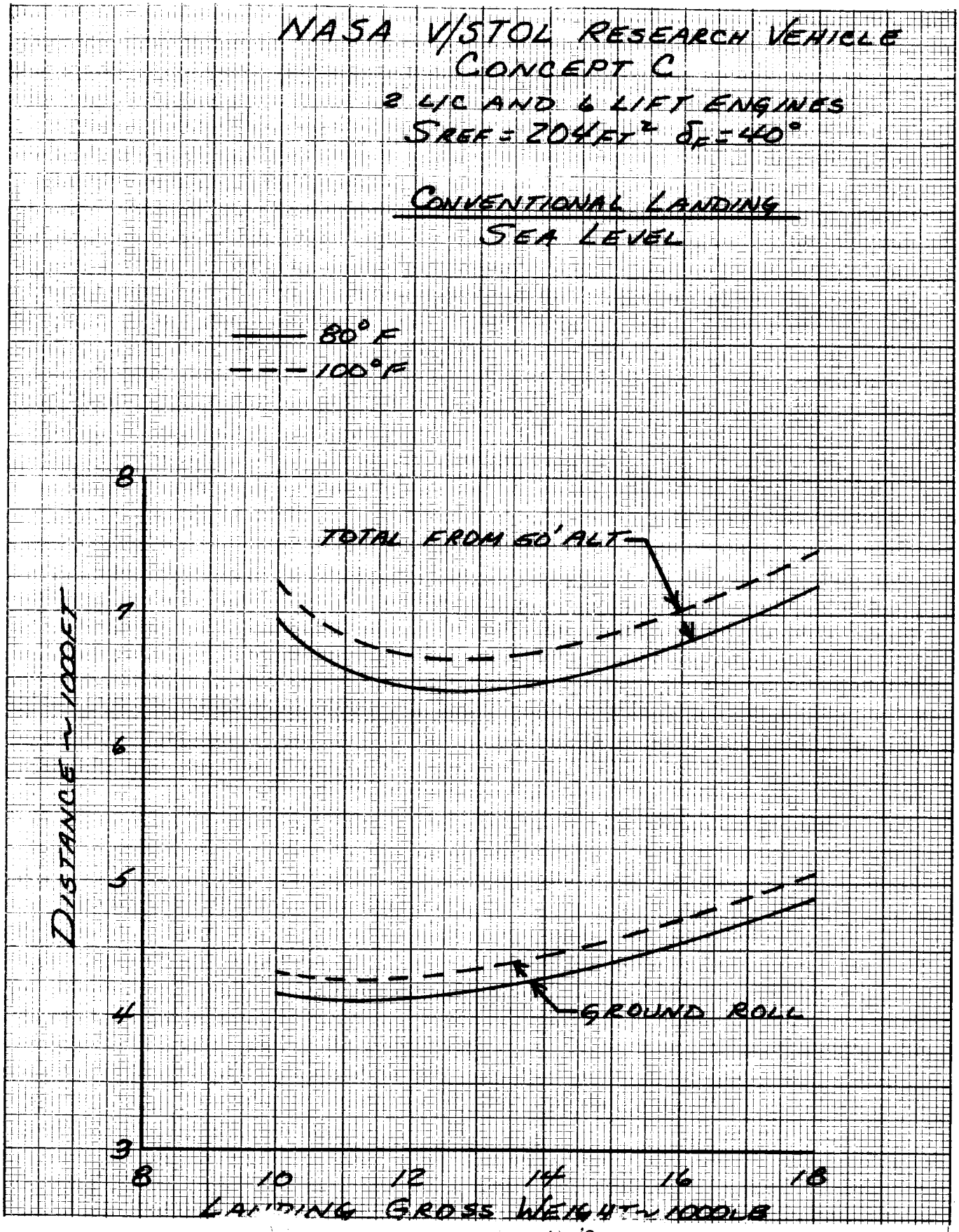

NASA V/STOL RESEARCH VEHICLE
CONCEPT C
2 L/C AND 6 LIFT ENGINES
SREF = 204 FT² δF = 40°
CONVENTIONAL LANDING
SEA LEVEL
80° F
100° F
TOTAL FROM 50' ALT
GROUND ROLL
DISTANCE ~ 1000 FT
LANDING GROSS WEIGHT ~ 1000 LB
8
7
6
5
4
3
8
10
12
14
16
18

Figure III-4

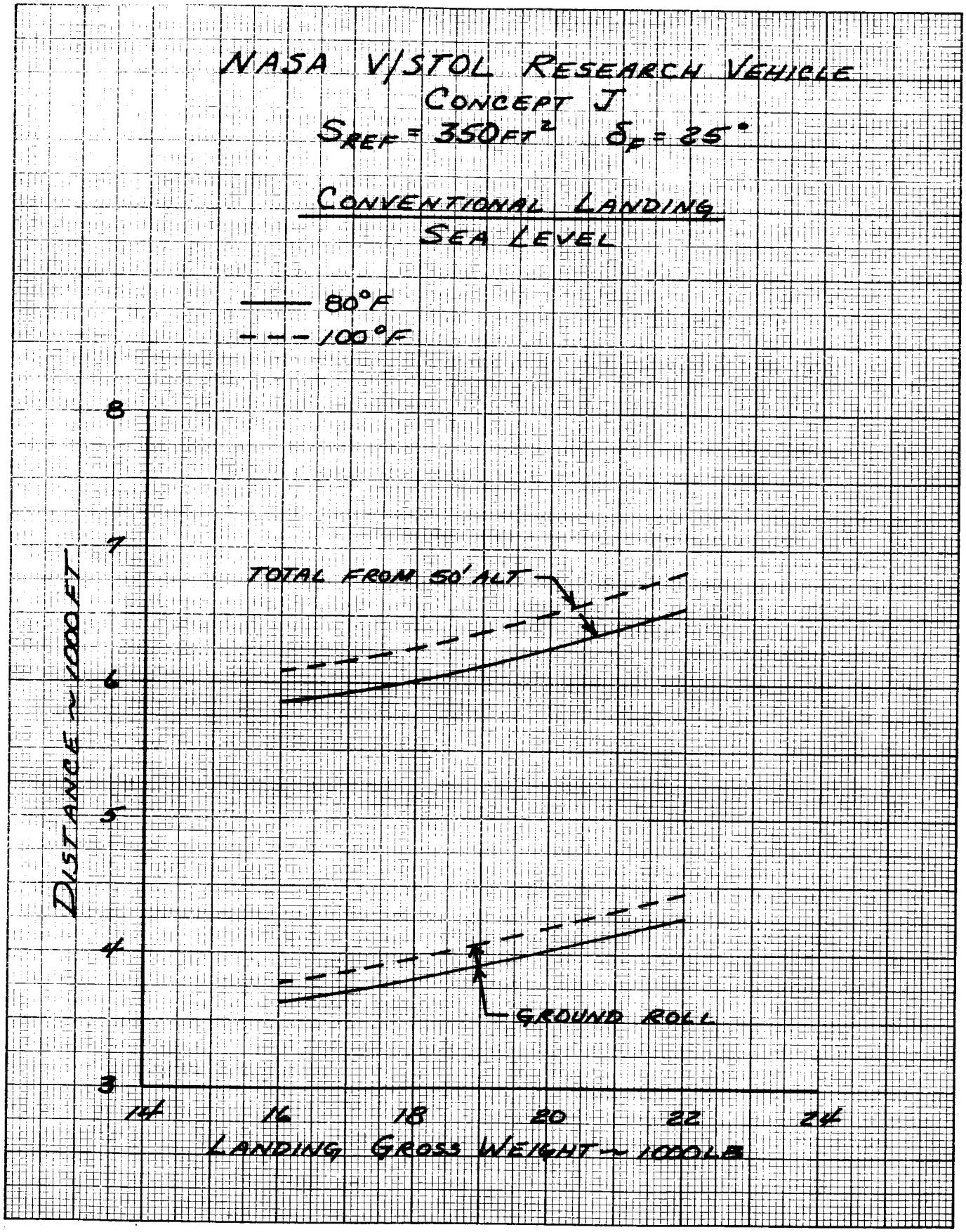


Figure III-46(2/2)

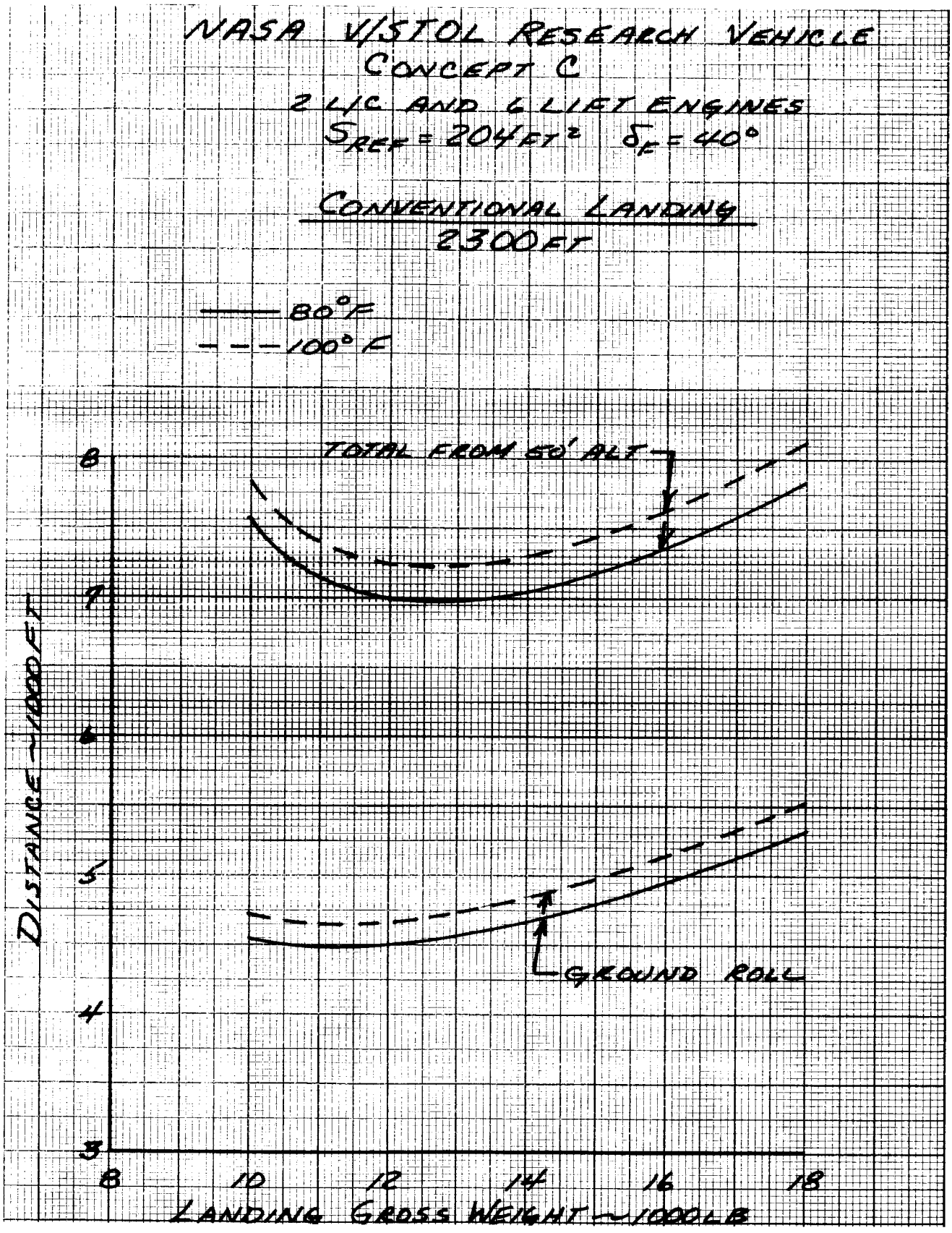


Figure III-47(1/2)

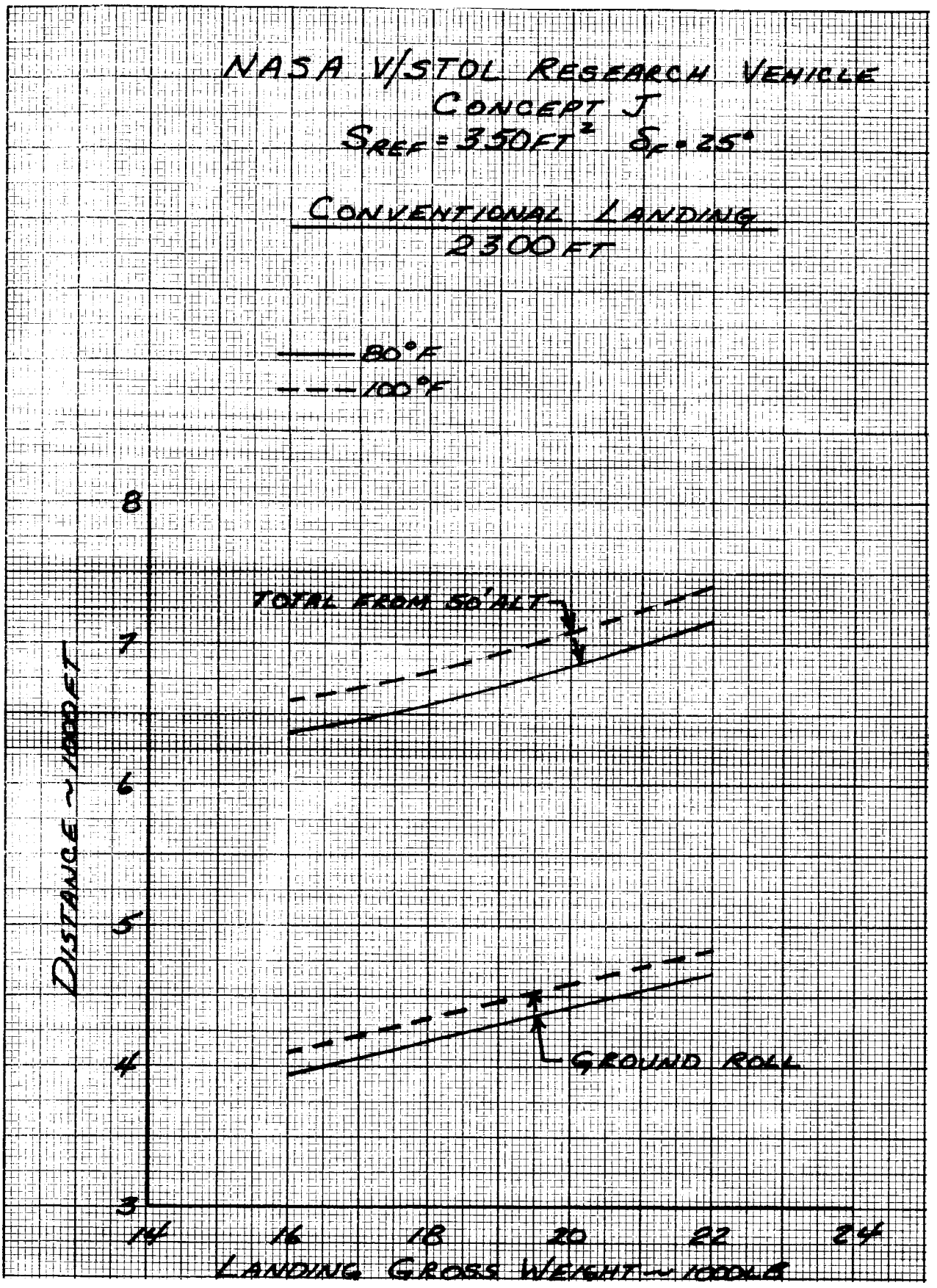


Figure III-47(2/2)

T = Thrust - lb

SFC = Specific fuel consumption - $\frac{lb/hr}{lb}$

A sample calculation of the specific cruise range as a function of speed is given in Table III-9.

b. Specific Range

The specific range is the nautical miles per pound of fuel. Calculation of the specific range over a range of airspeeds provides a maxima at which the specific range is a maximum for the momentary gross weight under study. This is the optimum range.

The variation of specific range with airspeed is presented in Figures II-48 through III-57 for Concepts C and J respectively. The maximum specific range is presented in Figure III-58.

8. Dive Recovery Data

The data in Figure III-59 is based on the following equation obtained from WADC Report M. R. Ser. No. EXP M 51/M823 for a study of design limit load factors.

$$\Delta h = 1.25 \left[\frac{V^2}{g} \log_e \left(\frac{n - \cos\gamma}{n - 1} \right) + V \sin\gamma \right]$$

where:

V = aircraft true speed in ft/sec

γ = dive angle measured from horizontal

n = load factor

Δh = altitude lost in dive recovery

The equation assumes constant speed during the maneuver and instantaneous application of load factor. The V Sinγ term approximates the altitude lost during load factor build-up. The time for this build-up, determined from test data, is approximately 2 seconds. A 25% safety factor is included to cover variations in technique and instrument error.

MODEL CONCEPT C

THRUST REQUIRED, FUEL FLOW & MILES PER POUND

Configuration CLEAN Alt.= 25000 Ft.

Gross Weight= 14000 Lbs. V_a = 601.8 σ = .4481 S = 204 Sq. Ft.

1	2	3	4	5	6	7	8	9	10	11	12	13
True Airspeed	qS	C_L	Mach Number	C_{D_o} Clean	K	C_{D_i}		C_D Total	Thrust Req.		Fuel Flow	Mi./LB.
		G.W./(2)	(1)/V_a			③² x ⑥		(5) + (7) + (8)	(2) x (9)		Fig.	(1)/(12)
									Lb.		Lb./Hr.	
361	40346	.347	.60	.0220	.075	.0090		.0311	1254		795	.2271
391	47297	.296	.65	.0216	.075	.0065		.0282	1335		851	.2297
421	54902	.255	.70	.0212	.075	.0049		.0261	1435		918	.2292
451	63063	.222	.75	.0210	.075	.0037		.0247	1560		999	.2259

Table III-9 Sheet 1

MODEL CONCEPT J

FUEL FLOW AND MILES PER POUND

ALTITUDE = 25000 FT.

G.W. 13000 LB					G.W. 16000 LB					G.W. 19000 LB				
V	T_{req}	RPM	W_f	Mi./LB.	V	T_{req}	RPM	W_f	Mi./LB.	V	T_{req}	RPM	W_f	Mi./LB.
		Fig.	Fig.	V/W_f			Fig.	Fig.	V/W_f			Fig.	Fig.	V/W_f
	Lb.		LB./HR.			Lb.		LB./HR.			Lb.		LB./HR.	
241	1003		1246	.1931	241	1210		1408	.1709					
271	1062		1318	.2054	271	1225		1454	.1863	271	1422		1636	.1655
301	1155		1426	.2110	301	1287		1542	.1949	301	1446		1696	.1773
331	1275		1570	.2108	331	1384		1674	.1976	331	1516		1808	.1830
361	1417		1748	.2065	361	1509		1842	.1959	361	1619		1960	.1841
										391	1742		2130	.1835

Table III-9 Sheet 2

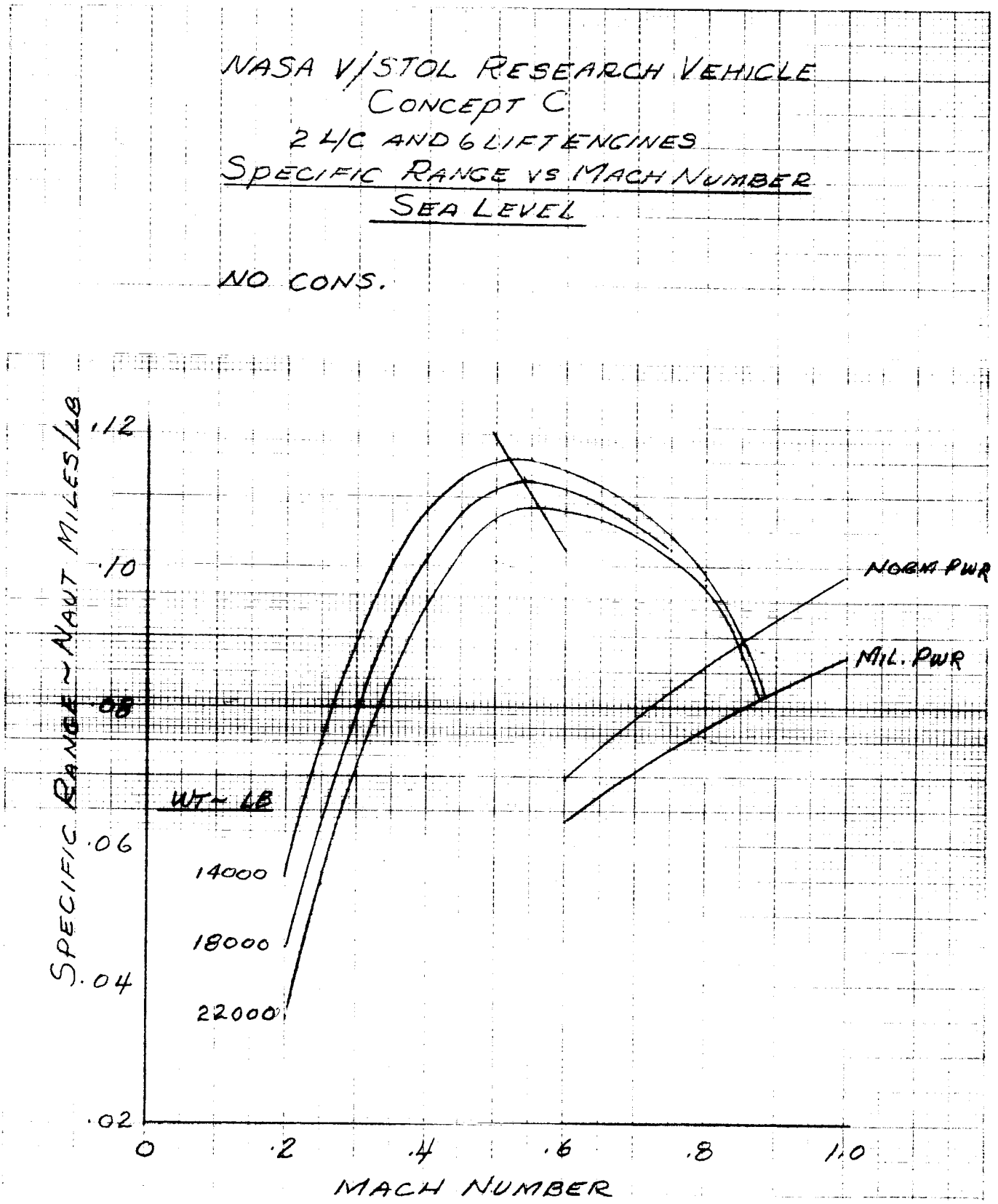


Figure III-48

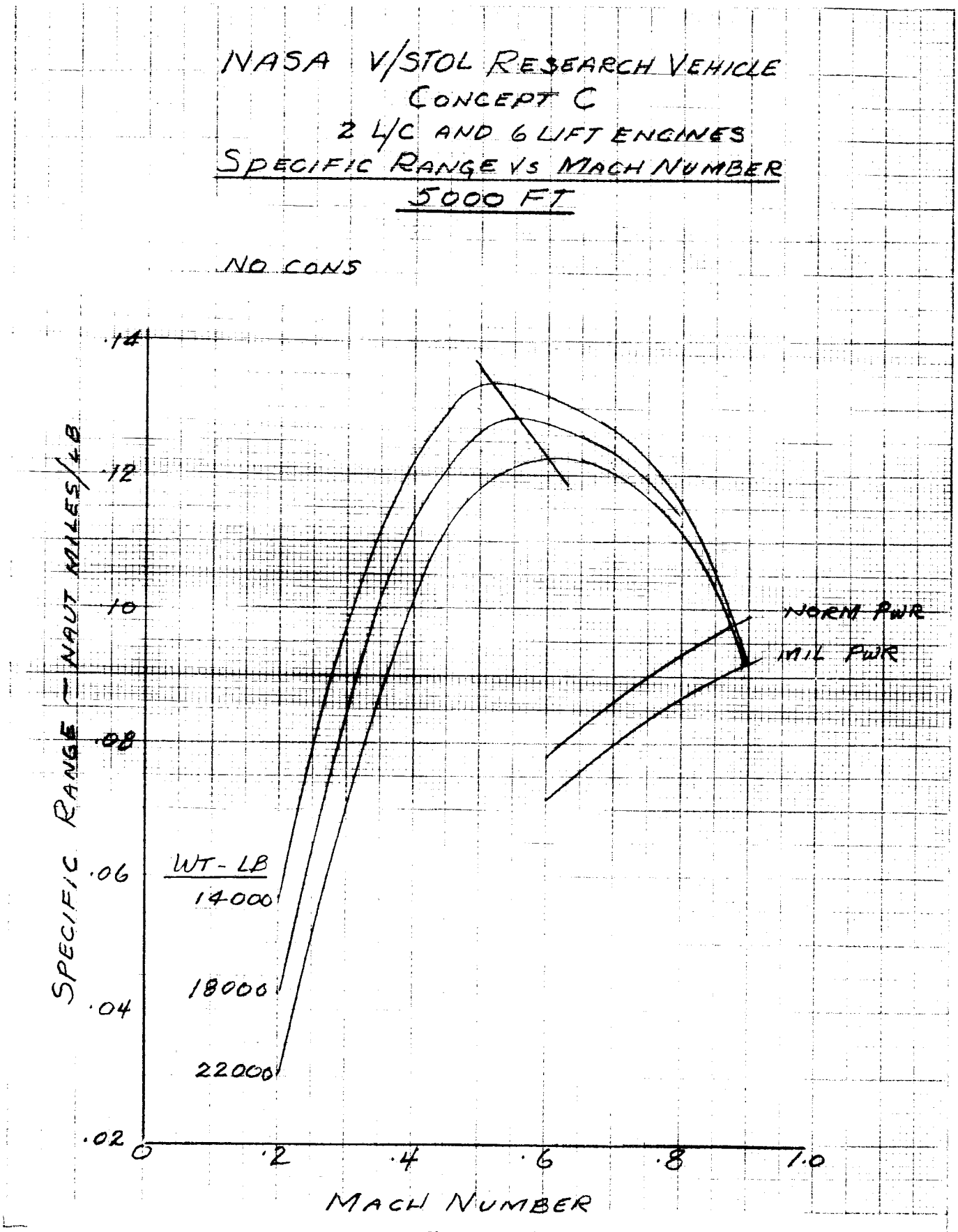


Figure III-49

III-49

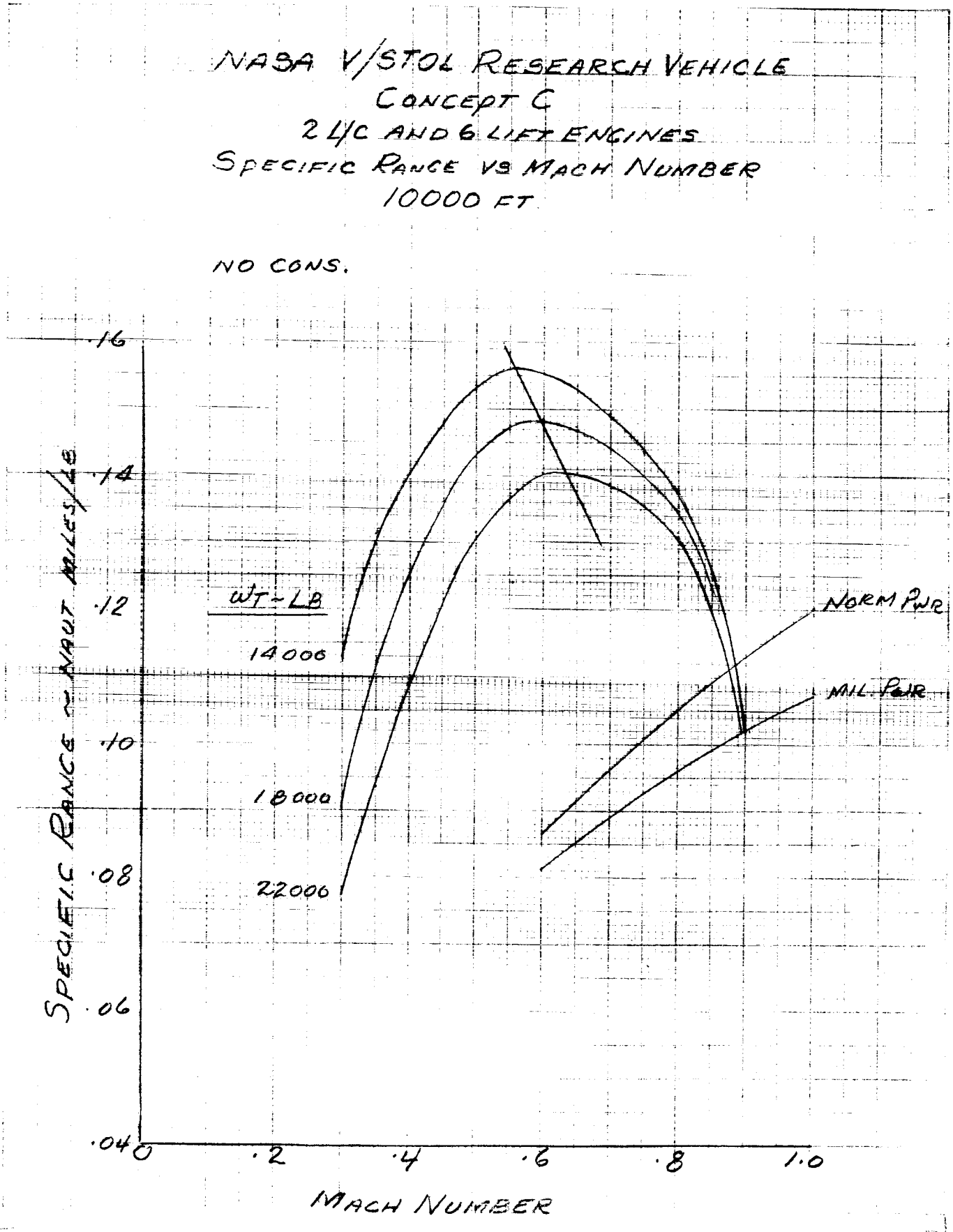


Figure III-50

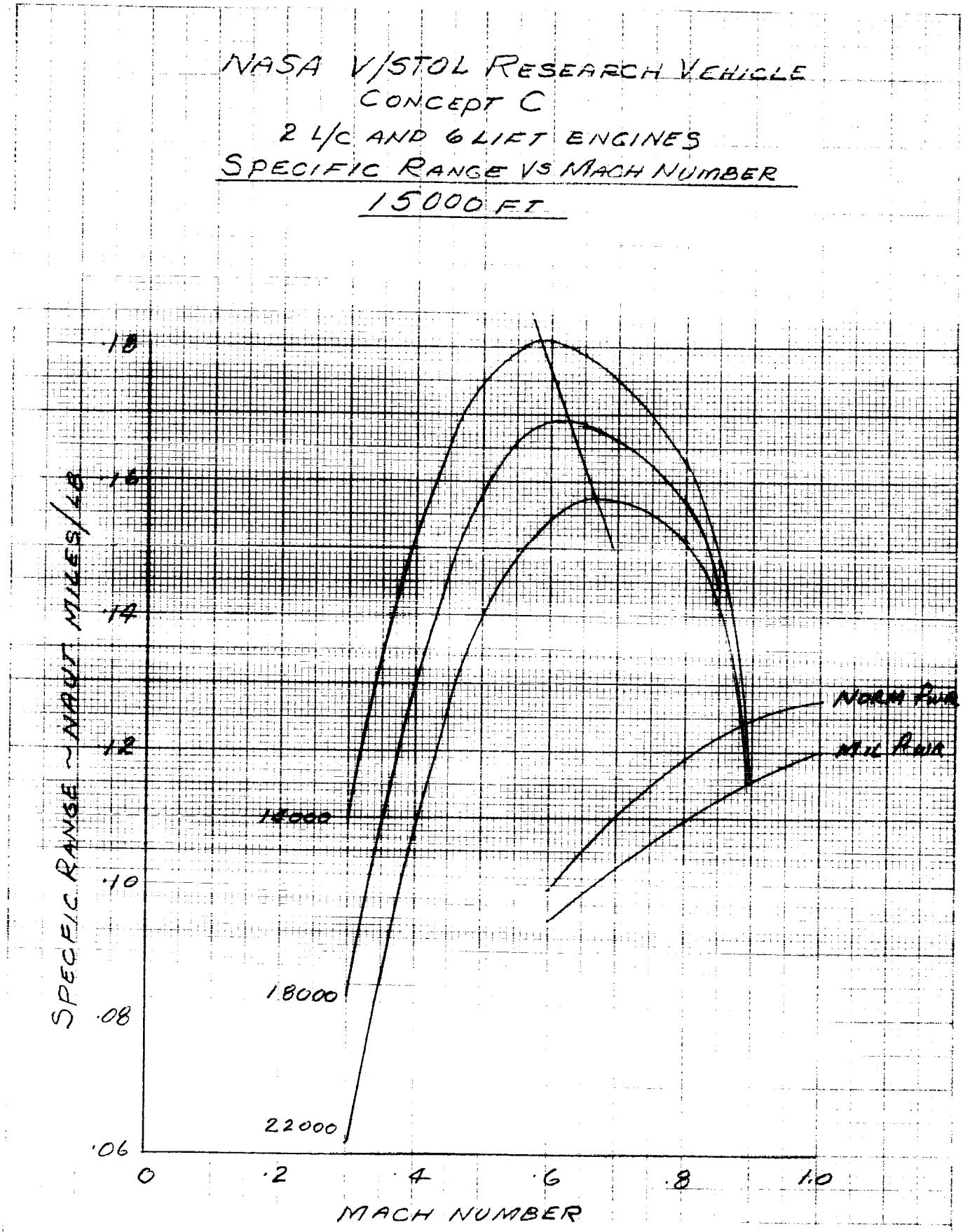


Figure III-51

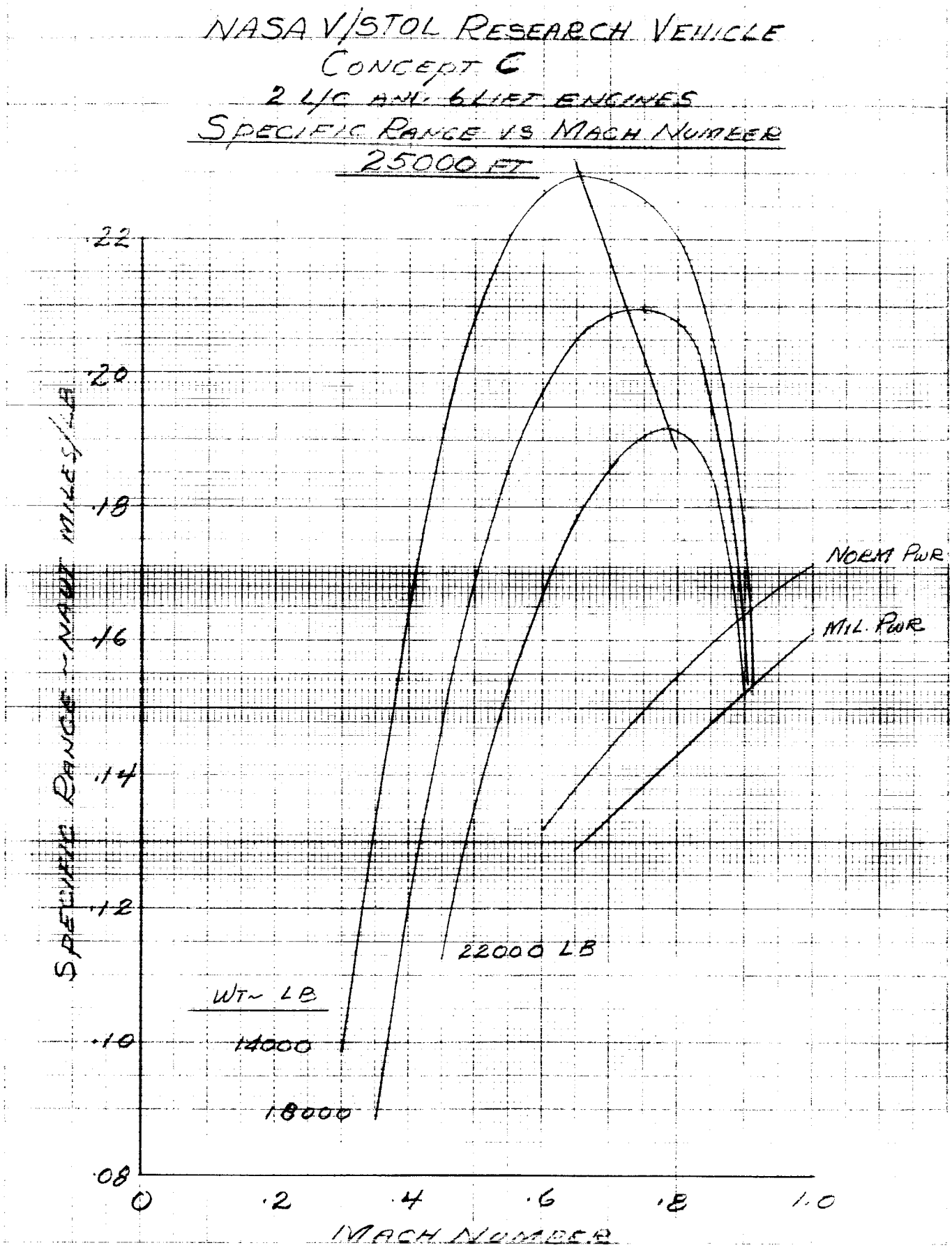
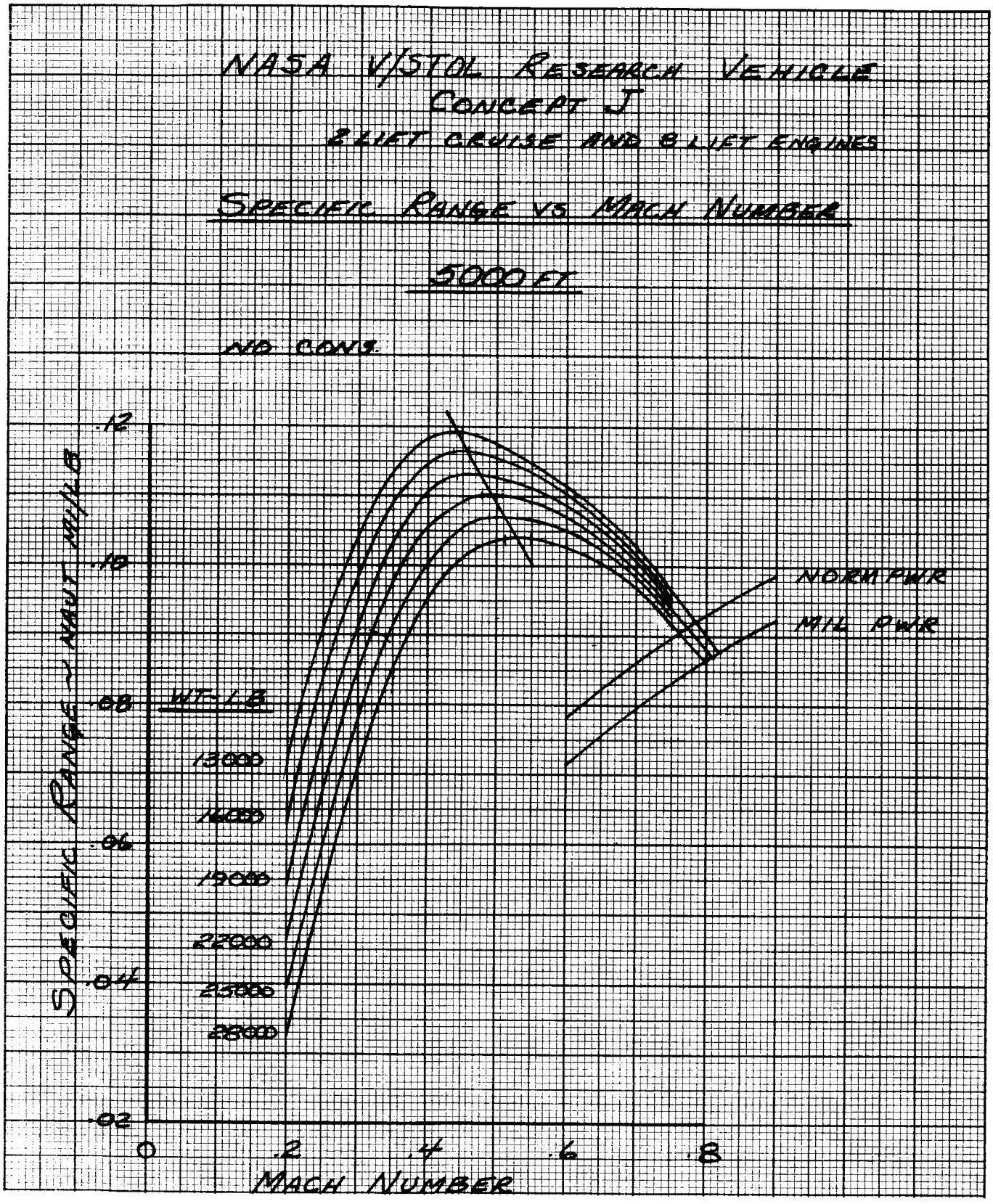


Figure III-52



54

Figure III-54

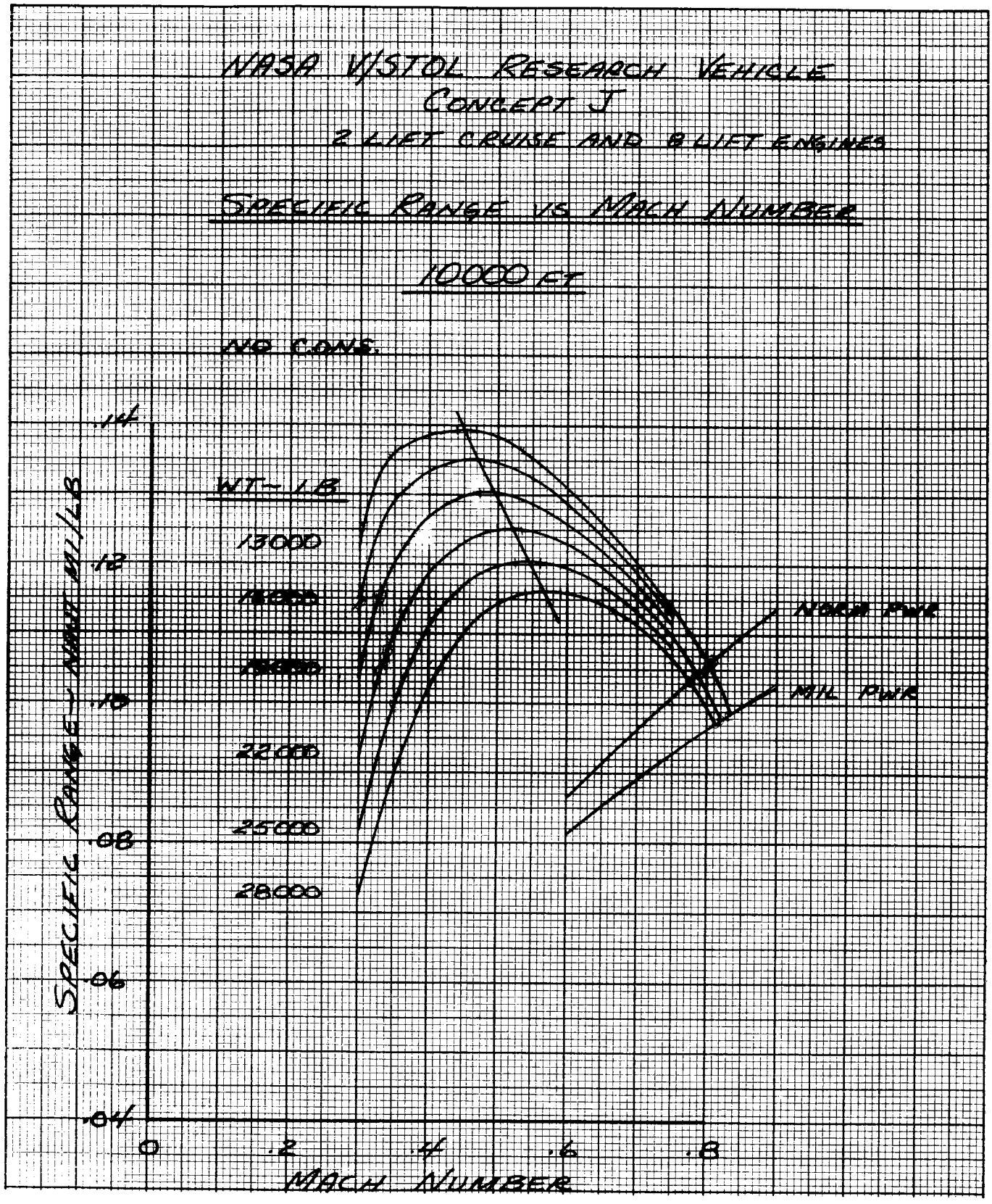
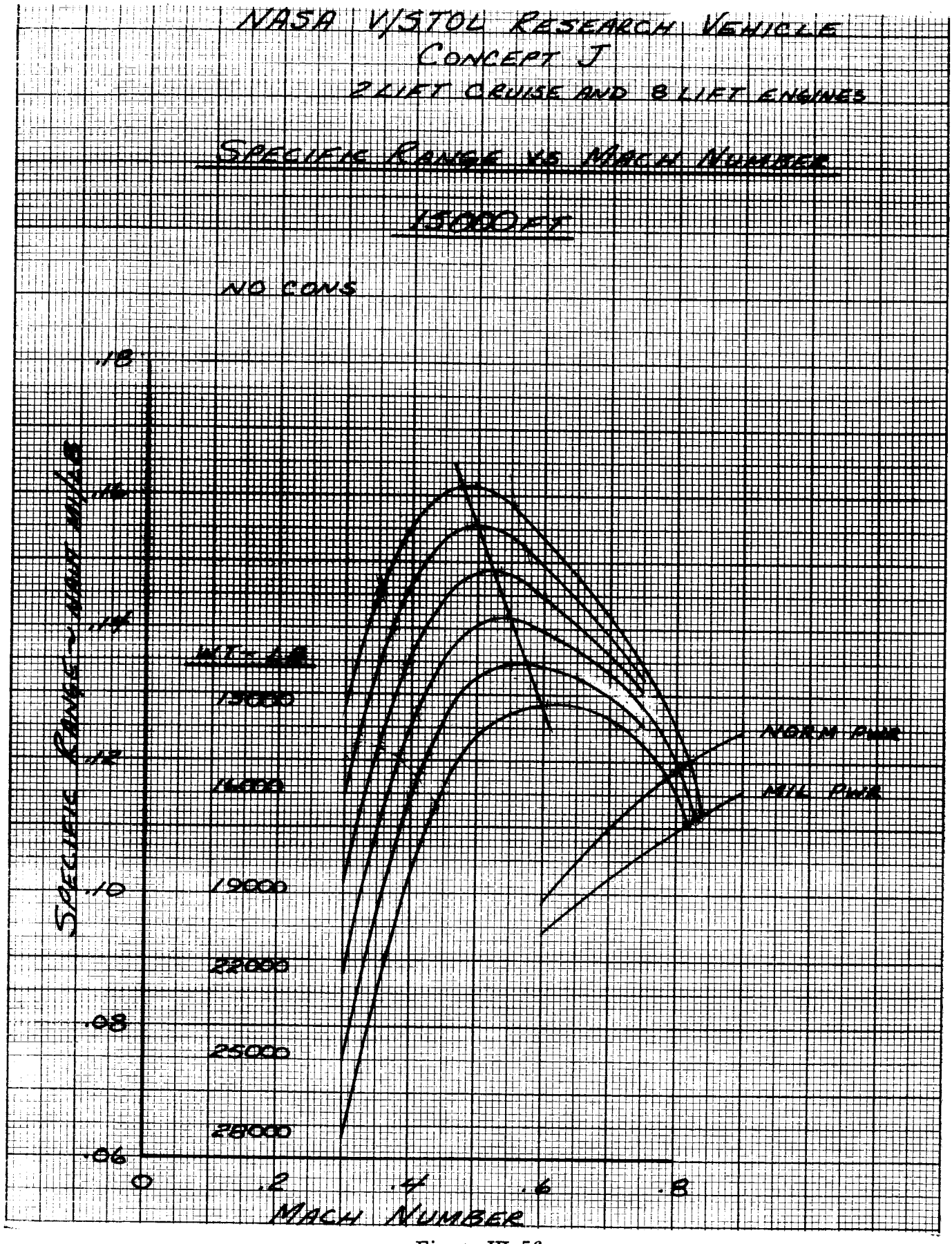


Figure III-55



III-56

Figure III-56

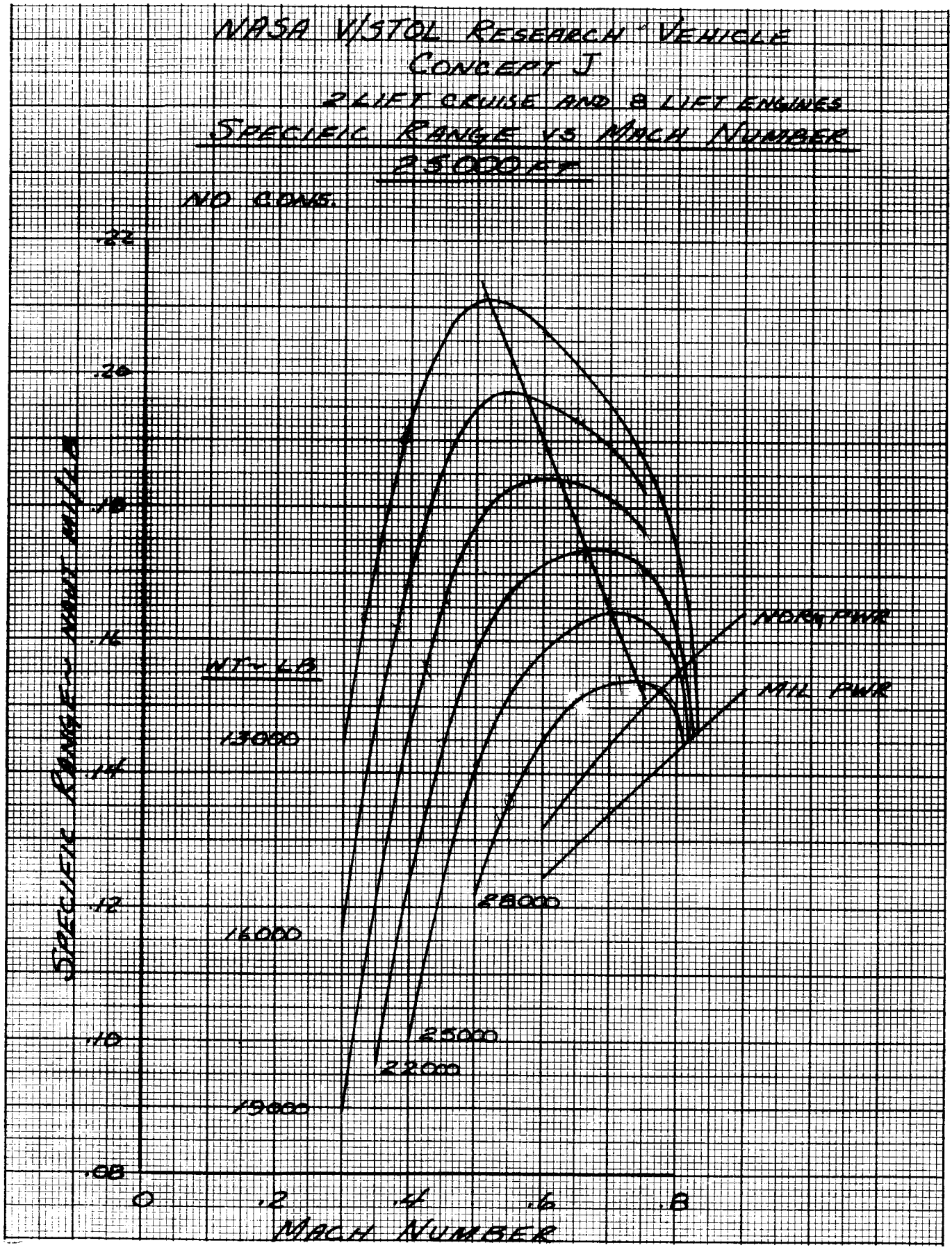
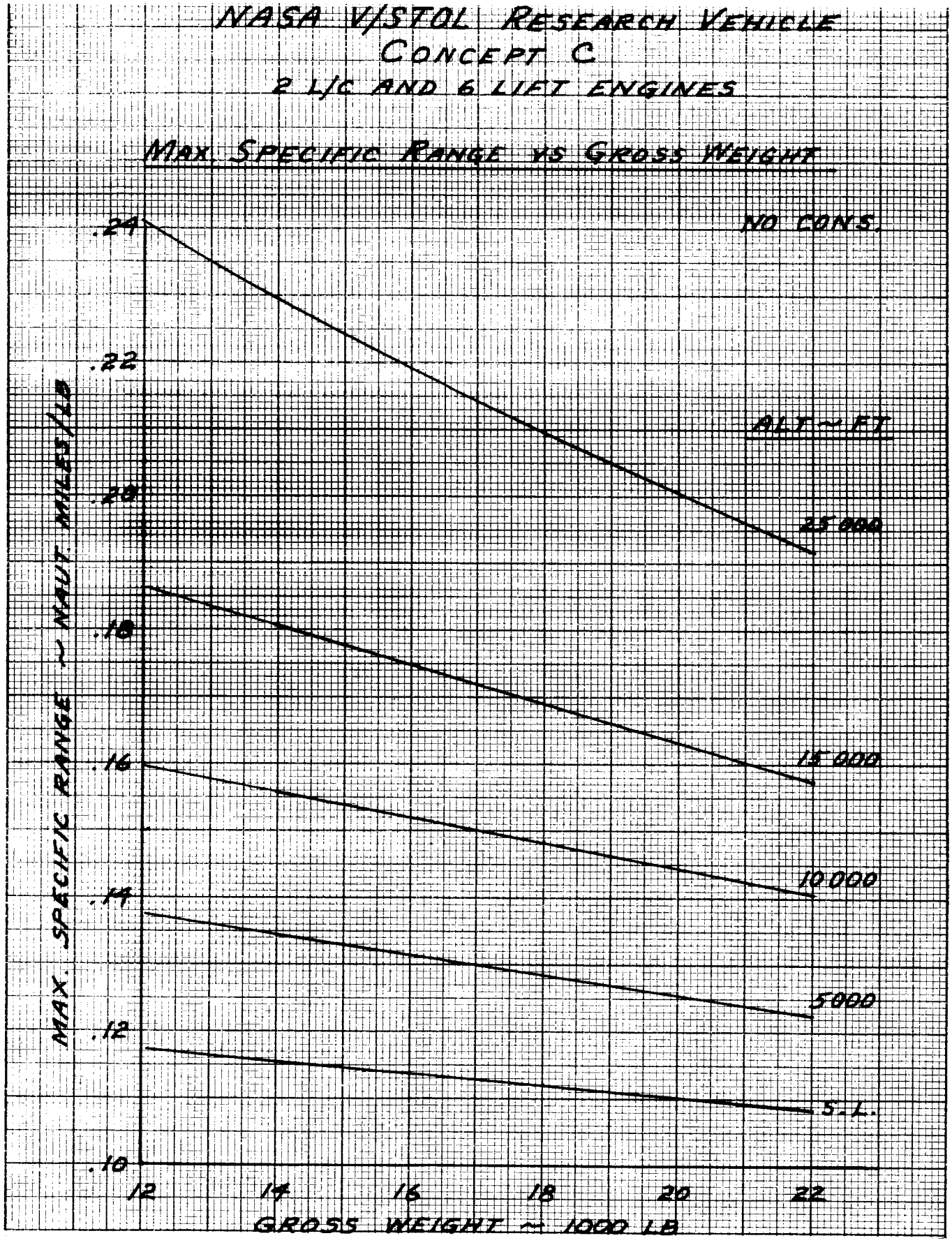


Figure III-57



III-58

Figure III-58(1/2)

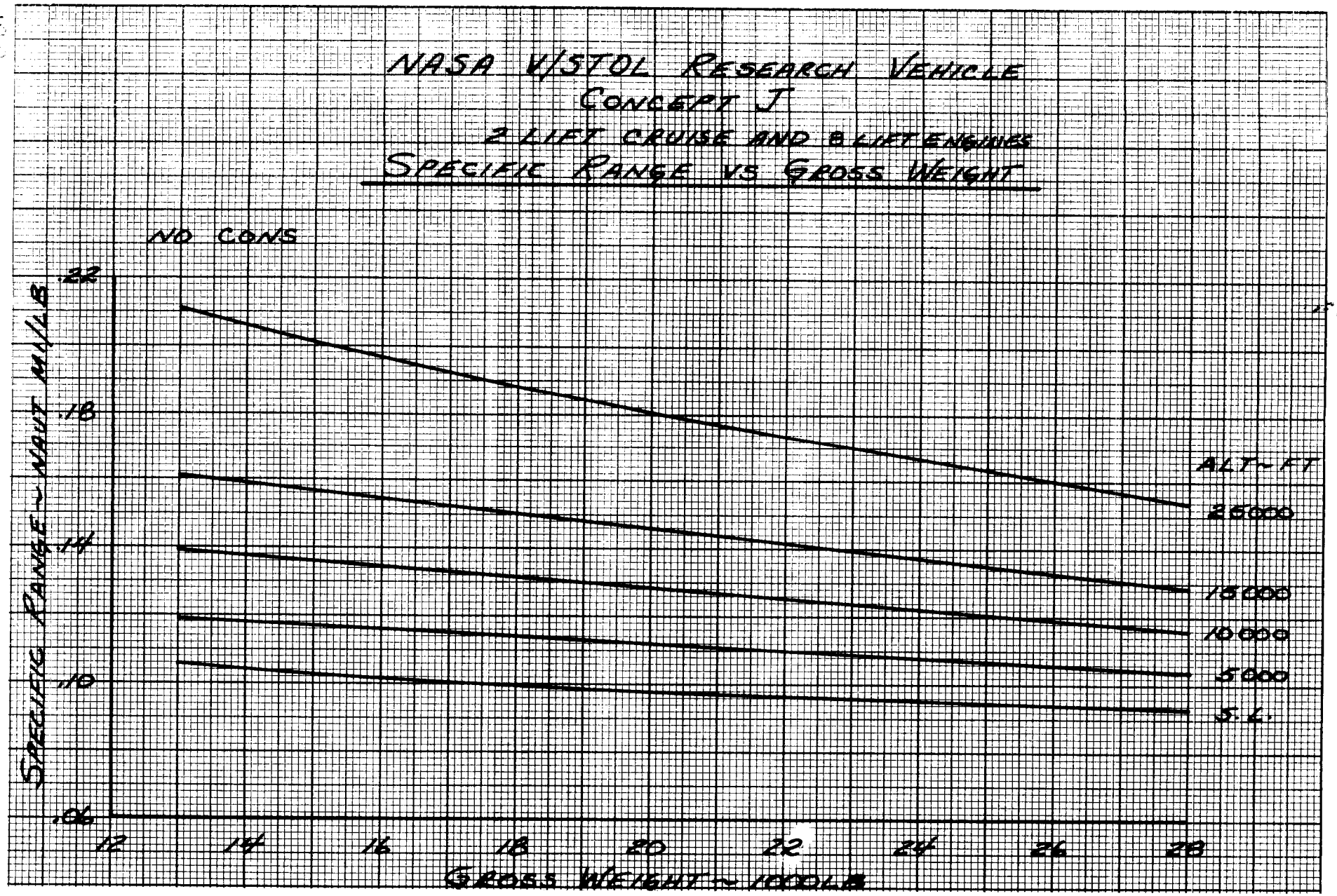


Figure III-58(2/2)

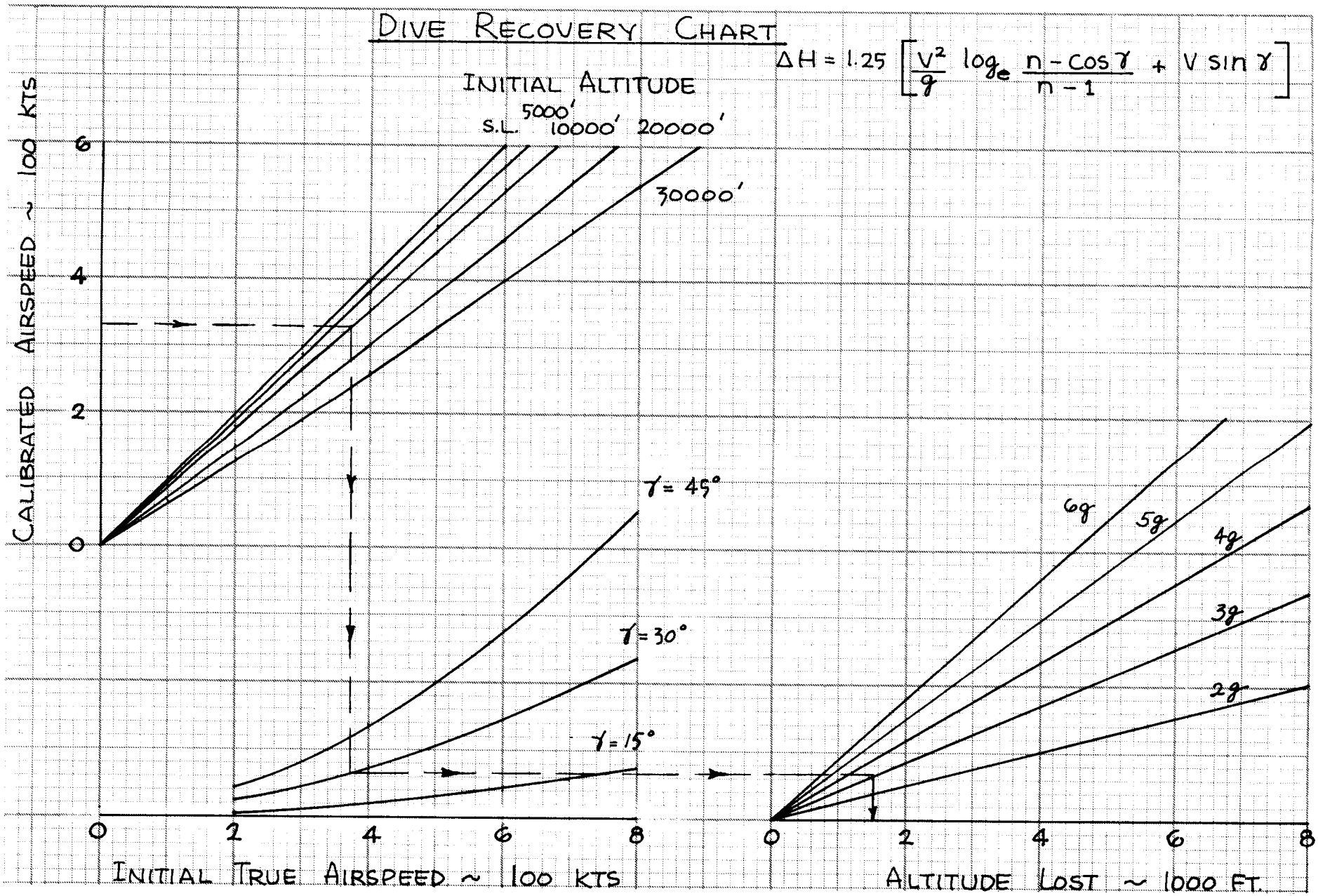


Figure III-59

9. Missions

a. Basic IFR Research Flight Pattern

To determine the potential characteristics of the proposal vehicle, a basic research flight mission has been evaluated under instrument flight regulations (IFR). This flight is studied in six segments as noted in Table III-10.

The instrument flight regulation research flight pattern meets the requirements for the basic vertical take-off and landing research mission at sea level, 80°F day. The altitude and fuel-time history are presented in the supplemental charts for the Standard Aircraft Characteristics in Section II.

Fuel consumption data has been increased by 5 percent for all engine power conditions as a service tolerance to allow for practicable operation. Allowances have also been made for power plant installation losses. The engine fuel is JP-4 (6.5 pounds per gallon).

(1) Pre-Flight

The pre-flight period starts at the time the pilot enters the airplane to the time the vehicle is in take-off configuration and position with take-off permission granted. During this interval, the lift-cruise engines are started and idled, the aerodynamic controls are checked, and the lift engines are started and trimmed for vertical take-off. All pre-flight check items are completed, and the vehicle is taxiied to the ready line to await take-off permission from the control tower.

(2) Vertical Lift-Off to Conversion

The flight segment from vertical lift-off to conversion for conventional flight starts with vertical lift-off to an altitude that is almost out of ground suck down effects. At this height, the thrust is vectored and modulated for acceleration to aerodynamic flight speed. At a safe margin to the stall speed, the lift engines are shut down and the aircraft converted to the conventional flight configuration.

TABLE III-10. BASIC IFR RESEARCH FLIGHT PATTERN

CONCEPT C (15300 lb) OPERATIONAL SEGMENT	FUEL (lb)	DISTANCE (ft)	TIME (Min)
Preflight	227	0	2.5
Vertical Lift-Off to Conversion	407	5580	1.25
Acceleration to Circuit Pattern	85	15730	0.92
Cruise in Circuit Pattern	167	97050	4.35
Deceleration to Conversion	129	36420	2.0
Conversion to Vertical Let-Down	1496	29420	5.25
TOTALS	2511	184200	16.27
Equivalent Hover Time @15300 lbs.	2511	0	9.9

CONCEPT J OPERATIONAL SEGMENT	FUEL (lb)	DISTANCE (ft)	TIME (Min)
Preflight	283	0	2.5
Vertical Lift Off to Conversion	504	5580	1.25
Acceleration to Circuit Pattern	103	19376	1.12
Cruise in Circuit Pattern	177	97050	4.35
Deceleration to Conversion	143	36420	2.0
Conversion to Vertical Let-Down	1906	29560	5.25
TOTALS	3116	187986	16.48
Equivalent Hover Time @19500 lbs.	3116	0	9.7

(3) Acceleration Climb to Cruise Altitude

The aircraft acceleratic at military power from conversion speed and altitude to cruise speed and altitude. Convention flight configuration clean ups are done at the beginning of this segment, including flaps up and gears up.

(4) Cruise

The aircraft cruises at normal power for optimum specific range in the pattern circuit to reenter the landing pattern.

(5) Deceleration to Conversion

Power is reduced and the aircraft is decelerated from cruise speed to conversion speed and altitude. Conventional flight landing preparations are done towards the end of this flight segment, including flaps down, and gear down.

(6) Conversion to Vertical Let-Down

Conversion is started from conventional flight by opening the lift-engine doors for ram air, setting the lift-cruise diverter valve and increasing the lift-cruise engine to military power to maintain speed at full power, and starting the lift engines with both ram air and engine bleed air. The thrust is trimmed and vectored for deceleration at a rate-of-descent of 500 to 800 ft/min and a speed of 80 knots.

On intercepting the localizer, a 6 degree glide slope is maintained to the outer marker. After the outer marker, the speed is decreased from 80 knots to 50 knots and the 6 degree glide slope is maintained to the middle marker. Recheck the landing list after the middle marker.

Continue deceleration to meet the threshold at 20 knots. Decelerate and translate to a position over the VTOL pad at zero velocity.

Execute a vertical let-down, land, idle then shut down all engines in accordance with the post-flight check list.

b. Basic Ferry Mission

The ferry mission starts with full internal fuel and is a conventional flight. Take-off and climb are performed at military power. The aircraft cruises at 25,000 ft altitude at optimum cruise speed. Range-free allowances include 5 minutes at normal power, sea level, for engine start, warm-up, and take-off, plus 20 minutes loiter at speeds for maximum endurance at sea level, plus 5 percent of initial fuel load for reserve. These are presented in the summary table, Table III-11.

Fuel consumption data has been increased by 5 percent for all engine power conditions as a service tolerance to allow for practicable operation. Allowances have also been made for powerplant installation losses. The engine fuel is JP-4 (6.5 pounds per gallon).

G. PERFORMANCE DATA: HOVER AND TRANSITION FLIGHT

1. Hover

a. Hover Endurance

The aircraft is designed for a minimum of 12 minutes of hover endurance at design VTO gross weight and 5 percent compressor air bleed. An engineering study was made during the Part 1 study phase to show the effect of hovering endurance in excess of 12 minutes on the aircraft size due to additional fuel and the associated tank and structure. The aircraft control power is affected by the gross weight and inertia of each structural configuration and engine arrangement considered.

Further discussions on the hover endurance is given in Section V.

b. VTOL Performance

Vertical flight performance is presented for sea level, 80°F day with maximum vertical acceleration, critical axis control, and single engine failure. The effects of elevated temperatures and altitude was also studied plus the effect of 35 knot ground winds.

TABLE III-11. BASIC FERRY MISSION

CONCEPT C (15570 lb) OPERATIONAL SEGMENT	FUEL (lb)	DISTANCE (n. Mi.)	TIME (Min)
Take-Off	462	0	5.0
Climb	411	38	5.2
Cruise	2280	503	71.5
Loiter	372	0	20.0
Reserve	185	0	0.0
TOTAL	3710	541	101.7

CONCEPT J (19830 lb) OPERATIONAL SEGMENT	FUEL (lb)	DISTANCE (N. Mi)	TIME (Min)
Take-Off	462	0	5.0
Climb	672	47	9.7
Cruise	2199	385	67.0
Loiter	752	0	20.0
Reserve	215	0	0.0
TOTAL	4300	432	101.7

The lift and total reaction control power available includes the effects of hot gas reingestion and base losses. The thrust available on Concept C is shown on Table III-12 for maximum control (10 percent bleed) at three power settings, and a lower control (6 percent bleed) at full power. The effects of altitude and temperature are shown for one elevated condition on Table III-13 for 2300 ft altitude, 100°F.

(1) Vertical Flight, Maximum Vertical Acceleration

With all engines operating at sea level on an 80° Fahrenheit day and with 50 percent of maximum control about all axes applied simultaneously, the ratio of net hover lift to design VTO gross weight shall not be less than 1.15 out of ground effect nor shall it be less than 1.05 in ground effect.

For Concept C, this is point E on Figure III-60. At sea level, 80°F conditions, there is sufficient lift and control to fulfill this requirement in or out of ground effects, normal operation. There is also sufficient additional margin for longitudinal trim for a 2 percent MAC change from the center-of-gravity to the center of lift.

For Concept J, this is point E on Figure III-61. At sea level, 80°F conditions, there is sufficient lift and control to fulfill this requirement in and out of ground effects, normal operation. Longitudinal trim due to center of gravity travel can be controlled by the additional control margin available out of ground effects; in ground effects, a decrease in the lift thrust results from the increased control requirements.

In all cases, this acceleration cannot be attained at 2300 ft, 100°F.

In the zero fuel condition, as shown by circle E, there is sufficient marginal control available for center-of-gravity trim.

(2) Vertical Flight, Critical Axis Control

With all engines operating at sea level on an 80° Fahrenheit day and with 80 percent of the maximum control about the most critical axis and

THRUST CONTROL ENVELOPE SL, 80°F

CONCEPT C ENGINES 8 TOTAL

n_c 2 L/C ENGINES [YJ85-GE-19] n_L = 6 L ENGINES [YJ85-GE-19]

1	POWER SETTING	1.000		0.800		0.600		1.000	
2	ENGINE	L/C	L	L/C	L	L/C	L	L/C	L
3	BLEED	MAX 10%	MAX 10%	MAX 10%	MAX 10%	MAX 10%	MAX 10%	NORM 6%	NORM 6%
4	THRUST/ENGINE	2120	2235	1696	1788	1272	1341	2190	2320
5	LOSS, (HOT GAS)	1.57%	0	1.57%	0	1.57%	0	1.57%	0
6	LIFT, (HOT GAS)	17580.		13351.		10549.		18230.	
4E	THRUST/ENGINE								
7E	LIFT, (HOT GAS)	15345		11563		9208		15910.	
8	CONTROL/ENGINE	248	248	212.5	214.5	178	181	150	150
9	CONTROL, (HOT GAS)	1974		1712		1442		1195	
8E	CONTROL/ENGINE								
10E	CONTROL, (HOT GAS)	1726		1495		1261		1045	
11	BASE LOSS IN GRD	0.10						0.10	
12	BASE LOSS OUT GRD	0.014						0.014	
13	BASE LOSS CONTROLS	0.02						0.02	

IN GROUND EFFECTS, NORMAL ENGINE OPERATION

14	NET LIFT	17099	13133	10436	17187
15	NET CONTROL	1934	1677	1413	1171

IN GRD EFFECTS, EMERGENCY OPERATION, SINGLE ENGINE FAILED

16E	NET LIFT	14929	11383	9110	15100
17E	NET CONTROL	1691	1465	1236	1024

OUT OF GROUND EFFECTS, NORMAL ENGINE OPERATION

18	NET LIFT	18611	14281	11343	18755
19	NET CONTROL	1934	1677	1413	1171

OUT OF GROUND EFFECTS, EMERGENCY OPERATION, SINGLE ENGINE FAILED

20E	NET LIFT	16249	12378	9902	16369
21E	NET CONTROL	1691	1465	1236	1024

Table III-12

THRUST CONTROL ENVELOPE 2300 FT, 100°F

CONCEPT C ENGINES 8 TOTAL

n_c 2 L/C ENGINES [YJ85-GE-19] n_L = 6 L ENGINES [YJ85-GE-19]

1	POWER SETTING		1.000		0.800		0.600		1.000	
2	ENGINE		L/C	L	L/C	L	L/C	L	L/C	L
3	BLEED		MAX 10%	MAX 10%	MAX 10%	MAX 10%	MAX 10%	MAX 10%	NORM 6%	NORM 6%
4	THRUST/ENGINE		1825	1923	1460	1538	1095	1154	1885	1990
5	LOSS, (HOT GAS)		1.57%	0	1.57%	0	1.57%	0	1.57%	0
6	LIFT, (HOT GAS)		15130		12101		9079		15650	
4E	THRUST/ENGINE									
7E	LIFT, (HOT GAS)		13207		10563		7925		13660	
8	CONTROL/ENGINE		216	216	186	187	155	159	131	131
9	CONTROL, (HOT GAS)		1721		1488		1239		1044	
8E	CONTROL/ENGINE									
10E	CONTROL, (HOT GAS)		1505		1301		1100		913	
11	BASE LOSS	IN GRD	0.10						0.10	
12		OUT GRD	0.014						0.014	
13		CONTROLS	0.02						0.02	

IN GROUND EFFECTS, NORMAL ENGINE OPERATION

14	NET LIFT	14741	11863	8977	14767
15	NET CONTROL	1686	1458	1233	1023

IN GRD EFFECTS, EMERGENCY OPERATION, SINGLE ENGINE FAILED

16E	NET LIFT	12868	10356	7851	12900
17E	NET CONTROL	1475	1275	1078	895

OUT OF GROUND EFFECTS, NORMAL ENGINE OPERATION

18	NET LIFT	16041	12903	9758	16113
19	NET CONTROL	1686	1458	1233	1023

OUT OF GROUND EFFECTS, EMERGENCY OPERATION, SINGLE ENGINE FAILED

20E	NET LIFT	14005	11265	8532	14065
21E	NET CONTROL	1475	1275	1078	895

Table III-13

ATTITUDE CONTROL REQUIREMENTS-CONCEPT C

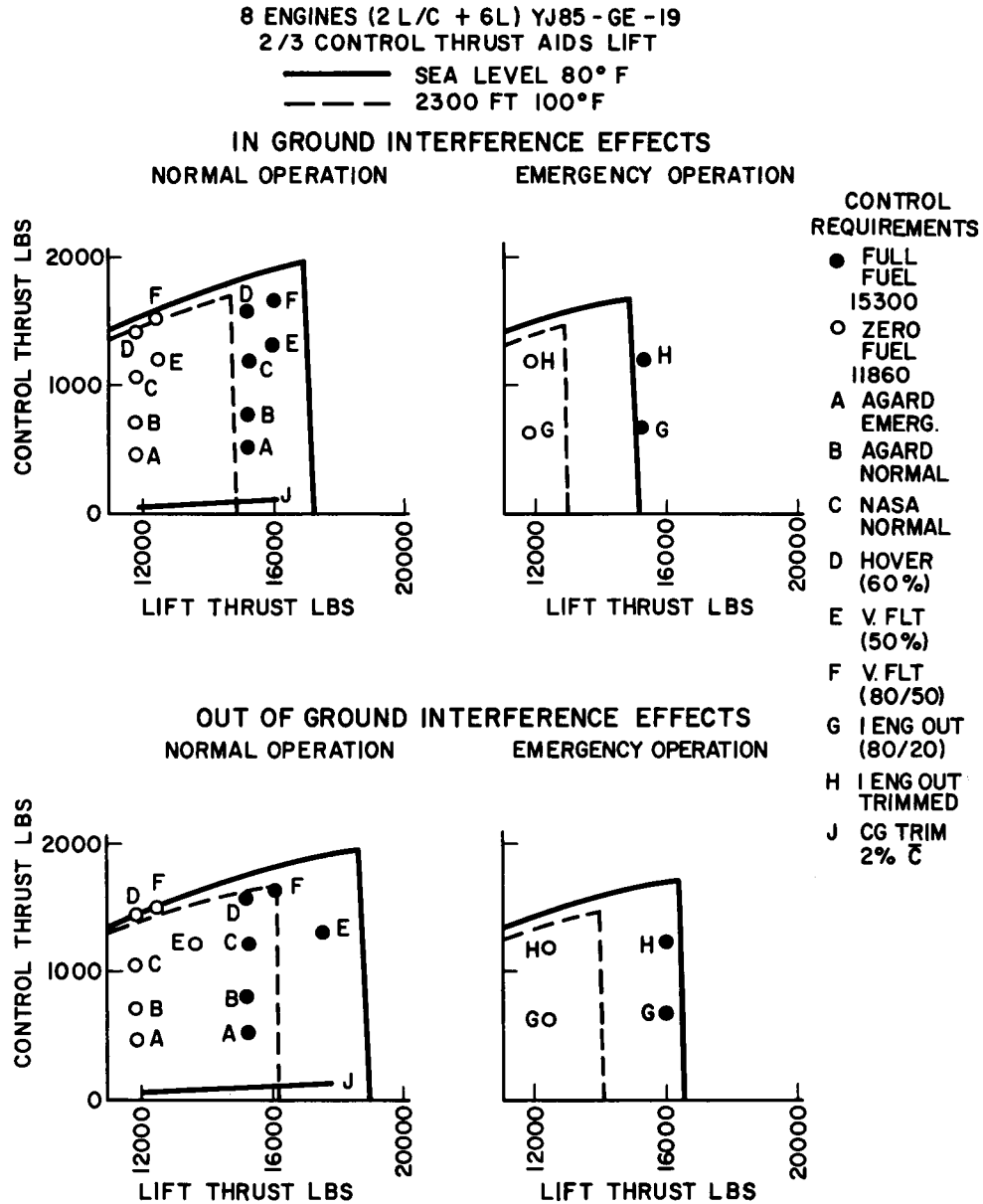


Figure III-60

ATTITUDE CONTROL REQUIREMENTS— CONCEPT J

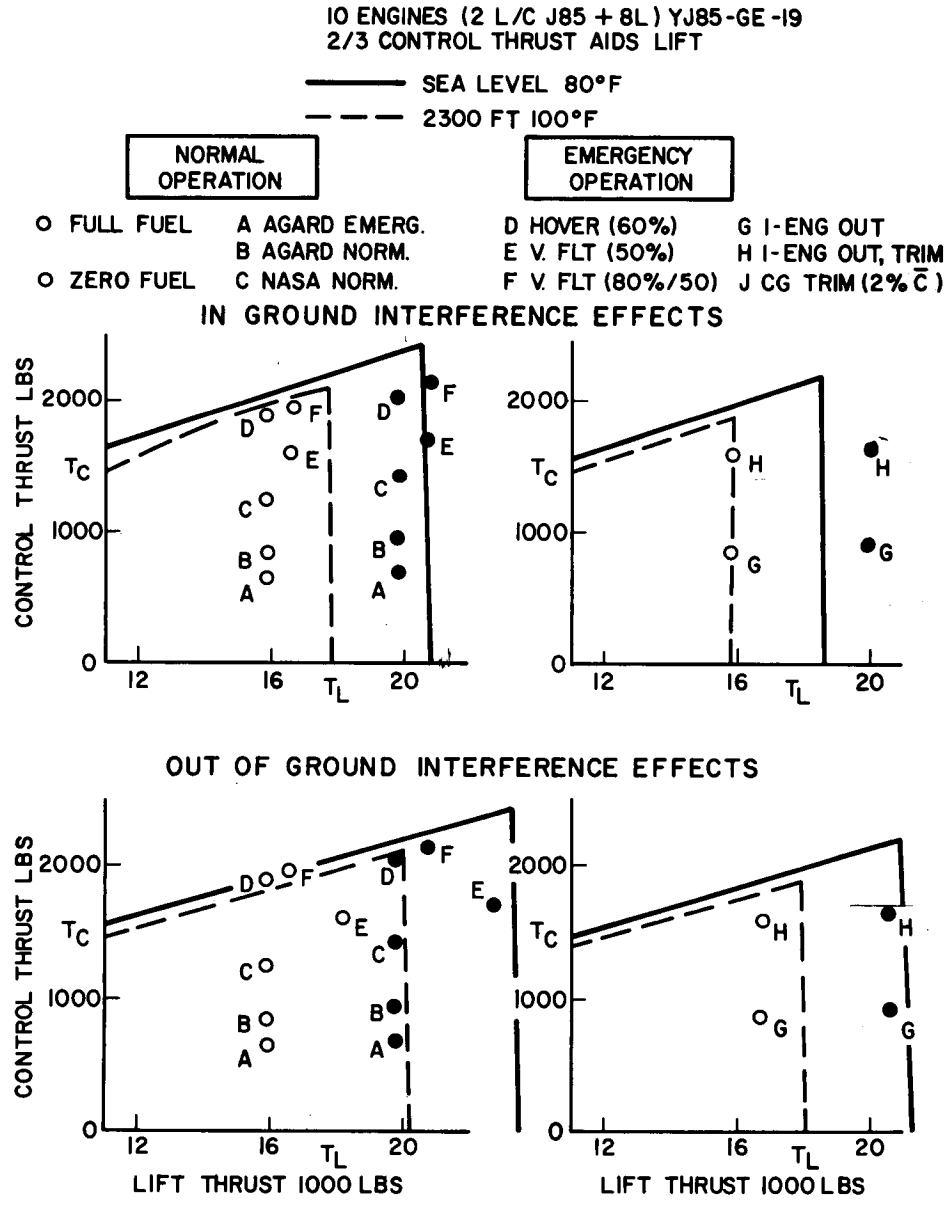


Figure III-61

50 percent about the other axes applied, the ratio of net hover lift to design VTO gross weight shall not be less than 1.05 in or out of ground effect.

For Concept C, this is point F on Figure III-60. At sea level, 80°F conditions, there is sufficient lift and control to fulfill this requirement in or out of ground effects, normal operation. There is also sufficient additional margin for longitudinal trim due to the location of the center of gravity to the center of lift.

For Concept J, this is point F on Figure III-61. At sea level, 80°F conditions, there is sufficient lift and control to fulfill this requirement in but not out of ground effects, normal operation. This is insufficient additional margin for complete longitudinal trim out of ground effects and no margin for additional trim in ground effects.

At 2300 ft, 100°F, critical axis control cannot be attained at the vertical flight lift requirements except for Concept C, out of ground effects. Even at this condition, no margin is available for longitudinal trim above the control requirement.

In the zero fuel condition, as shown by circle F, there is insufficient marginal control available for center-of-gravity trim. This condition is slightly improved in ground effects due to the higher thrust levels required for the same net lift.

(3) Vertical Flight, Single Engine Failed

The aircraft will be able to maintain controlled hovering and low-speed flight with a failure of any single engine at sea level on an 80° Fahrenheit day. Under these conditions it will have a ratio of net hover lift to design gross weight not less than 1.05 out of ground effect and using emergency ratings on the remaining engines. The aircraft can be controlled during and after an engine failure with the normal flight controls without the use of any automatic retrimming device and without requiring the pilot to shut down any other engine. During this condition, the aircraft shall retain a margin of at least 20 percent of the maximum control moment available before the failure about the pitch and yaw axes and 50 percent margin about the roll axis with the center-of-gravity in the most adverse position.

For Concept C, this is point H in Figure III-60, at emergency operation, out of ground effects at sea leve, 80°F. There is sufficient lift and control to fulfill this requirement out of ground effects, single engine out, emergency operation. The design VTO gross weight shows a growth potential above the 12 minutes of hover time presented in this figure.

For Concept J, this is point H in Figure III-61. There is no room for growth above the 12 minutes of hover time.

Insufficient lift is available at full VTO design weight at the elevated condition of 2300 ft, 100°F.

The increased control required due to center-of-gravity trim can be obtained with small losses in the net lift.

(4) Elevated Temperatures and Altitudes

An engineering study of the above VTOL performance was made to show the effect of elevated temperature (to 100 degrees Fahrenheit) and altitude (to 2300 feet).

(5) Hover in Winds up to 35 Knots

The aircraft capability to hover with sufficient control of rate of ascent and descent was studied including low-speed flight at all angles to a 35 knot wind. During vertical take-offs and landings the vertical thrust available out of ground effects is specified by AGARD Report 408, Paragraph 4.4. This is 1.05 time the aircraft weight for take-off and 1.15 time the aircraft weight for landing under the most adverse specified altitude-temperature conditions. It is assumed that 50 percent of the available control power is being used simultaneously about all these axes.

Capability relative to the vertical thrust and control power is given in item (1) at maximum vertical acceleration. During take-off, the application of full control about any one axis with 50 percent application about the remaining axis should not reduce the vertical thrust to less than the weight.

The pilot cannot obtain full control power about all axes simultaneously at any thrust margin. Roll control remains a problem area for a 35 knot wind at approximately 15 degrees yaw or sideslip due to the high roll stability of the aircraft in the hover configuration.

2. Glide Path Control at Constant Speed

At every speed below conversion speed, there is a configuration in which the aircraft is flyable continuously from military power to 1000 ft/min rate-of descent, without changing this configuration or retrimming by the pilot, and without encountering undesirable effects due to stalling or buffeting including feedback of unsteady forces on the controls.

The thrust response characteristics assure control of the vertical speed. During the final stages of a vertical landing, the vertical thrust response is demonstrated to be 21 percent of maximum value per second (which is greater than the engine specification value of 17.5 percent per second).

For demonstration purposes, the demanded thrust increase would be 10 percent of the landing weight at any power setting between hovering and 1000 ft/min rate-of-descent. The lift increase will be 60 percent of the demanded increase in no more than 0.3 seconds. In this situation, a 6 percent of maximum thrust is demanded, and a time rate of thrust increase to meet the AGARD Report 408 requirement would be 3.6 percent thrust in 0.3 seconds, or 12 percent thrust increase per second.

3. Acceleration: Hover to Aerodynamic Flight

Acceleration performance characteristics were determined from 2-degree of freedom trajectory studies in both level flight and climb. Two vectoring schedules were used:

1) The thrust of the two lift/cruise engines remained vertical while the thrust of the six lift engines was vectored 30 degrees aft of vertical.

2) The thrust of all eight engines was vectored 30 degrees aft of vertical.

The only deviation from the above vectoring schedule was made during the initial part of the level-flight acceleration from hover at constant angle-of-attack with the lift cruise thrust restored aft 30°. To maintain level flight at these conditions, the lift engines were vectored to a more vertical position at full throttle. At all other constant angles-of-attack in the level flight acceleration, a partial throttling was used. Full power was used for all climb acceleration. A list of the acceleration modes presented are given in Table III-14.

For transition flight speeds, the desired lateral control margins for trim are to be achieved to a maximum display angle equivalent to a 35-knot side wind but not to exceed 15 degrees sideslip. Thus, the 15-degree sideslip governs at low speeds and a 35-knot side wind governs at high speeds.

An examination of the presented results shows that the most efficient maneuver (shortest time to 1.5Vs) occurs for the level flight acceleration with angle-of-attack varying and all engines rectored 30 degrees aft of vertical. (Figure III-62.) The acceleration time from hover to 1.5 Vs is 24.5 seconds. This maneuver is also the simplest from a piloting standpoint, since no engine vectoring or throttle manipulations are required duringtransition. Also, the maximum trim moment for this maneuver is 18,000 ft-lbs, which is 60 percent of the available pitch control moment.

a. Constant Altitude

Level flight accelerations at constant altitude were studied at constant angles-of-attack and varying angle-of-attack. Sample results are shown on Figures III-61, -62 and -63. All accelerations were run at constant attitude angle until sufficient dynamic pressure was developed to maintain an angle of attack reference or line-of-sight reference (when angle-of-attack varies).

The longitudinal trim required is fairly high in Figure III-63, zero angle-of-attack. They are lower at -5 degree angle-of-attack, Figure III-64, but lateral control becomes increasely critical with speed under this flight mode. Figure III-62 represents a reasonable flight profile for constant altitude acceleration with varying angle-of-attack but always positive.

TABLE III-14. SUMMARY OF ACCELERATION MODES

Engine Tilt (°)		Thrust	Angles (°)			Figure
Lift	L/C		γ	θ	α	
30	0	Full	0	Vary	Vary	III-61
30	30	Full	0	Vary	Vary	III-61
30	0	Vary	0	0	0	III-62
Vary-30	30	Vary	0	0	0	III-62
Vary	0	Full	0	-5	-5	III-63
Vary	30	Full	0	-5	-5	III-63
30	0	Full	Vary	0	Vary	III-64
30	30	Full	Vary	0	Vary	III-64
30	0	Full	Vary	5	Vary	III-65
30	30	Full	Vary	5	Vary	III-65
30	0	Full	5	Vary	Vary	III-66
30	30	Full	5	Vary	Vary	III-66

b. Constant Attitude

Accelerations were studied at constant attitude angles with some of the results presented in Figures III-63, -64, -65, and -66. Comments on the constant angle-of-attack and flight path data presented on Figures 1 and 2 were reviewed in section subparagraph a, above. Review of the angle-of-attack variation with speed indicates that tilting the lift-cruise nozzle 30 degrees aft of vertical provides the most favorable angle-of-attack profile.

c. Constant Angle-of-Attack

Level flight accelerations were studied at constant angles-of-attack. This is not a desirable mode of flight based on cross wind considerations. The results of two studies are shown in Figures III-63 and III-64 which were reviewed in subparagraph a, above.

d. Positive Rate-of-Climb

Acceleration studies at a positive rate-of-climb were performed and four results are presented on Figures III-62 and III-67. Tilting all engines aft of the vertical provides the most favorable angle-of-attack variation with speeds for lateral control with cross wind effects.

4. Deceleration: Aerodynamic Flight to Hover

Sufficient deceleration studies have been performed to show that the aircraft is capable of decelerating from 1.2 power-off stall speed (flaps up, landing gear retracted) to hover in no more than 30 seconds (15 seconds desired). This deceleration is performed rapidly and safely, at approximately constant altitude, to stop and hover.

This maneuver is executed without restriction due to longitudinal control power, longitudinal trim, stalling or buffeting, or to engine thrust or response characteristics. The pilot need operate only the primary flying controls plus power setting and tilt of the thrust vector after the initial trim change required during conversion. The maneuver will generally be performed out-of-ground effects.

Following a failure in a stability augmentation system or a single critical lift engine, it is possible to execute the transition maneuver without restriction and to make an approach and landing.

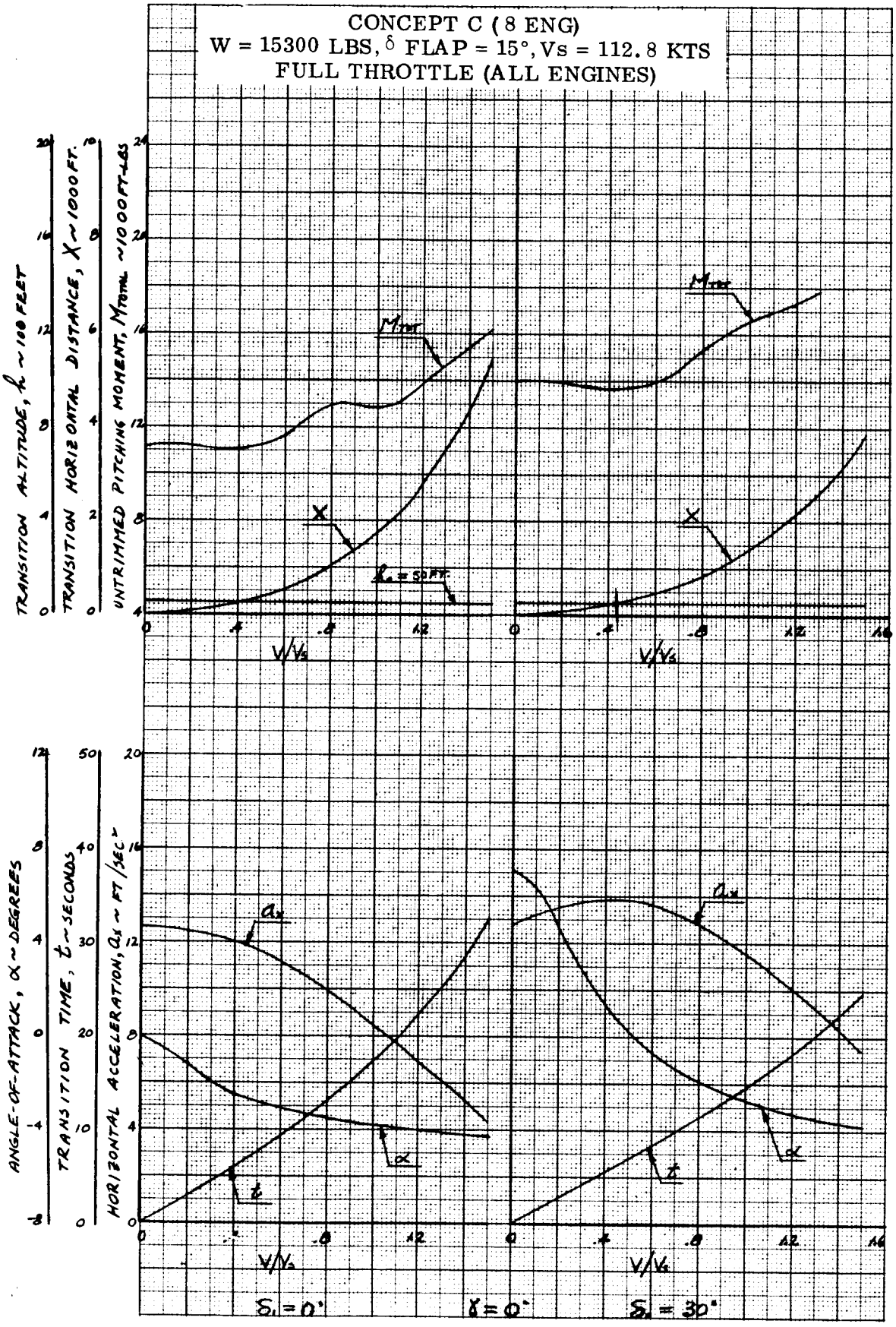


Figure III-62. Transition Performance-Constant Flight-Path Angle
($\gamma = 0°$, Variable α)

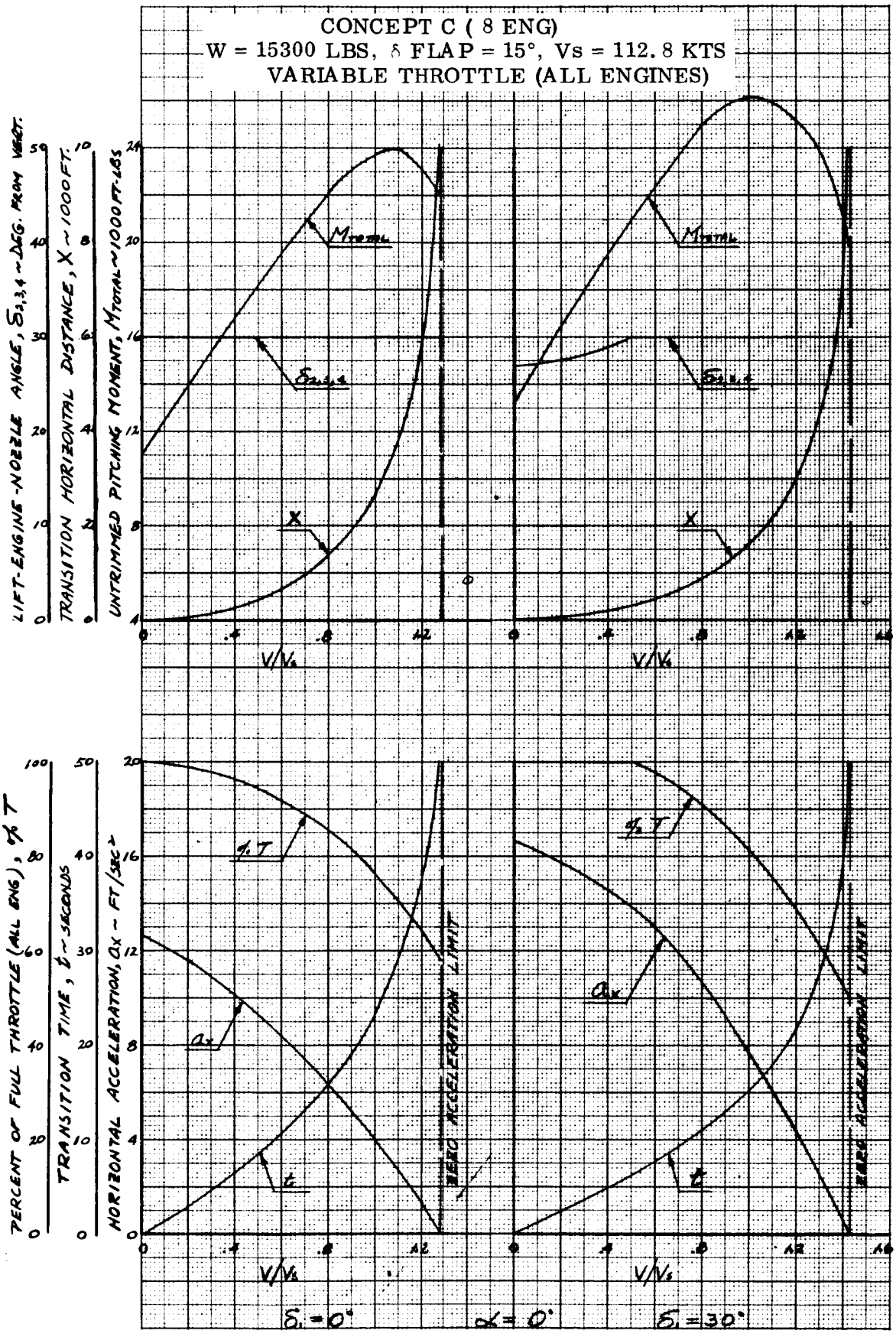


Figure III-63. Transition Performance-Level-Flight Acceleration
(γ = 0°, Constant α)

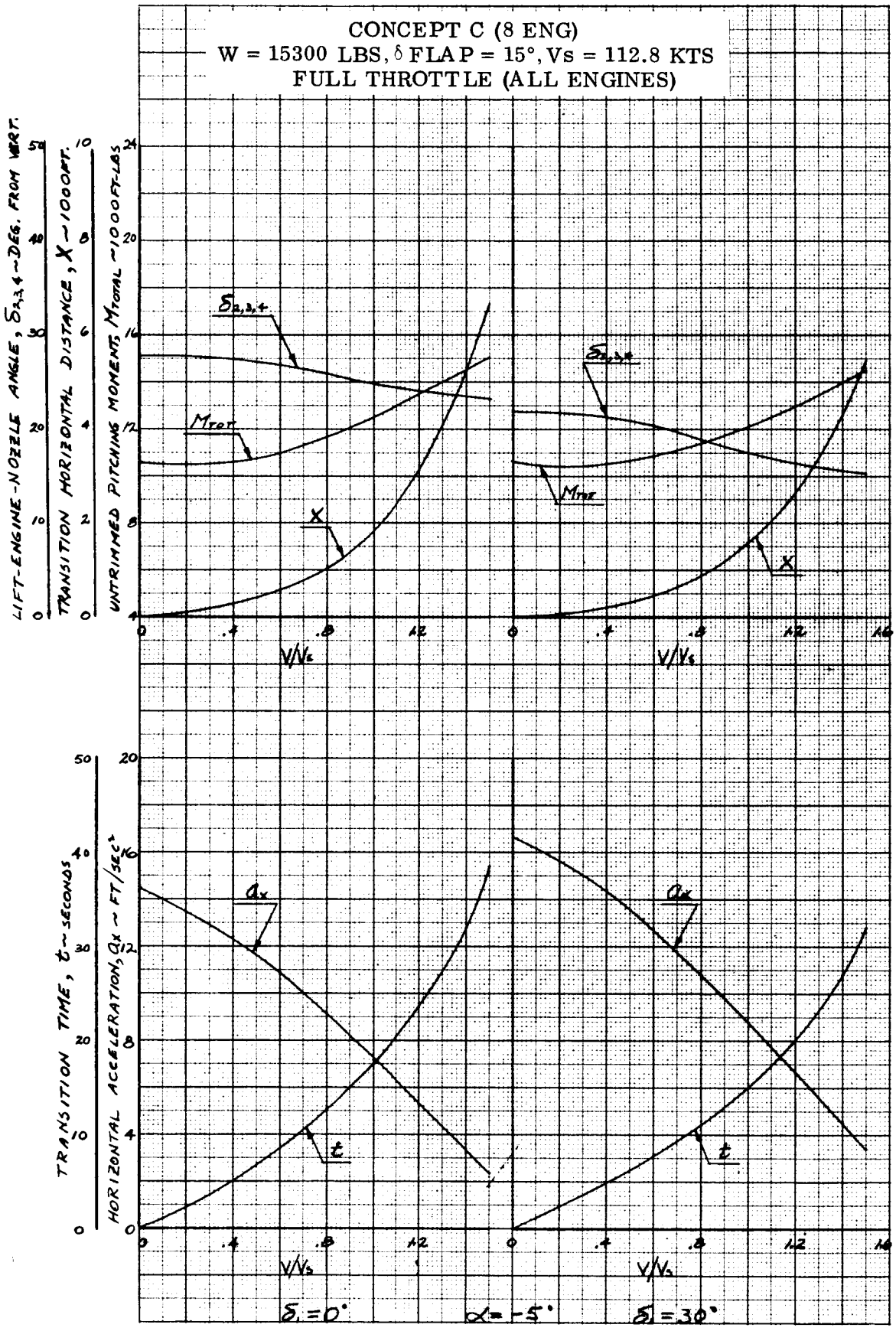


Figure III-64. Transition Performance-Level-Flight Acceleration
(γ = O°, Constant α)

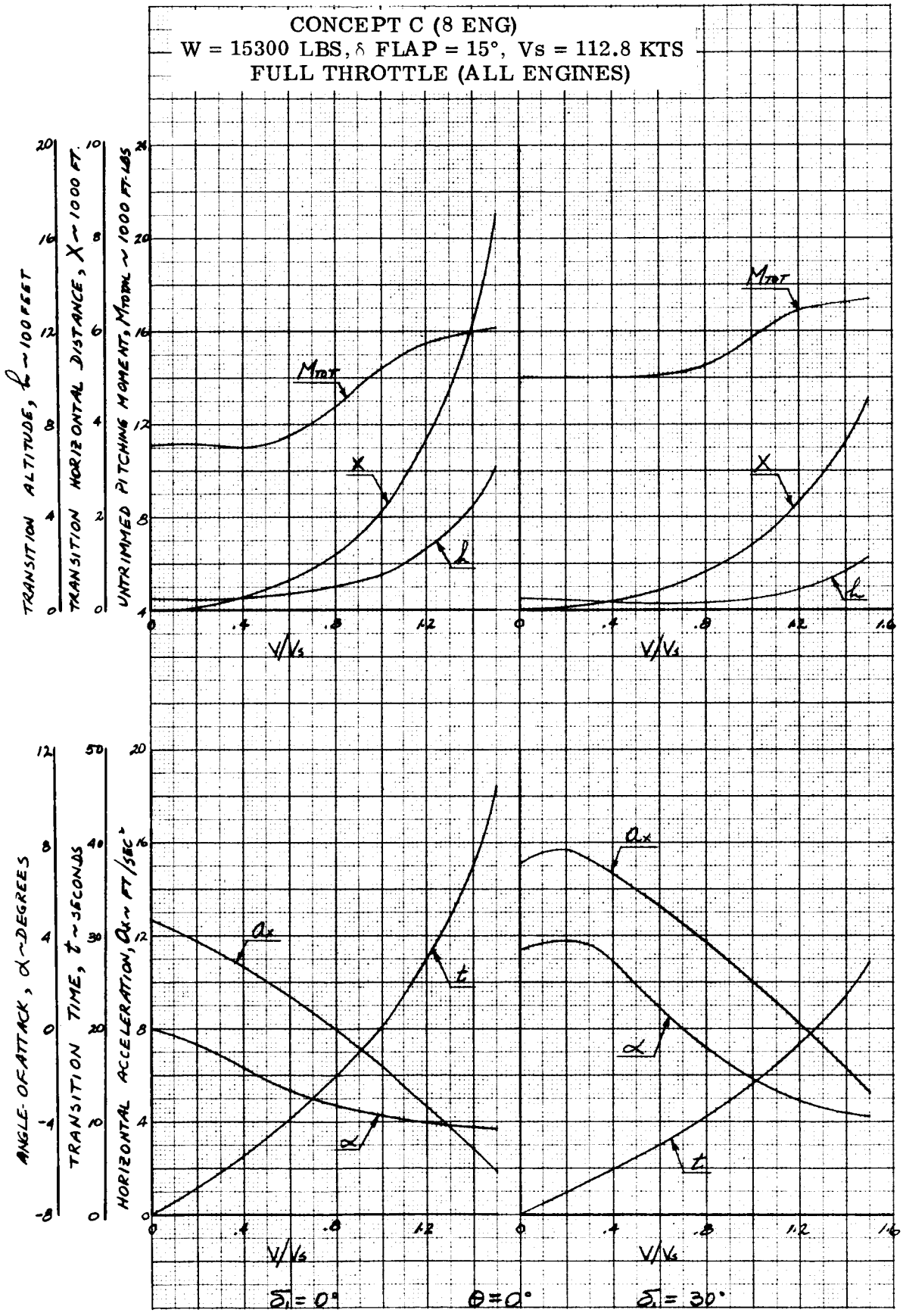


Figure III-65. Transition Performance - Constant Attitude Angle
($\theta = 0°$, Variable α)

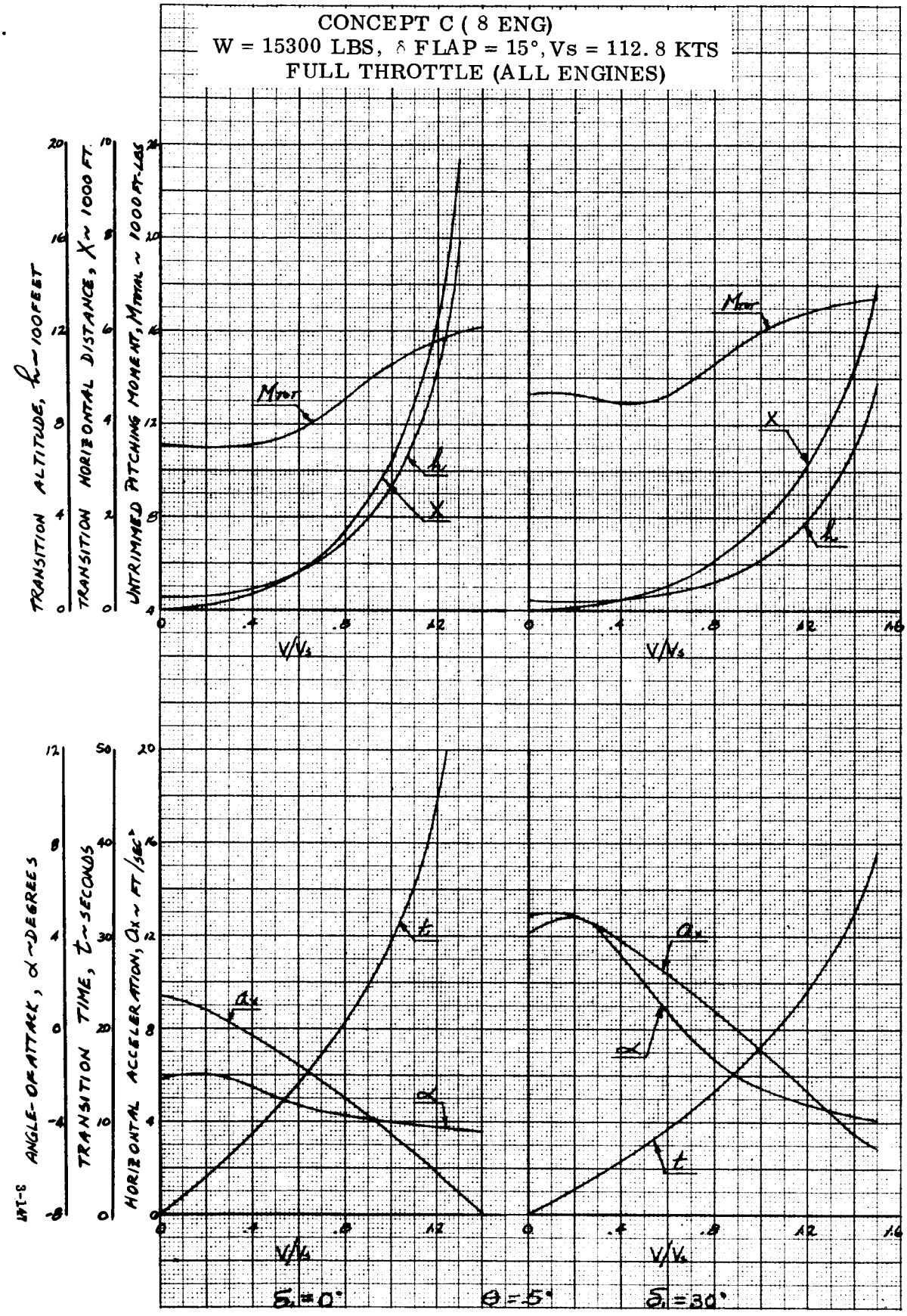


Figure III-66. Transition Performance-Constant Attitude Angle
(θ = 5°, Variable α)

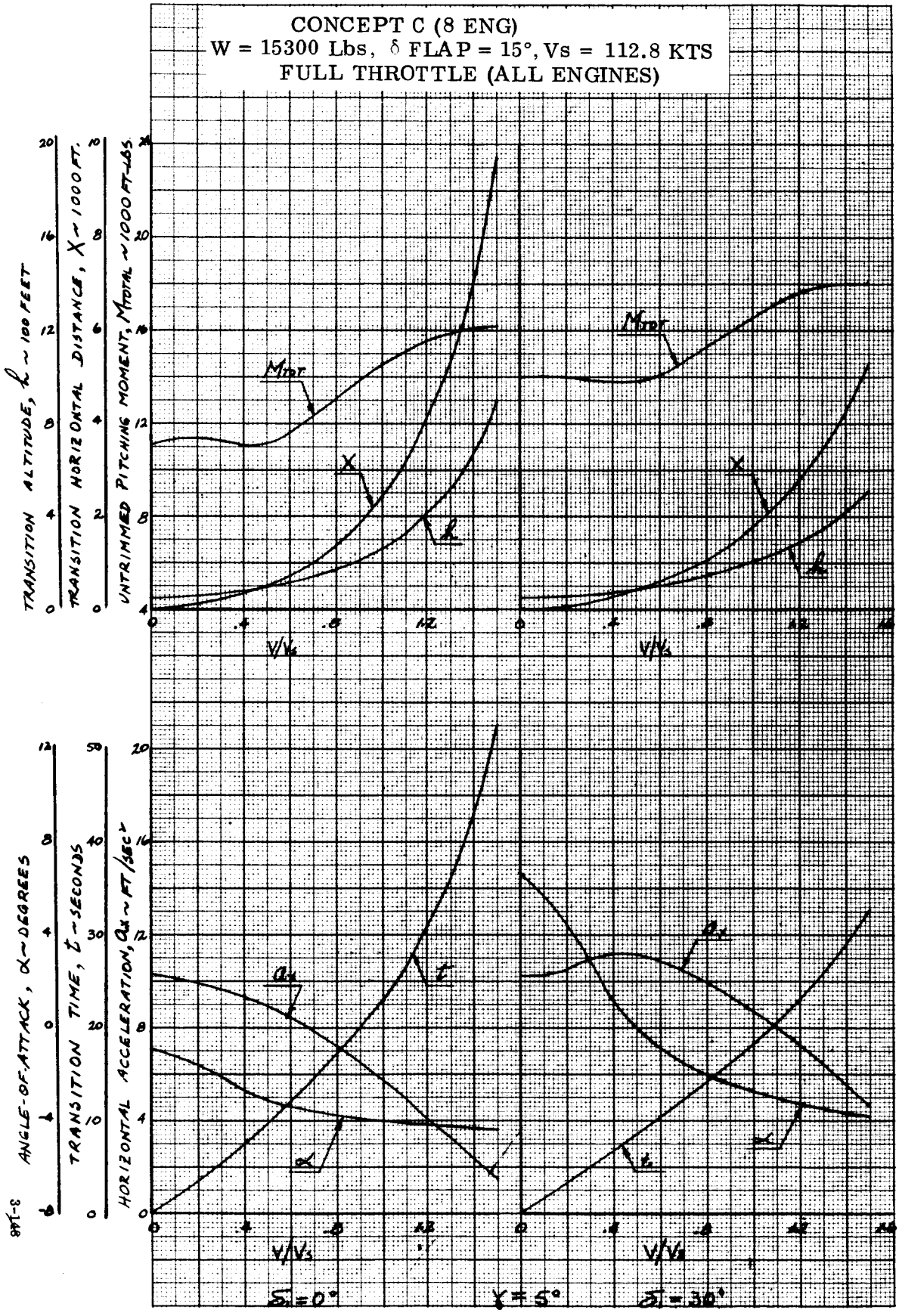


Figure III-67. Transition Performance-Constant Flight-Path Angle
($\gamma = 5°$, Variable α)

SECTION IV

PROPULSION AND THERMODYNAMICS DATA

A. INSTALLED ENGINE PERFORMANCE

The engines selected for this study were limited to the General Electric YJ85-GE-19 engines for use as lift-cruise engines, and both the General Electric YJ85-GE-19 and Rolls Royce RB162-81 engines for use as lift engines.

The basic engine data used to generate the installed performance was extracted from the General Electric Model Specification E1129 dated 1 November 1966 for the YJ85-GE-19 engine, and the Rolls-Royce Preliminary Project Performance Report No. PP-183. dated December 1965 for the RB162-81 engine.

Performance computations for the YJ85-GE-19 engine were performed on the IBM 7094 using the General Electric supplied computer program PCJ066. The computations for the RB162-81 engine were hand calculated in accordance with the methods shown in Rolls Royce Report No. PP-183.

Preliminary installed performance for both the YJ85-GE-19 and RB162-81 lift engine installations are presented in the reference 1 document. Due to a decision to delete the RB162-81 lift engine from further study, this report includes only the final engine performance for the YJ85-GE-19 engine.

The installed engine performance data that were used to determine the flight performance of the proposed aircraft is presented in Figures IV-3 through IV-28. These data were prepared by correcting the engine manufacturer's uninstalled engine performance data to include the performance losses associated with the installation. The data shown do not include the effects of hot gas reingestion, suckdown, or base losses. These losses are included in the calculation of the airplane performance and permissible VTO weight.

The installation factors and the corresponding effects upon engine performance are summarized in Figure IV-1.

1. Ambient Temperature

The effects of ambient temperature upon engine performance are included and computed in accordance with the methods prescribed by the engine manufacturer.

2. Induction System Total Pressure Recovery

a. Lift-Cruise Engines

The induction system for the lift-cruise engines is a side fuselage mounted subsonic inlet, including a boundary layer diverter, extending forward of the wing leading edge a sufficient distance to ensure an undisturbed inlet flowfield. The induction system for these engines will be designed to satisfy the following criteria:

1) Cowling critical Mach number compatible with that of the aircraft.
2) An inlet lip shape which will recover in the form of lip suction forces the full "additive" drag penalty associated with free stream tube diffusion in front of the inlet.
3) An inlet pressure recovery of nearly 100% throughout the flight regime.
4) An engine face velocity distribution compatible with the requirement specified by the engine manufacturer.

Although these requirements are quite stringent, sufficient theoretical and empirical data is available to ensure satisfactory performance and operation of the induction system.

A detailed inlet design and performance analysis was not prepared; however, the following assumptions were used:

1) Inlet pressure of 98% at all flight conditions
2) Full recovery of the additive drag penalty associated with stream tube diffusion

b. Lift Engines

The design criteria that the lift engine inlets must satisfy are as follows:

1) High inlet recovery at the static condition because the installed lift determines the aircraft VTO gross weight.
2) High inlet recovery near the flight transition speed. The pressure recovery of a "cross-flow" inlet diminishes slightly near the flight transition speed. This requirement is not critical because considerable wing lift becomes available as the aircraft accelerates to transition speed during take-off and the aircraft is lighter during landing.

3) An engine face velocity distribution compatible with the requirement specified by the engine manufacturer.

4) Although provision is made for starting the lift engines by utilizing compressor bleed air extracted from the lift-cruise engines, it is highly desirable to have sufficient inlet ram pressure available for ram starting of the lift engines during the landing transition sequence.

For the Concept C aircraft, a single ram scoop with two bellmouth engine inlets will be used for the forward two engines and a single ram scoop with four bellmouth engine inlets will be used for the four aft engines.

The design of the ram scoops will include balanced louvers in the aft section of the scoops which will be opened by engine suction at the static condition to reduce lip pressure losses and ensure high total pressure recovery. These louvers will be closed by ram pressure at the higher flight speeds near transition. In the open position, the trailing edge of the ram scoop will be raised slightly above the basic fuselage contour to exhaust the boundary layer that develops on the inner side of the ram scoops and to aspirate the boattail area on the rear side of the ram scoops. The bellmouth inlets to the engine ducting will have a minimum bellmouth/engine face diameter ratio of 0.25. This ratio will be increased wherever feasible with emphasis placed on the leading edge of the bellmouths.

There is sufficient model test data available to ensure that the design objectives are completely realistic. Republic has tested a "cross-flow" inlet configuration consisting of three bellmouth inlets in line (Reference 2). Douglas Aircraft Company, under subcontract to Continental Aviation and Engineering Corporation, has extensively tested both fuselage and podded engine installations with either in-line or side-by-side engine arrangements and has found that the data for all of these configurations were similar (Reference 3). Utilizing data from the subject report, a plot of total pressure recovery of a scoop type inlet is shown in Figure IV-2 for the YJ85-GE-19 installation. As can be readily seen, the data indicates a total pressure recovery of 99% throughout the take-off to transition speed range. Data from these two tests (References 2 and 3) support the design objectives of the NASA V/STOL aircraft.

The installed engine performance used for the aircraft performance analyses is predicted upon an assumed lift engine total pressure recovery of 98%.

3. Compressor Bleed Air Extraction

a. Lift-Cruise Engines

An assumed engine air flow at value of 1% is extracted from the compressor bleed discharge ports for the air conditioning package. During lift operation and lift engine air start cycle, 10% bleed is extracted from the bleed ports. The effect of engine bleed upon engine performance is included in the installed engine data and is calculated using the methods prescribed by the engine manufacturer.

b. Lift Engines

A continuous bleed of 10% is available from the YJ85-GE-19 engine with the exhaust nozzle sized for this quantity of bleed. This air is utilized for attitude control purposes only. The effect of this bleed upon engine performance is included in the installed engine data and is calculated using the methods prescribed by the engine manufacturer.

4. Horsepower Extraction

The installed lift-cruise engine data includes the effect of extracting an estimated 20 horsepower for hydraulic pumps and generator operation. No horsepower is extracted from the lift engines.

5. Diverter Valve Losses

A diverter valve similar to that used on the XV5A airplane will be installed on the lift-cruise engine in order to divert the exhaust gas either horizontally during cruise or vertically during take-off, hover, transition and landing. In addition, the valve will be capable of operation in any preset position for use during air start of the lift engines.

The performance losses associated with this valve were extracted from a General Electric report "Final Report-Diverter Valve Development Program," Contract No. AF33(600)-40862, Project No. 3066.

The total pressure loss coefficient, $\left(\frac{P_{T_{5.1}} - P_{T_{5.2}}}{q_{5.1}}\right)$ for diverted operation, corrected to a design turbine discharge Mach number of 0.5 was 0.24 for the test conditions which corresponds to a pressure drop of 3.5%. A pressure drop of 3.0% is calculated because the actual discharge Mach number for the YJ85-GE-19 engine is 0.45 at the sea level static, 80°F day condition at 10% bleed and 98% inlet recovery.

The installed engine performance includes a diverter pressure loss of 3.5% in the diverted mode and 3.1% in the cruise mode with diverter valve door gas leakage of 0.9% and 0.7%, respectively.

6. Engine Cooling

Cooling of lift engine and lift-cruise engine installations will be accomplished by means of ejector action of the engine exhausts. Past flight test experience at Republic with the F-84 and F-105 airplanes has demonstrated an equivalent thrust loss of from 1.0 to 1.5%. A thrust loss of 1.5% has been included in the calculation of the installed engine performance.

7. Exhaust Tailpipe

A curved duct to carry the diverted exhaust gas into the fuselage and down to the underside of the aircraft will be employed. Utilizing the methods of "SAE Aero-Space Applied Thermodynamics Manual," rev. January 1962, a pressure drop of 2.2% was calculated and is included in the installed performance of the lift-cruise engine during lifting mode of operation.

8. Vectoring Nozzle

The use of the General Electric pivoting sphere nozzle is assumed for both the lift-cruise and lift engine installations. The installed engine performance includes the nozzle performance supplied by General Electric. A thrust coefficient of 0.970 at zero deflection is used.

9. Sensitivity Analysis

In accordance with the requirements of the contract, a parametric investigation of the effects of a ±50% variation in the individual installation factors has been conducted. This investigation was conducted in two ways: (1) assuming that the exhaust area is sized to provide engine operation at both limit RPM and limit EGT at take-off power with the losses involved, and (2) assuming that the exhaust area is constant at the value which provides limit RPM and limit EGT at take-off power with the nominal losses. In both methods, one installation factor was varied ±50% while all of the other values were held at their nominal value. The results of these studies for the lift and lift-cruise engines are shown in Figures IV-29 through IV-36.

In addition to the above studies, the combined effect of all of the installation factors varying +50% and -50% were calculated with the exhaust area sized to provide limit RPM and limit EGT at take-off power. Figure IV-37 presents the result of this analysis and indicates a ±6.85 percent maximum thrust deviation.

10. References

1. Part II - Technical Data Substantiation Report - V/STOL Jet Operations Research Airplane Design Study Report No. FHR 3391-3I, 7 April 1967.

2. "AP-100 1/24th Scale Fan Inlet Tests," Republic Aviation Corporation Report No. ETR-12, July 1959.

3. "Tests to Establish Flow Distortion Criteria for Lift Engines," by B. I. Tyson, AIAA Journal of Aircraft, Vol. 2, No. 5, September-October 1965.

ENGINE INSTALLATION THRUST LOSS DATA—VERTICAL MODE

	LIFT CRUISE ENGINE ΔF_g		LIFT ENGINE ΔF_g	
IDEAL S.L. STD. DAY THRUST=3015				
80^0 F AMBIENT TEMP.	196		196	
INDUCTION SYSTEM RECOVERY (.98)	89		89	
COMPRESSOR BLEED AIR (10%)	440		440	
HORSEPOWER EXTRACTION (20 HP)	4			
DIVERTER VALVE PRESS. LOSS (3.5%)	56			
DIVERTER VALVE LEAKAGE (0.9%)	20			
TAILPIPE PRESS. LOSS (2.2%)	36			
PIVOTING SPHERE IN LIEU OF CONICAL NOZZLE	15		15	
COOLING (1.5% F_g inst.)	31		33	
TOTAL INSTALLATION LOSS	887		773	
INSTALLED THRUST		2128		2242

Figure IV-1

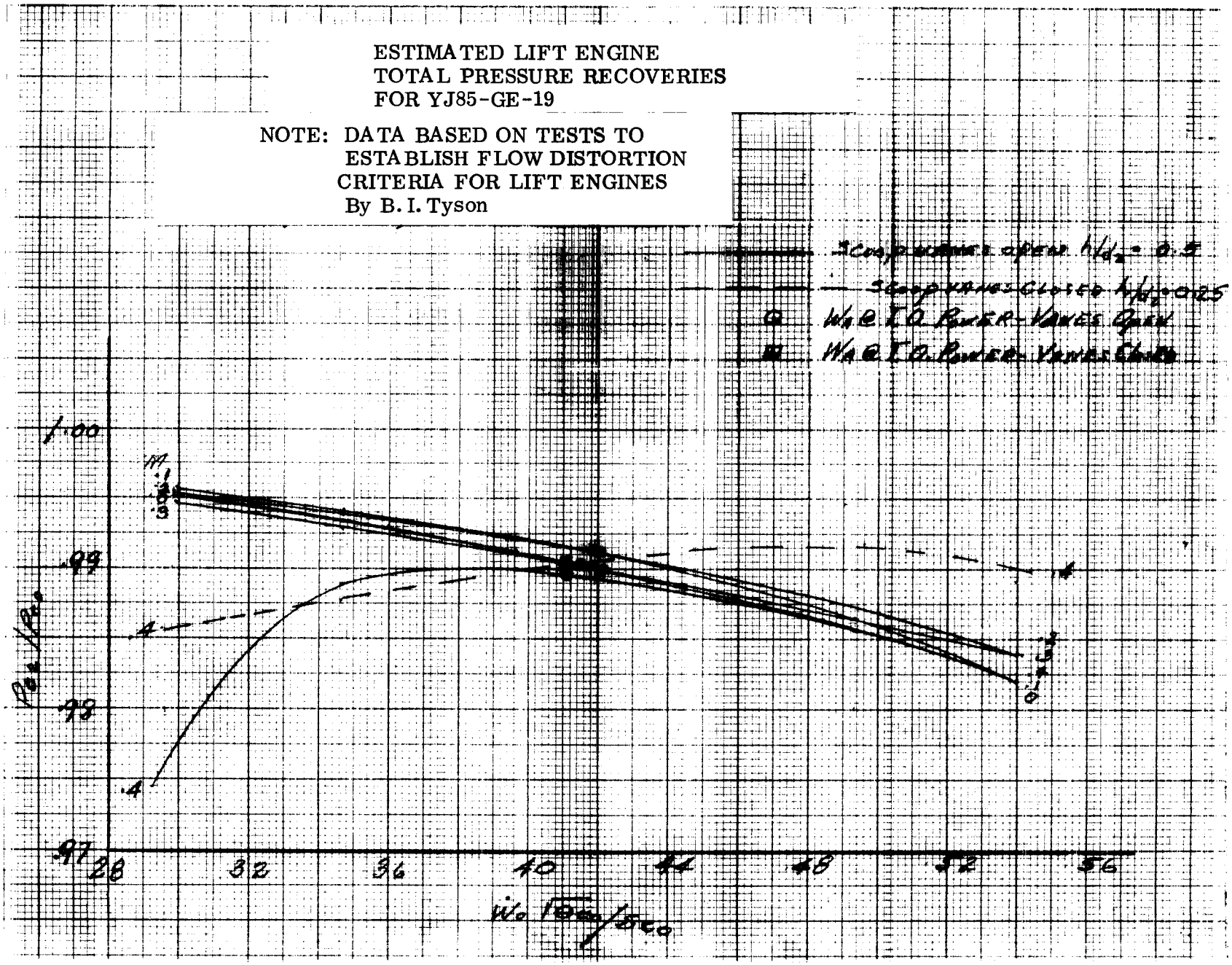

ESTIMATED LIFT ENGINE
TOTAL PRESSURE RECOVERIES
FOR YJ85-GE-19
NOTE: DATA BASED ON TESTS TO
ESTABLISH FLOW DISTORTION
CRITERIA FOR LIFT ENGINES
By B. I. Tyson
1.00
.99
.98
.97
28
32
36
40
44
48
52
56

Figure IV-2

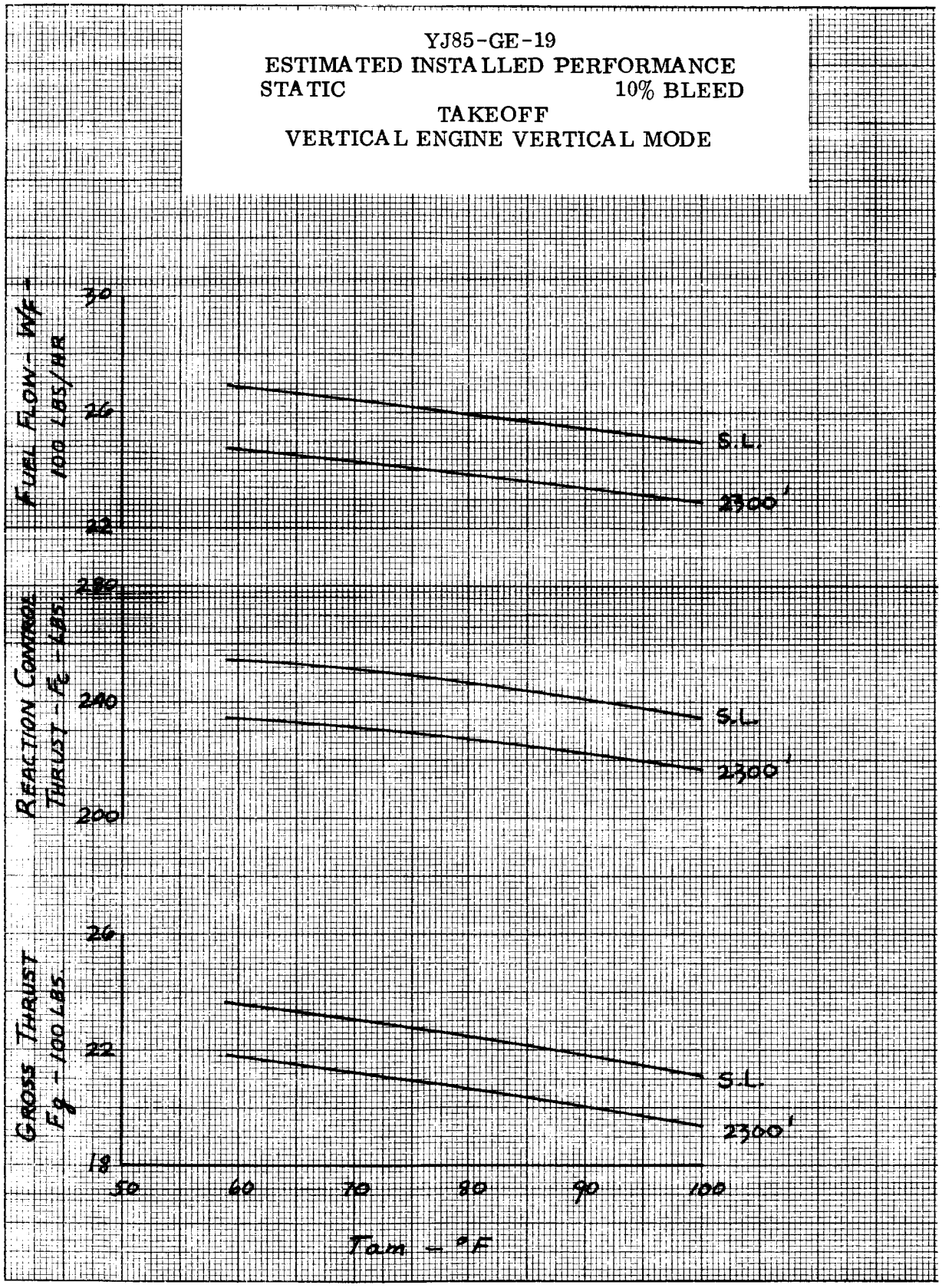

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
STATIC
10% BLEED
TAKEOFF
VERTICAL ENGINE VERTICAL MODE
FUEL FLOW - Wf - 100 LBS/HR
30
26
22
REACTION CONTROL THRUST - Fc - LBS.
280
240
200
GROSS THRUST Fg - 100 LBS.
26
22
18
S.L.
2300'
S.L.
2300'
S.L.
2300'
50
60
70
80
90
100
Tam - °F

Figure IV-3

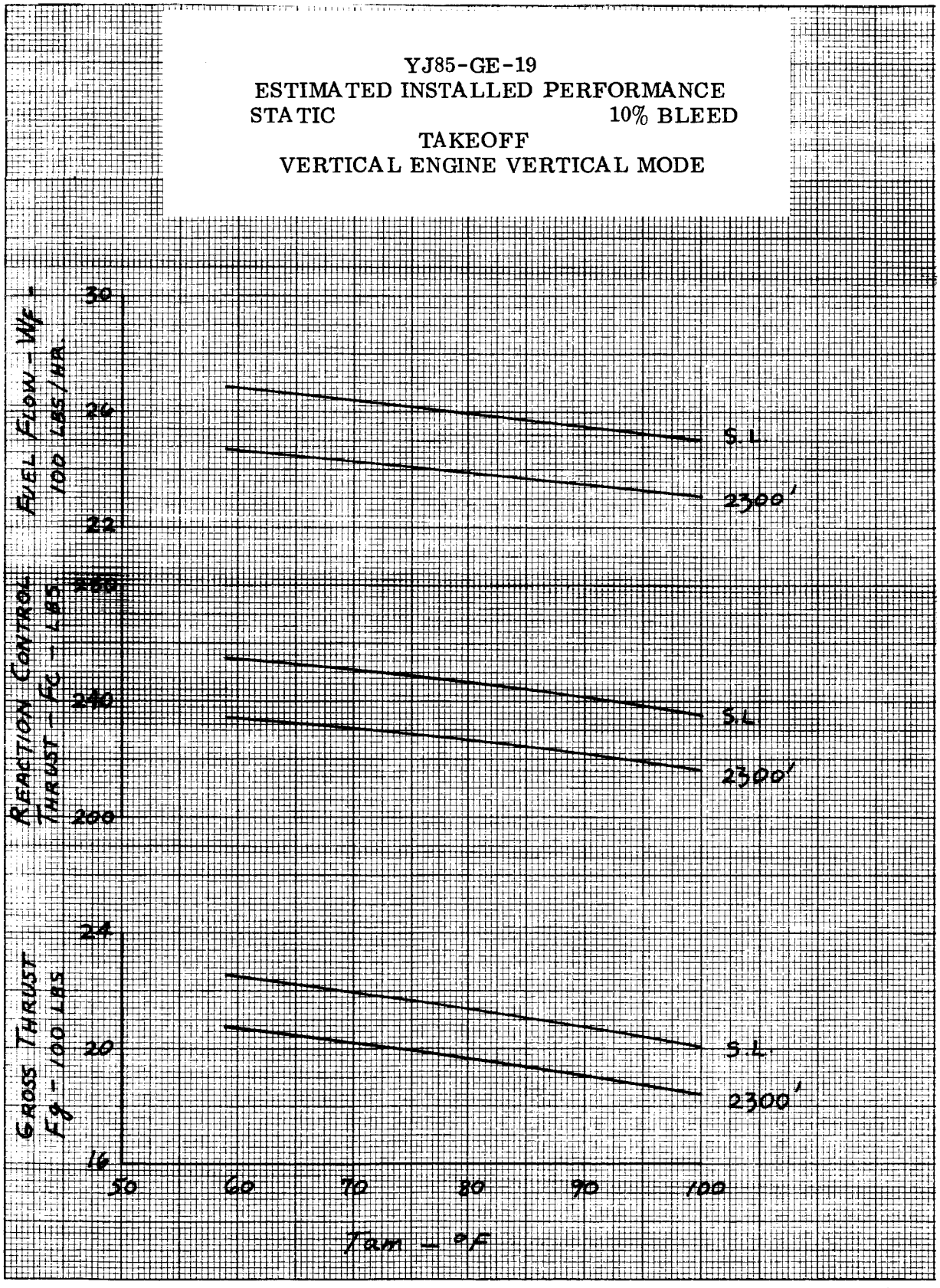

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
STATIC
10% BLEED
TAKEOFF
VERTICAL ENGINE VERTICAL MODE
FUEL FLOW - Wf - 100 LBS/HR
REACTION CONTROL THRUST - Fc - LBS
GROSS THRUST Fg - 100 LBS
30
26
22
280
240
200
24
20
16
50
60
70
80
90
100
TAM - °F
S.L.
2300'
S.L.
2300'
S.L.
2300'

Figure IV-4

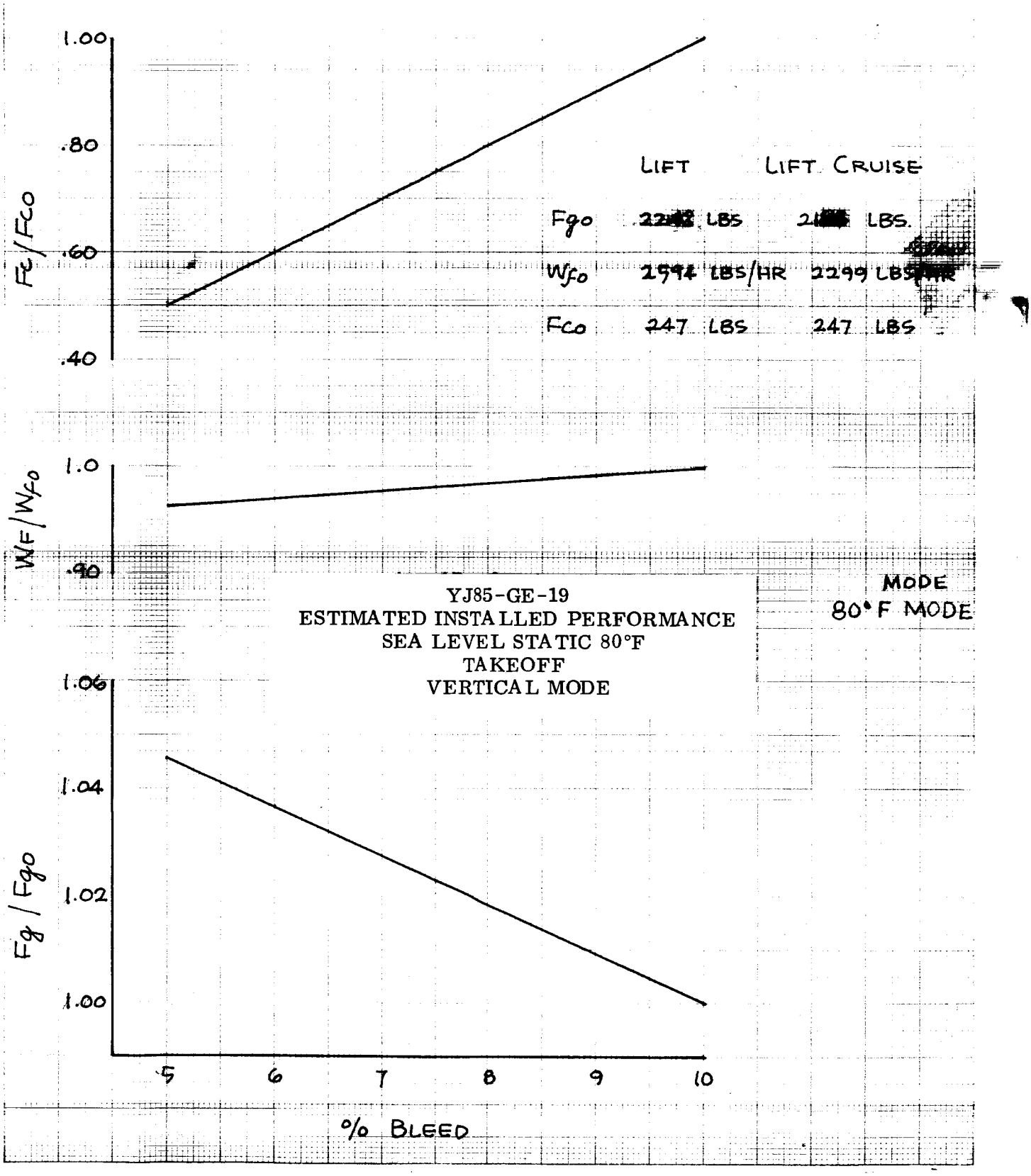

1.00
.80
.60
.40
Fc/Fco
LIFT
LIFT CRUISE
Fgo
LBS
LBS.
Wfo
2994 LBS/HR
2299 LBS
Fco
247 LBS
247 LBS
1.0
.90
WF/Wfo
YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
SEA LEVEL STATIC 80°F
TAKEOFF
VERTICAL MODE
MODE
80°F MODE
1.06
1.04
1.02
1.00
Fg/Fgo
5
6
7
8
9
10
% BLEED

Figure IV-5

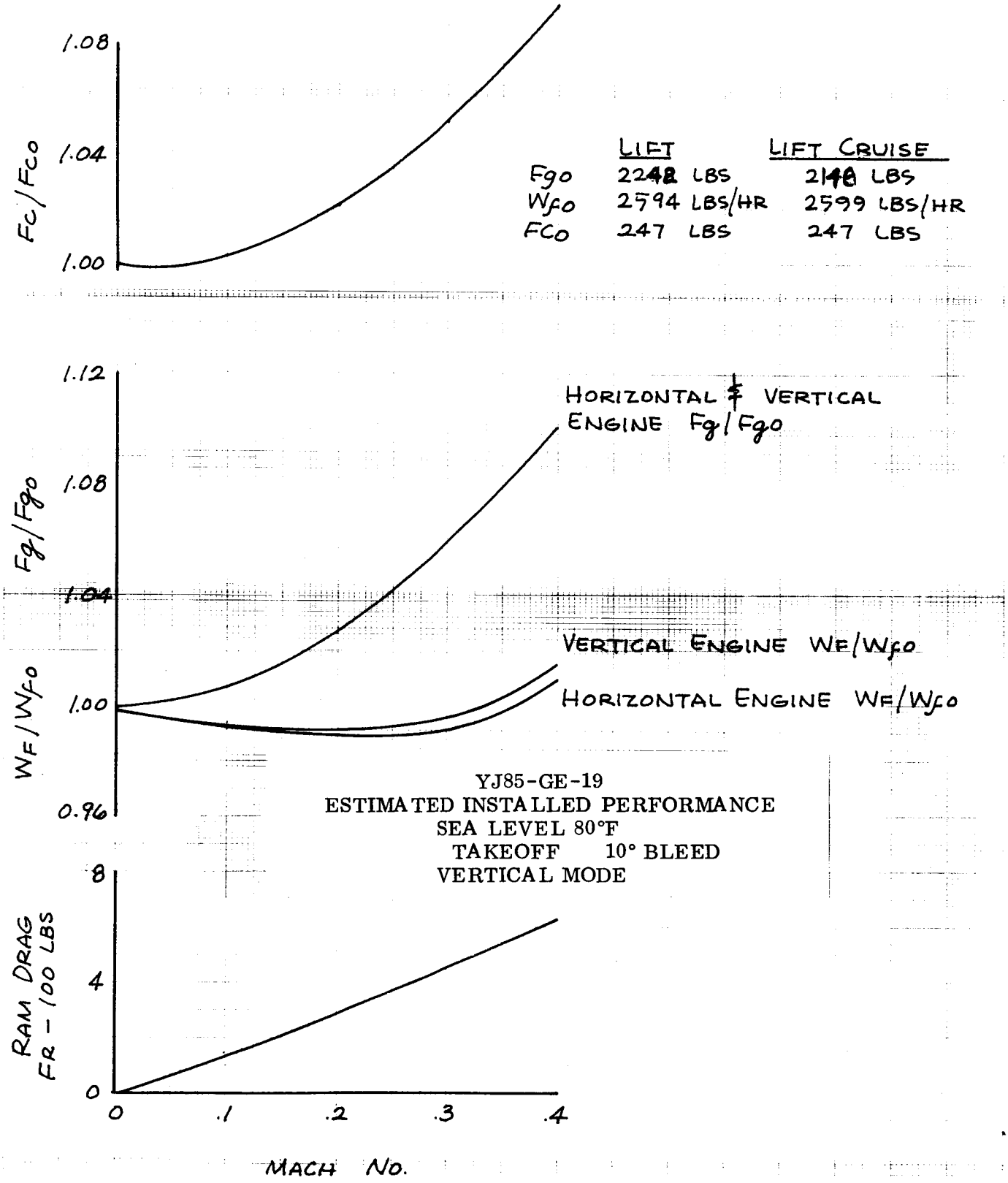

1.08
1.04
1.00
Fc/Fco
LIFT
LIFT CRUISE
Fgo 2248 LBS 2148 LBS
Wfo 2594 LBS/HR 2599 LBS/HR
Fco 247 LBS 247 LBS
1.12
1.08
1.04
1.00
0.96
Fg/Fgo
Wf/Wfo
HORIZONTAL & VERTICAL ENGINE Fg/Fgo
VERTICAL ENGINE Wf/Wfo
HORIZONTAL ENGINE Wf/Wfo
YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
SEA LEVEL 80°F
TAKEOFF 10° BLEED
VERTICAL MODE
8
4
0
RAM DRAG FR - 100 LBS
0
.1
.2
.3
.4
MACH NO.

Figure IV-6

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
MILITARY POWER
STANDARD DAY

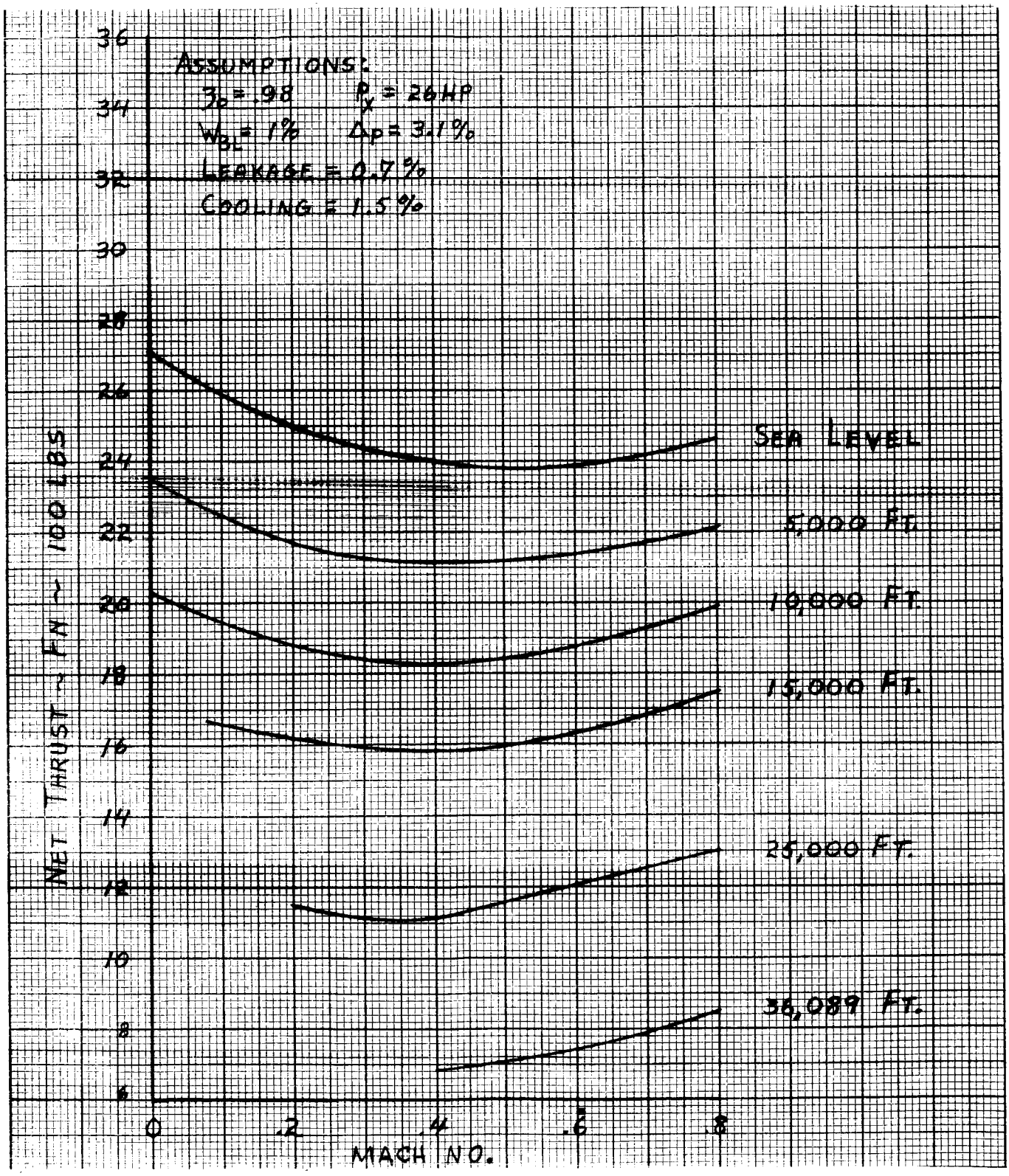


Figure IV-7

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
MILITARY POWER
STANDARD DAY

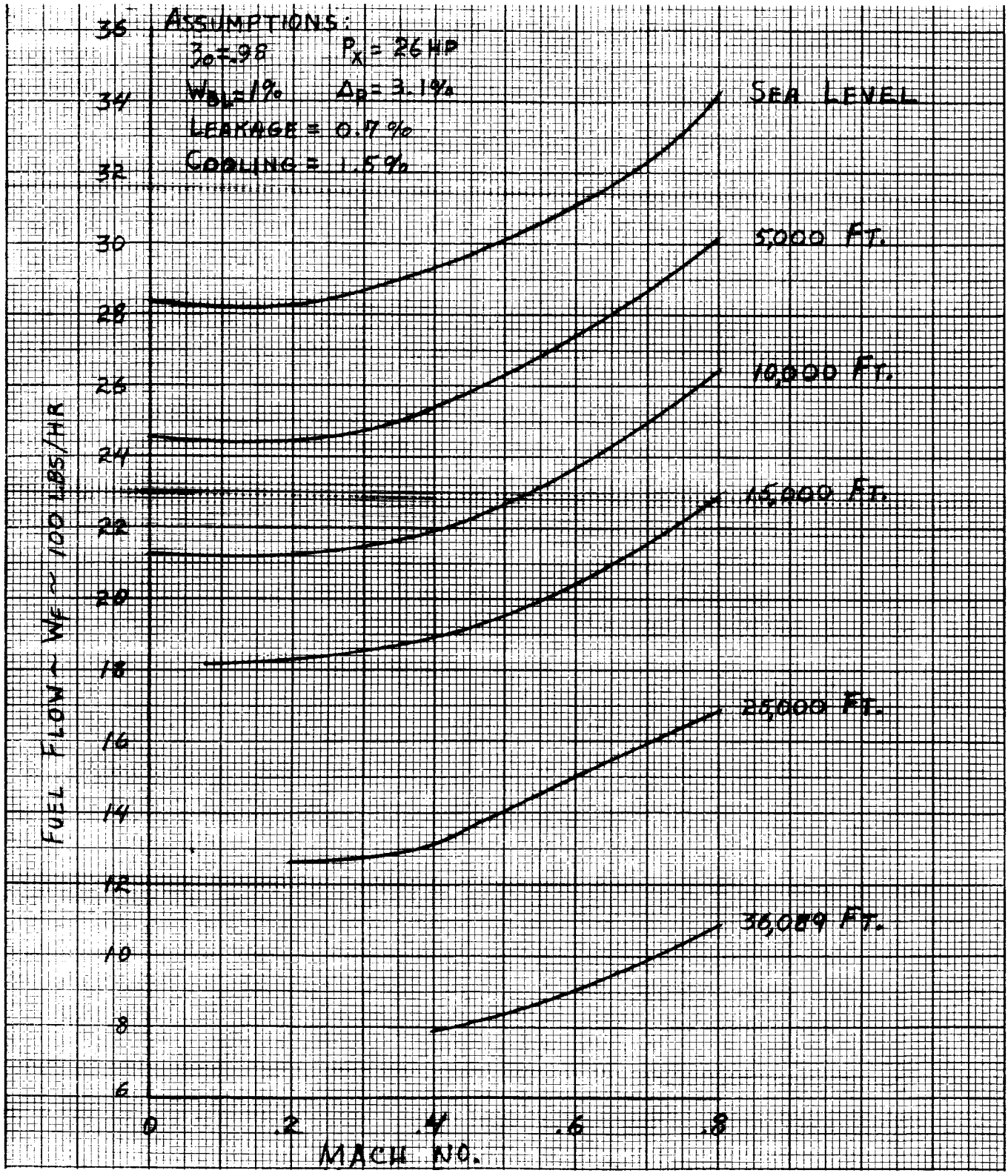


Figure IV-8

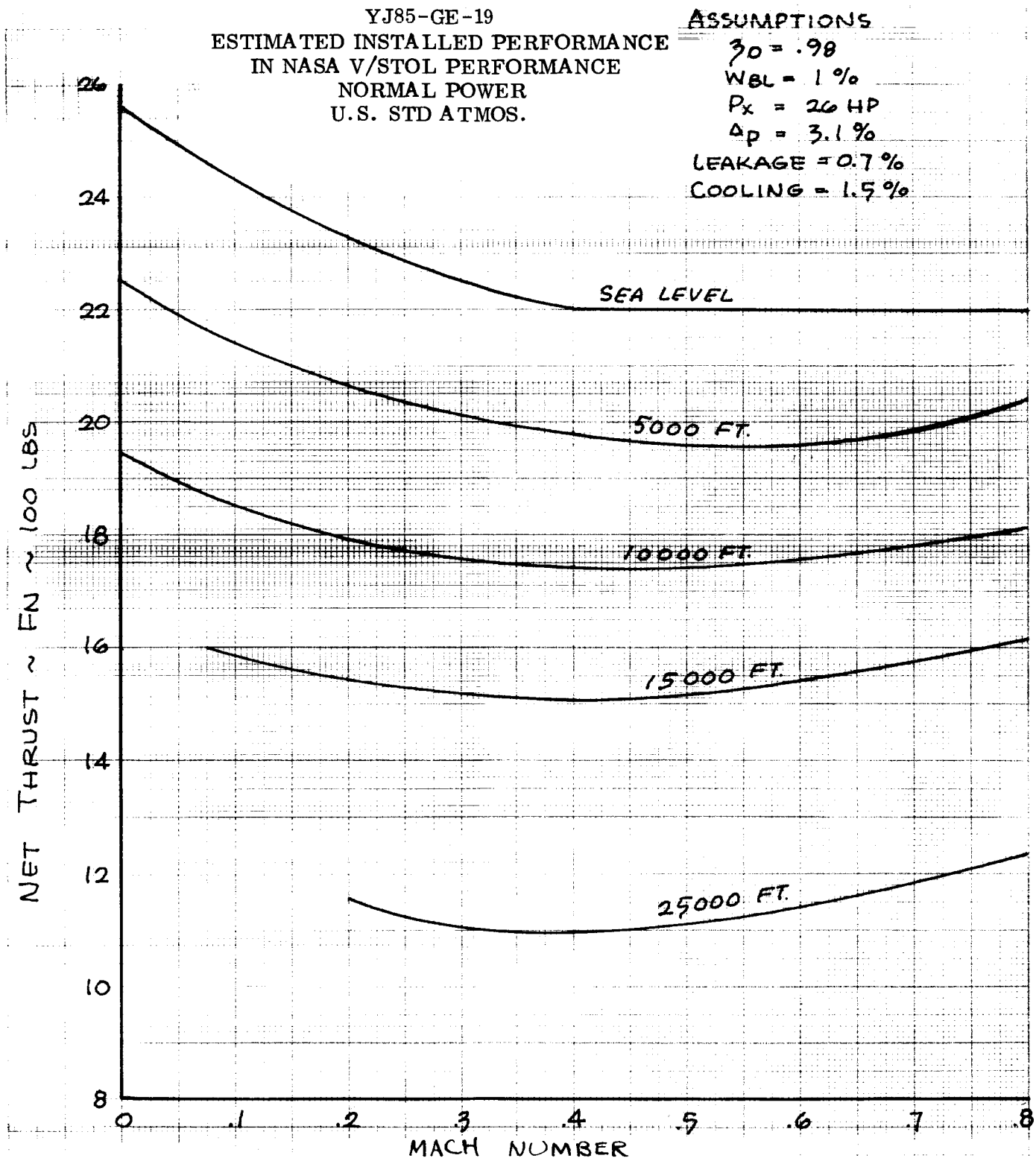

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL PERFORMANCE
NORMAL POWER
U.S. STD ATMOS.
ASSUMPTIONS
η_0 = .98
W_{BL} = 1 %
P_x = 26 HP
Δp = 3.1 %
LEAKAGE = 0.7 %
COOLING = 1.5 %
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
25000 FT.
NET THRUST ~ FN ~ 100 LBS
26
24
22
20
18
16
14
12
10
8
0
.1
.2
.3
.4
.5
.6
.7
.8
MACH NUMBER

Figure IV-9

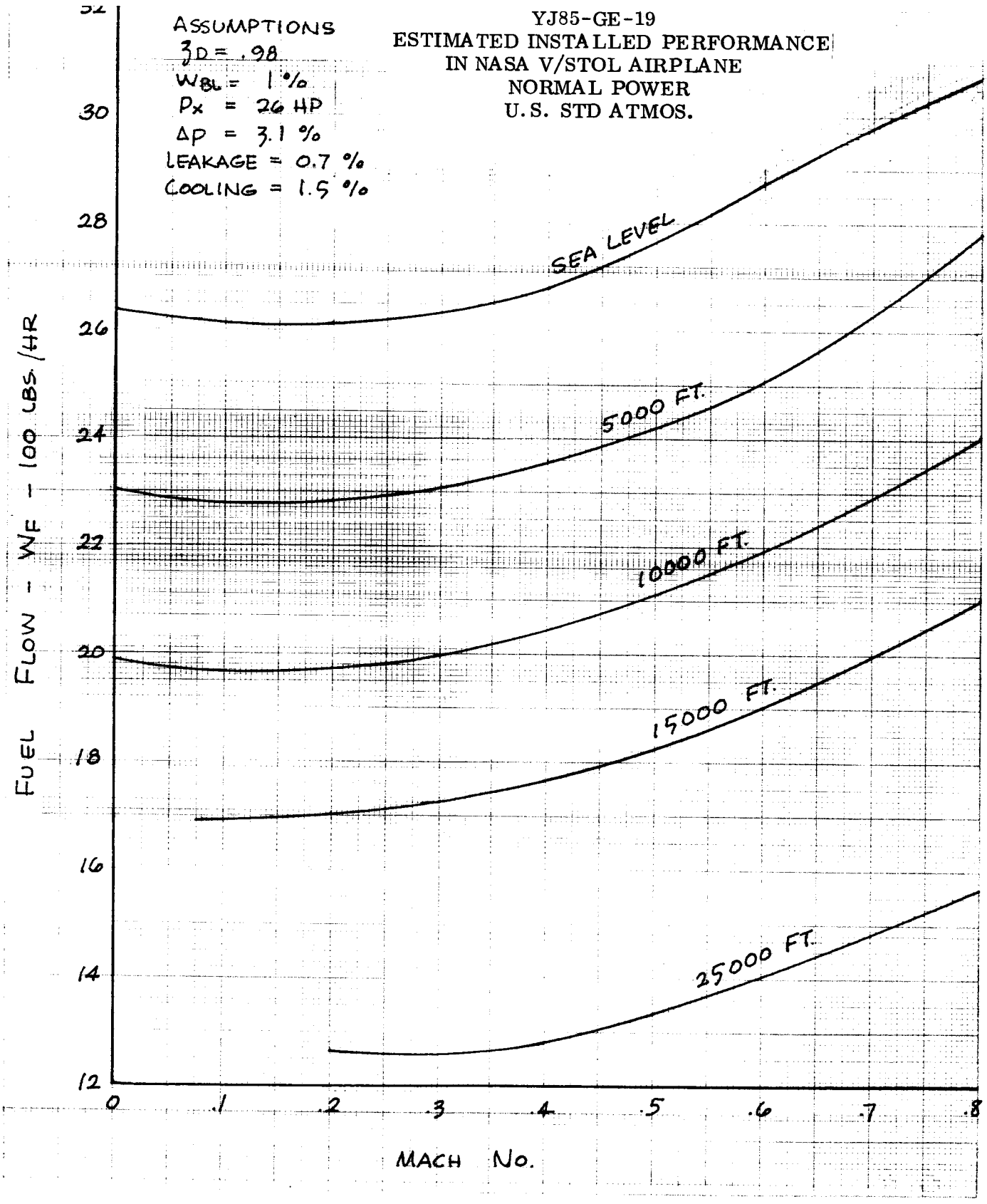

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
NORMAL POWER
U.S. STD ATMOS.
ASSUMPTIONS
ηD = .98
WBL = 1 %
Px = 26 HP
ΔP = 3.1 %
LEAKAGE = 0.7 %
COOLING = 1.5 %
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
25000 FT.
FUEL FLOW - WF - 100 LBS./HR
32
30
28
26
24
22
20
18
16
14
12
0
.1
.2
.3
.4
.5
.6
.7
.8
MACH No.

Figure IV-10

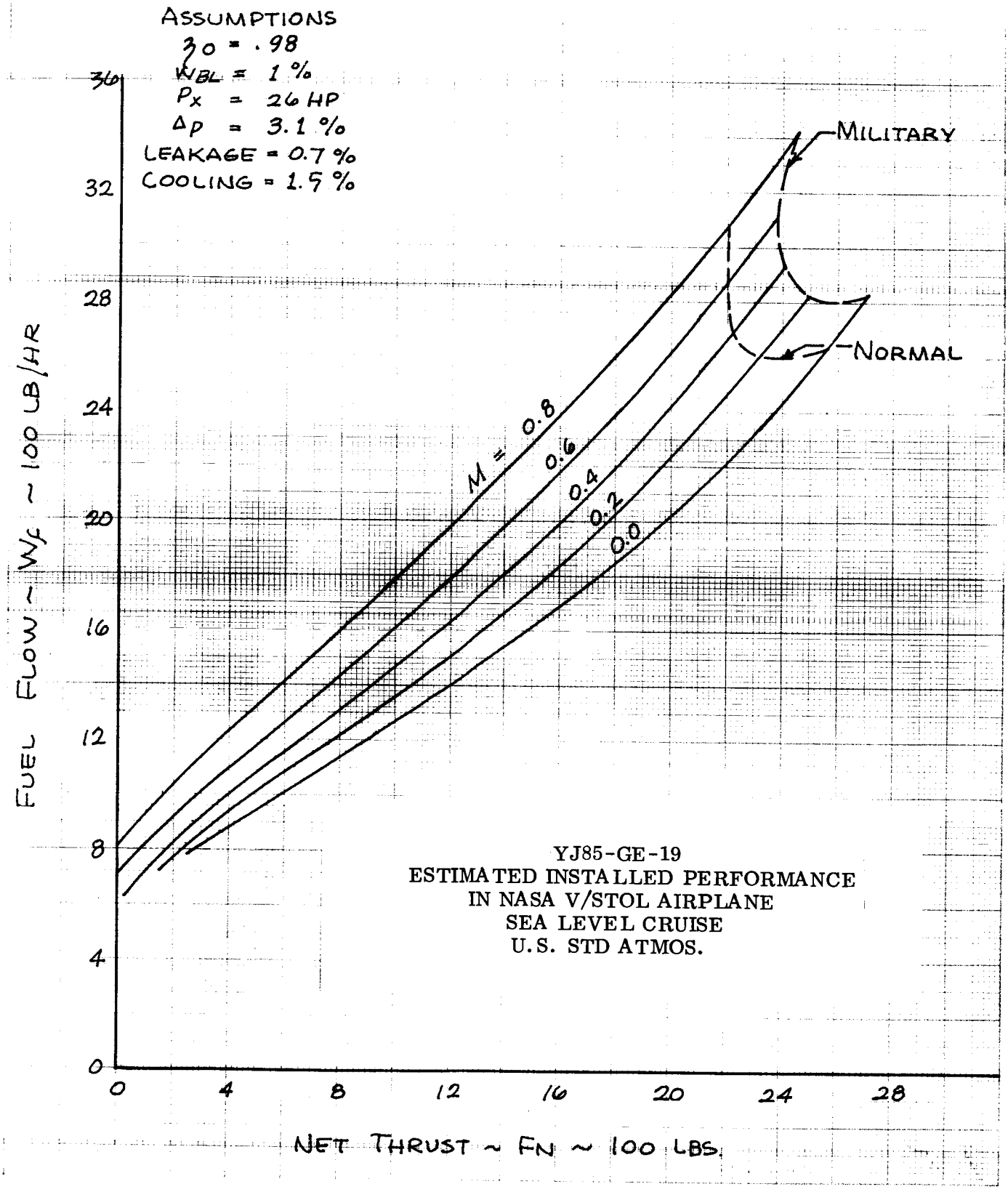

ASSUMPTIONS
η_0 = .98
W_{BL} = 1 %
P_X = 26 HP
Δp = 3.1 %
LEAKAGE = 0.7 %
COOLING = 1.5 %
MILITARY
NORMAL
M = 0.8
0.6
0.4
0.2
0.0
YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
SEA LEVEL CRUISE
U.S. STD ATMOS.
FUEL FLOW ~ W_f ~ 100 LB/HR
NET THRUST ~ F_N ~ 100 LBS.
0
4
8
12
16
20
24
28
32
36

Figure IV-11

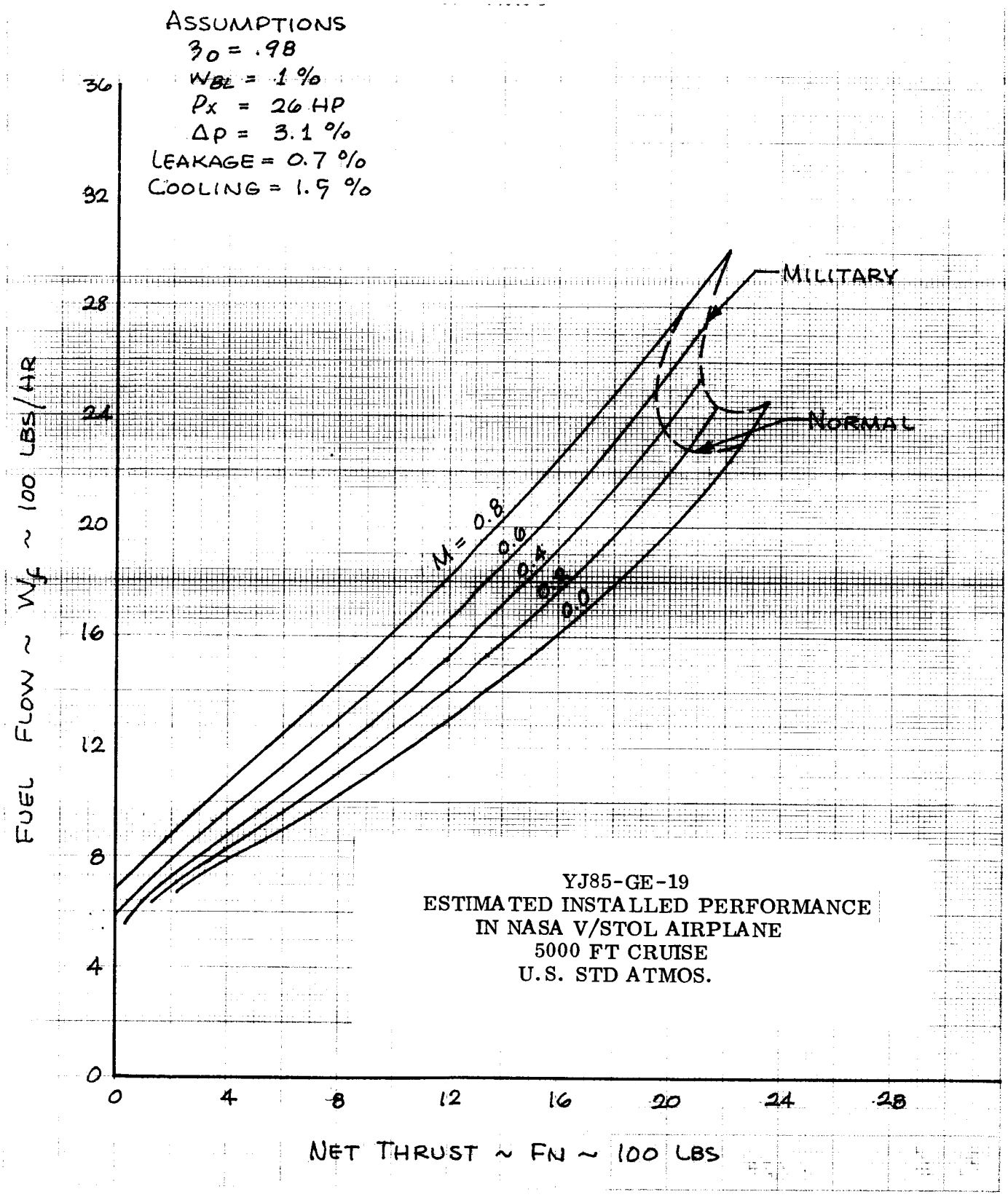

ASSUMPTIONS
η_0 = .98
W_{BL} = 1 %
P_X = 26 HP
Δp = 3.1 %
LEAKAGE = 0.7 %
COOLING = 1.5 %
MILITARY
NORMAL
M = 0.8
0.6
0.4
0.2
0.0
FUEL FLOW ~ W_f ~ 100 LBS/HR
NET THRUST ~ FN ~ 100 LBS
0
4
8
12
16
20
24
28
32
36
YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
5000 FT CRUISE
U.S. STD ATMOS.

Figure IV-12

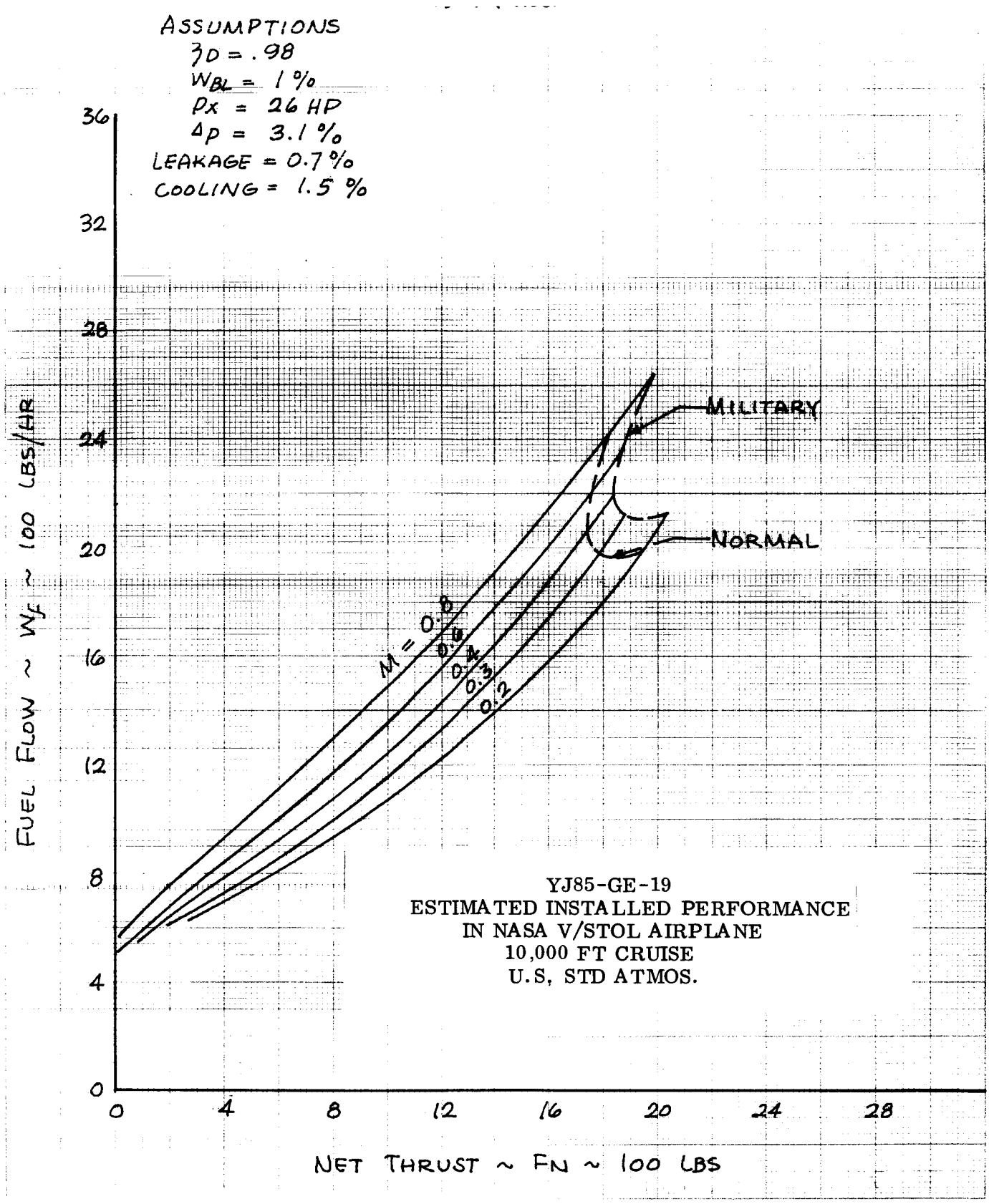

ASSUMPTIONS
$\eta_D = .98$
$W_{BL} = 1\%$
$P_X = 26$ HP
$\Delta p = 3.1\%$
LEAKAGE = 0.7%
COOLING = 1.5%
MILITARY
NORMAL
M = 0.8
0.6
0.4
0.3
0.2
YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
10,000 FT CRUISE
U.S. STD ATMOS.
FUEL FLOW ~ W_f ~ 100 LBS/HR
NET THRUST ~ F_N ~ 100 LBS
0
4
8
12
16
20
24
28
32
36

Figure IV-13

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
15,000 FT. CRUISE
U.S. STD ATMOS.

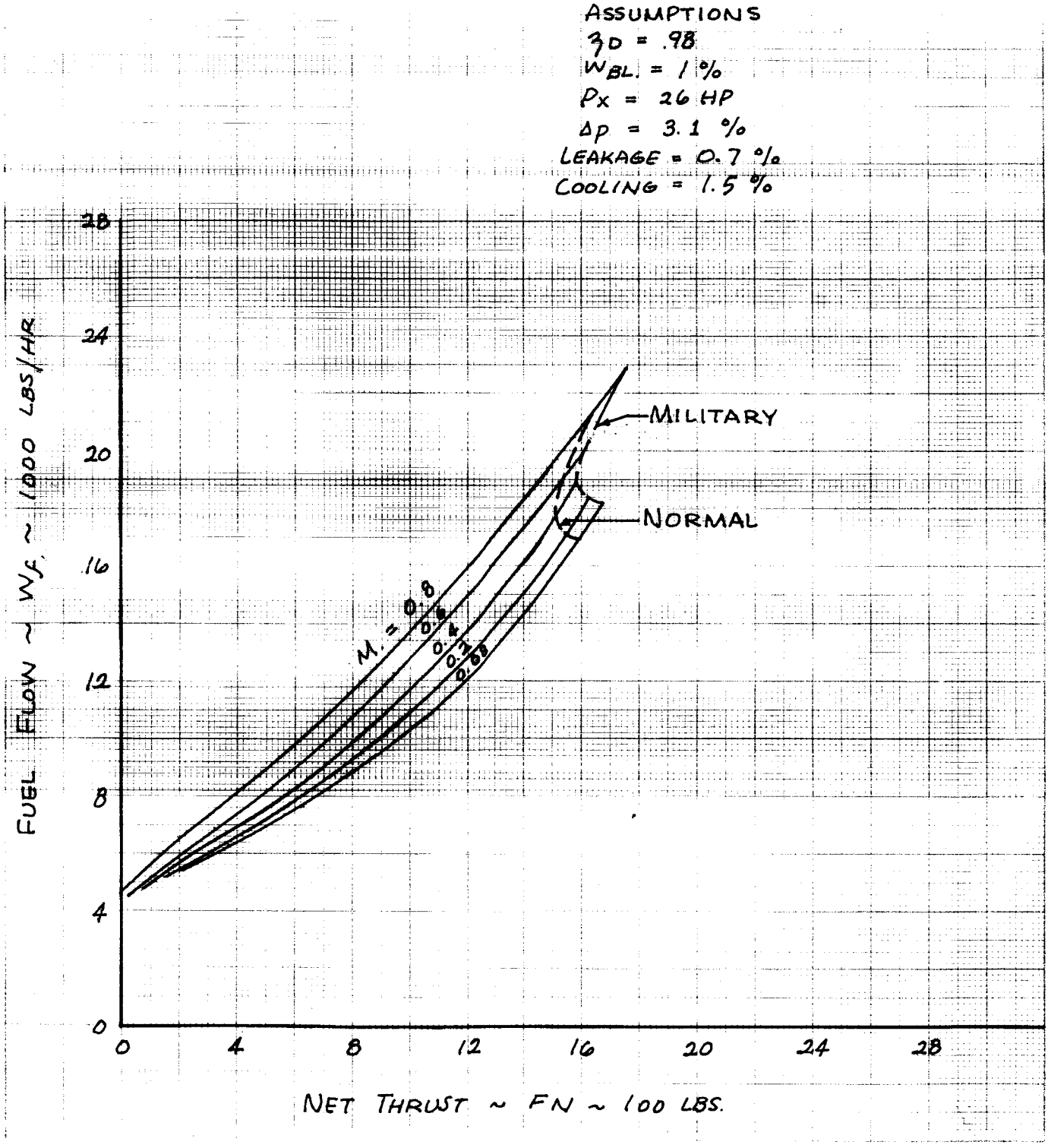


Figure IV-14

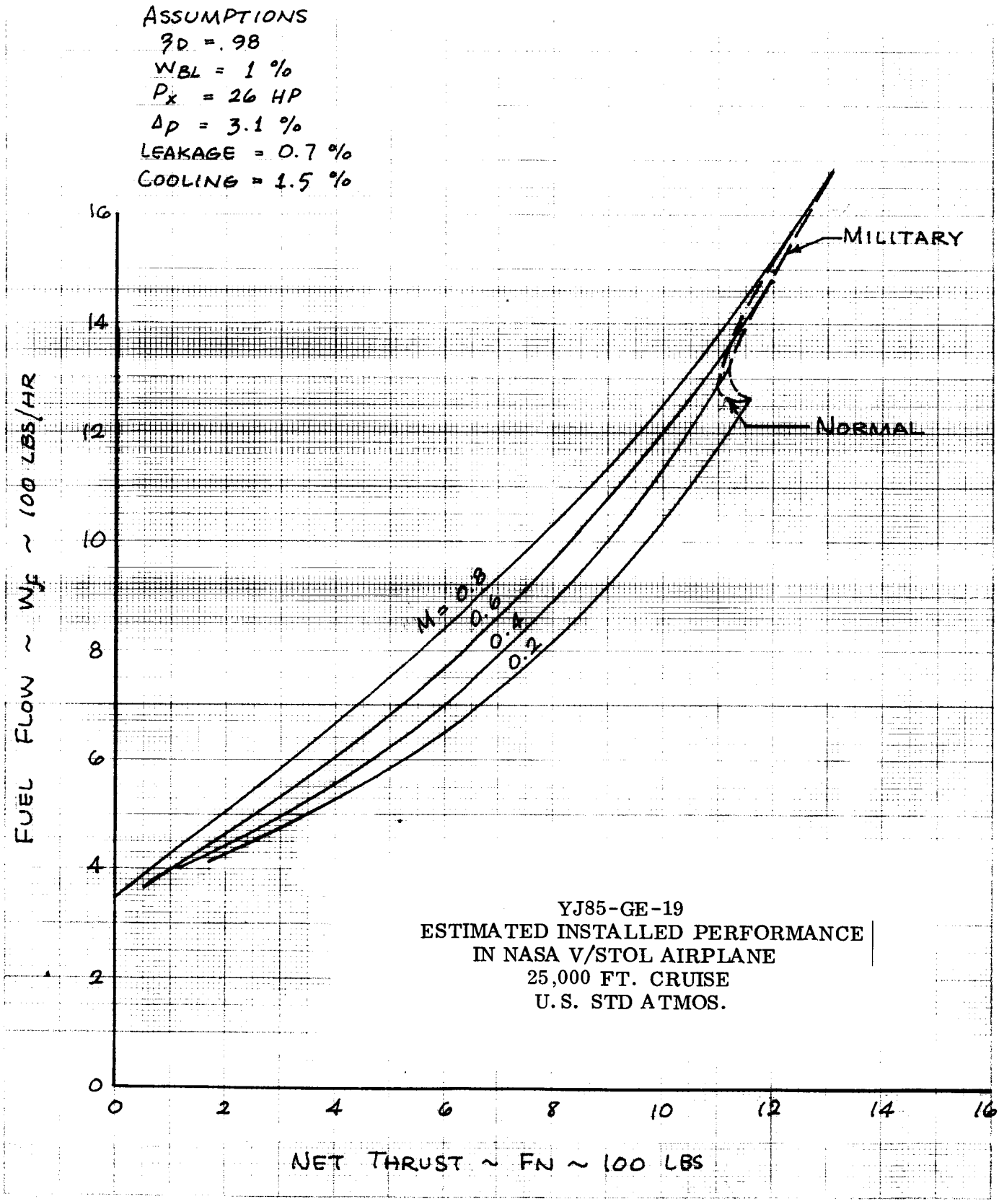

ASSUMPTIONS
η_D = .98
W_{BL} = 1 %
P_x = 26 HP
Δp = 3.1 %
LEAKAGE = 0.7 %
COOLING = 1.5 %
MILITARY
NORMAL
M = 0.8
0.6
0.4
0.2
YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
25,000 FT. CRUISE
U.S. STD ATMOS.
FUEL FLOW ~ W_f ~ 100 LBS/HR
NET THRUST ~ FN ~ 100 LBS
0
2
4
6
8
10
12
14
16

Figure IV-15

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
MILITARY POWER

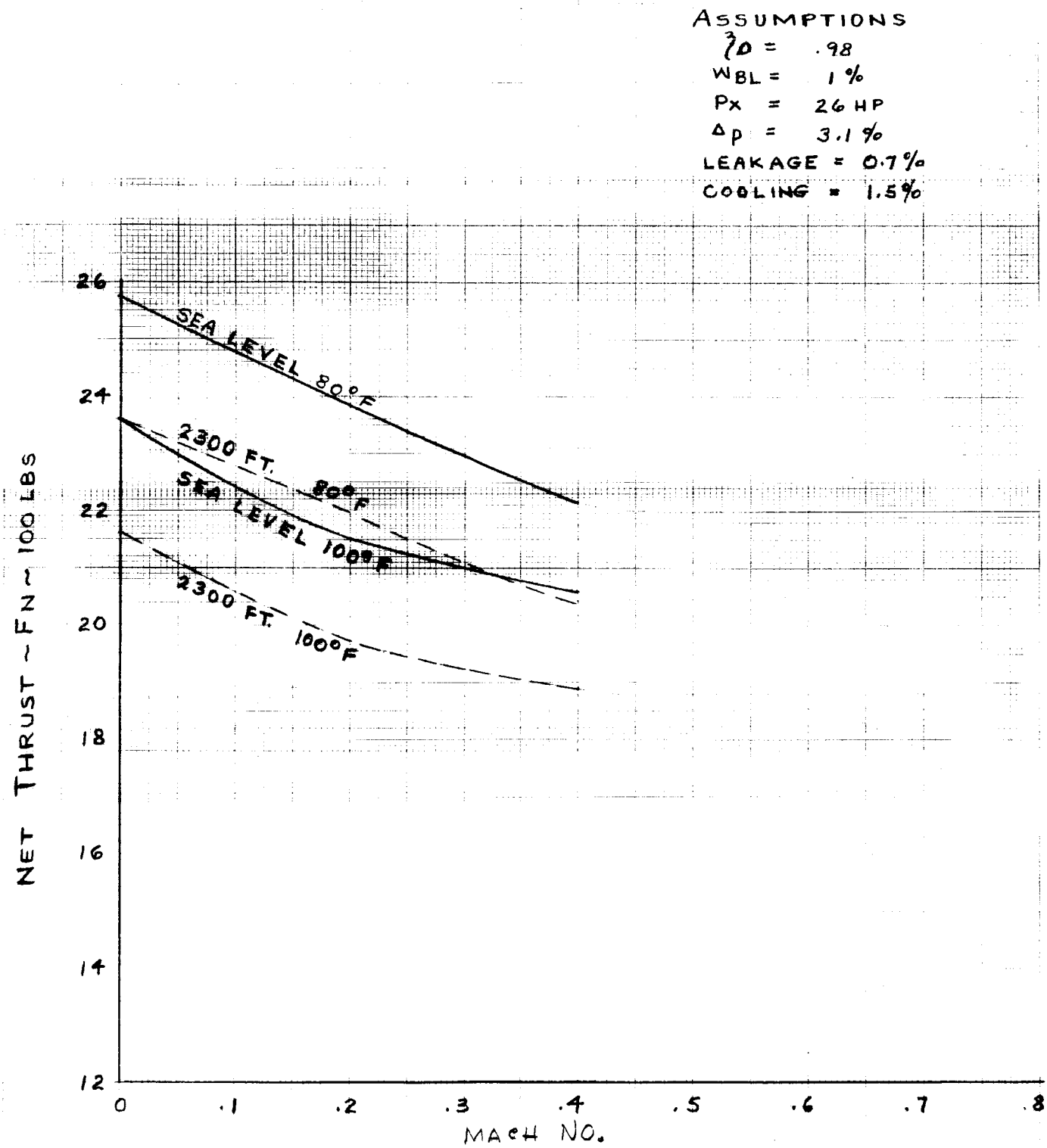


Figure IV-16

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
MILITARY POWER

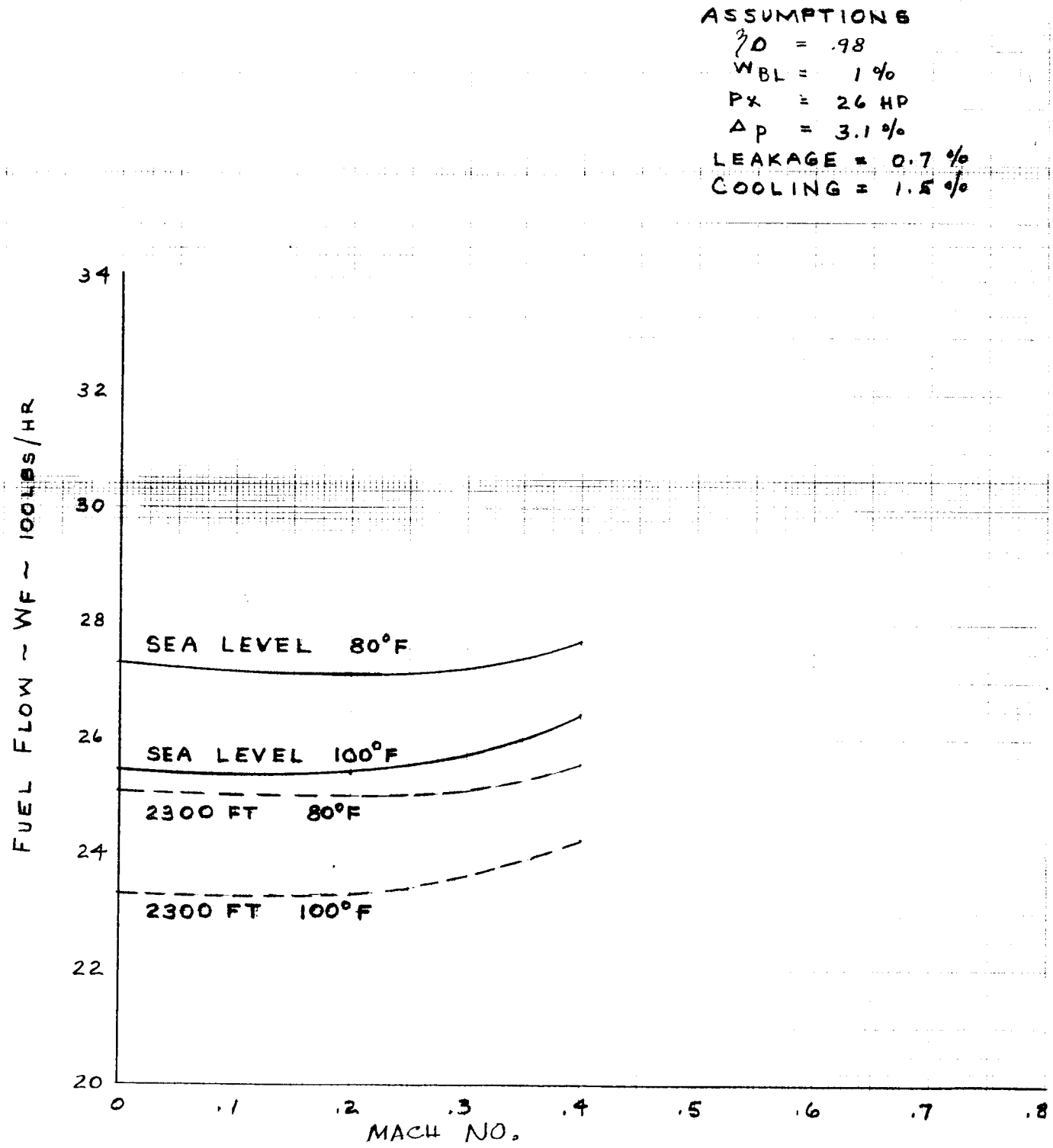


Figure IV-17

YJ85-GE-19

ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE 2500 FT.
CRUISE 80° F. DAY

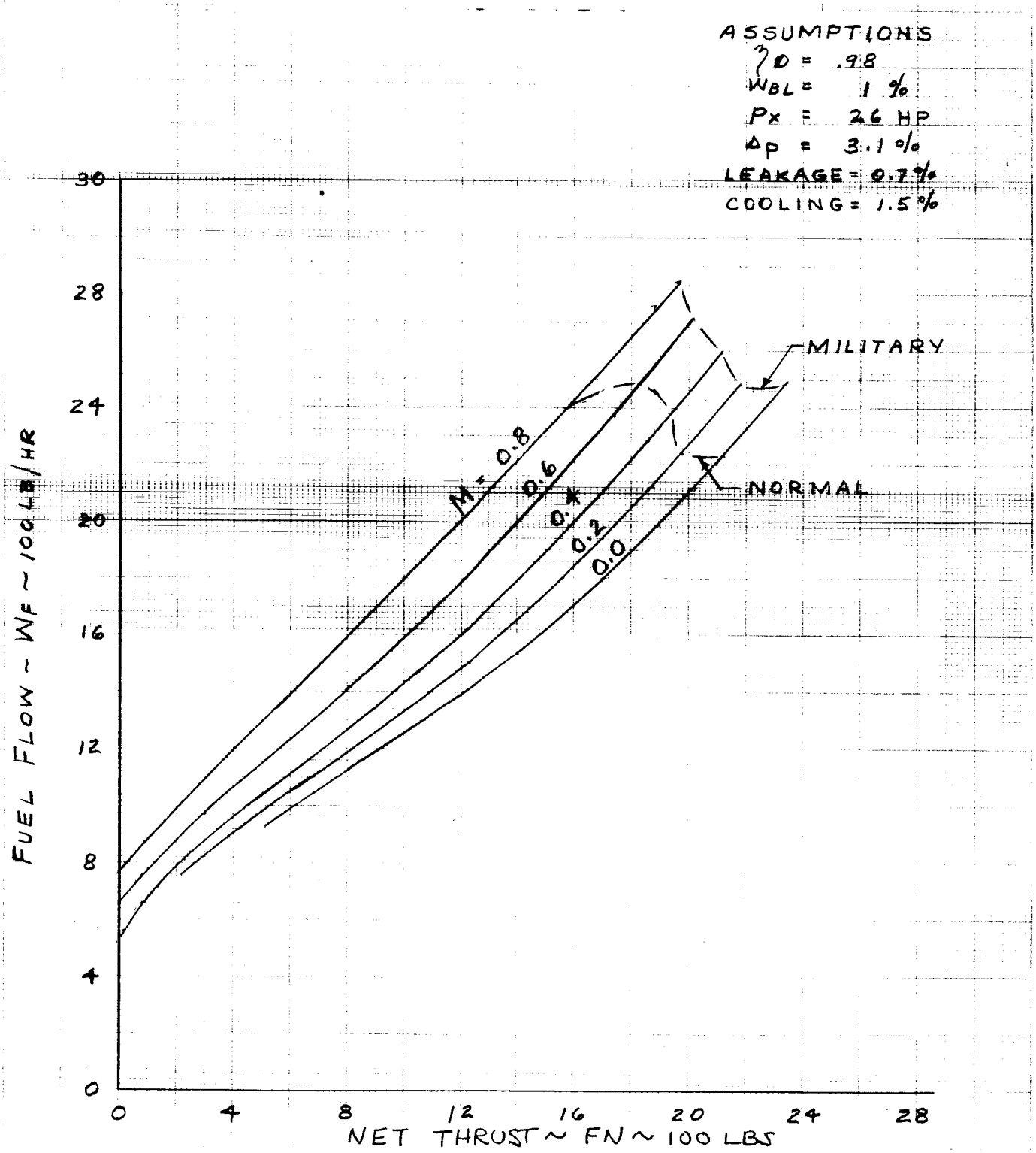


Figure IV-18

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
IN NASA V/STOL AIRPLANE
5000 FT. CRUISE 80° F. DAY

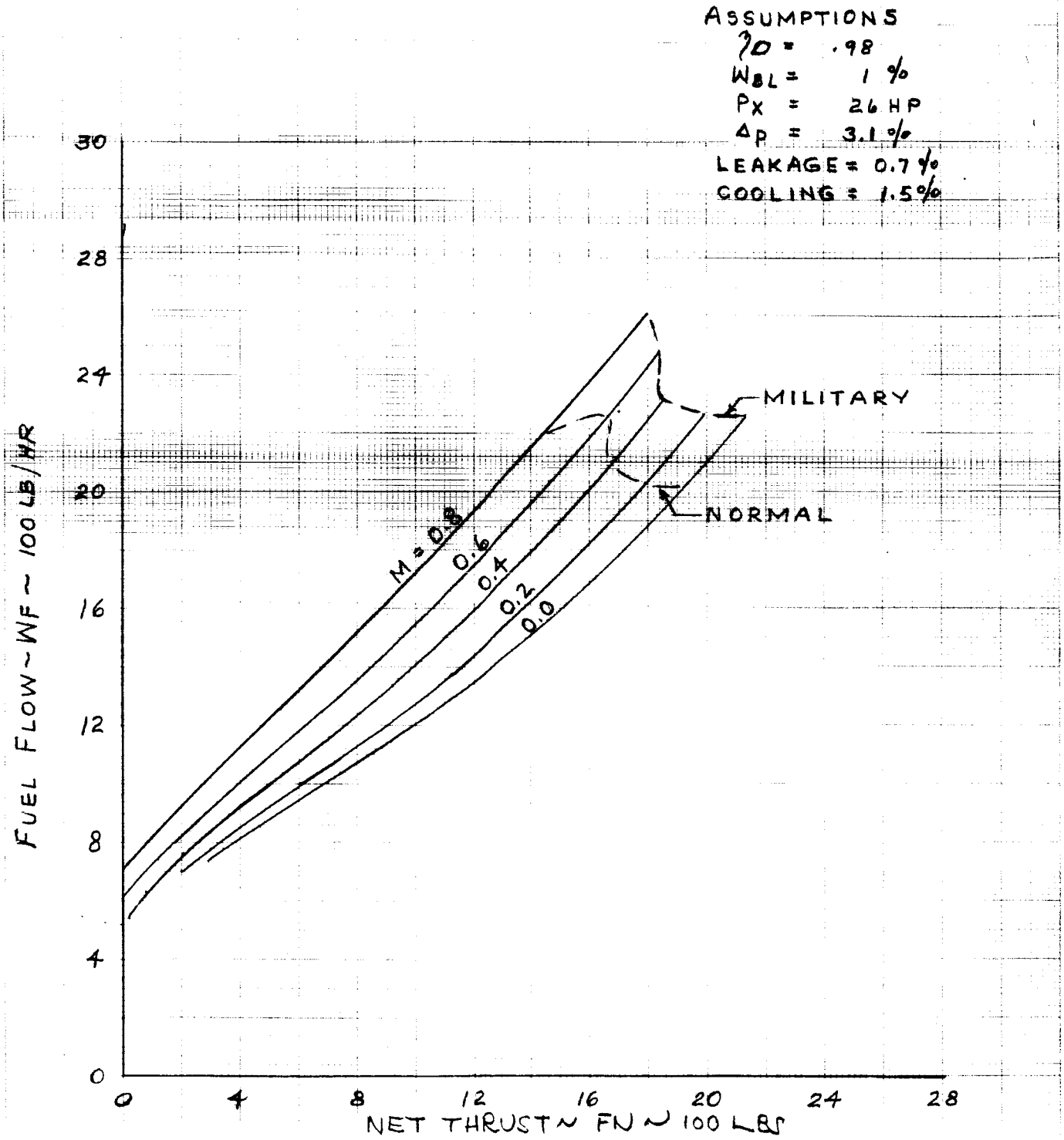


Figure IV-19

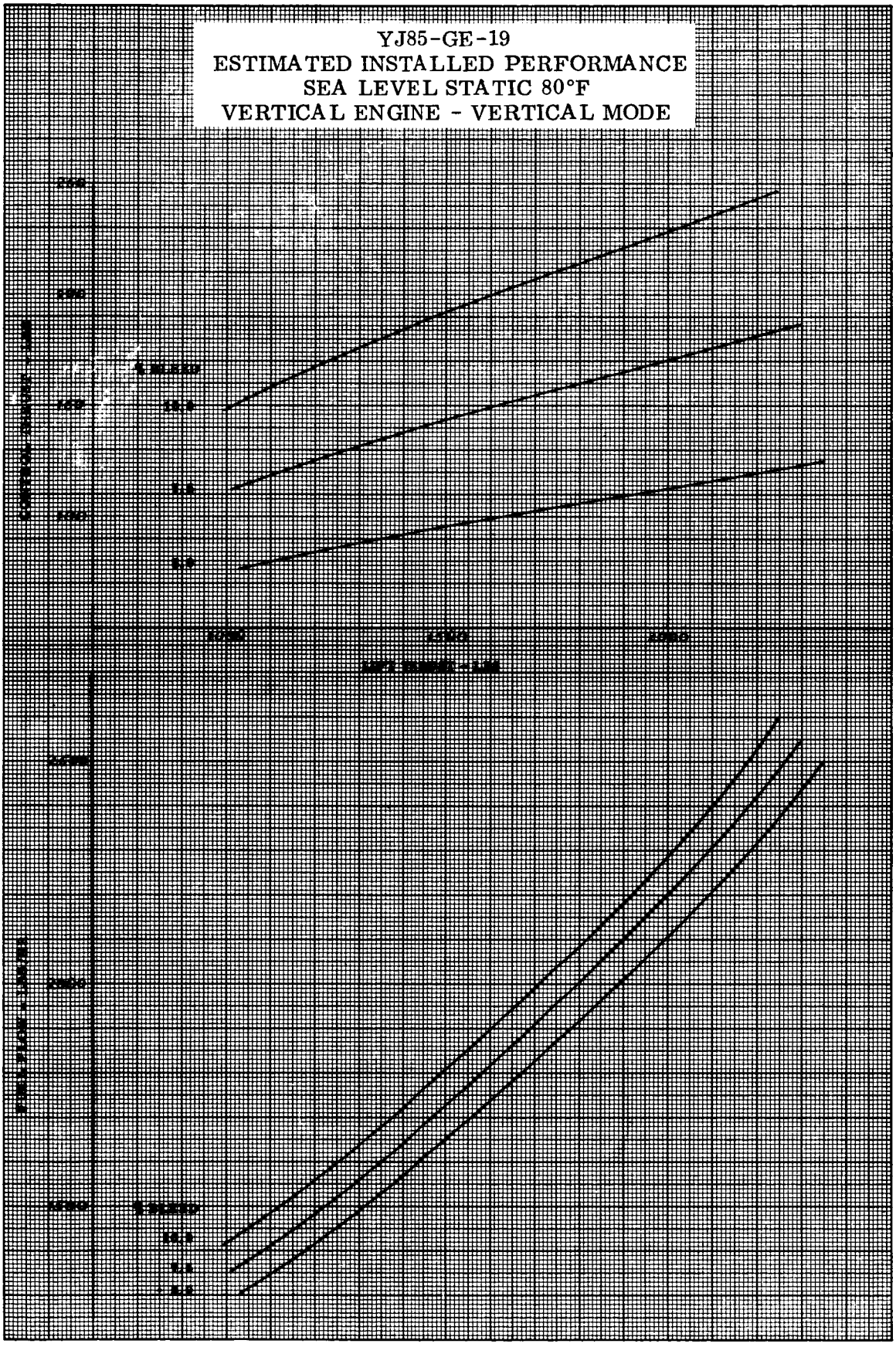

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
SEA LEVEL STATIC 80°F
VERTICAL ENGINE - VERTICAL MODE

Figure IV-20

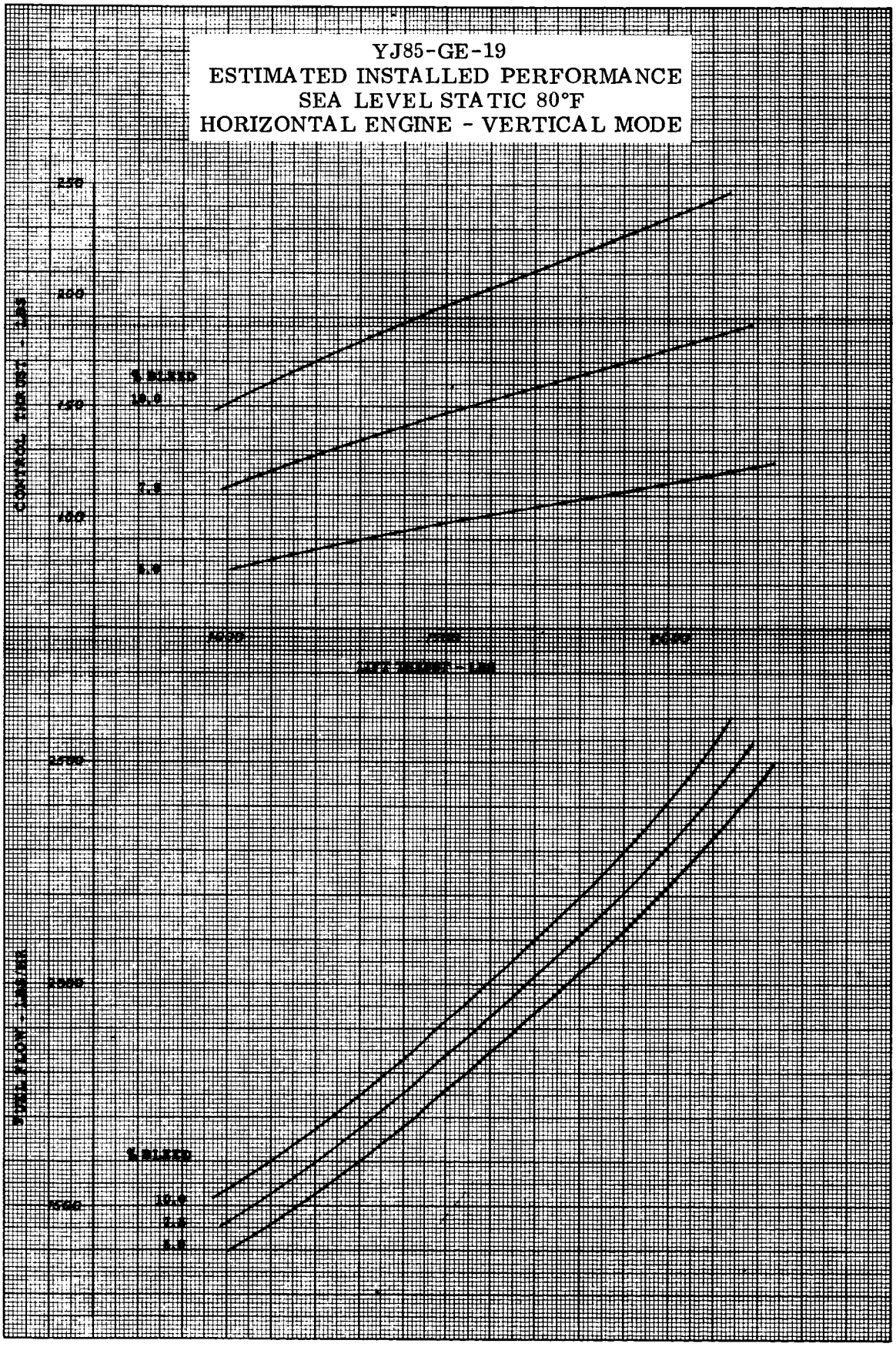

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
SEA LEVEL STATIC 80°F
HORIZONTAL ENGINE - VERTICAL MODE
250
200
150
100
% BLEED
10.0
7.0
5.0
2500
2000
1500
% BLEED
10.0
7.0
5.0

Figure IV-21

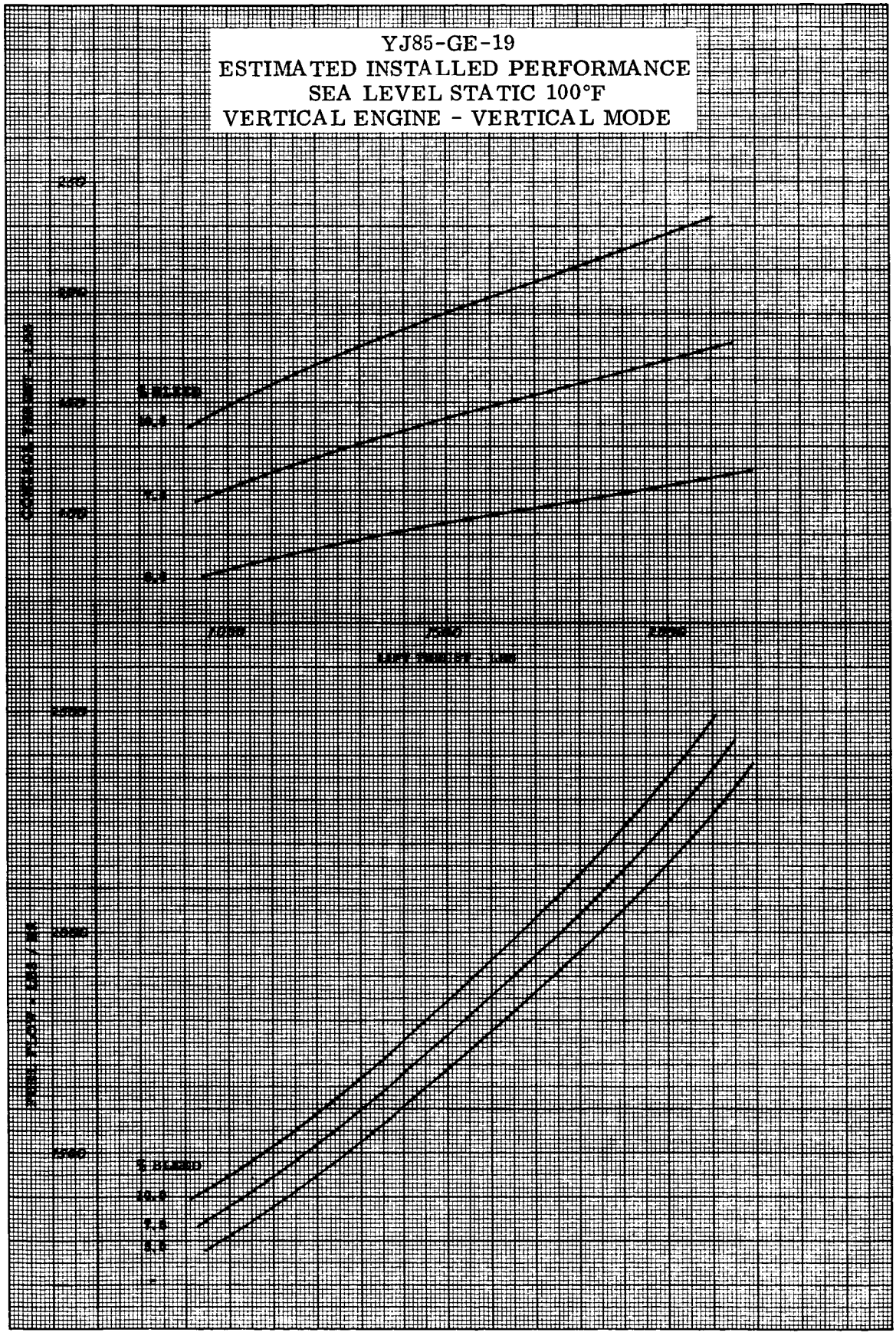

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
SEA LEVEL STATIC 100°F
VERTICAL ENGINE - VERTICAL MODE

Figure IV-22

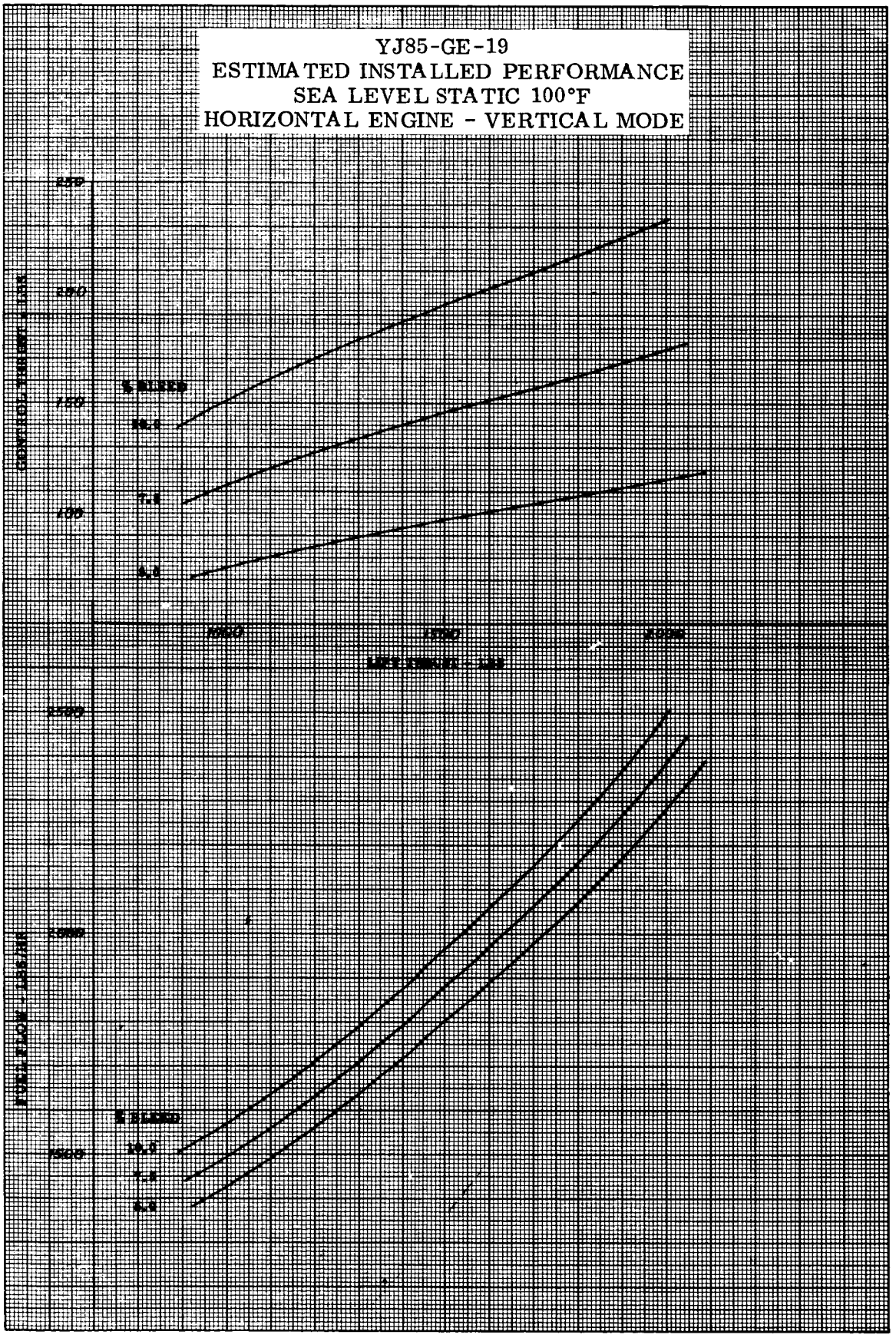

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
SEA LEVEL STATIC 100°F
HORIZONTAL ENGINE - VERTICAL MODE

Figure IV-23

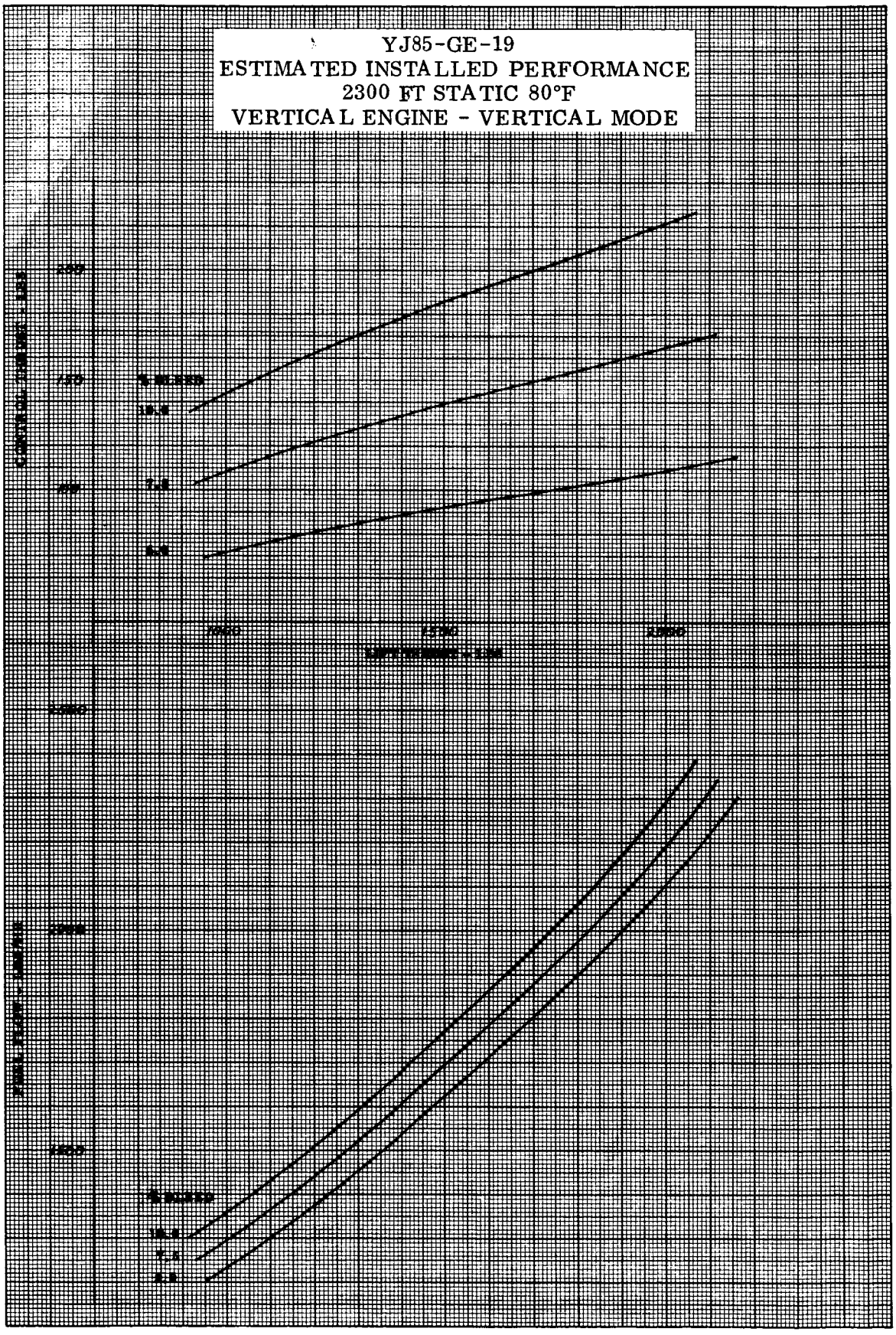

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
2300 FT STATIC 80°F
VERTICAL ENGINE - VERTICAL MODE

Figure IV-24

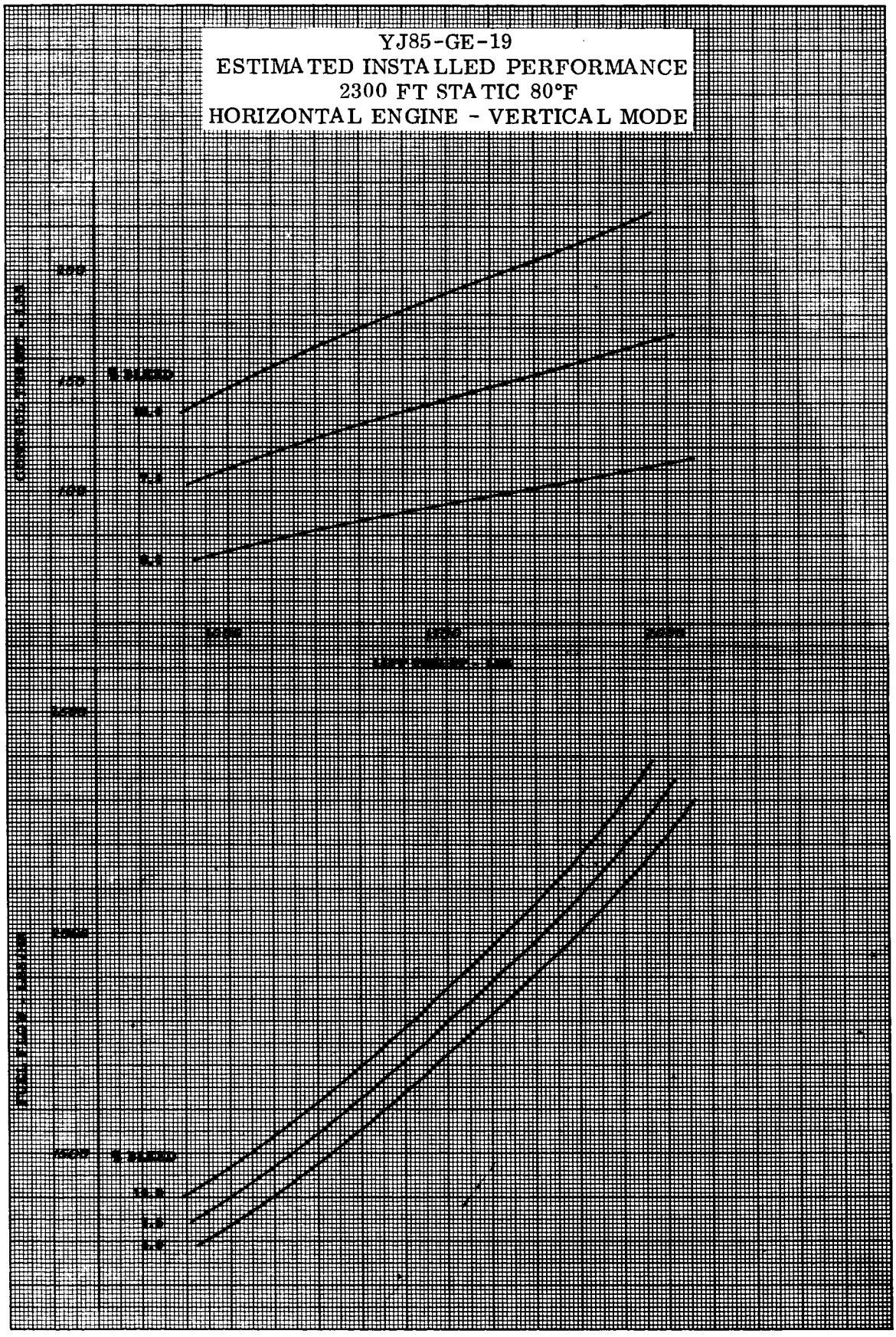

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
2300 FT STATIC 80°F
HORIZONTAL ENGINE - VERTICAL MODE

Figure IV-25

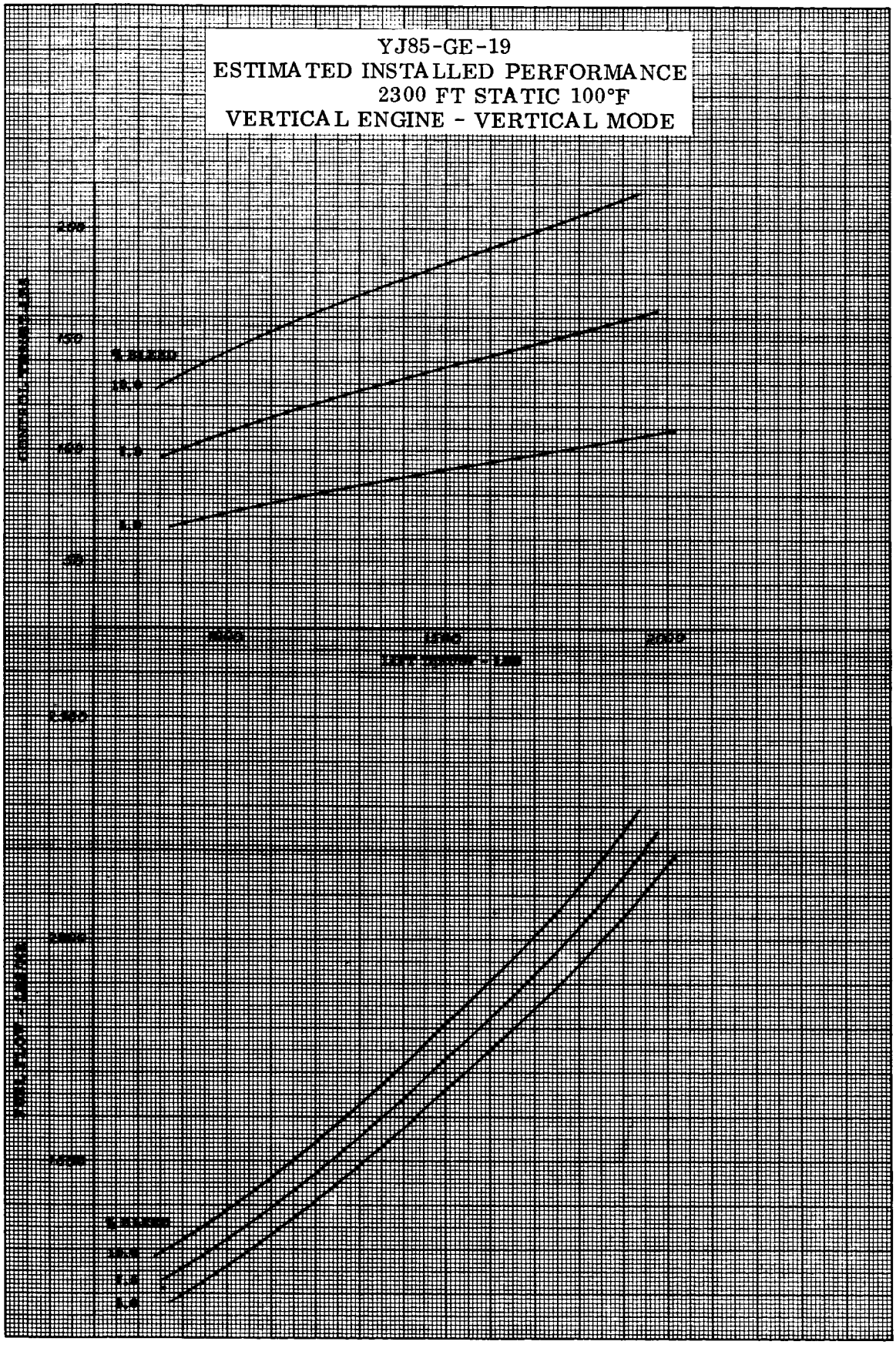

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
2300 FT STATIC 100°F
VERTICAL ENGINE - VERTICAL MODE

Figure IV-26

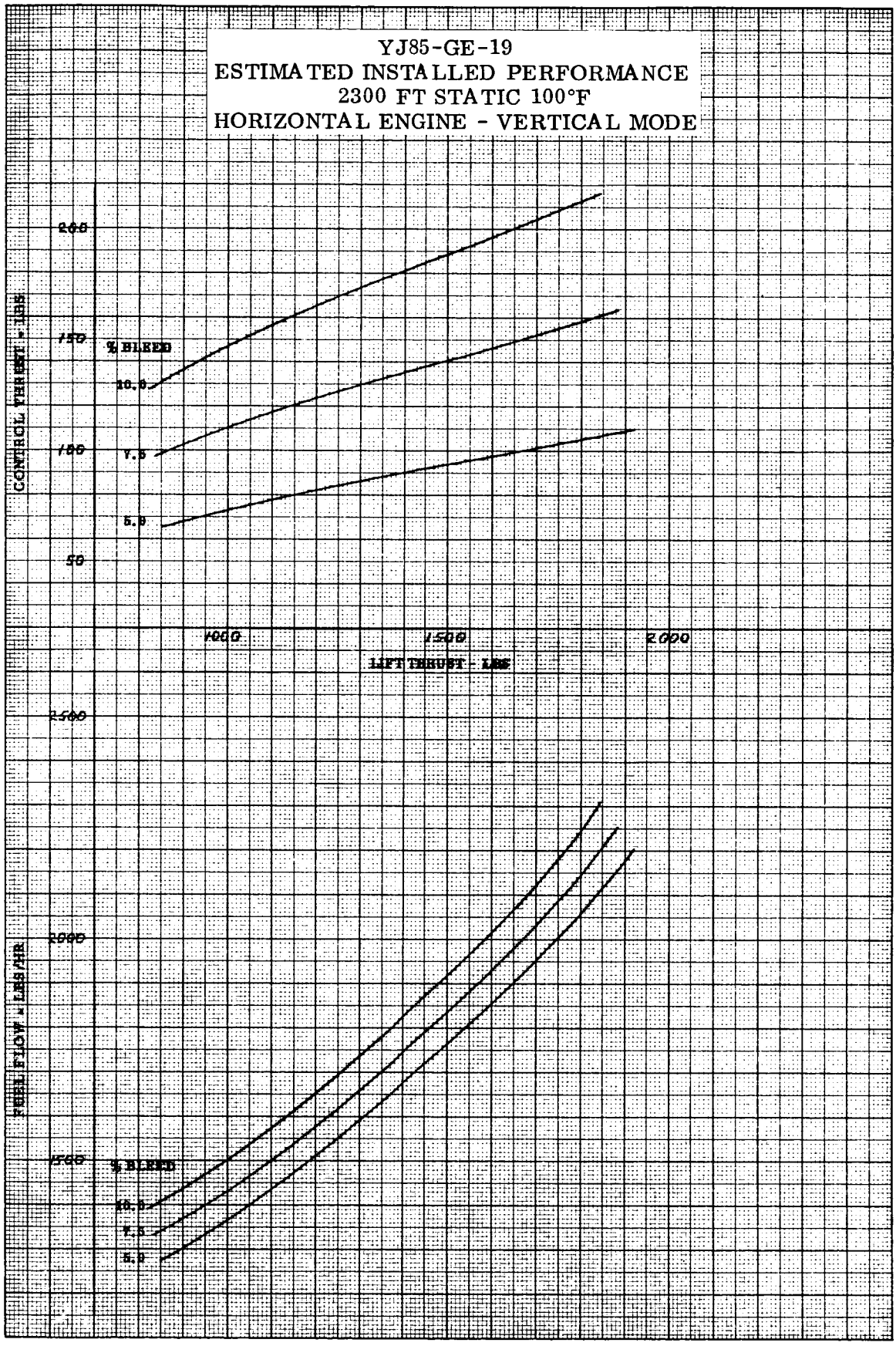

YJ85-GE-19
ESTIMATED INSTALLED PERFORMANCE
2300 FT STATIC 100°F
HORIZONTAL ENGINE - VERTICAL MODE
CONTROL THRUST - LBS
200
150
100
50
% BLEED
10.0
7.5
5.0
1000
1500
2000
LIFT THRUST - LBS
FUEL FLOW - LBS/HR
2500
2000
1500
% BLEED
10.0
7.5
5.0

Figure IV-27

YJ85- GE-19
INSTALLED GROSS THRUST vs. SFC
SEA LEVEL STATIC 80° F. DAY

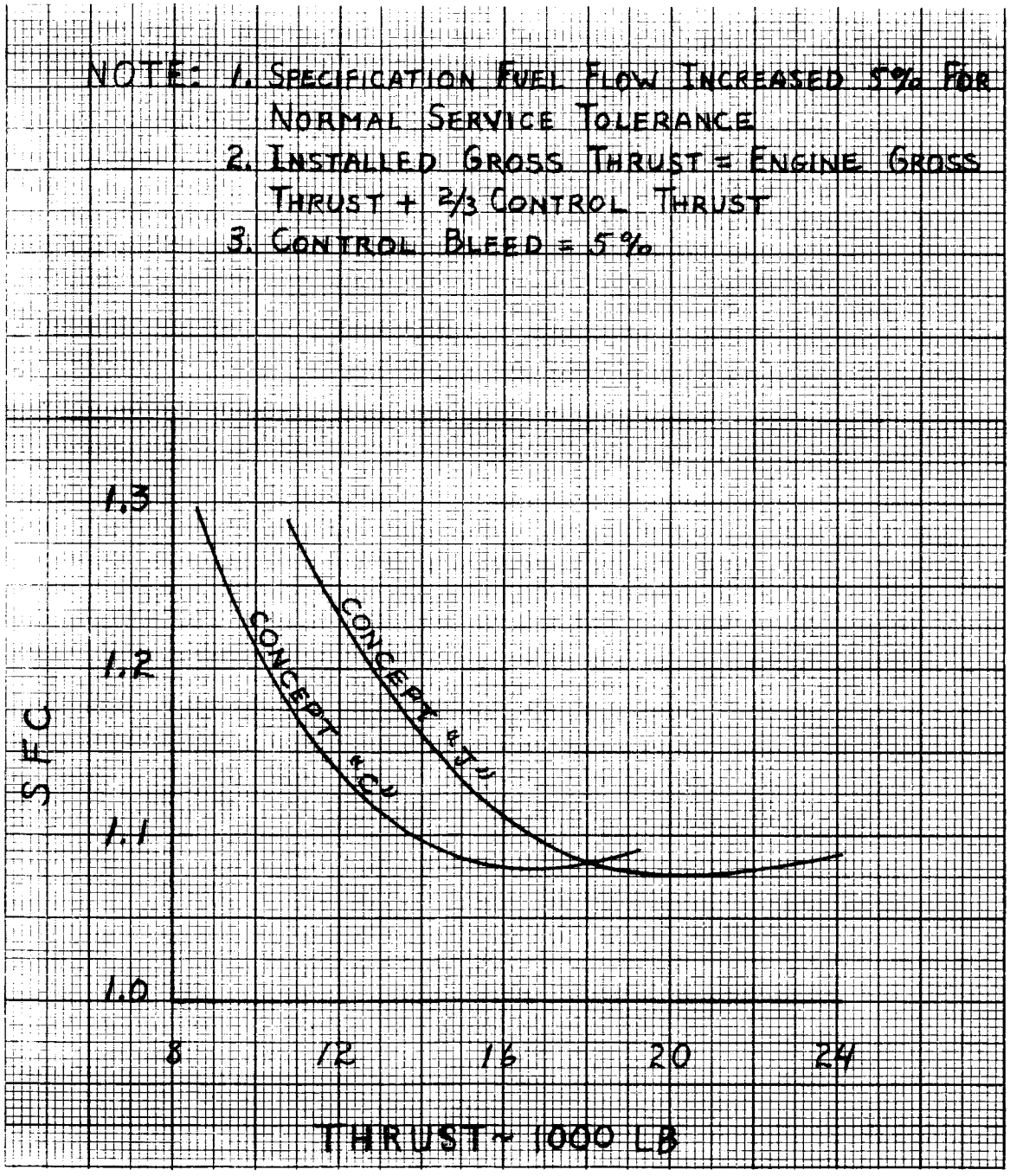


Figure IV-28

INSTALLED PERFORMANCE YJ85-GE-19 NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARIATION
VERTICAL MODE VERTICAL ENGINE OPTIMIZED A 8

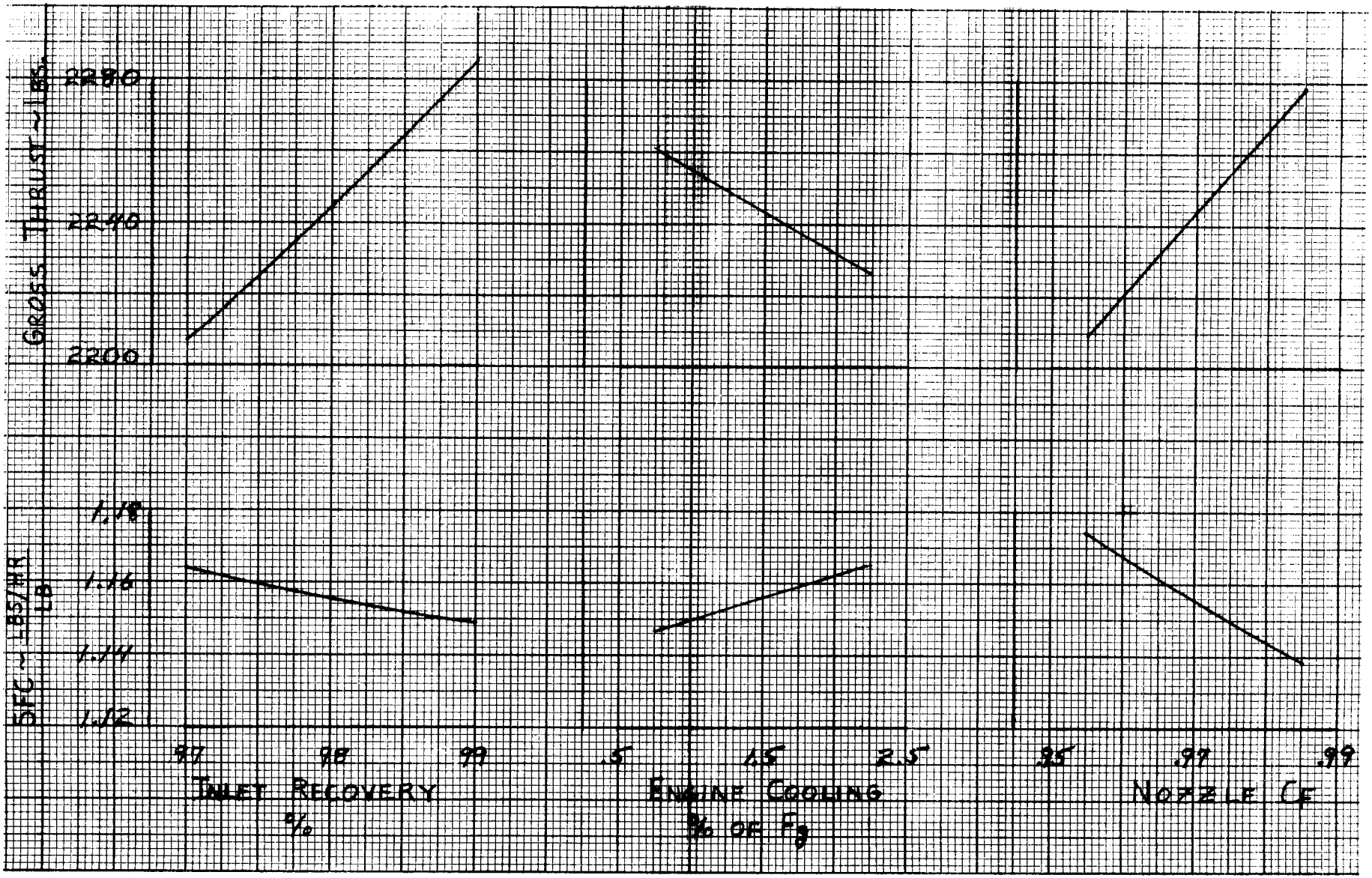


Figure IV-29

INSTALLED PERFORMANCE YJ85- GE-19 IN NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARATION

HORIZONTAL ENGINE VERTICAL MODE OPTIMIZED A 8

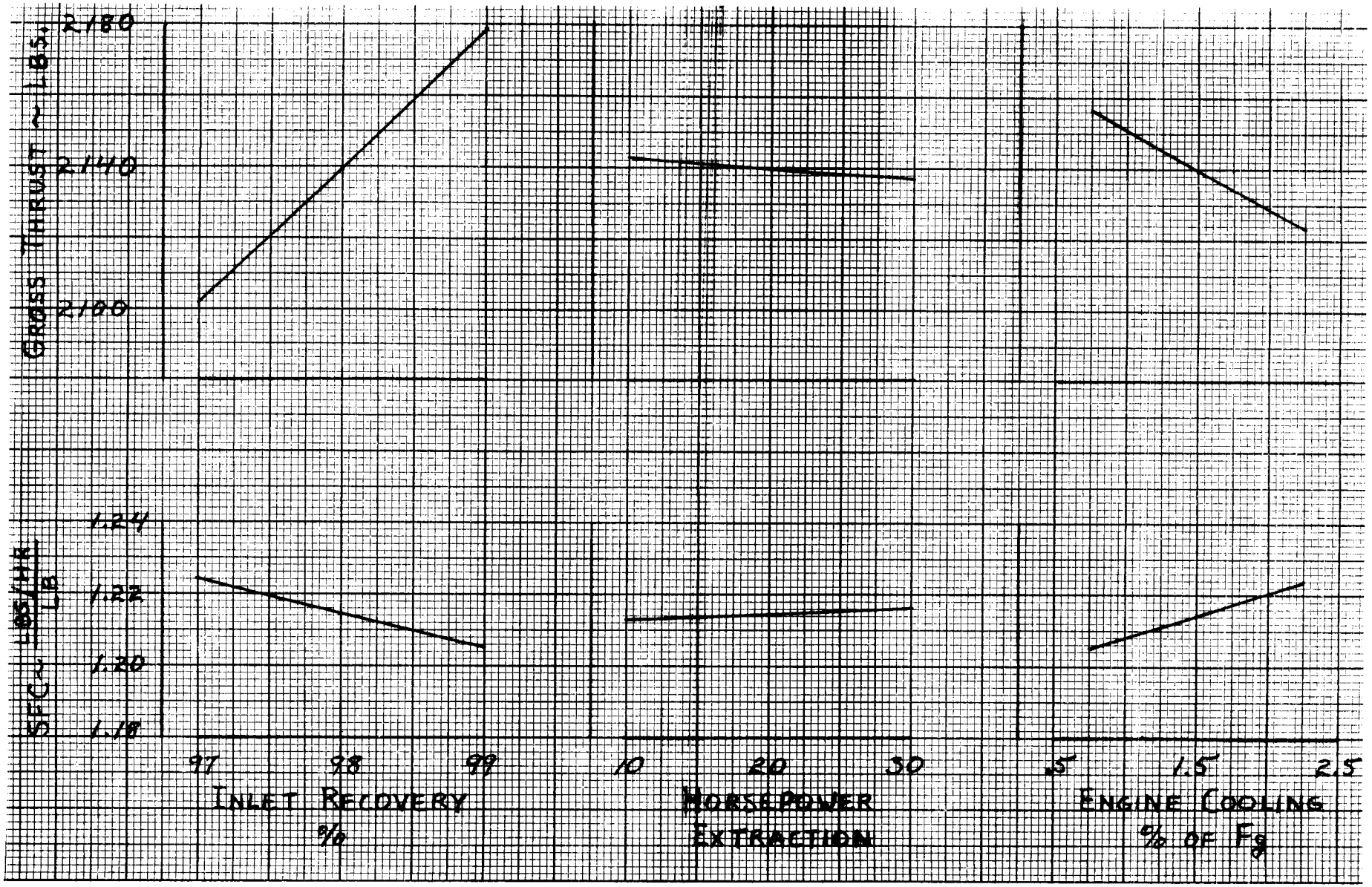


Figure IV-30

INSTALLED PERFORMANCE YJ-85-GE-19 IN NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARATION
HORIZONTAL ENGINE VERTICAL MODE OPTIMIZED A 8

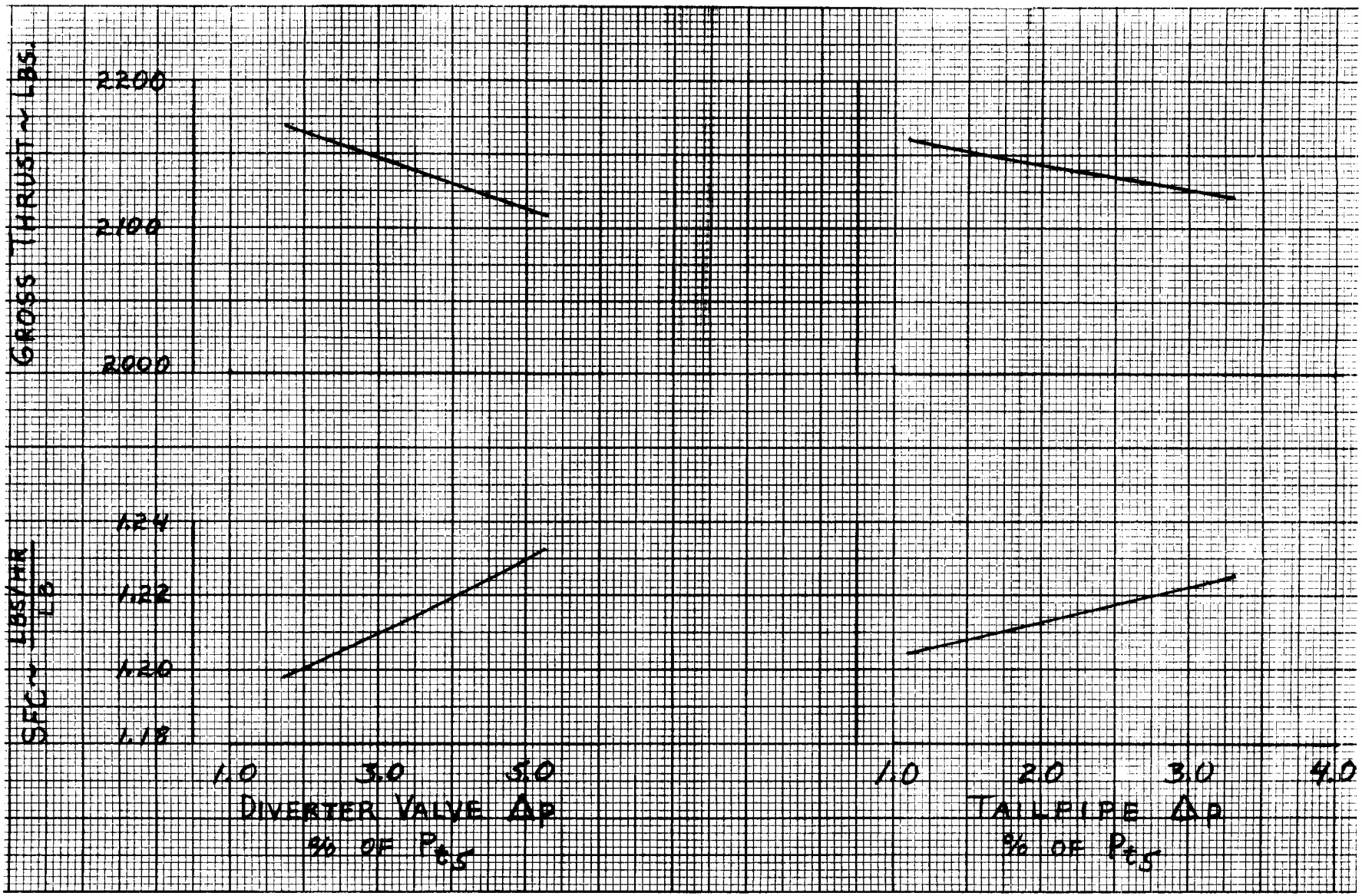


Figure IV-31

INSTALLED PERFORMANCE IN YJ-85 -GE-19 IN NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARIATION
HORIZONTAL ENGINE VERTICAL MODE OPTIMIZED A 8

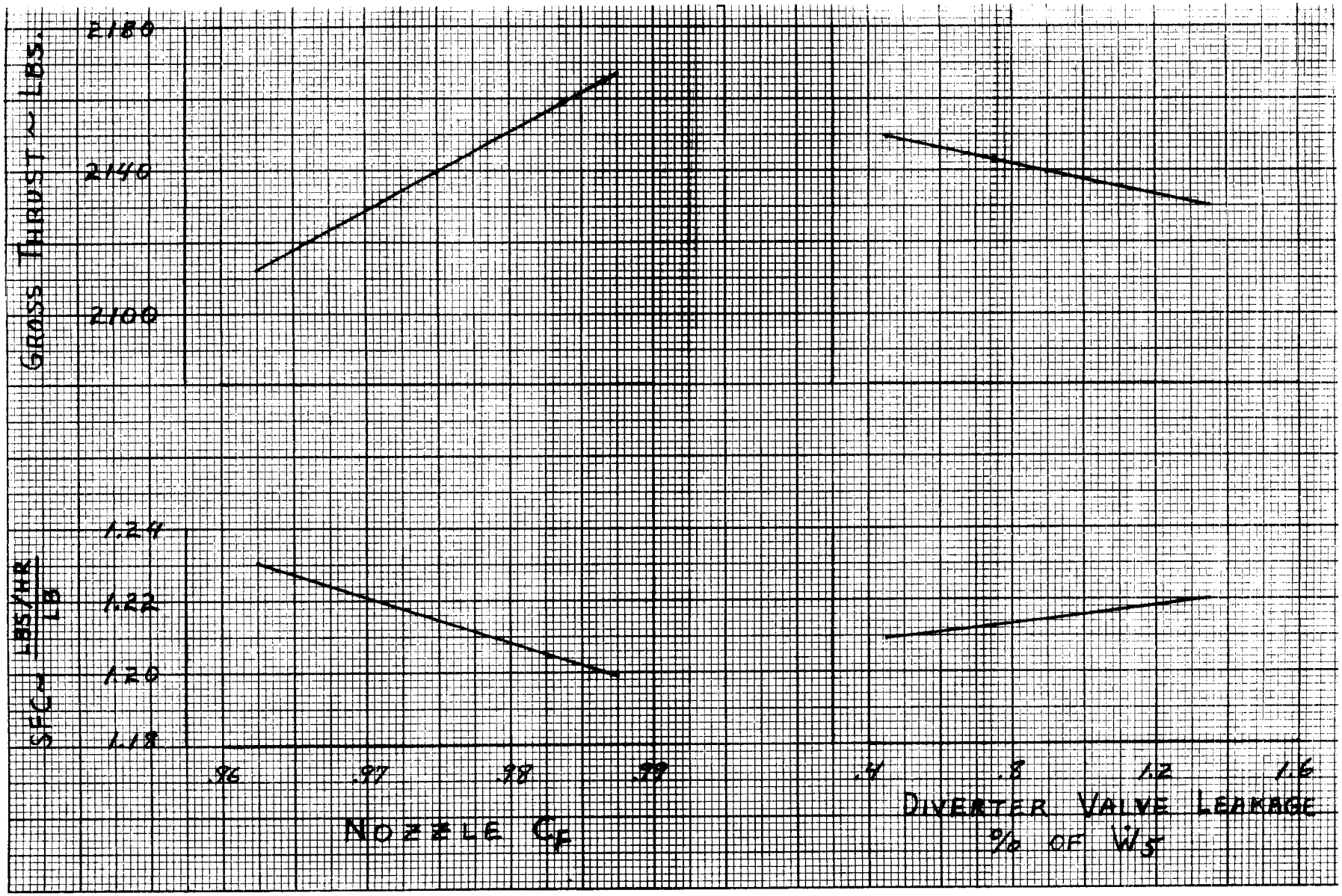


Figure IV-32

INSTALLED PERFORMANCE YJ85-GE-19 IN NASA V/STOL
SENSITIVITY TO ±50% INSTALLATION LOSS VARIATION
VERTICAL ENGINE VERTICAL MODE
CONSTANT A8

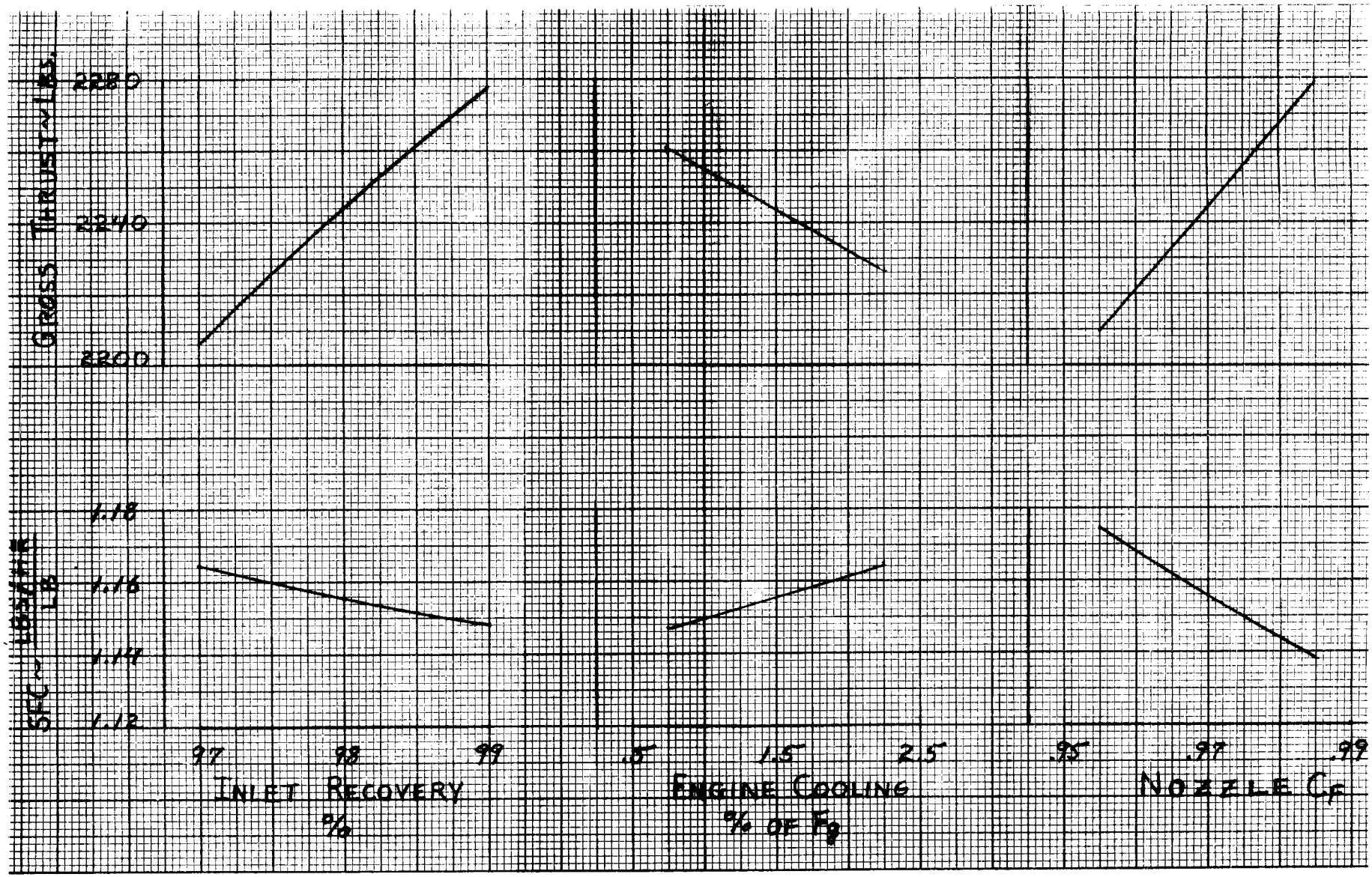


Figure IV-33

INSTALLED PERFORMANCE YJ85-GE-19 IN NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARIATION
HORIZONTAL ENGINE VERTICAL MODE
CONSTANT A8

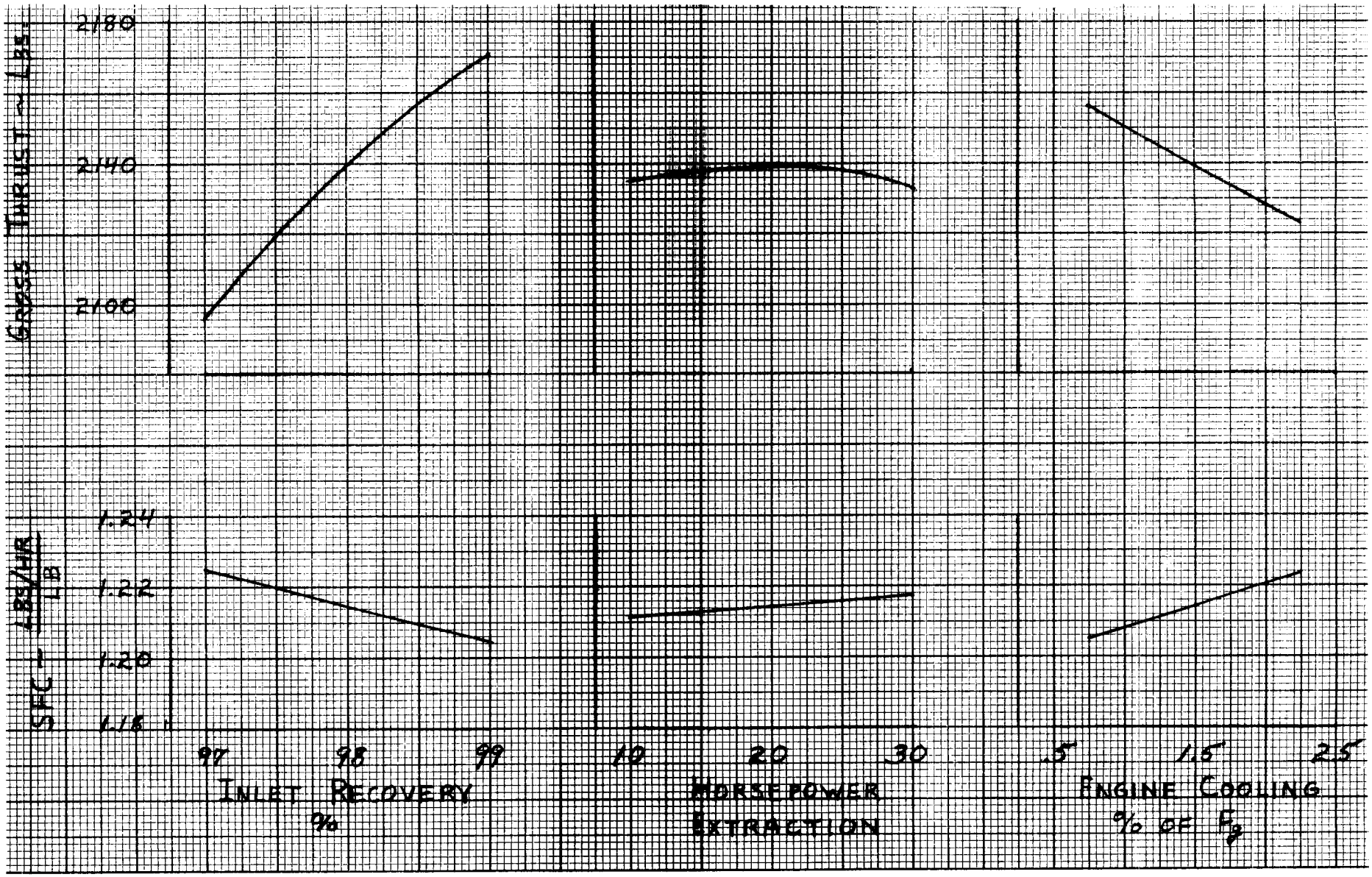


Figure IV-34

INSTALLED PERFORMANCE YJ85-GE-19 IN NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARIATION
HORIZONTAL ENGINE VERTICAL MODE
CONSTANT A8

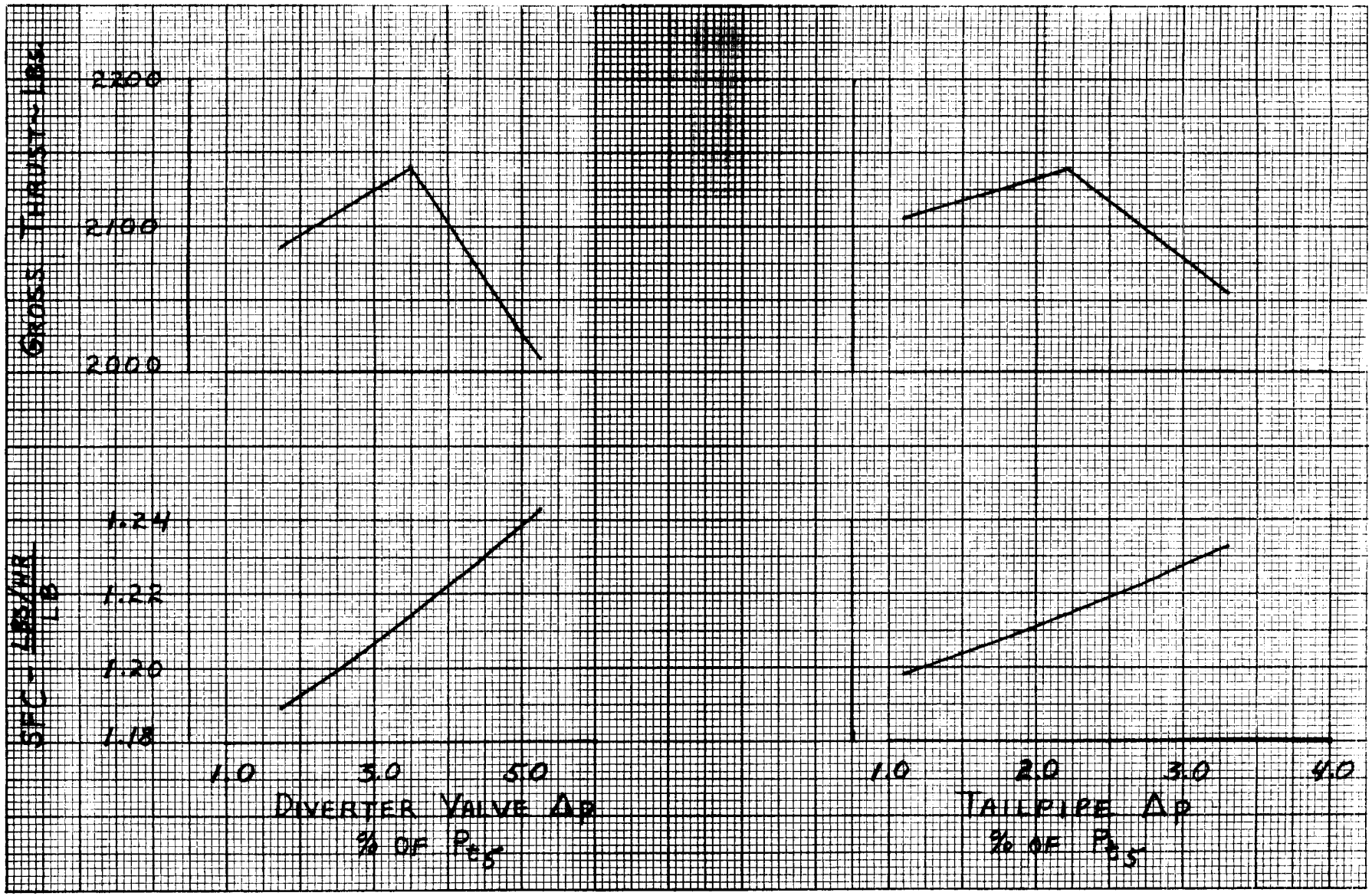


Figure IV-35

INSTALLED PERFORMANCE YJ85-GE-19 IN NASA V/STOL
SENSITIVITY TO ± 50% INSTALLATION LOSS VARIATION
HORIZONTAL ENGINE VERTICAL MODE
CONSTANT A8

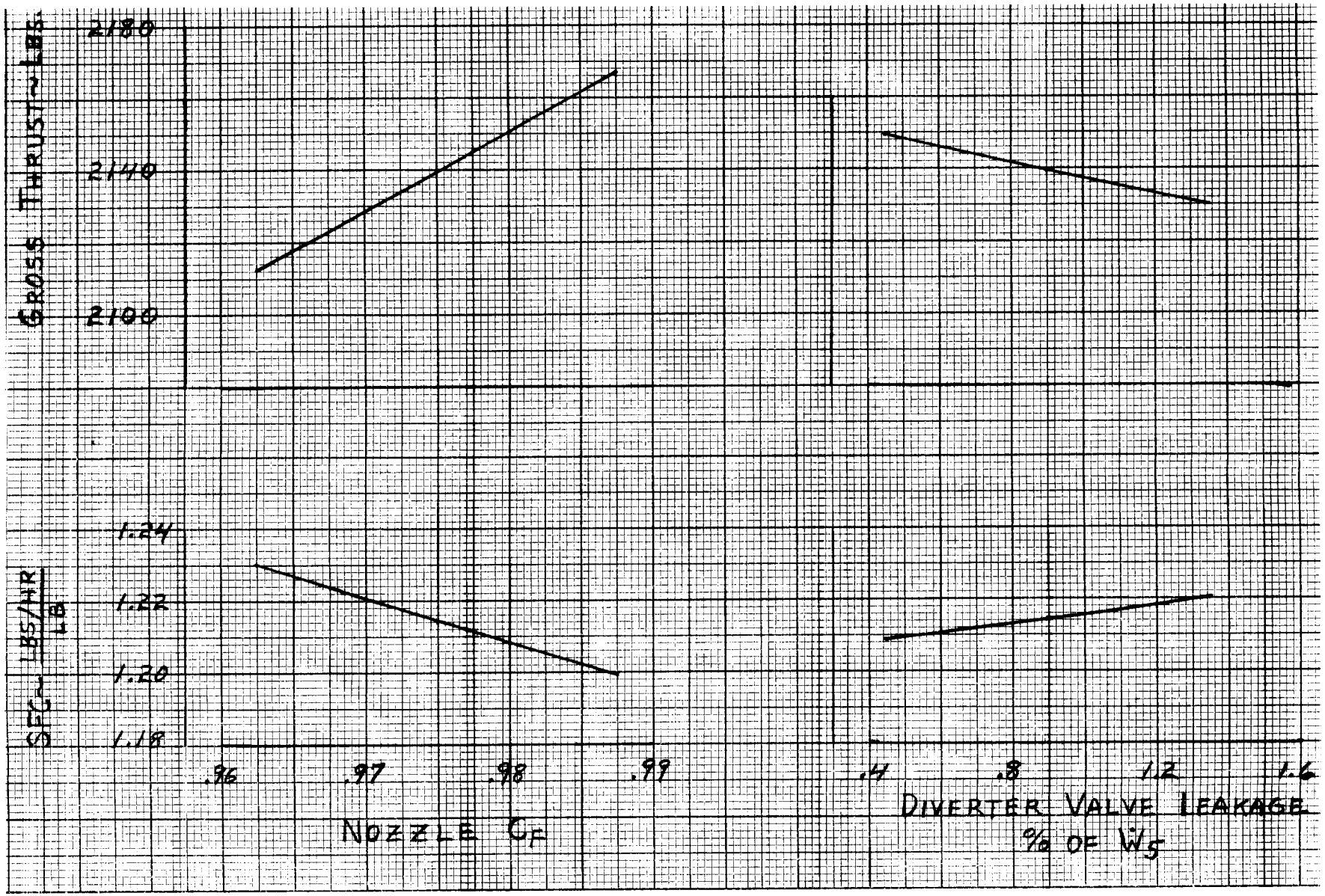


Figure IV-36

ENGINE THRUST LOSS SENSITIVITY

LIFT-CRUISE ENGINE THRUST LOSS, ΔF_g LB

	-50%	NOMINAL	+50%
INDUCTION SYSTEM RECOVERY	50	89	127
HORSEPOWER EXTRACTION	1	4	7
DIVERTER VALVE PRESS. LOSS	26	56	86
DIVERTER VALVE LEAKAGE	10	20	30
TAILPIPE PRESS. LOSS	16	36	56
PIVOTING SPHERE IN LIEU OF CONICAL NOZZLE	+13	15	43
COOLING	15	31	48
80^0 F AMBIENT TEMP	196	196	196
COMPRESSOR BLEED AIR (10%)	440	440	440
TOTAL INSTALLATION LOSS, LB	741	887	1033
INSTALLED THRUST, LB	2274	2128	1982
% THRUST DEVIATION FROM NOMINAL	+6.85%	-	-6.85%
REQUIRED NOZZLE AREA, A_{nozzle} -- SQ. IN.	112.59	117.04	122.01

Figure IV-37

B. REACTION CONTROL SYSTEM

The analytical design of the engine bleed air operated attitude control system for the Concept C aircraft as configured on drawing AD001G18A60 has been completed and its performance has been estimated at the following operating points:

1) minimum aircraft weight (minimum fuel)
2) maximum aircraft weight (full fuel)
3) combined full power lift and lift/cruise engines

1. System Design

Figure IV-38 presents a schematic of the attitude control system for the Concept C aircraft. This aircraft concept assumes the use of two (2) General Electric YJ85-GE19 lift/cruise (L/C) engines and six (6) General Electric YJ85-GE19 lift (L) engines. The design and performance of the control system was accomplished through the use of the IBM 7094 computer program (modified) as discussed, in detail, in Reference 4.

Curve A of Figure IV-39 was used to describe the engine compressor bleed port loss characteristics of the lift and lift/cruise engines. Curve B of this figure, extracted from Reference 5, presents the revised bleed port characteristics of these engines. This curve was not used in the presented performance because G.E. states that it must be substantiated by test for other than straight manifolds. However, calculations were performed to determine the percent change in control thrust when using this curve relative to curve A. The estimated changes are as follows:

Engine Comp. Bleed - %	Thrust Change - %
3.4	-0.4
5.2	+0.35
7.5	+1.30
10.0	+6.20

Since the engine bleed manifolds as depicted on drawing AD001G18A60 are not straight it can be anticipated that the percent change in the presented control thrust when using curve B as the bleed port characteristic will be less than the above listed changes at the corresponding percent engine bleed.

The bleed air distribution ducting of drawing AD001G18A60 was analyzed and duct equivalent velocity head losses were assigned to friction, bends, intersections, expansion joints and transition sections occurring in the system in accordance with the procedure outlined in Reference 6. Table IV-1 presents the magnitude of the assigned component losses and the total equivalent velocity head loss ($\Delta H/q$) assigned to each of the four major ducts of the distribution system (engine bleed, roll, nose pitch, tail pitch-yaw). Since the computer program does not distinguish between bleed air extracted from the lift and lift/cruise engines the bleed air duct possessing the largest $\Delta H/q$ was used in the program which, for this aircraft concept, is the lift/cruise engine bleed air duct. As stated in Reference 4, the roll ducts are assumed to be symmetrical; hence, the presented roll duct $\Delta H/q$ is for either of the wings.

The attitude control thrust requirements presented in Section III of this report were used as the design requirements for the control system. Figure IIB-7 of Reference 4 was used in conjunction with the system analysis to establish the optimum inlet mach number to each of the four distribution ducts. The analysis indicate that the engine bleed air ducts must be designed to their maximum allowable are if the area of the roll, pitch and pitch-yaw duct are to be minimized. The area of the bleed duct was established based upon the maximum allowable area on drawing AD001G18A60. An optimum duct inlet Mach number of approximately .25 was established for the roll, nose pitch, and tail pitch-yaw duct. This inlet Mach number was used to determine the minimum duct area consistent with the design thrust requirements. The next larger commercially obtainable duct was selected to minimize cost and simplify procurement. Table IV-2 presents the established duct diameters.

After the distribution duct sizes had been established the sensitivity of this system was determined relative to variations in duct $\Delta H/q$. The sensitivity analysis was performed at sea level with total engine lift thrust equal to minimum aircraft weight since the bleed air velocity - hence, pressure loss - will be maximum within the ducts at this condition. The assigned $\Delta H/q$ for each duct was varied ± 50 percent. Table IV-3 presents the results of this analysis.

Analysis of the control requirements of the attitude control system, presented in Section VI of this report, established that the lift and lift/cruise engines

should be bled at a nominal 5 percent when the controls are in the neutral position, with the bleed air being distributed as follows:

a) one-sixth to each of the roll control jets

b) one-third to the nose pitch control jet

c) one-third to the tail control jet

d) zero to the yaw control jets

Single asis operation was established as being as follows:

> When full control thrust is demanded from one control jet the opposite control jet on that axis is closed. The control jets on the remaining two axes are in their neutral control position.

Since the combined pitch-yaw thrust requirement is less demanding on the system than the single axis thrust requirement this requirement did not influence the system design except for the tail pitch-yaw duct diameter.

The above system configuration was used to establish the maximum control jet nozzle area, the neutral control jet nozzle area and to compute system performance. The control jets are located at the extremities of each of the control axes, with the yaw jets located at the tail only, and operate as variable area convergent nozzles with a constant adiabatic efficiency of 96 percent. The maximum required nozzle area for each control jet is presented on Table IV-2 along with the required single axis control jet area and the corresponding neutral control jet area per jet.

2. System Performance

The single axis attitude control system performance was estimated at the following three operating points:

1) Total lift thrust equal to minimum aircraft weight

2) Total lift thrust equal to maximum aircraft weight

3) Full combined lift and lift/cruise engine power

The control system was assumed to exhibit the previously discussed characteristics. Maximum engine compressor bleed was limited to 10 percent. This limitation

produced a 2.4 percent deficit in yaw control thrust at minimum aircraft weight. However, this deficit can be eliminated by reducing the neutral control nominal engine compressor bleed from 5 percent to 4.8 percent. Table IV-4 presents the estimated control system performance at the above listed operating points. The design point for the roll and pitch nozzle areas is minimum aircraft weight whereas the design point for the yaw control nozzle area is maximum aircraft weight.

3. References

4. Part II - Technical Data Substantiation Report, "V/STOL Jet Operations Research Airplane Design Study, Report No. FHR 3391-3I, 7 April 1967.

5. General Electric Model Specification E1129, dated 1 November 1966.

6. "SAE Aero-Space Applied Thermodynamics Manual," Rev. January 1962.

7. Faires, V.M., "Thermodynamics," The MacMillan Company, 1962.

Concept C - Attitude Control System Schematic (Ref. AD001618A60)

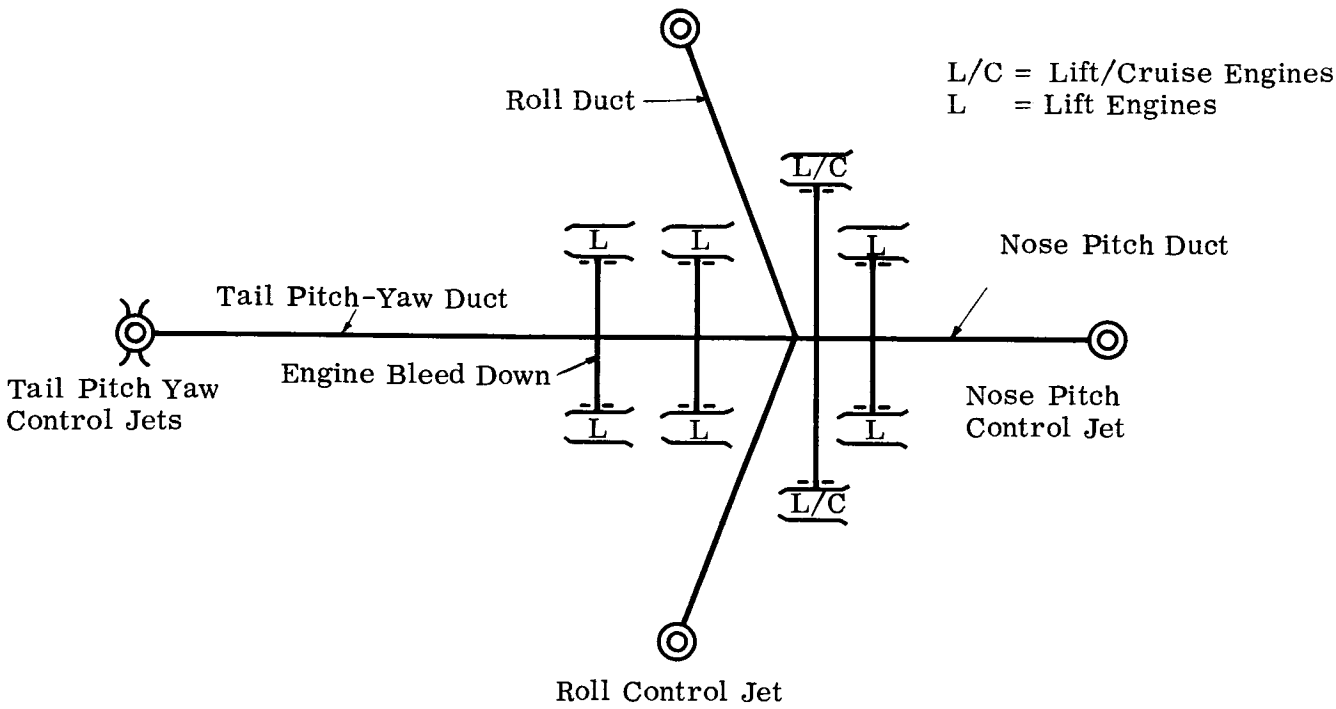


Figure IV-38

% $\Delta P/P_E$ Vs. % W_E/W_A

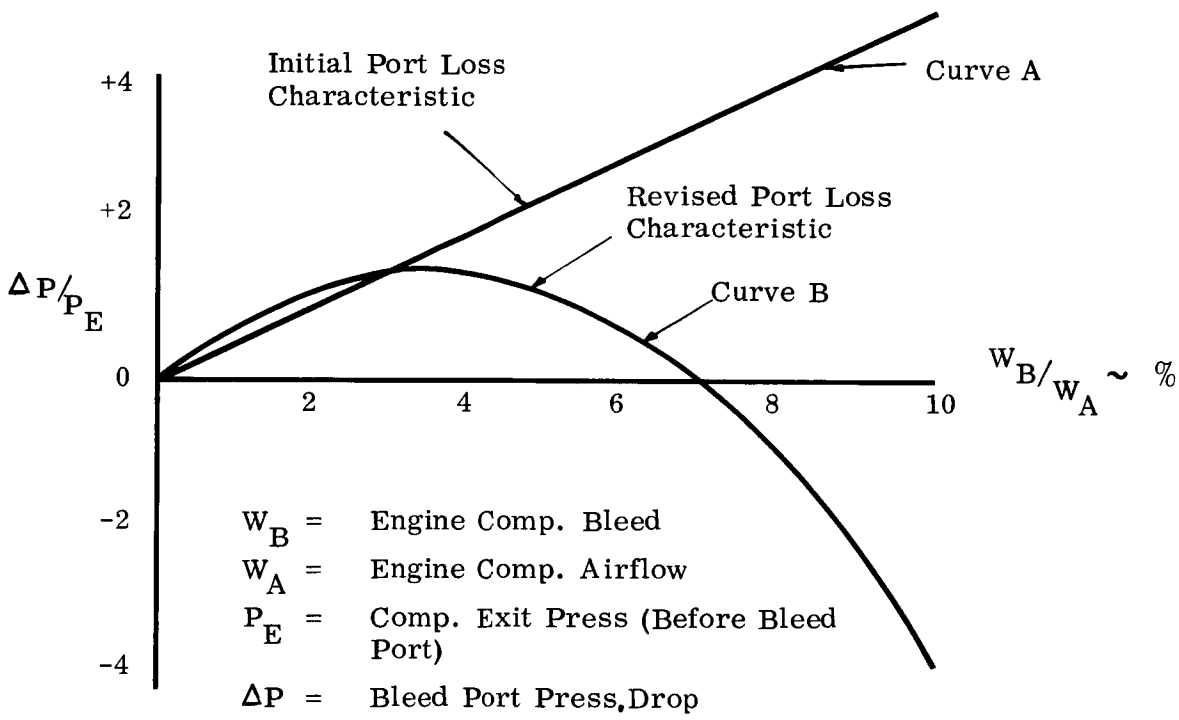


Figure IV-39

TABLE IV-1

ATTITUDE CONTROL SYSTEM -
EQUIVALENT VELOCITY HEAD LOSS

Distribution Duct	Engine Bleed	Roll	Nose Pitch	Tail Pitch-Yaw
*Friction $\sim (4f \frac{L}{D})$	.08	.62	.39	.29
Bends	.60	.63	.66	.00
Intersections	.95	.52	.40	.40
Expansion Joints	.20	.90	.90	.90
Transition Sections	.00	.15	.15	.15
Total	1.83	2.82	2.50	1.74

* $f = \frac{.046}{(N_{Re})^{.2}}$ Reference 4, page 414

L = duct length ~ ft

D = duct diameter ~ ft

N_{Re} = Reynolds number

TABLE IV-2

ATTITUDE CONTROL SYSTEM -
DUCT DIAMETER AND CONTROL JET NOZZLE AREA

Distribution Duct	Roll	Nose Pitch	Tail Pitch-Yaw	Engine Bleed (per engine)
Duct Diameter ~ In.	5.5	8.0	9.0	3.5

Control Jet	Roll	Pitch	Yaw
Maximum Jet Nozzle Area - In.^{2}	13.0	24.0	15.0
Single Axis Roll Nozzle area ~ in.^{2}	13.0	4.0	0.0
Single Axis Pitch Nozzle area ~ in.^{2}	2.0	24.0	0.0
Single Axis Yaw Nozzle area ~ in.^{2}	2.0	4.0	15.0

TABLE IV-3

ATTITUDE CONTROL SYSTEM -
SYSTEM SENSITIVITY

Ambient: Sea level, 80°F; Aircraft weight/max. lift thrust = .615
Engines: 2L/C + 6L

Control Axis	Engine Bleed Duct	Roll	Pitch		Yaw
			Nose	Tail	
Duct Diameter ~ in.	3.5	5.5	8.0	9.0	-
Engine Comp. Bleed ~ %	10	3.8	7.6	7.6	5.2
Design $\Delta H/q$	1.83	2.82	2.50	1.74	1.74
Attitude Control Thrust ~ lbs	-	581	1141	1167	817
Design $\Delta H/q$ + 50%	2.75	4.23	3.75	2.61	2.61
Attitude Control Thrust ~ lbs	-	562	1102	1152	814
Design $\Delta H/q$ ~ lbs	.915	1.41	1.25	.87	.87
Attitude Control Thrust ~ lbs	-	594	1170	1181	821

Note: Zero bleed airflow to all axes except the axis being analysed

TABLE IV-4

ATTITUDE CONTROL SYSTEM -

ESTIMATED PERFORMANCE

Ambient: Sea level, 80°F; Engines: 2 L/C + 6L
Aircraft Weight: Min. = 11860 lbs, max. = 15,300 lbs
F_{NT} = Max. combined lift thrust, F_N = combined lift thrust

Single Axis Control	F_N/F_{NT}	(Min. Aircraft Weight) .615	(Max. Aircraft Weight) .795	(Full Engine Power) 1.0
Roll	Thrust required ~ lbs	558	553	-
	Engine comp. bleed ~ %	6.9	7.3	7.5
	Roll thrust ~ lbs	570	703	840
	Neut. pitch thrust ~ nose/tail	225/225	280/280	325/330
	Neut. yaw thrust per jet	0	0	0
Pitch	Thrust required ~ lbs	1135	1255	-
	Engine comp. bleed ~ Nose atch/tail pitch	8.8/9.1	9.25/9.4	9.5/9.5
	Pitch thrust ~ nose/tail	1142/1160	1348/1352	1575/1605
	Neut. roll thrust per jet	110	140	170
	Neut. yaw thrust per jet	0	0	0
Yaw	Thrust required ~ lbs	794	872	-
	Engine comp. bleed ~ %	9.5	10.0	10.0
	Yaw thrust ~ lbs	775	890	1065
	Neut. roll thrust per jet	105	140	150
	Neut. pitch thrust ~ nose/tail	230/210	290/280	320/310
Neutral Controls	Engine comp. bleed ~ %	4.4	4.8	5.0
	Neut. roll thrust per jet	115	155	174
	Neut. pitch thrust ~ nose/tail	251/229	321/310	370/358
	Neut. yaw thrust per jet	0	0	0

C. VECTORING SYSTEM

1. General

The selection of the thrust vectoring system for the NASA V/STOL lift and lift-cruise engines was based on considerations of performance, cost, and availability. The most attractive system based on these considerations is the General Electric improved pivoting sphere nozzle.

2. Types of Thrust Vectoring Systems Considered

a. Lift Engines

Thrust vectoring systems considered for the lift engines were: compensating and downstream louvers, rotating cascades, and three versions of the pivoting sphere.

b. Lift-Cruise Engines

Thrust vectoring systems considered for the lift-cruise engines included all of those considered for the lift engines plus the Pratt & Whitney rotating nozzle, the G.E. three-bearing swivel nozzle, and a block and turn system.

All of the above thrust vectoring systems are covered in detail in FHR 3391-3I "Part II - Technical Data Substantiation Report" dated 7 April 1967.

3. Final System Selection

The final system selected, the G.E. single plane ±30° improved pivoting sphere was selected primarily on the basis of its higher performance over a range of deflection angles, due primarily to its high thrust coefficients and minimal effective exhaust area variation (Figure IV-40). It is also light weight, simple to actuate, and because of its simplicity, should be highly reliable.

The improved pivoting sphere nozzle is essentially a hemisphere cut by a plane normal to the radius to the pole of the hemisphere. The circular hole cut out by the plane forms the exhaust nozzle. The hemisphere is mounted at the equator on two diametrically opposed trunnions which permit it to pivot as required in the fore and aft direction. A wiping seal prevents the gas flow from leaking back into the mounting area and actuation is accomplished with a hydraulic piston mounted on the engine.

Figure IV-41 presents the performance of the improved pivoting sphere nozzle installed on the lift and lift-cruise engines.

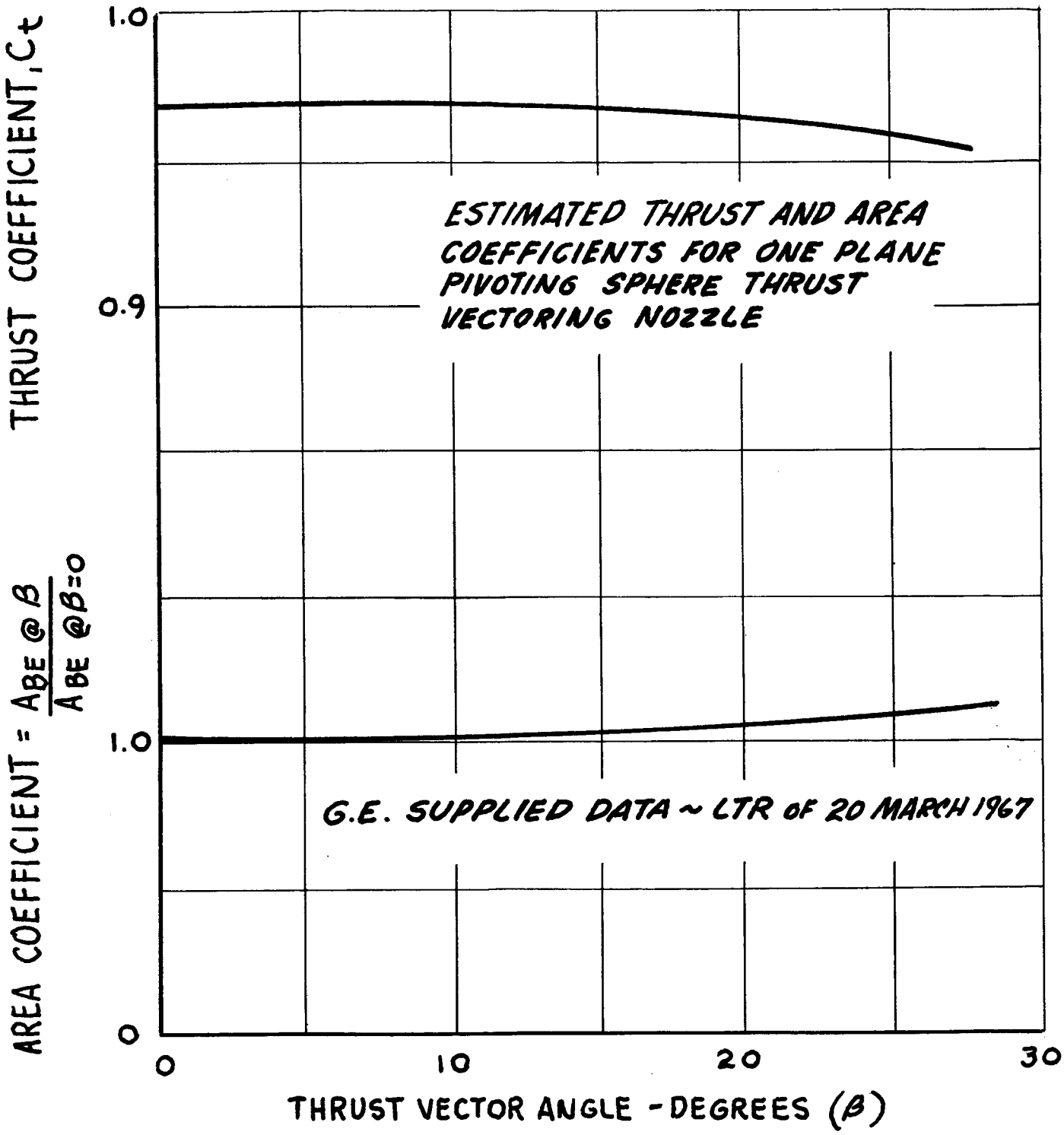
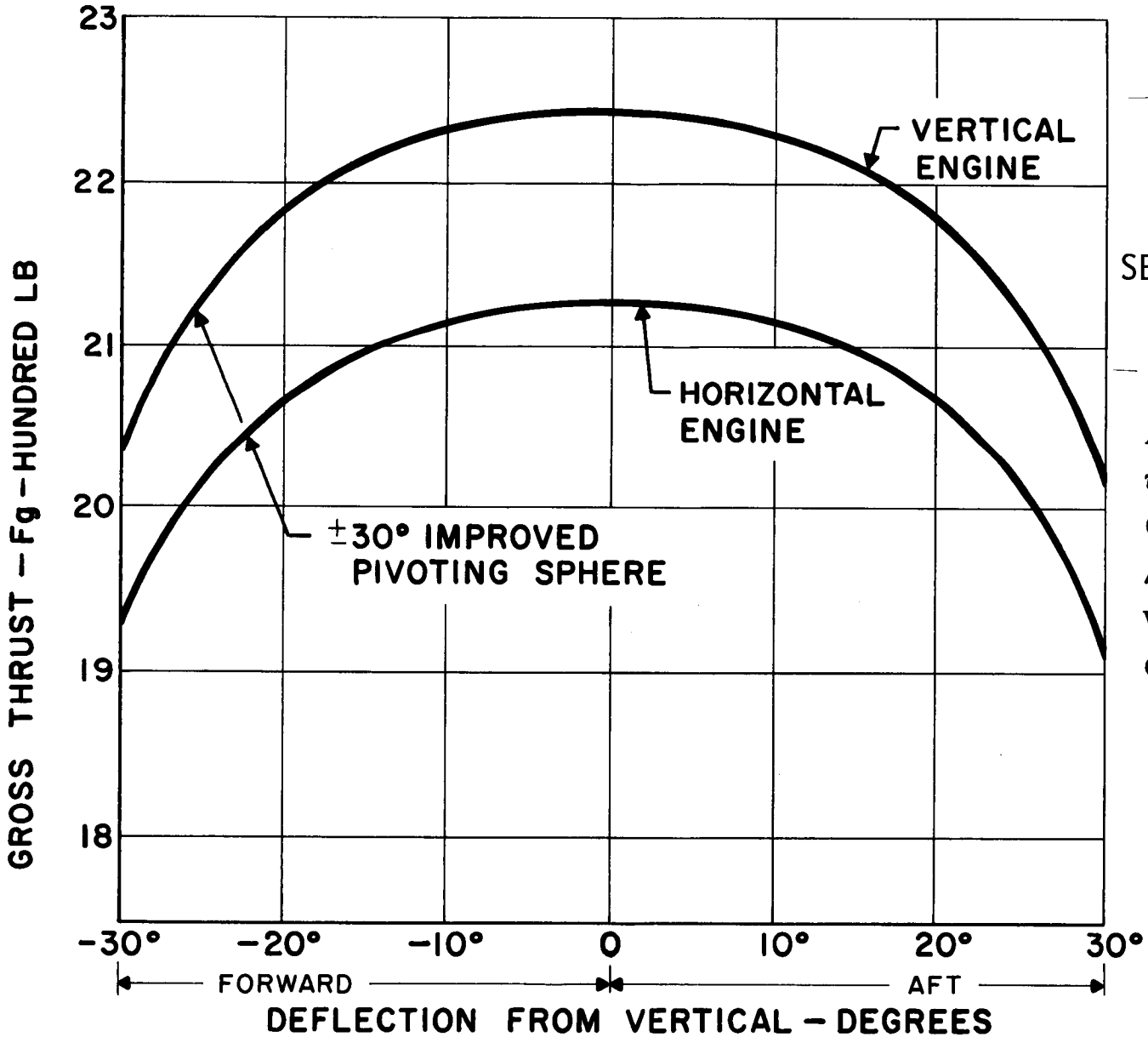

THRUST COEFFICIENT, C_t
1.0
0.9
ESTIMATED THRUST AND AREA COEFFICIENTS FOR ONE PLANE PIVOTING SPHERE THRUST VECTORING NOZZLE
AREA COEFFICIENT = $\frac{A_{BE} @ \beta}{A_{BE} @ \beta = 0}$
1.0
G.E. SUPPLIED DATA ~ LTR OF 20 MARCH 1967
0
0
10
20
30
THRUST VECTOR ANGLE - DEGREES (β)

Figure IV-40

THRUST VECTORING PERFORMANCE



YJ85 - GE - 19

VERTICAL MODE

SEA LEVEL STATIC 80°F DAY

TAKE-OFF SETTING

ASSUMPTIONS

η_D = .98

COOLING = 1.5%

ΔA_8 - AS APPLICABLE

W_{BL} = 10%

C_F - AS APPLICABLE

Figure IV-41

D. ENVIRONMENTAL CONTROL SYSTEM

1. Function

The Environmental Control System (ECS) for this aircraft controls the temperature (cooling and heating) and humidity within the cabin; defogs and removes rain from selected cabin transparent surfaces; controls forward and aft electronic bay (cooling) temperature; supplies oxygen to the crew during VTOL and emergency operation; and controls emergency ventilation to the cabin and electronic bays. The ECS requirements of this aircraft are identical to those of conventional aircraft except during VTOL operation, at which time the cabin atmosphere is subject to become contaminated with engine exhaust gases, dust and debris. Since the duration of time at which the cabin atmosphere is subjected to the above hazards is very short relative to total mission time, the ECS will be selected to satisfy conventional operational requirements and modifications to the system will be made or precautions taken to negate the aforementioned hazards.

2. Air Conditioner Selection and Operation

The open loop air cycle machine (ACM) air conditioner was selected as the air conditioner for this aircraft. The major factors that led to this selection are as follows:

1) The ACM air conditioner possesses the lowest total penalty factor of any feasible system for conventional aircraft of comparable size, configuration, and operating regime to the NASA requirement.

2) It is amenable to the required modifications to negate the hazards of injested engine exhaust gas, dust and debris.

3) It can accommodate all the required functions of the ECS.

4) Since this type of air conditioner is extensively used in conventional aircraft of this size and operational regime, it is shown that applicable off-the-shelf units are available at a considerable saving in development time and cost.

Figure IV-42 is a schematic of the ACM air conditioner. This schematic indicates the manner in which the requirements of Paragraph 4.2.11.3(a), (b), and (c) of NASA document L-7151, Exhibit A will be fulfilled. The ACM is

operated by high pressure air which is bled from the aircraft lift/cruise engine compressors. The pressure of this air is reduced to an acceptable level as it passes through the pressure limiter. The heat of engine compression is removed from the bleed air in the ram air heat exchanger and the cooled high-pressure air is then expanded through a turbine where its temperature is further reduced to the required level. Any moisture appearing in the turbine exhaust is removed in the moisture separator (optional) before it enters the cabin. The temperature reduction across the turbine is made possible by the absorption of the turbine's work by the connected compressor which is loaded by ram air passing through the ram heat exchanger. Cabin temperature modulation is realized by mixing hot bleed air taken upstream to the ram heat exchanger with the turbine exit air, as indicated. Temperature modulated bleed air is also used for cabin transparent area defog and rain removal. Emergency ventilation is supplied by ram air, as indicated. Pure oxygen or an appropriate mixture is used metabolically by the crew for VTOL and emergency operation.

3. ECS Modifications for Crew Safety

A study was conducted to determine the extent of system modifications and precautions necessary to negate the hazards of ingested engine exhaust gases, dust, and debris. Subparagraph 9, and Table IV-6 present a summary of the results of this study. The encircled scheme represents the scheme accepted by NASA at the Part I presentation of this Design Study. In essence, the total modifications to the ECS to guard against the hazards of ingested engine exhaust gases, dust and debris consist of the following:

a) The installation of filters in the lift/cruise engine compressor bleed air lines to remove smoke, dust and debris.

b) Sealing the cabin (entails minor positive pressure relative to ambient) to assure zero leakage of smoke and dust into the compartment. (Cabin pressurization is not warranted for this aircraft).

c) The use of pure oxygen or an appropriate mixture supplied to the crew for metabolic purposes through the use of high speed helmets (see Note 1 for description of this helmet)

Note 1: Identification: Robertshaw-Fulton High Speed Helmet USAF Type HGU-15/P; Navy Type AOH-I

Note 1: (Continued)

This helmet has an internally mounted oxygen breathing system and is designed to replace the standard flight helmet, oxygen mask and oxygen mask retainer kit with a single, integrated assembly. The helmet is designed to cover the entire head and face and can be worn with standard flight clothes. It has a full face, clear visor and a tinted visor. A spray bar distributes breathing oxygen over the clear visor for defogging purposes. An operational field test of 250 units has been conducted by the U. S. Navy and a report is being prepared.

This helmet is recommended for use in this program because it protects the eyes of the crew as well as their lungs from the effects of contaminated cabin atmosphere.

4. Competitive Air Conditioners Studied

During the study that resulted in the selection of the ACM air conditioner for this application the vapor cycle system and the evaporative cooling system were analyzed and were rejected for the following major reasons:

Vapor Cycle System

a) Requires excessive power to operate the vapor compressor
b) High condenser sink temperature around the aircraft
c) Low reliability due to mechanical complexity
d) Excessive weight penalty for aircraft of this size
e) Requires a separate cabin heating, defog and rain removal system
f) Does not considerably reduce or alleviate the exhaust gas contamination problem for aircraft of this size

Evaporative Cooling System

a) High use rate of evaporant, hence excessive weight penalty
b) Unfavorable evaporant logistics
c) Requires development of an evaporator
d) Requires a separate heating, defog and rain removal system

e) Does not considerably reduce or alleviate the exhaust gas contamination problem

5. Aircraft Heat Load Analysis

The cabin heat load study is based upon aircraft Concept C cabin configuration. The design conditions were taken as specified in paragraph 4.2.11.3 of the NASA document L-7151, Exhibit A (sea level, 110°F ambient, 85°F cabin). Figure IV-43 presents the results of this study. The component heat loads were estimated through the use of applicable sections of WADC Technical Report 55-254. All external and internal cabin walls were assumed to be covered with one inch thick insulation. The solar radiation portion of the design heat load, 62 percent, as indicated on Figure IV-43, reflect the use of a larger than normal transparent area in the cabin, the use of which was necessary to satisfy the specified cabin visual requirements. This component of the heat load was calculated through the use of the following equation:

$Q_s = \tau G_t A_p$ (equation 16, page 20, WADC TR 55-254)

$\tau = 0.79$ - transmission factor (Fig. 9, page 20, WADC TR 55-254)

$G_t = 358$ Btu/hr ft^2 - total solar radiation (Fig 10, page 20, WADC TR 55-254)

$A_p = 41.13$ ft^2 - projected area of transparent surface normal to the sum at its zenith (Aircraft Concept C layout)

$Q_s = 0.79\ (358)\ (41.13) = 11,632$ Btu/hr $= 193.9$ Btu/min

One promising scheme of reducing this portion of the heat load consists of placing a shade in the top of the rear canopy and the top rear half of the forward canopy for use when the sun is at or near its zenith on a hot day. Shades so placed will not impede the horizontal (forward and side) view of the crew yet will reduce this portion of the heat load approximately 32 percent (20 percent overall).

After a study of the average specific humidities recorded throughout the United States, and particularly in the state of Virginia, the specific humidity as defined by paragraph 3.1.1 of MIL-P-18927C (133 grains/lb of dry air) was selected as the design specific humidity. Although Figure IV-42 indicates the use

of a moisture separator in the system, the use of this component cannot be determined until the specific off-the-shelf ACM air conditioner is designated.

Analysis indicates that heating of the cabin will not present any difficulties because of the magnitude of the cabin electral load and because of the use of one inch thick insulation to cover the cabin bulkheads and floors. The heating load is estimated to be 112 Btu/min at 25,000 feet altitude, minimum aircraft speed.

Electronic equipment that requires relatively close temperature control is located in the forward electronics bay and cooled with discharge air from the sealed cabin. The remainder of the equipment which comprises the bulk of the electronics gear is located in the aft electronics bay. The altitude ceiling of this aircraft and the design operating temperature of this equipment places it as category 1 (+) as defined by MIL-E-5400F. Hence, ram air will be used to cool this equipment during forward flight and forced air cooling (blower) will be used during VTOL and ground operation. The required blower is a readily available "off-the-shelf" item.

6. "Off-the-Shelf" ACM Survey

A manufacturer survey was conducted to locate suitable off-the-shelf ACM air conditioners for this application. As a result of this survey, three units were located that possess the necessary capacity. These units are presented, in the order of preference, as Table IV-5. The first choice unit (part number 682420-1) is used in the North American T-39 aircraft; the second choice unit (part number 682565-1-1) is used in the Canadair CL-41 aircraft; and the third choice unit (part number 107551) is presently undergoing qualification tests for use in the Lockheed-California produced Advanced Aerial Fire Support System (AAFSS-rigid rotor helicopter). The alternate #1 system is used if analysis indicates that the maximum engine bleed air temperature precludes the use of the T-39 and the CL-41 unit, both of which utilize aluminum plate-fin heat exchangers whereas the AAFSS unit uses a steel tubular heat exchanger. This analysis will be conducted by the air conditioner manufacturer at the time of procurement of the unit (s). All the candidate systems incorporate provisions for modulating system exit conditioned air temperature from full cooling to full heating.

TABLE IV-5 - "OFF-THE-SHELF" ACM AIR CONDITIONERS

Bleed Air Supply: GE YJ85-GE19 Engine

Ambient Temperature = 110°F, Sea Level Static**

Order of Preference	#1	#2	#3	Alt. #1
Manufacturer	Airesearch Div. Garrett Corp.	Airesearch Div. Garrett Corp.	Stratos Div. F-H Corp.	Stratos Div. F-H Corp.
Part Number	682420-1	682565-1-1	107551	107551
Units Per Aircraft	1	1	1	2
Present (P) or Scheduled (S) Use	NAA T-39(P)	Cana. CL-41(P)	AAFSS(S)	AAFSS(S)
Unit Weight-lbs	25.0	25.2	19.35	19.35
Min. Est. Power Setting at 85°F Cabin ~ %(F_N/F_{NT}*)	86.2 (.483)	88.5 (.57)	95.5 (.87)	85.75 (.47)
Lift/Cruise Engine Bleed at Min. Pwr. Setting for 85°F Cabin ~ %	.58	.45	.27	.45

*F_N = Combined Lift Thrust

F_{NT} = Max. Combined Lift Thrust

**All unit performance estimated at zero ram air

7. ACM Auxiliary Provisions

All the candidate "off-the-shelf" ACM air conditioners also possess provisions to supply conditioned air to the canopy defog and rain removal systems. The defog system will be configured on the aircraft since its effectiveness is primarily a function of specific cabin configuration and required visibility. The configuration of the rain removal system will be determined by test conducted on the transparent area of a model cabin. Figure IV-44 depicts the configuration of the ECS control panel located in the forward and aft crew stations.

8. ECS Ground Operation

An inherent characteristic of the ACM air conditioner is that cooling capacity decreases as engine power setting is reduced (reduction in compressor bleed air pressure). Hence, during system checkout at engine idle the ACM will be supplied with high pressure air from an auxiliary ground source. Analysis indicates that all the candidate off-the-shelf units will supply adequate cooling when so operated (zero ram air).

9. Additions to Existing ACM Air Conditioner

Table IV-6 presents five possible system additions to reduce or eliminate cabin atmosphere contamination due to engine exhaust gas ingestion. Each scheme can assume several possible configurations to obtain varied effects upon cabin contamination. Major advantages and disadvantages of the five presented schemes are listed below.

Scheme	Advantages	Disadvantages
1	Requires minimum cost and cabin changes to incorporate.	1) Unacceptable because it subjects the crew to the indicated elements. 2) Possibility of crew eye irritation due to ingested gases.
2	Protection from ingested exhaust gas because of the use of the high-speed helmet (e.g. USAF HGU-15/P)and a sealed and insulated cabin. The recommended minimum-standards system.	Discharges possible contaminated air directly into the cabin.
3	The rate of stored air/oxygen is minimized, making possible the use of stored oxygen/air gas.	1) Crew is required to wear ventilated suits. 2) Crew is subjected to possible contaminated air if they remove helmets.
4	1) Excellent cabin atmosphere for crew comfort. 2) Does not require high-speed helmet or oxygen mask.	1) Requires cryogenic storage because of large gas flow rates. 2) Cryogenic fluid system and evaporator expensive to develop. 3) Use of pure-oxygen cabin atmosphere hazardous.
5	Maximum crew comfort and safety. Minimum expendable use rate and system complexity. The most desirable system of the five modifications.	Requires cabin heat exchanger and humidity control assembly, and a control system.

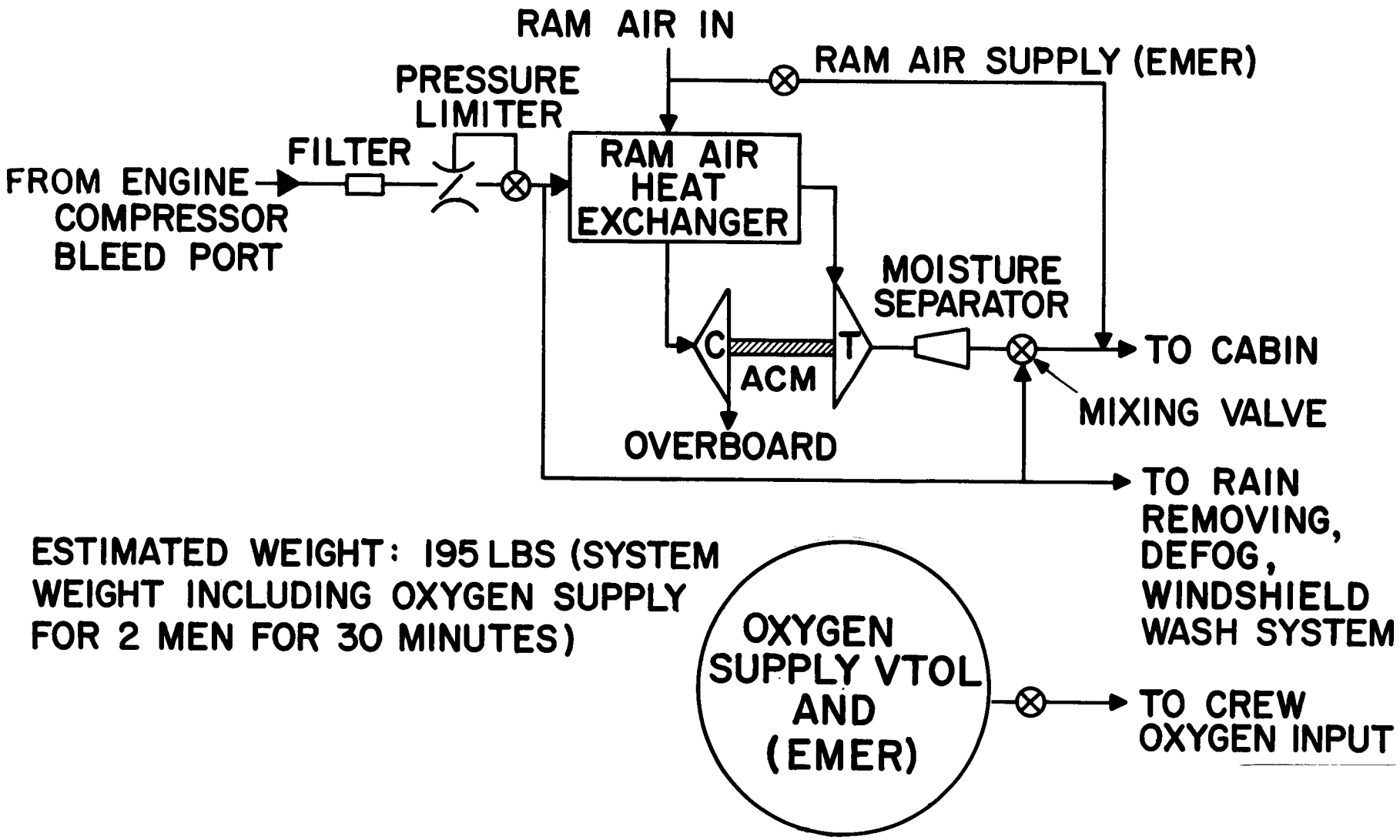
Table IV-6
COCKPIT ENVIRONMENT

ADDITIONS TO EXISTING AIR CYCLE MACHINE (ACM) AIR CONDITIONER

	CREW SAFETY	CREW COMFORT	(lbs) MODIFICATION WEIGHT PENALTY	MODIFICATION COST INCREMENT	POSSIBLE CREW EFFECTIVENESS IMPAIRMENT DUE TO: GASES	HEAT	HUMIDITY	SMOKE, DUST & DEBRIS	ADDITIONAL ENVIRONMENTAL CONTROL SYSTEM REQUIREMENTS: SEALED POSITIVE PRESSURE CABIN	INSULATED CABIN	STORED OXYGEN OR AIR	CABIN HEAT EXCHANGER ASSEMBLY	CABIN HUMIDITY CONTROL	FILTERS	COMMENTS
1 OXYGEN MASK (ACM SHUT OFF TO CABIN DURING VTOL)	POOR	POOR	2	0	✓	✓	✓	✓							OXYGEN MASK -- GFE
2 HIGH SPEED HELMET (ACM TURBINE DISCHARGES DIRECTLY INTO CABIN)	GOOD	GOOD	58	$11,000	✓		✓		✓	✓				✓	MINIMUM STANDARDS RECOMMENDED SYSTEM HIGH SPEED HELMET -- GFE
3 STORED AIR/OXYGEN VENTILATED SUIT HIGH SPEED HELMET (ACM SHUT OFF TO CABIN DURING VTOL)	EXCELLENT	FAIR	110	$10,000	✓			✓		✓	✓			✓	VENTILATED SUIT -- GFE HIGH SPEED HELMET -- GFE STORED GAS VOLUME VERY HIGH
4 STORED LIQUID OXYGEN USE OF HIGH SPEED HELMET NOT MANDATORY (ACM SHUT OFF TO CABIN DURING VTOL)	FAIR	EXCELLENT	100	$100,000					✓	✓	✓				LIQUID OXYGEN STORAGE. SYSTEM COMPLEX. PURE OXYGEN CABIN
5 CLOSED LOOP CABIN SYSTEM USE OF HIGH SPEED HELMET NOT MANDATORY (ACM SHUT OFF TO CABIN DURING VTOL)	EXCELLENT	EXCELLENT	80	$60,000					✓	✓	✓	✓	✓		MAXIMUM COMFORT, MAXIMUM SAFETY, MINIMUM USE RATE OF EXPENDABLES MOST DESIRABLE SYSTEM

AIR CONDITIONER SYSTEM SCHEMATIC

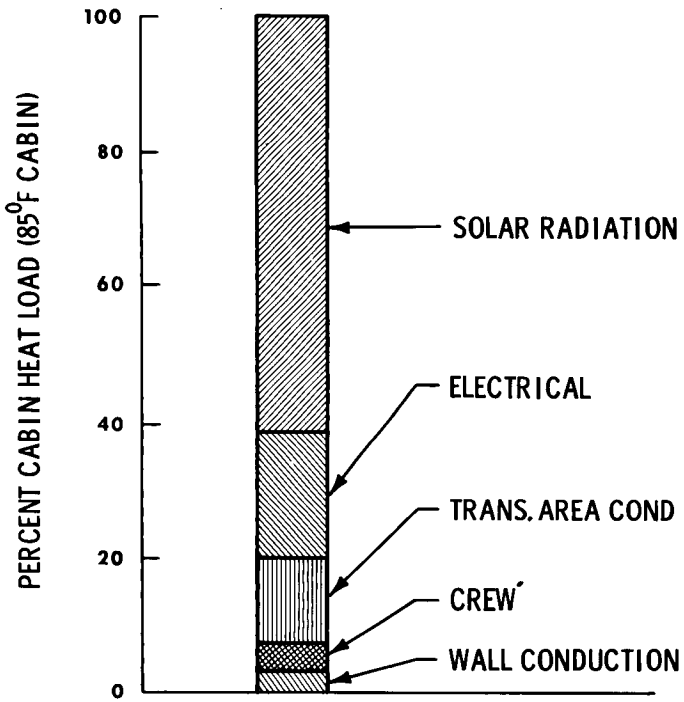
AIR CYCLE MACHINE (ACM)

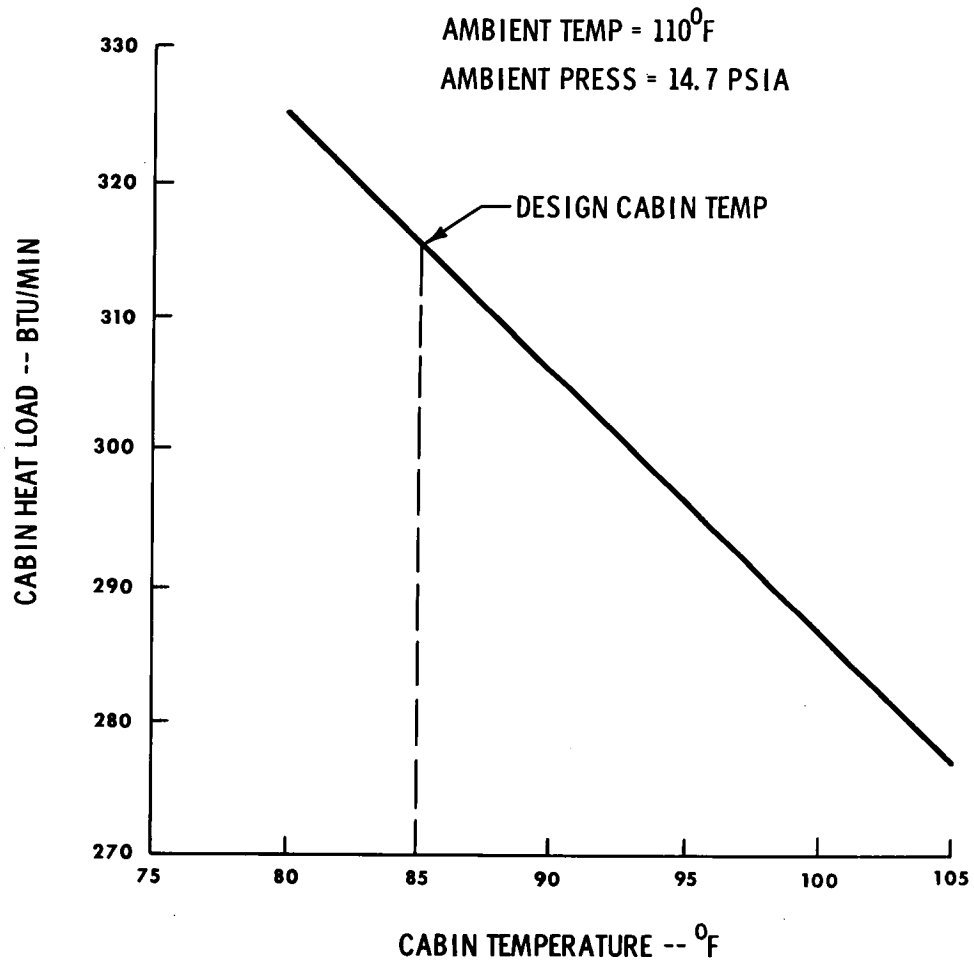


ESTIMATED WEIGHT: 195 LBS (SYSTEM WEIGHT INCLUDING OXYGEN SUPPLY FOR 2 MEN FOR 30 MINUTES)

Figure IV-42

AIR CONDITIONING REQUIREMENTS





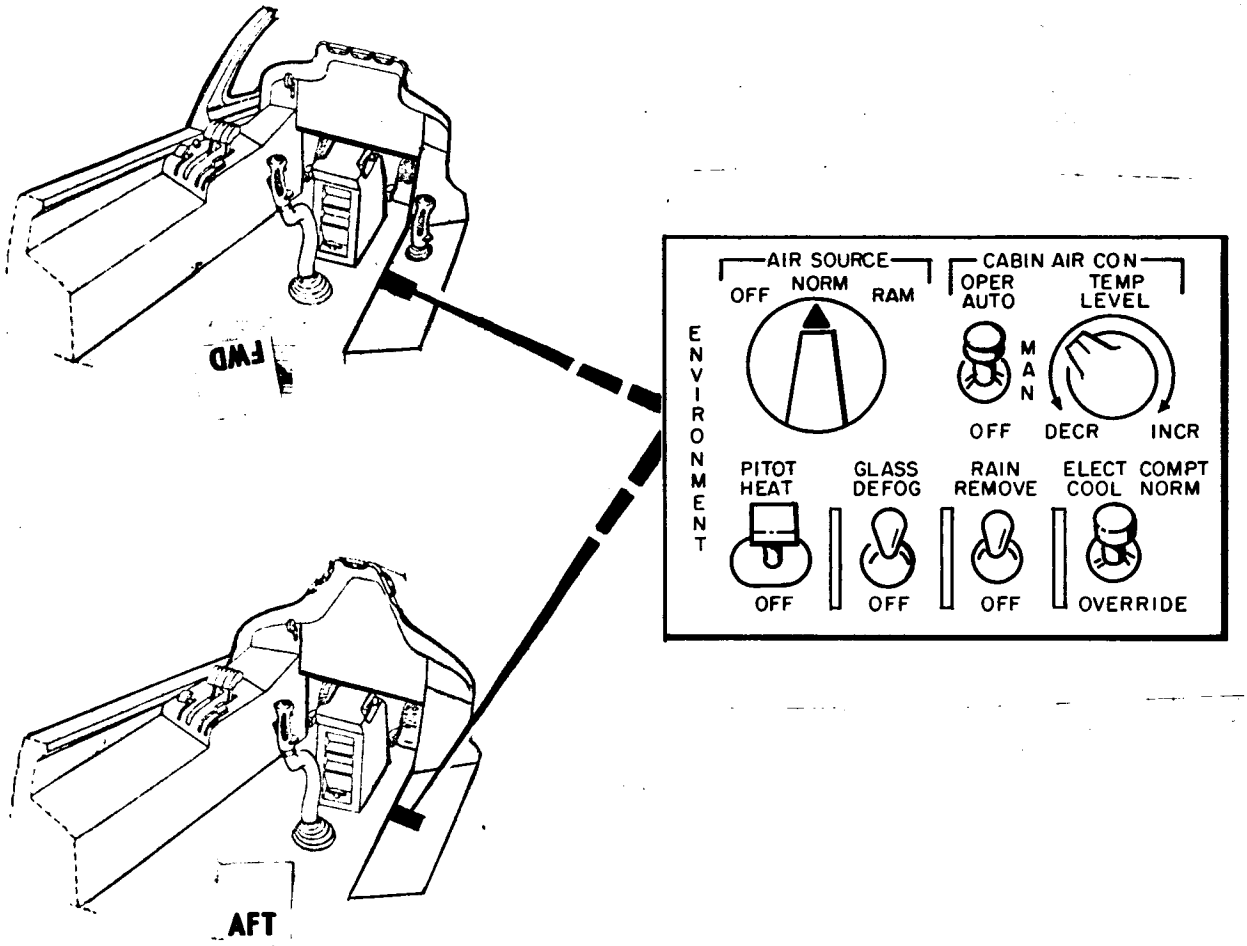

FWD
AFT
ENVIRONMENT
AIR SOURCE
OFF
NORM
RAM
CABIN AIR CON
OPER
AUTO
TEMP
LEVEL
MAN
OFF
DECR
INCR
PITOT
HEAT
GLASS
DEFOG
RAIN
REMOVE
ELECT
COOL
COMPT
NORM
OFF
OFF
OFF
OVERRIDE

Figure IV-44

E. ENGINE EXHAUST EFFECTS

1. Engine Inlet Hot Gas Ingestion

The extent to which engine exhaust gas will be inducted into the engine inlets during ground VTO operation at zero local wind velocity depends upon

1) Arrangement of the engine exhaust nozzles
2) Location of the engine inlets
3) Aircraft geometry

Critically involved in the relative locations of the exhaust nozzles is the formation of hot gas fountains which tend to flow up over the fuselage and nacelles into the engine air intakes. The nozzle arrangement of Concept C is shown in Figure IV-45 which indicates the nozzle axes spacing in terms of nozzle diameters. Though the spacings are such that fountains would likely be formed between isolated pairs of jets, the side-by-side in-line array will tend to minimize ambient air entrainment between the jets thus providing coalescence. In addition, the jets may be so canted as to encourage coalescence.

The lift engine inlets are located on top of the fuselage straddling the centerline in a region that is substantially over the wing. The latter should act as a fence in the event vertical flow by exhaust gas be induced.

The L/C side inlets are generally more susceptible to ingestion and a moderate increase in inlet temperature is anticipated. This concept has an H/D_e value of approximately one which is very close to the test point value of 1.17 in Figures 14, 15, 16, and 17 of Reference 8 upon which the inlet temperature rise predictions are based. It is estimated that the side inlet rise will be $5^{o}F$. The top inlets are not expected to suffer any inlet temperature increment under static conditions.

2. Ground Flow Fields

Experimental studies of impinging jet flow fields were described in References 9 and 10. In Figure IV-46 is presented the temperature field on the ground plane with a single vertical impinging jet (Reference 9 Figure 11) demonstrating the relative insensitivity of the H/D_E parameter (the tests were run for values of 1.1 and 17.8). Superimposed on this plot is temperature field data shown in Reference 10, Figure 8 for 3 in-line jets. It is seen that there is strong concurrence of the data. Other data from Reference 10 for varying throttle settings and an elongated nozzle (not shown in Figure IV-46) were also compared, and yielded further substantiation of the generality of the plots shown. Accordingly, the triple nozzle plot was chosen as the basis for deriving representative temperature fields for Concept C. The method of calculation is indicated below. The computed ground fields are shown in Figure IV-47.

3. Computed Estimate of Ground Flow Field

At present, insufficient data are available for estimating range of error of calculated values and, accordingly, the results are best applicable for comparing different configurations rather than providing "hard" design values. It is considered that testing with a large scale model is in order for generating the latter.

Let:

$$\theta/\theta_j = \frac{T-T_a}{T_j-T_a} = \frac{T-80}{1339-80} = \frac{T-80}{1259}$$

where:

θ/θ_j = temperature parameter in Figure IV-46 shows triple nozzle data from Reference 10, Figure 8 plotted on Reference 9, Figure 11

T = local gas temperature (^oF)

T_j = engine jet exhaust temperature = $1339\,^oF$

T_a = $80\,^oF$

From Reference 11, p. 13 and Reference 12,

$$\frac{\Delta T}{T_j\,(^oR)} = \frac{V}{V_J}$$ is taken as a coarse approximation for determining local velocities.

where:

ΔT = local gas temperature rise

T_J = nozzle jet temperature (^oR)

V = local gas velocity

V_J = nozzle exhaust velocity - 1900 ft/sec

Hence, for Concept C

R/D_J	θ/θ_J	T	$\frac{\Delta T}{T_j\,(^oR)}$	V
1	0.6	835°F	0.42	800 ft/sec
2	0.35	520	0.245	465
3	0.26	407	0.182	345
4	0.2	332	0.14	266
5	0.17	294	0.12	226

where:

R = distance from effective jet center

D_J = effective diameter of symmetrical 4 jet cluster for Concept C

The elevation profiles of the isotherms shown in Figure IV-47 are based upon scaling curves in Reference 10, Figures 2-7.

4. Tire Heating

The flow field of the ground flow gas in the immediate vicinity of the tires is not readily defined in detail. For purposes of computing heat transfer rates easily, the procedure adopted considers the heat transfer model to consist of a short cylinder immersed in a gas flowing parallel to the cylindrical axis. On this basis the heat transfer coefficients have been computed for Concept C with YJ-85 engines in the following manner:

Approximating from Figure IV-47

$$T_g = 350^\circ F = 810^\circ R, \quad V_g = 270 \text{ ft/sec.}$$

$$\rho = \frac{14.7 \times 144}{53.3 \times 810} = 0.049 \text{ lb/ft}^3$$

For air at 350°F,

$$\mu = 0.52 \times 10^{-6} \text{ lb-sec/ft}^2$$

Flow length along tire = 0.67 ft (assumed)

$$\text{Reynolds Number} = \frac{0.049 \times 270 \times 0.67}{32.2 \times 0.52 \times 10^{-6}} = 5.3 \times 10^5$$

For turbulent flow parallel to cylinder axis with a Reynold's number smaller than 10^7, the following equation may be employed:

$$\frac{h}{3600\rho \, VC} = \frac{0.0296}{(R.N.)^{0.2} P_R^{2/3}}$$

this reduces for air to the simpler expression:

$$h = \frac{0.0212}{T^{.5}} \quad \frac{(PV)^{.8}}{X^{.2}}$$ (Ref. 13 sec. 1, page 1.C-61)

where:

X = ft

h = BTU/hr-ft^2-°F

T = °R

P = lb/ft^2

V = ft/sec

Substituting in the above

$$h = 32.4 \text{ BTU/hr-ft}^2\text{-}^\circ\text{F}$$

The tire is also assumed to have a rubber with nylon 1 inch thick carcass having the following thermo-physical properties:

Thermal Conductivity $K = 1.33 \text{ BTU-in/hr-ft}^2\text{-}^\circ\text{F}$

Specific Heat $= 0.5 \text{ BTU/lb-}^\circ\text{F}$

Density $= 70 \text{ lbs/ft}^3$

Assuming the inside surface of the tire to be adiabatic, the temperature time response of the carcass was computed based upon the above data employing Republic's Thermal Analyzer IBM Digital Program 61H605. The resultant temperatures are shown in Figure IV-48. The carcass outer surface rises to 320°F in 5 minutes at which time the inside surface has reached 210°F. We have been advised by aircraft tire manufacturers that such temperature histories pose no significant problem to the tires. In addition, the ground flow field temperatures and velocities in the study were based upon full engine throttle settings with all jets vertical. This condition can, in reality, only persist for a very short time before the aircraft leaves the ground. Accordingly, under normally anticipated circumstances, the tires would be in the ground field for a matter of seconds under which conditions the outside surface might not much exceed 200°F. It is, therefore, anticipated that there would be no tire overheat problem with the Concept C configuration.

5. Skin External Heating

Rising hot gas fountains induced by the collision of ground gas flow between adjacent vertical nozzles can be expected to flow up around the fuselage and under the wings. This high temperature, high velocity gas will heat the vehicle skin and will ultimately result in skin temperatures above

the limits established by structural strength considerations. The heating effect persists so long as the vehicle is on or close to the ground with engine operation at high throttle setting and the jets vertical.

Accordingly, it may be necessary to limit the time in this operating mode so that excessive skin temperatures are avoided. In Reference 11, Figure 19, are shown isotherms ($\Delta T/T_j$) around fuselage, landing gear, and wing of a model VTOL aircraft determined through appropriate temperature measurements made during a simulated full throttle (jet temperature = 1260°F) " on the ground" condition. The model has two separated vertical nozzles whose exit planes are approximately a height of h/De = 2 above the ground plane. The Concept "C" aircraft can be expected to suffer a more severe skin heating environment since its h/De is a little greater than one, and has 8 vertical nozzles. The Reference 11 results show a value of $\Delta T/T_j$ = .20 (ΔT = local gas temperature increment over ambient and T_j = jet absolute temperature) following the contour of the fuselage upward and along the wing root lower surface. Close to the ground in the vicinity of the jets a $\Delta T/T_j$ of 0.4 was measured. In view of the relatively low h/De value of Concept "C" and the large number of nozzles in close proximity, it is felt this latter value of $\Delta T/T_j$ is more likely to occur along the fuselage in the region of the lift engines in Concept "C". Accordingly, the lower side fuselage skin and bottom wing root surfaces can be bathed in 800°F local gas temperatures (based on T_j = 1339°F) at gas velocities (it is assumed V_j = 1900 ft/sec and that $V/V_j \approx \Delta T/T_j$,(see References 11 and 12) of approximately 760 ft/sec.

The time temperature response of a 0.06 inch thick aluminum skin based on such heating conditions, is shown in Figure IV-49). (800° curve) in addition to responses for other local gas temperatures included in the parametric study (each according to its corresponding local gas velocity). It may be seen that for the anticipated 800°F local gas temperature the skin

(considering the back surface to be adiabatic) will reach its limit temperature of 350°F in approximately 20 seconds. Under similar heating conditions, a 0.03 inch thick titanium skin would reach a limit of 500°F in 18 seconds (not shown in Figure IV-49).

The time for the skin to reach limiting temperature may be prolonged by applying insulation to the heated surface. Figure IV-50 shows the skin temperature-time response with a protective layer of 3/16 inch Johns Manville Microfiber insulation with a 0.015 inch steel sheet cover. It is seen that for 800°F local gas temperature it will take well over two minutes for the skin to reach 350°F.

6. Internal Structural Heating

It is highly desirable that the Concept "C" aircraft be capable of operating out of ground effect (OGE) hover mode so long as is desired with no limitation other than the fuel capacity. To achieve this design goal, thermal protection is to be applied where necessary to maintain acceptable temperatures throughout the aircraft considering the OGE hover as a quasi-steady state thermal condition.

In the vicinity of the engines, local structure receives heat from the hot engine surfaces, and the basic thermal design problem is to provide insulation, shielding, and cooling air flow required to maintain tractable temperature levels.

The analysis has been based upon the maximum YJ-85 engine surface temperatures quoted by the manufacturer and are as follows:

Aft End of Compressor	500°F
Combustion Chamber	850°
Turbine	1200°
Exhaust Cone	1300°

The initial investigation concerned itself with what appeared to be the most severe heating area, i.e. the structure between the side by side pairs of lift engines. This structure having heat sources on both sides can be expected to reach the highest temperature levels (aside from the portions of the bottom of the fuselage which can be bathed in exhaust gases close to full jet temperatures and must consequently be steel). It may be readily appreciated that with no heat sink, this structure would, ultimately, under steady state conditions reach the temperature of the adjacent heat source, regardless of insulation provisions. Accordingly, it is necessary that the jet exhaust region be designed to induce a cooling airflow between the engine and structure through ejector action. This airflow would enter the upper region of the engine compartments through louvers and flow downward over engine and structure to be discharged with the engine exhaust.

It was quickly ascertained that cooling airflow alone is inadequate to achieve required temperatures. A cooling airflow equal to 1% of engine flow could not bring "between engine" structure below 700°F in the combustion region or below 1000°F in the turbine and exhaust cone regions. It is therefore essential that some form of insulation be provided in addition to the cooling airflow.

Two insulating techniques were examined. The first is that of applying an insulating blanket directly on the engine; the second is installing an insulated shroud in the annular space between the engine structure with the cooling airflow split equally between the inside and outside of the shroud.

Analysis confirmed previous experience in showing that a particular structure temperature level can be maintained by means of a shroud using only approximately half the insulation thickness required when the insulation

is applied directly to the engine skin. For example, a one inch thick blanket of Johns Manville Min-K insulation on the engine exhaust cone with 3/4% engine cooling air results in a 390°F temperature on the adjacent structure whereas a 1/2" Min-K shroud (contained within two very thin steel foil sandwiching layers) results in a temperature of 360°F.

In the present design a titanium structure is employed which permits higher temperatures and it is recommended that a shroud arrangement with approximately 3/8 inch thick Min-K and 3/4% total cooling air per engine be employed which will result in about 500°F on the structure between engines in the region of the exhaust cones. The shroud will extend from the aft end of the compressor to the region where exhaust cone transitions to the exhaust nozzle. In the immediate vicinity of the nozzles, there is not sufficient room for a complete shroud between engine and structure, and special local application of insulation is necessary.

The vehicle skin and associated structure on the outboard side of the engine compartments enjoys the assistance of ambient air natural convection cooling when the aircraft is hovering and free of ground effect. This results in lower temperatures than those determined for the structure on the inboard side of the engine. Other regions of structural heating which have been examined are between lift engine compartments and cruise engine tail pipe, and the route of the cruise engine diverter ducts. The final selection of design temperatures are indicated in the structural design section of this report.

7. References

8. H. C. McClemore and C. C. Smith, Jr., LWP-332, "Hot-Gas Ingestion of Large-Scale Jet VTOL Fighter-Type Models"

9. W. A. Abbott, N.G.T.E.: M.391, "Studies of Flow Fields Created by Vertical and Inclined Jets When Stationary or Moving Over a Horizontal Surface"

10. D. Reeves, N.G.T.E.: M.176, "Concerning the Heating Effects of the Jet Exhaust Gases from Vertical Take-Off Aircraft"

11. R. F. Speth and P. E. Ryan, AIAA Paper 66-740, "A Generalized Experimental Study of Inlet Temperature Rise and Near Field Temperature Contours of Jet V/STOL Aircraft in Ground Proximity"

12. H. B. Squire, Aircraft Engineering, March 1950, "Jet Flow and Its Effects on Aircraft"

13. Aero-Space Applied Thermodynamics Manual, Society of Automotive Engineers

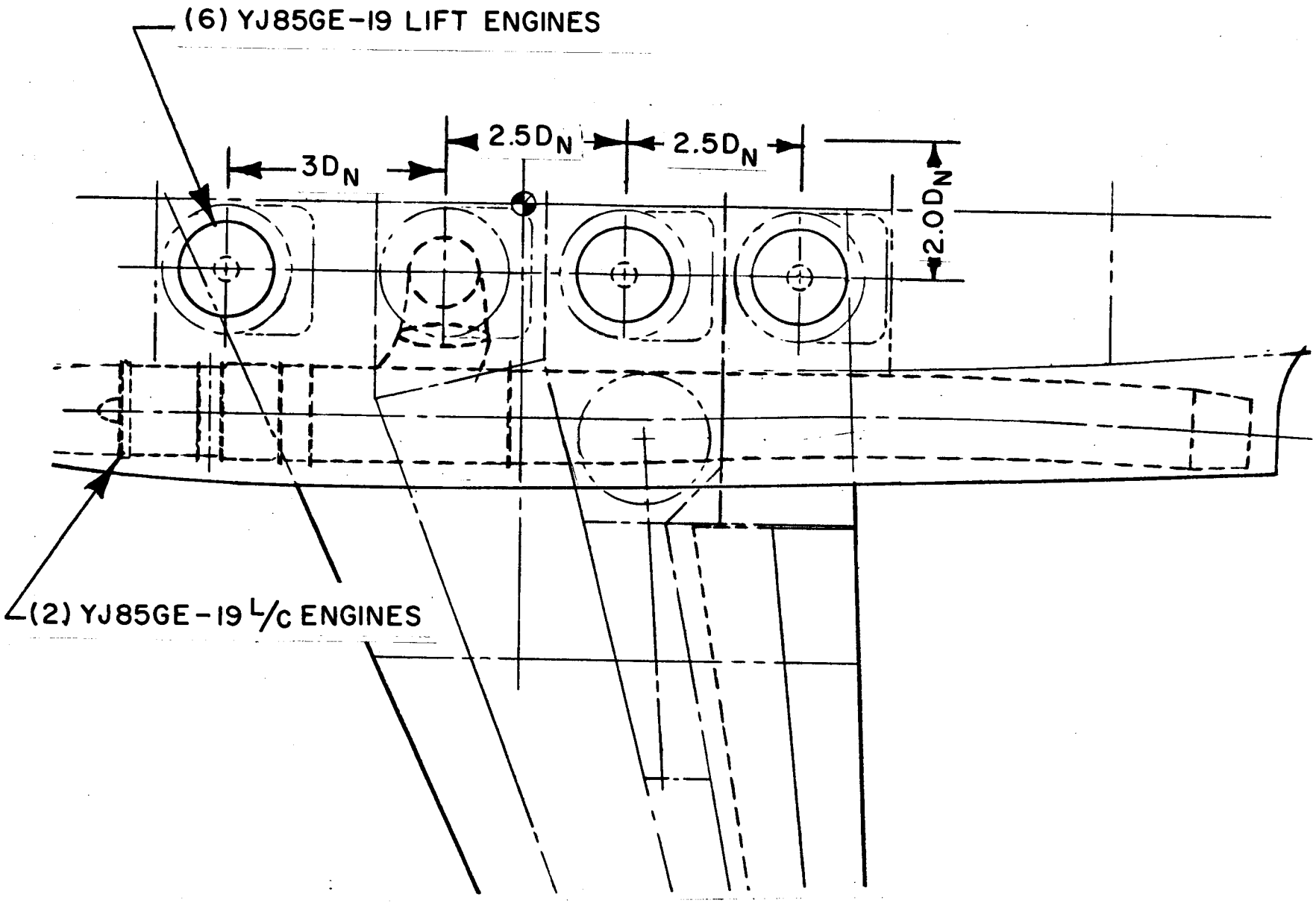


Figure IV-45 Nozzle Arrangement, Concept C

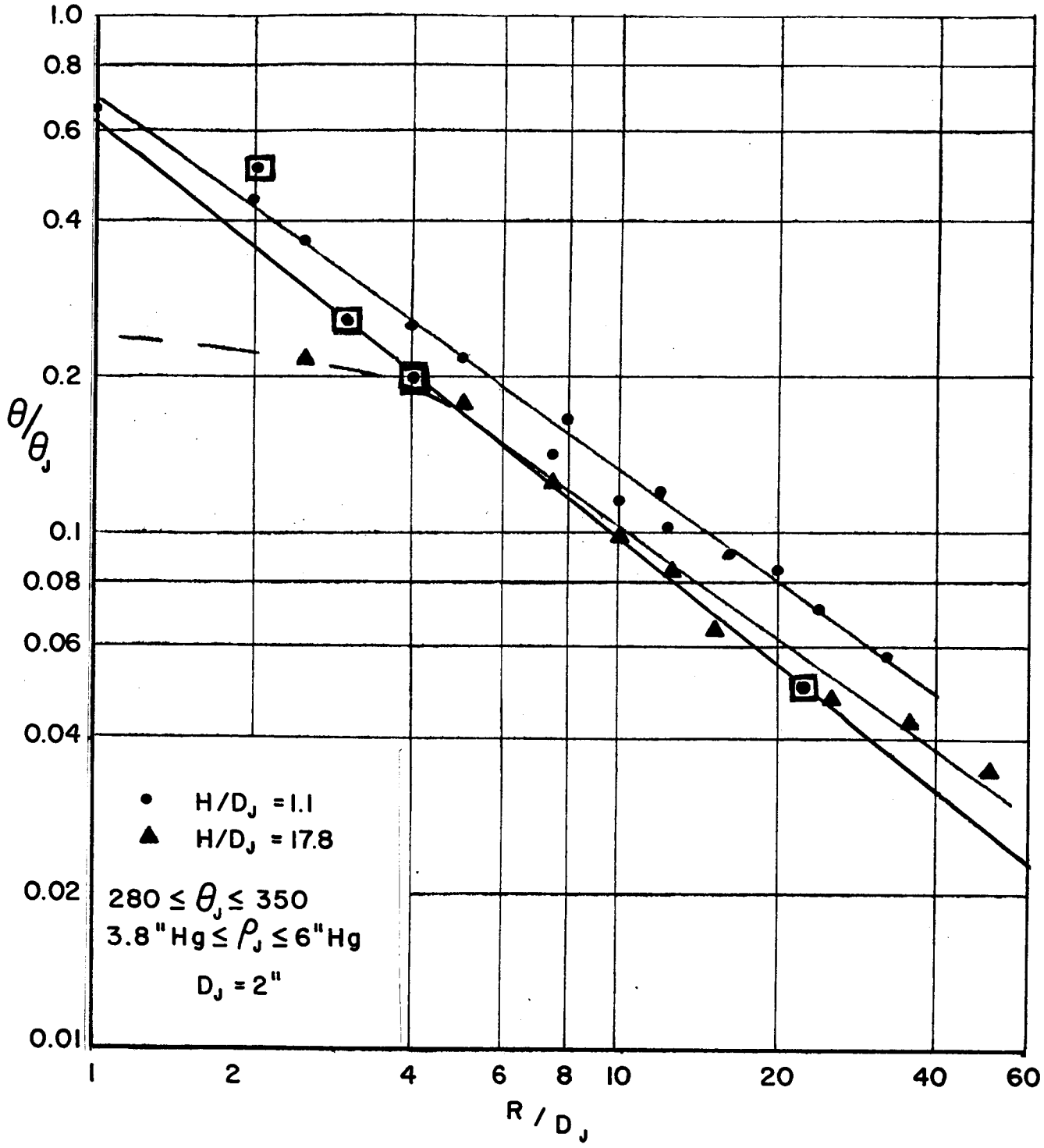


Figure IV-46 Decay of Maximum Temperature Effect of Nozzle Height Single Nozzle Vertical
(Reproduced from Figure 11 of Reference 9. Data for in-line Triple Nozzle from Figure 8 of Reference 10. Added and indicated by ⊡ .)

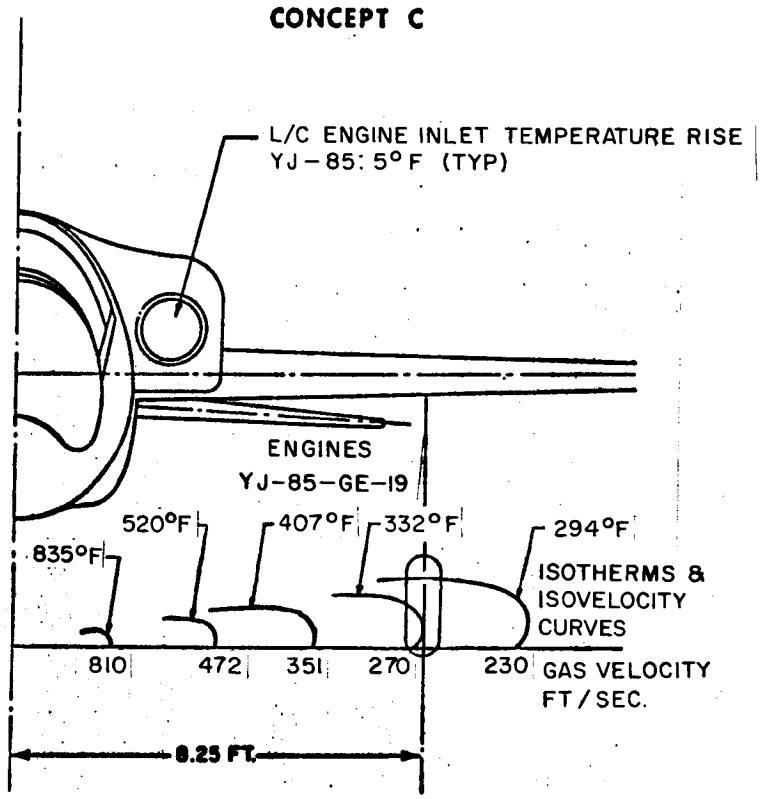


Figure IV-47 · **Ground Temperature and Velocity Environment**

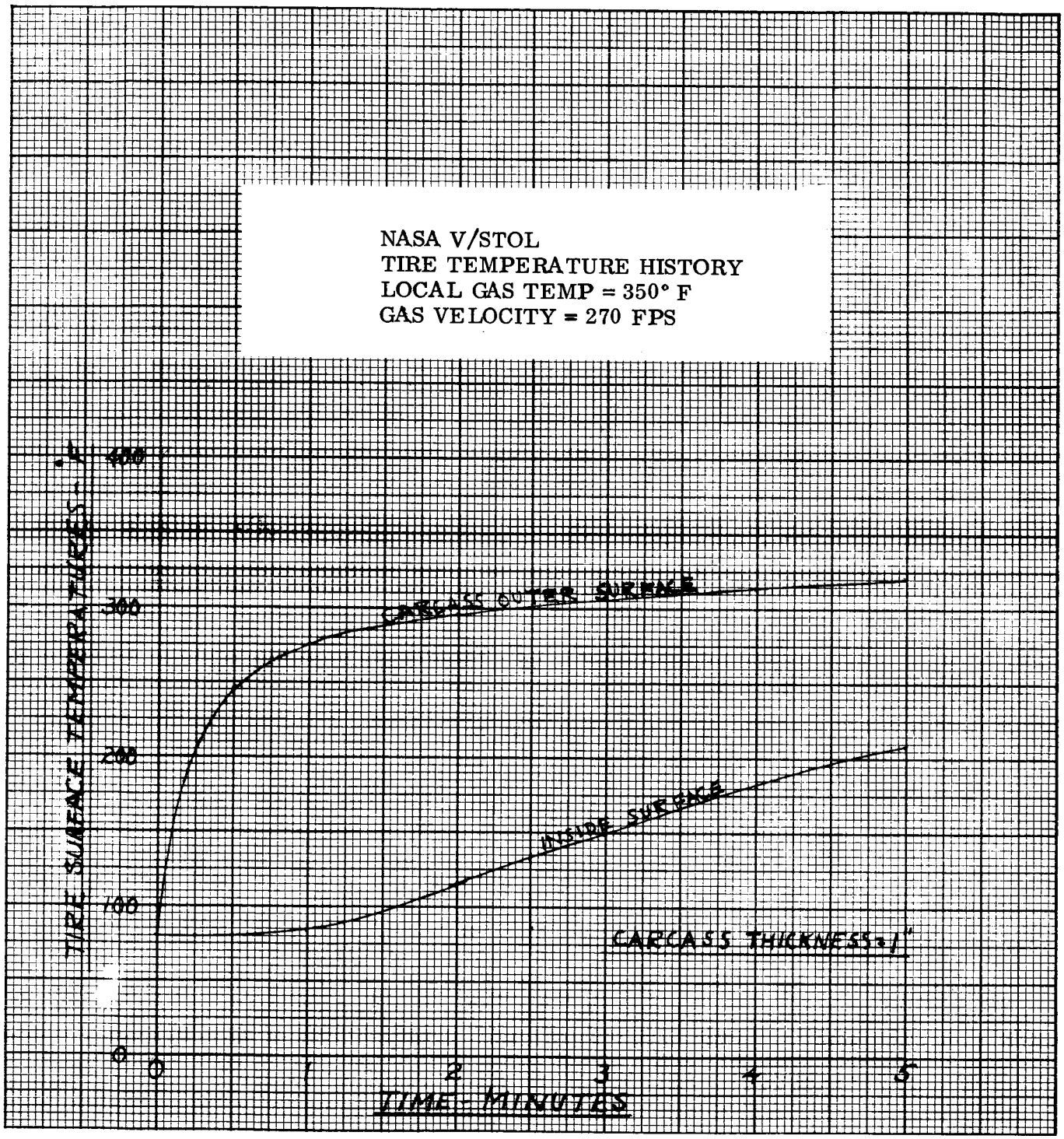

NASA V/STOL
TIRE TEMPERATURE HISTORY
LOCAL GAS TEMP = 350° F
GAS VELOCITY = 270 FPS
TIRE SURFACE TEMPERATURES - °F
400
300
200
100
0
CARCASS OUTER SURFACE
INSIDE SURFACE
CARCASS THICKNESS = 1"
0
1
2
3
4
5
TIME - MINUTES

Figure IV-48

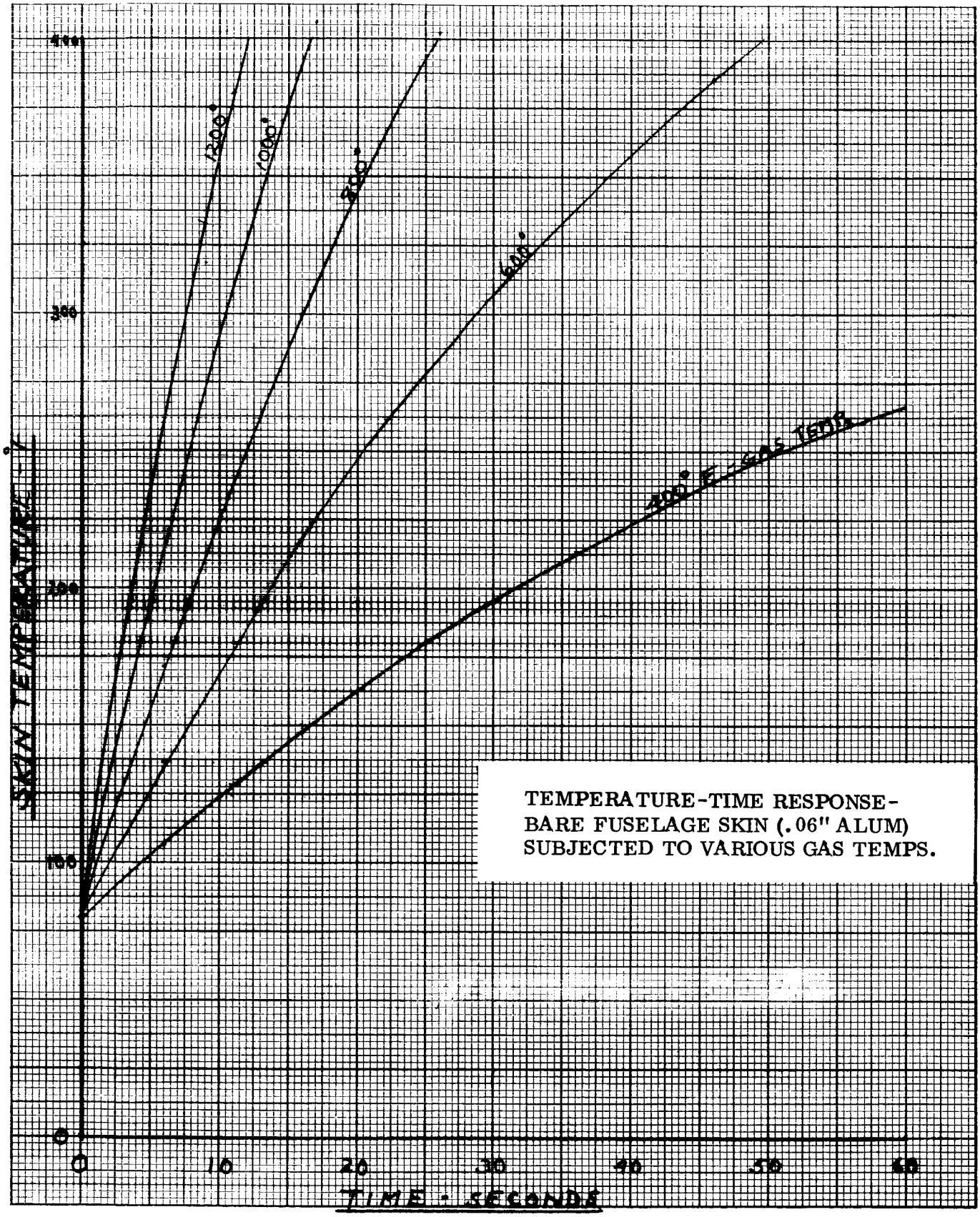

TEMPERATURE-TIME RESPONSE-
BARE FUSELAGE SKIN (.06" ALUM)
SUBJECTED TO VARIOUS GAS TEMPS.
1200°
1000°
800°
600°
400° F - GAS TEMPS
SKIN TEMPERATURE °F
400
300
200
100
0
0
10
20
30
40
50
60
TIME - SECONDS

Figure IV-49

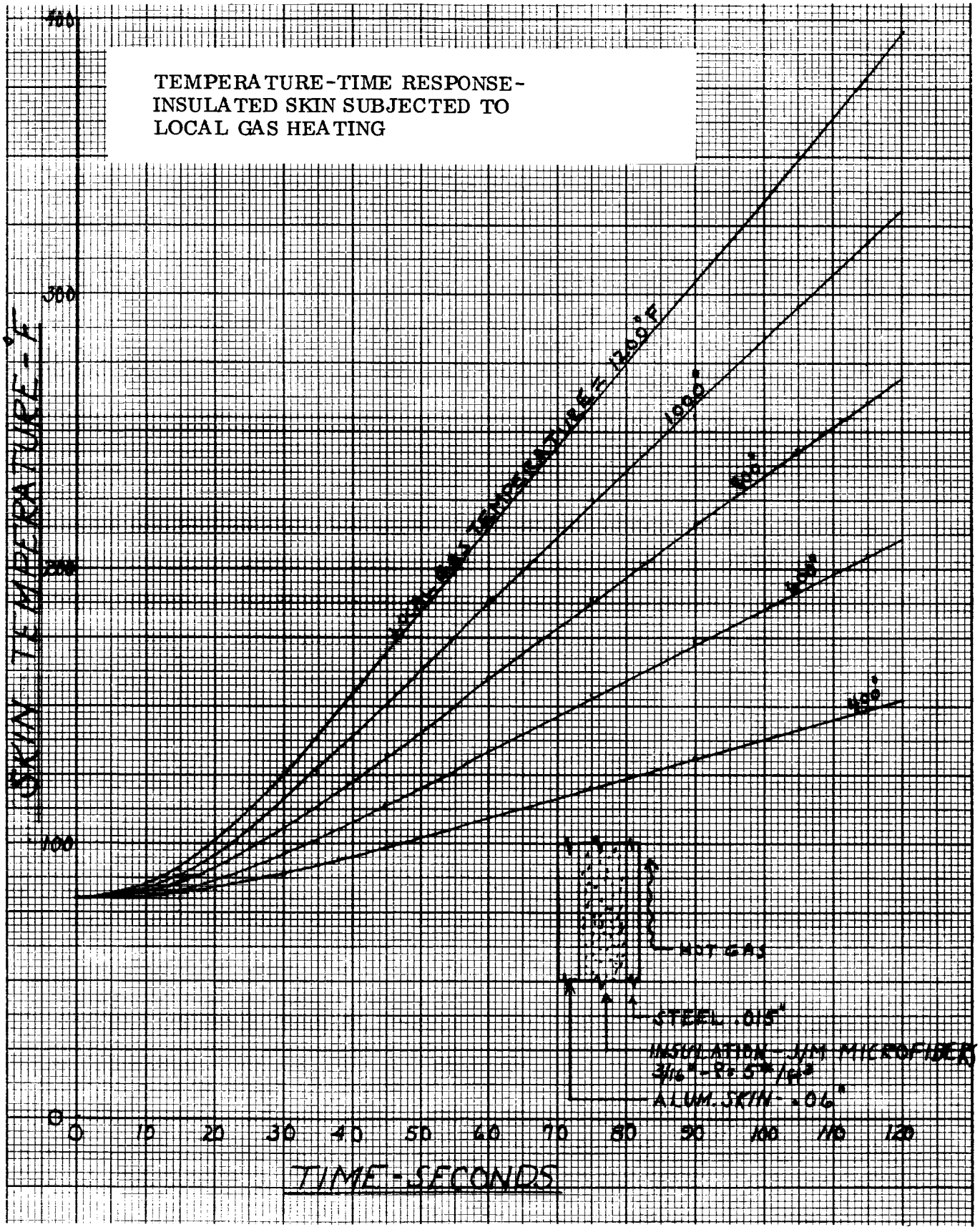

TEMPERATURE-TIME RESPONSE-
INSULATED SKIN SUBJECTED TO
LOCAL GAS HEATING
SKIN TEMPERATURE - °F
TIME - SECONDS
HOT GAS
STEEL .015"
INSULATION - 1/4" MICROFIBER
3/16" - 8.5 #/ft³
ALUM. SKIN - .06"

Figure IV-50

F. LIFT ENGINE AIR START

This study is directed toward satisfying NASA requirements for a simultaneous air start of all lift engines quoted in paragraph 4.2.4.3(b) of the Reference 14 document.

The first objective was to determine the minimum airspeed required to windmill the lift engine to the desired RPM for ignition. Utilizing the data presented in the Reference 15 and 16 documents, in addition to the installation factors associated with a lift engine scoop door(s) type installation, Figure IV-51 was generated. It is shown that a minimum equivalent airspeed of 225 knots is required in order to achieve a ram air start. Superimposed on Figure IV-51 is NASA test data extracted from the Ref. 4 document. This data indicates a somewhat higher windmill RPM can be achieved which is the equivalent of from 10 knots to 35 knots lower airspeed requirement for a ram air start. It is quite obvious that the YJ85GE-19 engine cannot be ram air started at the NASA required airspeeds (approx. 120 kts, flaps down).

Since the YJ85GE-19 engine is equipped with an air port for turbine impingement starting purposes, a study was undertaken utilizing the cruise engines' compressor bleed air as a supplementary source to ram air. The first step was to determine the supplementary bleed air horsepower required to achieve a windmill start (12% RPM). This was calculated based on G.E. requirement of 1.67 lb/sec flow at 43 psia and 360°F quoted in paragraph 3.4.16.1 of the Reference 16 document in addition to the ram air horsepower available as shown on Figure IV-51. Utilizing the bleed air data presented in Reference 16, available bleed air horsepower was calculated at varying air speeds and engine power settings. Figure IV-52 presents a plot of available and required bleed air horsepower. The intersection of horsepower available and required will result in a satisfactory windmill start. Figure IV-53 is a plot of the intersecting horsepower lines of Figure IV-52 and is presented as a plot of thrust available from two (2) YJ85GE-19 cruise engine vs. minimum airspeed required for a satisfactory simultaneous start of 4, 6, 7 and 8 YJ85GE-19 lift engines.

Figure IV-54 presents the airspeeds and corresponding airplane drag values at various airplane gross weights for both the flaps up and approach flaps positions (dashed lines). Since a minimum airspeed of 225 knots is required to ram air start the YJ85GE-19 engine, this method alone will not satisfy the NASA requirements.

A simultaneous air start of the six YJ85GE-19 lift engines can be achieved by utilizing the bleed air of the two lift-cruise engines for lift engine turbine impingement as an assist to ram air. At reduced cruise thrust levels, bleed air horsepower can be maximized by presetting the lift-cruise engine diverter valve in an intermediate position (thrust decay). The solid lines represent the minimum airspeed requirement with bleed air assist for both normal and thrust decay lift-cruise engine operation. Since the desired airspeed (dashed lines) for a simultaneous air start is above the minimum airspeed requirement (solid lines), a simultaneous air start can be achieved by Concept C.

A brief review was made of the information contained in the Reference 18 document pertaining to Continental Aviation's starting system design study. Although the conclusion reached in the subject report states "Simultaneous starting of multiple lift engines can be satisfactorily achieved with systems utilizing either bleed air or hydraulic power provided by the main propulsion engines," this does not appear to be the case for our specific concepts. Continental's study assumed that two (2) main propulsion engines were used to start a bank of five (5) lift engines whereas our concepts utilize either six (6) or eight (8). In addition Continental assumed that the main propulsion engines delivered air at a 12:1 pressure ratio whereas the YJ85GE-19 engines deliver air at approximately 6:1. Other systems such as cartridge turbine drive can achieve the simultaneous start; however, these are both heavy and costly.

It is recommended that the ram air and bleed air impingement starting system be utilized for the NASA V/STOL vehicle.

References

14. NASA Statement of Work L-7151 dated 6-13-66 - Addendum Number 1 dated 8-9-66
15. G. E. CJ610 Installation Manual
16. G. E. Model Specification E1129 - YJ85GE-19 dated 11-1-66
17. NASA Ames YJ85-5 Windmill RPM Test - NASA letter dated 1-31-67, Mr. David H. Hickey to Mr. Elliot Schiller
18. Continental Aviation Report No. 967 dated 6/65 - Lift Engine Installation Technology (Section IV - Starting Systems Design Study)

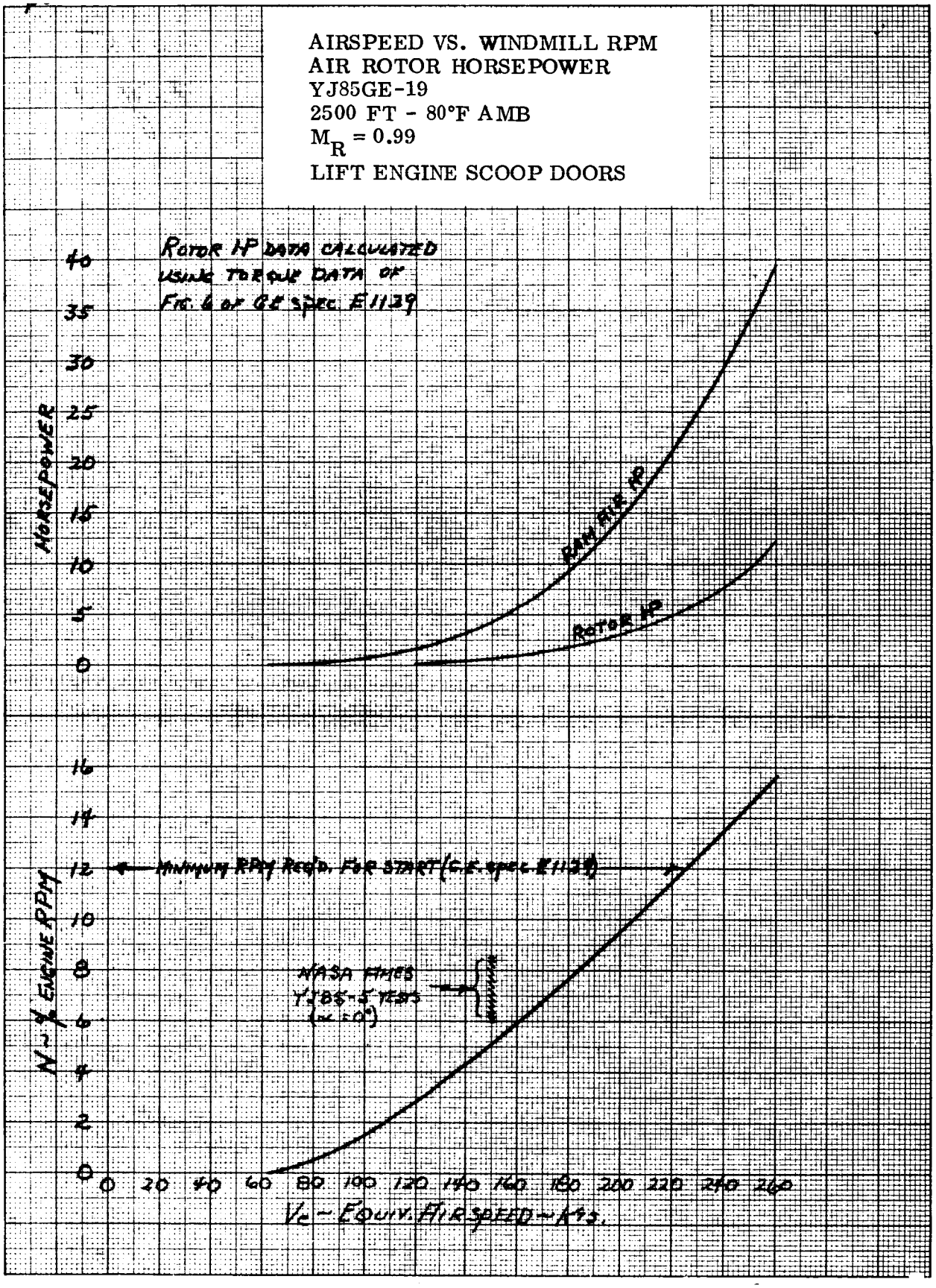

AIRSPEED VS. WINDMILL RPM
AIR ROTOR HORSEPOWER
YJ85GE-19
2500 FT - 80°F AMB
M_R = 0.99
LIFT ENGINE SCOOP DOORS
ROTOR HP DATA CALCULATED USING TORQUE DATA OF FIG. 6 OF GE SPEC. E1139
HORSEPOWER
RAM AIR HP
ROTOR HP
MINIMUM RPM REQ'D. FOR START (G.E. SPEC. E1139)
NASA FIRES YJ85-5 TESTS
N ~ % ENGINE RPM
0 20 40 60 80 100 120 140 160 180 200 220 240 260
V_e ~ EQUIV. AIRSPEED ~ KTS.

Figure IV-51

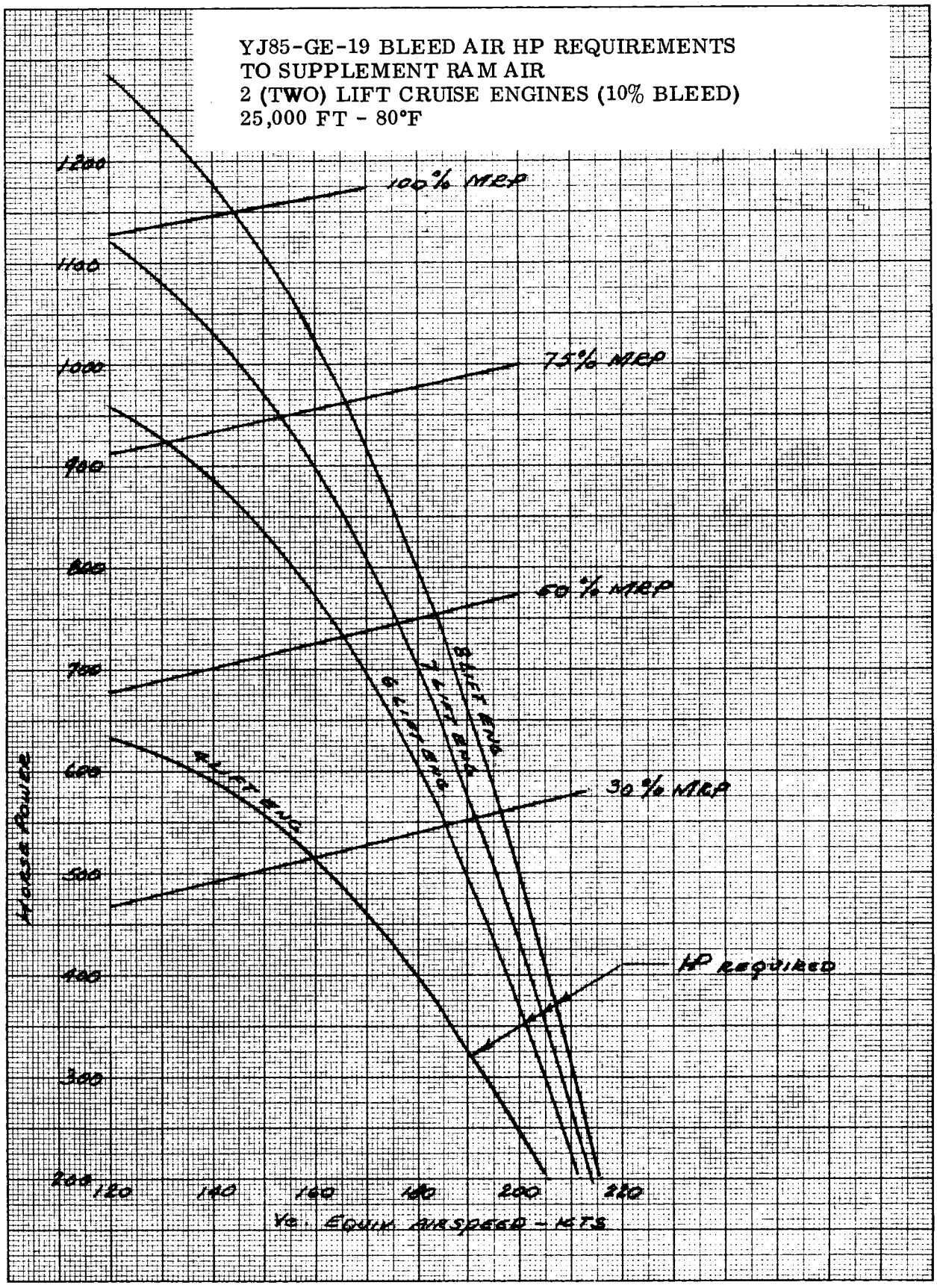

YJ85-GE-19 BLEED AIR HP REQUIREMENTS
TO SUPPLEMENT RAM AIR
2 (TWO) LIFT CRUISE ENGINES (10% BLEED)
25,000 FT - 80°F
100% MRP
75% MRP
60% MRP
30% MRP
HP REQUIRED
LIFT FAN
LIFT FAN
LIFT FAN
LIFT FAN
HORSE POWER
1200
1100
1000
900
800
700
600
500
400
300
200
120
140
160
180
200
220
Vo. EQUIV AIRSPEED - KTS

Figure IV-52

YJ85GE-19 THRUST VS MIN. AIR SPEED
FOR WINDMILL START OF YJ85GE-19
LIFT ENGINES

NOTE:

1) 2 YJ85GE-19 LIFT CRUISE ENGINES USING 10% BLEED FOR IMPINGEMENT START
2) SCOOP DOORS FOR LIFT ENGINES
3) 2500 FT ~ 80°F

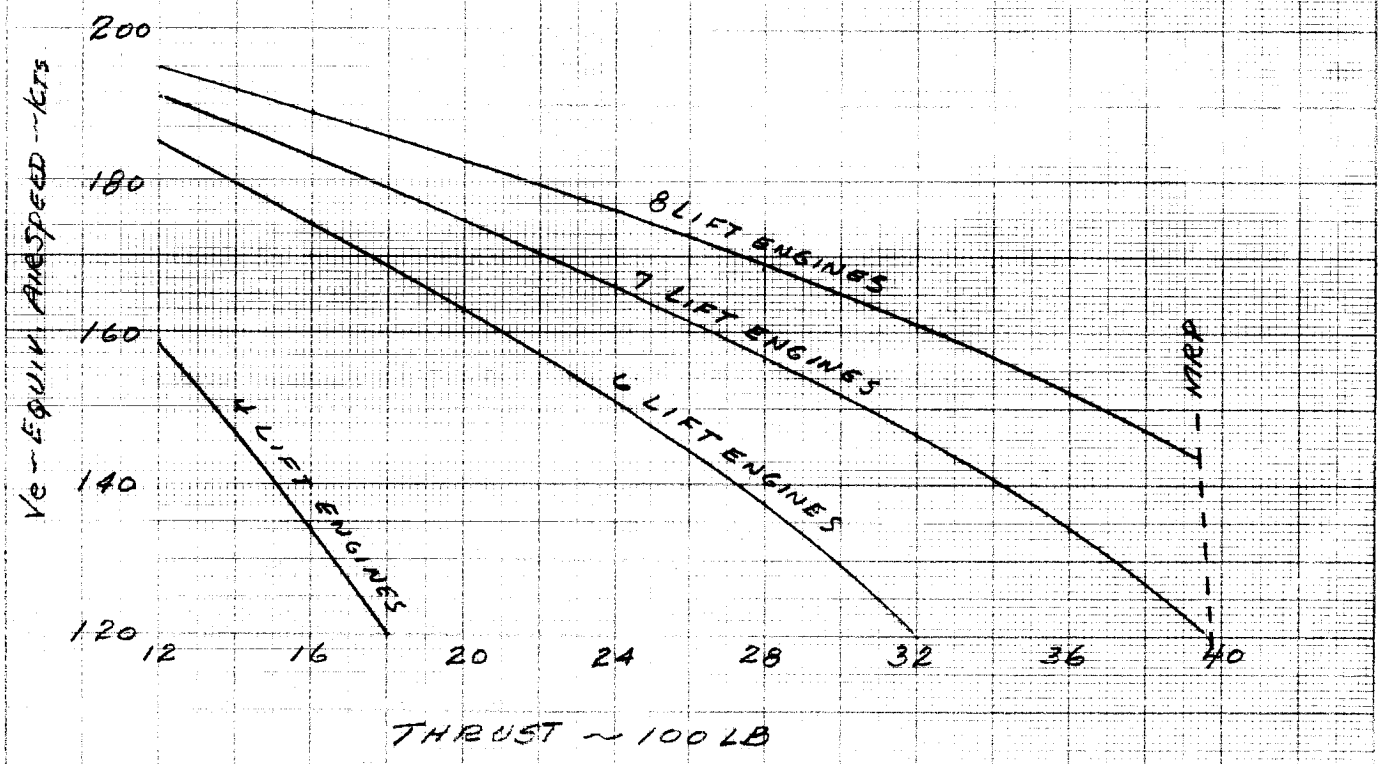
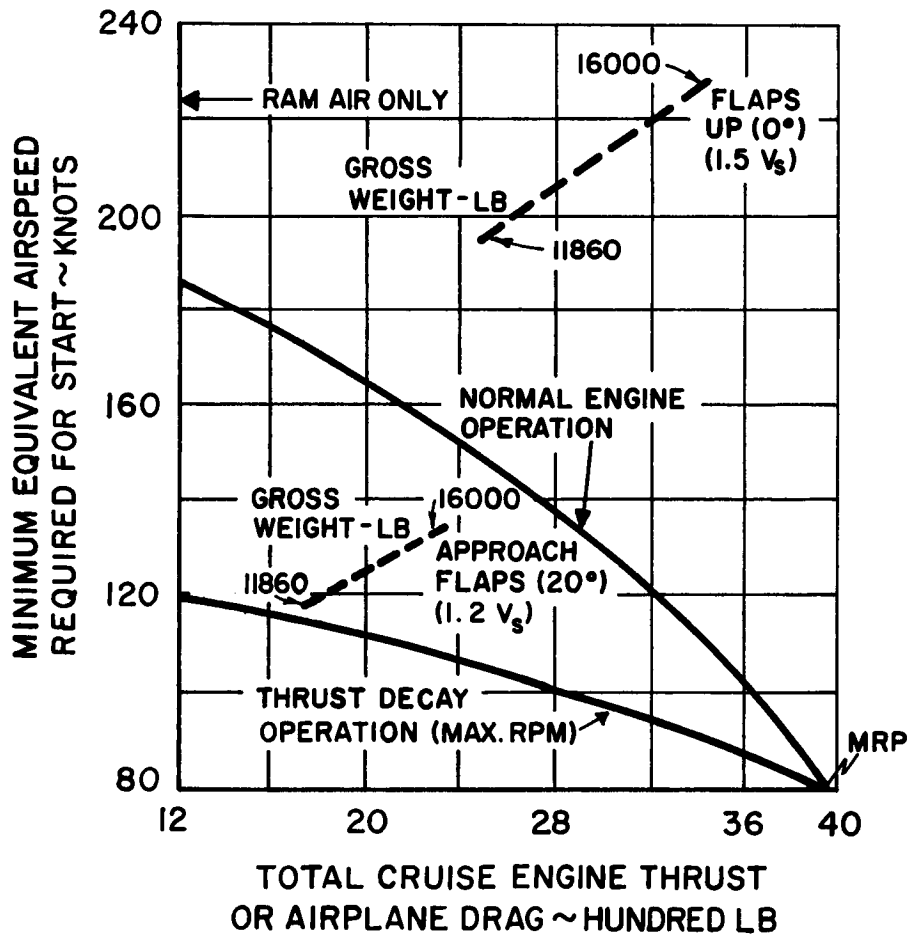


Figure IV-53

LIFT ENGINE AIR START CONCEPT C



NOTES:

1. 2 YJ85GE-19 LIFT-CRUISE ENGINES USING 10% BLEED FOR IMPINGEMENT START
2. SCOOP DOORS FOR LIFT ENGINES
3. 2500 FT ~ 80^0 F AMBIENT

——— MINIMUM AIRSPEED REQ'D

- - - - - AIRPLANE DRAG

Figure IV-54

G. FUEL SYSTEM

The fuel system selected is shown in Figure IV-55 and is designed for use of JP-4 or JP-5 fuel. The fuel is contained in two fuselage bladder tanks of approximately equal capacity located symmetrically about the aircraft center-of-gravity. Each tank independently supplies fuel to three lift engines and one lift-cruise engine. No fuel transfer system is employed. A double-ended electrically driven pump (in a negative "G" sump) and a hydraulically driven pump, each capable of delivering the tank fuel requirements, are installed in each tank. A pressure-sensing differential crossfeed valve is installed between the two engine feed systems. Therefore, one pump inoperative in each tank or two pumps inoperative in one tank will not adversely affect mission accomplishment. Suction feed is provided through the pumps.

C. G. control will be maintained by the nearly equal fuel consumption of the engines in the symmetrical feed system grouping as shown in the schematic. A tank interconnection system is provided to equal tank levels in the event of a malfunction condition. Each engine has a fuel shut-off valve, and each feed line contains a fuel filter with emergency by-pass and contamination indication. Pressure switches are installed at the pumps and in each feed line to provide malfunction indication.

Each tank is refueled through a gravity fill port and de-fueled through the emergency balance system. A vent line, quantity gaging system and independent low level indication are provided in each tank.

The use of polyurethane open-cell reticulated foam is presently being considered for the tanks to reduce slosh, enhance crash resistance, and provide tank inertion.

FUEL SYSTEM SCHEMATIC-CONCEPT C

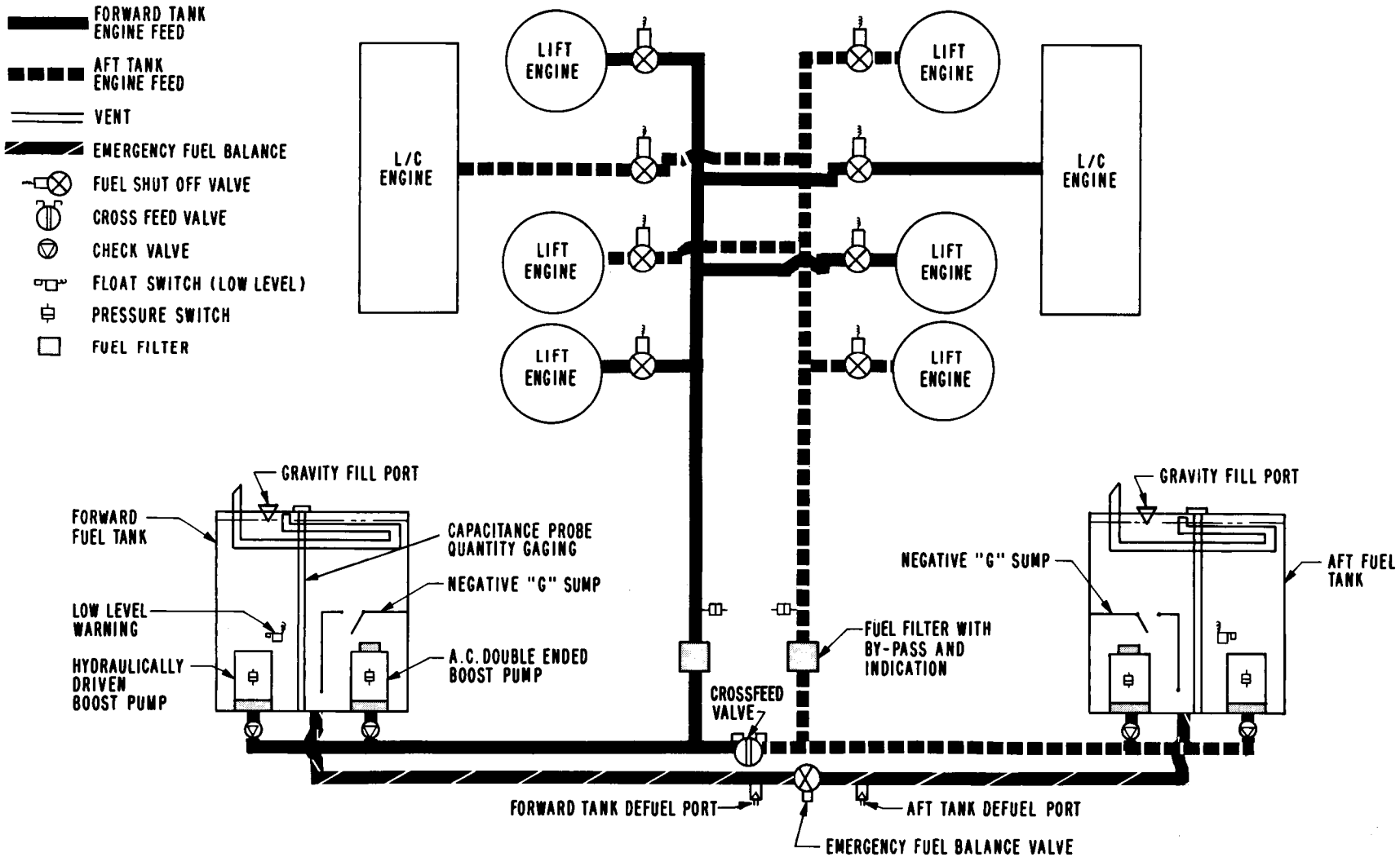


Figure IV-55

H. THRUST REVERSER INSTALLATION

Paragraph 5.1.2(f) and 4.2.4.2 of the NASA V/STOL work statement indicates the desirability of a fully modulating thrust reverser for thrust vectoring when operating as a pure lift-engine configuration.

A review of thrust reverser test data obtained on several aircraft installations (Ref. on Figure IV-56) revealed that external target and tailpipe cascade thrust reversers will satisfy both the modulating thrust and performance requirements. Among the problems inherent in reversing the cruise jets are structural overheating due to jet impingement or attachment to aircraft surfaces and ingestion of the hot exhaust gas into inlets in the jet wake. It was concluded that the final selection of the reverser type should be based on installation considerations rather than performance.

Forward and reverse thrust output from the two lift-cruise engines is shown in Figure IV-56. Although test data has indicated values as high as 84 percent of the engine gross thrust may be reversed, a value of 40 percent has been used. This value of 40 percent was selected so as to prevent structural overheat during reversal. It is seen that at zero forward speed the thrust reverser can be modulated between 200 and 2100 pounds of reverse thrust (idle to MRP). Also shown on Figure IV-56 is the engine inlet temperature rise at zero forward speed and the minimum forward speed for no temperature rise when operating the thrust reverser with the engines at military rated power (MRP). For Concept C lift-cruise engines, a rise of approximately 135 degrees F is estimated at zero forward speed with no rise occurring at approximately 62 knots forward speed.

A cascade type reverser has been selected for the Concept C installation as depicted on the propulsion installation drawing. It has been decided that local structural provisions will only be made for possible thrust reverser installation in the field. Decision to perform installation shall be an option after initiation of the flight research program.

THRUST REVERSER PERFORMANCE

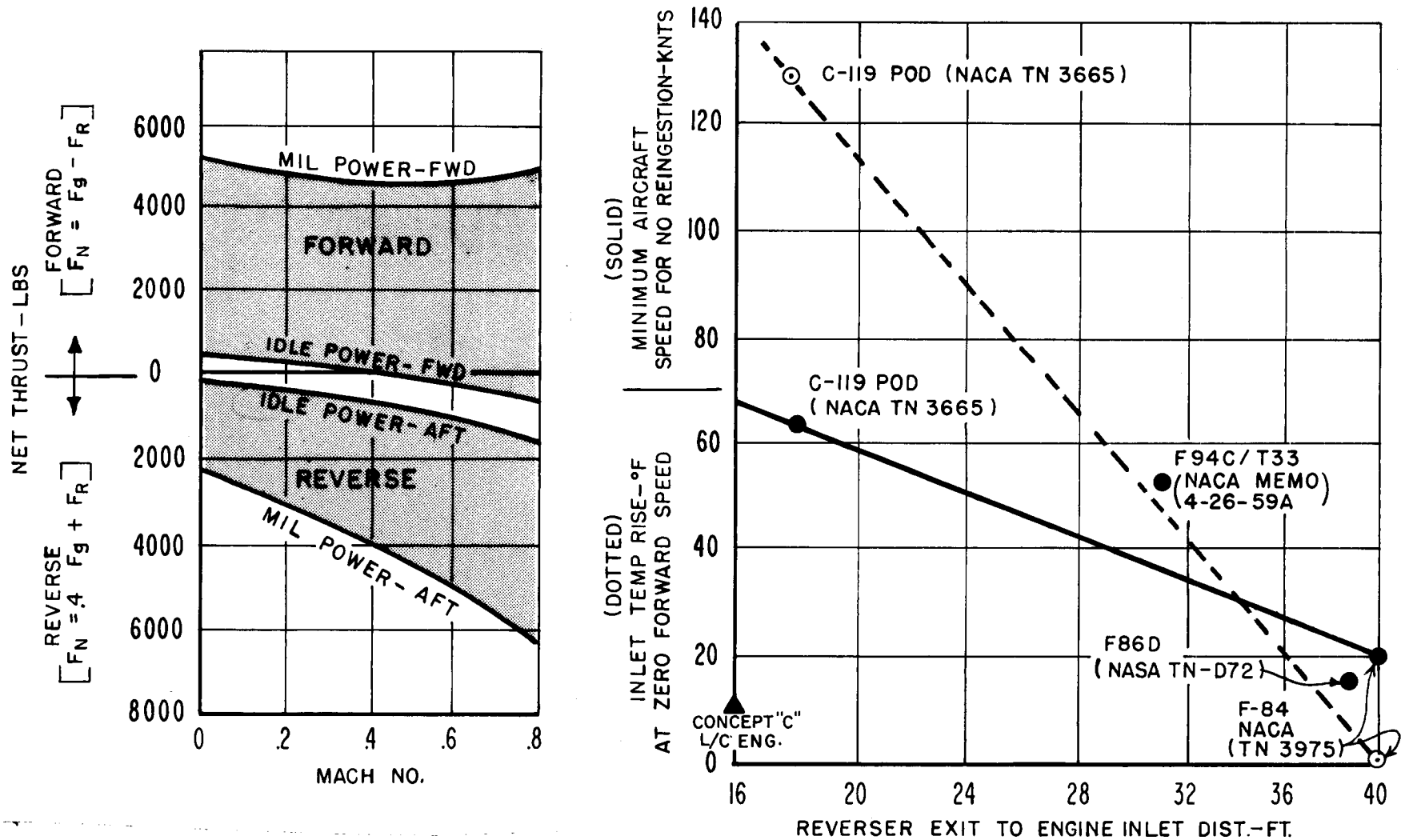


Figure IV-56

I. FIRE DETECTION AND EXTINGUISHING SYSTEM

A fire protection system shall be installed in the aircraft which has the function of both detecting and extinguishing a fire. A review of various systems has been made resulting in the selection of the surveillance fire detection (Fire Eye) and wet tube fire extinguishing system.

1. Fire Detection System

Three types of detection systems are available, namely, unit, continuous and surveillance (UFD, CFD and SFD, respectively).

The Unit Fire Detection System (UFD) consists of single temperature probes and operates on the principle of expansion and contraction of the outer shell caused by an increase or decrease in temperature which in turn results in actuation of enclosed contacts.

The Continuous Fire Detection System (CFD) consists of a wire sensing element and a control unit. The sensing element is connected in a closed loop circuit and provides the input signal to the control unit. Upon reaching the temperature at which the sensing element is set, the input impedence to the control unit is lowered thus activating the magnetic amplifier and initiating an alarm signal.

The Surveillance Fire Detection System (SFD) consists of a infrared radiation sensor and a control unit. The sensor detects near-infrared radiation from flames sending a signal to the control unit which in turn initiates an alarm signal.

2. Fire Extinguishing System

Based on the results of a Study of the Effectiveness of a Lightweight, Fixed, Fire-Extinguishing System performed by Fenwal Incorporated under U.S. Navy Contract N600(19)59752, the Fenwal lightweight system was selected in lieu of the conventional system for use in the airplane.

The lightweight system uses the distribution tubes of the extinguishing system to store the agent (wet tube) and a propellant gas generator to expel the agent whereas the conventional system utilizes spherical container(s) for agent storage. Tare weight savings of 50 to 70 percent are obtained wherever the lightweight

system replaced the spherical containers.

The results of the fire tests indicated that the lightweight system extinguished fires as effectively as a conventional system.

SECTION V

WEIGHT AND BALANCE

CONCEPT C

A. WEIGHT, BALANCE, AND INERTIA SUMMARY

A summary of the weight, balance, and moments of inertia for the selected Concept C design is presented in Table V-1.

TABLE V-1. WEIGHT, BALANCE, AND INERTIA SUMMARY

	lb	%G.W.
Structure	3,563	22.9
Propulsion and Nacelle	4,794	30.8
Power Systems	1,235	7.9
Equipment Groups	830	5.3
Contingency	106	0.7
Weight Empty	10,528	67.6
Crew	400	2.6
Oil, Oxygen and Trapped Fuel	132	0.9
Payload (VSS and Research Equipment)	800	5.1
Hover Fuel	3,440	22.1
VTO Gross Weight	15,300	98.3
Ramp Fuel	270	1.7
Ramp Gross Weight	15,570	100.0

	V.T.O. Condition	Zero Fuel Condition
Weight	15,300 lb	11,860 lb
Center-of-Gravity	31.7% M.A.C.	31.3% M.A.C.
I_x (Roll)	6,144 Slug-ft^2	5,505 Slug-ft^2
I_y (Pitch)	34,280 Slug-ft^2	28,296 Slug-ft^2
I_z (Yaw)	37,857 Slug-ft^2	31,505 Slug-ft^2

B. GROUP WEIGHT STATEMENT

The Group Weight Statement, Form AN-9103-D, is presented on the following pages.

AN-9103-D
SUPERSEDING
AN-9103-C

NAME
DATE 5/31/67

PAGE 1
MODEL
REPORT

GROUP WEIGHT STATEMENT

ESTIMATED - CALCULATED - ACTUAL
(Cross out those not applicable)

CONTRACT NO. NASA V/STOL ~ CONCEPT C
AIRPLANE, GOVERNMENT NO. (14 MINUTE HOVER)
AIRPLANE, CONTRACTOR NO.
MANUFACTURED BY FAIRCHILD HILLER - REPUBLIC DIV

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY	LIFT / CRUISE	LIFT
	MODEL	YJ85-GE-19	YJ85-GE-19
	NO.	2	6
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

AN-9103-D
NAME
DATE 3/31/67

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 2
MODEL
REPORT

1	WING GROUP		724
2	CENTER SECTION - BASIC STRUCTURE	72	
3	INTERMEDIATE PANEL - BASIC STRUCTURE		
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS 20 LBS.)	398	
5			
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)	42	
7	AILERONS (INCL. BALANCE WEIGHT LBS.)	34	
8	FLAPS - TRAILING EDGE	110	
9	- LEADING EDGE		
10	SLATS		
11	SPOILERS	68	
12	SPEED BRAKES		
13			
14			
15	TAIL GROUP		364
16	STABILIZER - BASIC STRUCTURE	234	
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.)	87	
18	SECONDARY STRUCTURE (STAB. & FINS)		
19	ELEVATOR (INCL. BALANCE WEIGHT LBS.)		
20	RUDDERS (INCL. BALANCE WEIGHT LBS.)	43	
21			
22			
23	BODY GROUP		1850
24	FUSELAGE OR HULL - BASIC STRUCTURE	909.6	
25	BOOMS - BASIC STRUCTURE		
26	SECONDARY STRUCTURE - FUSELAGE OR HULL	457.4	
27	- BOOMS		
28	- SPEEDBRAKES		
29	- DOORS, PANELS & MISC.	482.0	
30			
31	ALIGHTING GEAR GROUP - LAND (TYPE:)		625

32	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS	
33					
34	NOSE - FUSELAGE	22	68	21	111
35	MAIN - WING	184	266	64	514
36					
37					
38					
39					

40	ALIGHTING GEAR GROUP - WATER				
41	LOCATION	FLOATS	STRUTS	CONTROLS	
42					
43					
44					
45					

46	SURFACE CONTROLS GROUP & V.T.O. CONTROLS		676
47	COCKPIT CONTROLS	69	
48	AUTOMATIC ~~PILOT~~ STABILIZATION	40	
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS LBS.)	313	
50	V.T.O DUCTING & NOZZLES	254	
51	ENGINE SECTION OR NACELLE GROUP		276
52	INBOARD		
53	CENTER	48	
54	OUTBOARD	228	
55	DOORS, PANELS & MISC.		
56			
57	TOTAL (TO BE BROUGHT FORWARD)		4515

AN-9103-D
NAME
DATE 5/31/67

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 3
MODEL
REPORT

1	PROPULSION GROUP				4518
2		~~AUXILIARY~~ LIFT		~~MAIN~~ L/C	
3	ENGINE INSTALLATION		2322		974
4	AFTERBURNERS (IF FURN. SEPARATELY)				
5	ACCESSORY GEAR BOXES & DRIVES				
6	SUPERCHARGERS (FOR TURBO TYPES)				
7	AIR INDUCTION SYSTEM		30		
8	EXHAUST SYSTEM		404		344
9	COOLING SYSTEM				
10	LUBRICATING SYSTEM				
11	TANKS				
12	COOLING INSTALLATION				
13	DUCTS, PLUMBING, ETC.				
14	FUEL SYSTEM				300
15	TANKS - PROTECTED				
16	- UNPROTECTED			92	
17	PLUMBING, ETC.			208	
18	WATER INJECTION SYSTEM				
19	ENGINE CONTROLS		60		20
20	STARTING SYSTEM		48		16
21	PROPELLER INSTALLATION				
22					
23					
24	AUXILIARY POWER PLANT GROUP				
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP				150
26	HYDRAULIC & PNEUMATIC GROUP				154
27					
28					
29	ELECTRICAL GROUP				405
30					
31					
32	ELECTRONICS GROUP				185
33	EQUIPMENT			148.4	
34	INSTALLATION			36.6	
35					
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)				
37	FURNISHINGS & EQUIPMENT GROUP				352
38	ACCOMMODATIONS FOR PERSONNEL			304	
39	MISCELLANEOUS EQUIPMENT			25	
40	FURNISHINGS				
41	EMERGENCY EQUIPMENT			23	
42					
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP				143
44	AIR CONDITIONING			139	
45	ANTI-ICING			4	
46					
47	PHOTOGRAPHIC GROUP				
48	AUXILIARY GEAR GROUP				
49	HANDLING GEAR				
50	ARRESTING GEAR				
51	CATAPULTING GEAR				
52	ATO GEAR				
53					
54					
55	~~MANUFACTURING VARIATION~~ CONTINGENCY				106
56	TOTAL FROM PG. 2				4315
57	WEIGHT EMPTY				10528

AN-9103-D
NAME
DATE 5/31/67

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

PAGE 4
MODEL
REPORT

1	LOAD CONDITION			VTO (14 MIN)		RAMP	
2							
3	CREW (NO. 2)			400		400	
4	PASSENGERS (NO.)						
5	FUEL	Type	Gals.				
6	UNUSABLE			56		56	
7	INTERNAL			3440		3710	
8							
9							
10	EXTERNAL						
11							
12	BOMB BAY						
13							
14	OIL			60		60	
15	TRAPPED						
16	ENGINE						
17							
18	FUEL TANKS (LOCATION)						
19	WATER INJECTION FLUID (GALS)						
20							
21	BAGGAGE						
22	CARGO						
23							
24	ARMAMENT						
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.			
26							
27							
28							
29							
30							
31							
32	AMMUNITION						
33							
34							
35							
36							
37							
38							
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)						
*40	BOMB OR TORPEDO RACKS						
41							
42							
43	OXYGEN			16		16	
44							
45							
46	EQUIPMENT (VSS & RESEARCH)			800		800	
47	PYROTECHNICS						
48	PHOTOGRAPHIC						
49							
*50	OXYGEN						
51							
52	MISCELLANEOUS						
53							
54							
55	USEFUL LOAD			4772		5042	
56	WEIGHT EMPTY			10528		10528	
57	GROSS WEIGHT			15300		15570	

*If not specified as weight empty.

AI-9103-D
NAME
DATE 5/31/67

GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

PAGE 5
MODEL
REPORT

1	LENGTH - OVERALL (FT.)				HEIGHT - OVERALL - STATIC (FT.)			
2		Main Floats	Aux. Floats	Booms	Fuse or Hull	Inboard	Nacelles Center	Outboard
3	LENGTH - MAX. (FT.)				42.9			
4	DEPTH - MAX. (FT.)				5.8			
5	WIDTH - MAX. (FT.)				4.7			
6	WETTED AREA (SQ. FT.)				575			
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							
8	FUSELAGE VOLUME (CU. FT.)		PRESSURIZED			TOTAL		
9						Wing	H. Tail	V. Tail
10	GROSS AREA (SQ. FT.)					204.0	65	36
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)							
12	SPAN (FT.)					32.0	6.0	7.4
13	FOLDED SPAN (FT.)							
14								
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES)					25.0		
16	- AT % CHORD LINE (DEGREES)							
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES)					117.5	66	84
18	- MAX. THICKNESS (INCHES)					12.6		
***19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)							
20	- MAX. THICKNESS (INCHES)							
***21	THEORETICAL TIP CHORD - LENGTH (INCHES)					35.2	32	32
22	- MAX. THICKNESS (INCHES)					6.34		
23	DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)							
24	TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.)							
25	AREAS (SQ. FT.) Flaps	L.E.		T.E. 25.2				
26	Lateral Controls	Slots		Spoilers 12.6		Ailerons 10.4		
27	Speed Brakes	Wing		Fuse. or Hull				
28								
29								
30	ALIGHTING GEAR (LOCATION)							
31	LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)							
32	OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)							
33	FLOAT OR SKI STRUT LENGTH (INCHES)							
34	ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)							
35	HYDRAULIC SYSTEM CAPACITY (GALS.)							
36	FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected		No. Tanks	****Gals. Unprotected	
37	Fuel - Internal	Wing						
38		Fuse. or Hull				2	570	
39	- External							
40	- Bomb Bay							
41								
42	Oil					8	8	
43								
44								
45	STRUCTURAL DATA - CONDITION				Fuel in Wings (Lbs.)	Stress Gross Weight		Ult. L.F.
46	FLIGHT				0	15300		5.625
47	LANDING							
48								
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL							
50	CATAPULTING							
51	MIN. FLYING WEIGHT							
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)							
53	WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)							
54	STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)							
55	PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)							
56								
57	AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)							

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

U. S. GOVERNMENT PRINTING OFFICE : 1963 O - 200000

C. DETAIL WEIGHT STATEMENT

The Detail Weight Statement, Form AN-9102-D, is presented on the following pages.

NAME
DATE 5/31/67

DETAIL WEIGHT STATEMENT

ESTIMATED - CALCULATED - ACTUAL

(Cross out those not applicable)

CONTRACT NO. NASA V/STOL ~ CONCEPT C

AIRPLANE, GOVERNMENT NO. (14 MINUTE HOVER)

AIRPLANE, CONTRACTOR NO.

MANUFACTURED BY FAIRCHILD HILLER ~ REPUBLIC DIV.

		[redacted]	[redacted]
ENGINE	MANUFACTURED BY	LIFT/CRUISE	LIFT
	MODEL	YJ85-GE-19	YJ85-GE-19
	NO.	2	6
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

AN-9102-D
NAME
DATE 5/31/67

WING GROUP
BASIC STRUCTURE

PAGE 2
MODEL
REPORT [redacted]

	Center Section	Interm. Panel	Outer Panel		
CODE NO.					
UPPER - SPAR CAP - [redacted]			30		
- [redacted]					
- [redacted]					
- AUXILIARY					
- INTERSPAR COVERING			84		
SPANWISE STIFFENERS					
- JOINTS, SPLICES & FASTENERS					
LOWER - SPAR CAP - [redacted]			30		
- [redacted]					
- [redacted]					
- AUXILIARY					
- INTERSPAR COVERING			54		
SPANWISE STIFFENERS					
- JOINTS, SPLICES & FASTENERS					
SPAR WEB & STIFFENERS - [redacted]			40		
- [redacted]					
- [redacted]					
- AUXILIARY					
- JOINTS, SPLICES & FASTENERS					
INTERSPAR - RIBS			46		
- BULKHEADS					
- CHORDWISE STIFFENERS					
LEADING EDGE - COVERING			30		
- STIFFENERS					
- RIBS			14		
- AUXILIARY SPARS					
- JOINTS, SPLICES & FASTENERS					
TRAILING EDGE - COVERING			23		
- STIFFENERS					
- RIBS			12		
- AUXILIARY SPARS			15		
- JOINTS, SPLICES & FASTENERS					
TIPS			20		
FIREWALL (STRUCTURAL)					
CARRY-THRU STRUCTURE	72				
TOTALS - BASIC STRUCTURE	72		398		
TOTAL (TO BE BROUGHT FORWARD)					470

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WING GROUP
SECONDARY STRUCTURE
(DOORS, PANELS & MISCELLANEOUS)

PAGE 3
MODEL
REPORT

		* Location	** Type Power	Area Sq. Ft.	Structure	Operating Mechanism: Mechanism & Controls	Power Trans.	Actuator	Lock Mech.	Emerg.
1										
2										
4	CODE NO.									
5	WING FOLD									
6										
7										
8	DOORS & FRAMES-LANDING			21.0	42					
9										
10										
11	- BOMB									
12										
13										
14	- GUN									
15										
16	- AMMUNITION									
17										
18	- ROCKET									
19										
20	- LIFE RAFT									
21										
22	- ESCAPE									
23										
24	- ACCESS									
25										
26										
27										
28	PANELS-(NON STRUCTURAL)									
29										
30										
31										
32										
33										
34										
35										
36										
37										
38										
39										
40										
41										
42										
43										
44										
45										
46										
47										
48										
49	WALKWAYS, STEPS, GRIPS									
50										
51	FAIRING & FILLETS									
52										
53	EXTERIOR FINISH									
54										
55										
56	TOTALS				42					
57	TOTAL - SECONDARY STRUCTURE (TO BE BROUGHT FORWARD)									42

*Indicate location for major doors by C.S., O.P., I.P., etc.

**H-Hydraulic, E-Electrical, P-Pneumatic; power transmission from main distribution point to actuating unit.

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WING GROUP
CONTROL SURFACES

PAGE 4
MODEL
REPORT

	Ailerons Inb'd	Ailerons Outb'd	T.E. Flaps Inb'd	T.E. Flaps Outb'd	L.E. Flaps or Slats Inb'd	L.E. Flaps or Slats Outb'd	Spoilers	Speed Brakes	
1									
2									
3									
4 CODE NO.									
5 SPARS									
6									
7									
8									
9 RIBS									
10									
11									
12 COVERING & STIFFENERS									
13									
14									
15 T.E. STRIPS									
16									
17 FABRIC & DOPE									
18									
19									
20									
21 TABS									
22									
23									
24									
25 TORQUE TUBES									
26									
27									
28									
29 BALANCE WEIGHTS & SUPPORTS									
30									
31 AERO. SEAL									
32									
33									
34 CONTROL HORNS									
35									
36									
37 ACCESS DOORS (NON STRUCT.)									
38									
39 HINGES & PINS									
40 EXTERIOR FINISH									
41 TOTALS - SURFACE	( )	(29)	(72)	( )	( )	( )	(60)	( )	( )
42									
43 CONTROL SURFACE SUPPORTS									
44 HINGES									
45 BRACKETS									
46 TRACKS									
47 CARRIAGES									
48									
49									
50									
51									
52 TOTALS - SUPPORTS	( )	(5)	(38)	( )	( )	( )	(8)	( )	( )
TOTALS (LINES 41 & 52)									
54 TOTALS - CONTROL SURFACES	34		110				68		
55 TOTAL									
56 TOTALS FROM PGS. 2 & 3								212	
57 TOTAL - WING GROUP								512	
								724	

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TAIL GROUP
BASIC STRUCTURE

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MODEL
REPORT

	Stabilizer		Fins			
	C.S.	O.P.	Center	Outer	Dorsal	
CODE NO.						
4 UPPER - SPAR CAP - ~~FRONT~~		7				
5 - ~~INTERMEDIATE~~						
6 - ~~REAR~~						
7 - ~~AUXILIARY~~						
8 - INTERSPAR COVERING		19	18			
9 SPANWISE STIFFENERS						
10 - JOINTS, SPLICES & FASTENERS						
11						
12						
13						
14 LOWER - SPAR CAP - ~~FRONT~~		7				
15 - ~~INTERMEDIATE~~						
16 - ~~REAR~~						
17 - ~~AUXILIARY~~						
18 - INTERSPAR COVERING		19	18			
19 SPANWISE STIFFENERS						
20 - JOINTS SPLICES & FASTENERS						
21						
22						
23						
24 SPAR WEB & STIFFENERS - ~~FRONT~~		16	12			
25 - ~~INTERMEDIATE~~						
26 - ~~REAR~~						
27 - ~~AUXILIARY~~						
- JOINTS, SPLICES & FASTENERS						
29						
30						
31						
32 INTERSPAR - RIBS		8	8			
33 - BULKHEADS						
34 - CHORDWISE STIFFENERS						
35						
36 LEADING EDGE - COVERING		12	20			
37 - STIFFENERS						
38 - RIBS		8	9			
39 - AUXILIARY SPARS						
40 - JOINTS, SPLICES & FASTENERS						
41						
42						
43 TRAILING EDGE - COVERING		15				
44 - STIFFENERS						
45 - RIBS		10				
46 - AUXILIARY SPARS		7				
47 - JOINTS, SPLICES & FASTENERS						
48 TRAILING EDGE - HONEYCOMB		17				
49						
50 TIPS		4	2			
51						
52 TORQUE TUBE		69				
53 FITTINGS		116				
55 TOTALS						
56 TOTALS - BASIC STRUCTURE		234	87			
57 TOTAL (TO BE BROUGHT FORWARD)						321

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TAIL GROUP
SECONDARY STRUCTURE
(DOORS, PANELS & MISCELLANEOUS)

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MODEL
REPORT

1	* Location	** Type Power	Area Sq. Ft.	Structure	Operating Mechanism: Mechanism & Controls	Power Trans.	Actuator	Lock Mech.	Emerg.
4 CODE NO.									
5 DOORS & FRAMES - LANDING									
6									
7									
8									
9									
10 ACCESS									
11									
12									
13									
14									
15									
16									
17									
18									
19									
20									
21									
22									
23									
24									
25									
26									
27									
28									
29 PANELS - (NON STRUCTURAL)									
30									
31									
32									
33									
34									
35									
36									
37									
38									
39									
40									
41									
42									
43									
44									
45 WALKWAYS, STEPS, GRIPS									
46									
47									
48									
49 FAIRING & FILLETS									
50									
51									
52 EXTERIOR FINISH									
53									
54									
55									
56 TOTALS									
57 TOTAL - SECONDARY STRUCTURE (TO BE BROUGHT FORWARD)									0

*Indicate location for major doors by "Surface" - Stab., Fin, etc.

**H-Hydraulic, E-Electrical, P-Pneumatic; power transmission from main distribution point to actuating unit.

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TAIL GROUP
CONTROL SURFACES

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MODEL
REPORT

		Elevators	Rudders Center	Rudders Outer		
1						
2						
3	CODE NO.					
4 SPARS						
5						
6						
7						
8 RIBS						
9						
10						
11						
12 COVERING & STIFFENERS						
13						
14						
15 T. E. STRIPS						
16						
17 FABRIC & DOPE						
18						
19						
20						
21 TABS						
22						
23						
24						
25 TORQUE TUBES						
26						
27						
28 STRUCTURE			27			
29 BALANCE WEIGHTS & SUPPORTS			10			
30						
31 AERO. SEAL						
32						
33						
34 CONTROL HORNS						
35						
36						
37 ACCESS DOORS (NON STRUCTURAL)						
38						
39 HINGES & PINS						
40 EXTERIOR FINISH						
41						
42 TOTALS - SURFACE		( )	(37)	( )	( )	( )
43						
44	CONTROL SURFACE SUPPORTS					
45 HINGES			6			
46 BRACKETS						
47						
48						
49						
50						
51						
52 TOTALS - SUPPORTS		( )	(6)	( )	( )	( )
53 TOTALS (LINES 42 & 52)			43			
54 TOTALS - CONTROL SURFACES						
55 TOTAL						43
56 TOTALS FROM PGS. 5 & 6						321
57 TOTAL - TAIL GROUP						364

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BODY GROUP
BASIC STRUCTURE

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MODEL
REPORT

					Fuselage or Hull				Booms
			CODE NO.						
			SECTION						
*4 BULKHEADS & FRAMES	Stations				NOSE	COCKPIT	FUEL & ENG	AFT	
5					3.6	52.0	225.4	20.9	
6									
7									
8									
9									
10									
11									
12									
13									
14									
15									
16									
17									
18									
19									
20									
21									
22									
23									
24 MINOR FRAMES					3.2	20.0	69.9	20.1	
25 JOINTS, SPLICES & FASTENERS									
26 OVERTURN STRUCTURE									
27									
28 COVERING - UPPER BETWEEN LONGERONS					10.8	49.0	183.0	48.0	
29 - SIDE BETWEEN LONGERONS									
30 - LOWER BETWEEN LONGERONS									
31									
32 COVERING LONGITUDINAL STIFFENERS - UPPER BETW. LONG.						5.1	23.4	6.1	
33 - SIDE BETW. LONG.									
34 - LOWER BETW. LONG.									
35									
36									
37 LONGERONS - UPPER					5.2	20.7	31.7	9.7	
38 - LOWER									
39									
40									
41 LONGITUDINAL PARTITIONS - (STRUCTURAL)									
42 NOSE WHEEL WELL BEAMS						10.4			
43 FLOORING & SUPPORTS - (BASIC STRUCTURE)						46.5			
44 INTAKE DUCT SUPPORT BEAM							7.5		
45									
46									
47 FIREWALL - (STRUCTURAL)							38.4		
48									
49 KEELSONS									
50 KEEL									
51									
52 CHINE & SPRAY STRIPS									
53 STEP ASSEMBLY									
4 STAIRWAY - (STRUCTURAL)									
55 TOTALS					22.8	203.7	578.3	104.8	
56 TOTALS - BASIC STRUCTURE							909.6		
57 TOTAL (TO BE BROUGHT FORWARD)									909.6

*List all main & watertight bulkheads & frames individually. Minor frames may be combined.

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BODY GROUP
SECONDARY STRUCTURE

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MODEL

REPORT

		Fuselage or Hull				Booms	Speed Brakes
1							
2	CODE NO.						
3	SECTION						
4							
5	ENCLOSURES (EXCLUDING TURRET ENCLOSURES)						
6	CANOPY		168.0				
7	CANOPY-OPERATING MECHANISM						
8	-RAILS		170.0				
* 9	-CYLINDERS, PLUMBING, FLUID						
10							
11	GUNNER - TAIL						
12							
13	BOMBARDIER						
14	SIGHTING BLISTERS						
15							
16	WINDSHIELD (EXCLUDING BULLET PROTECTION)		44.0				
17							
18	WINDOWS & PORTS INCL. FRAMES						
19							
20							
21							
22							
23							
24							
25							
26							
27							
28	FLOORING & SUPPORTS (SECONDARY STRUCTURE)		50.0				
29							
30							
31	STAIRWAYS & LADDERS (FIXED)						
32							
33							
34	STERNPOST & FITTINGS						
35	NOSE BUMPER (HULL)						
36	RUBBING STRIPS						
37							
38							
39							
40	TAIL CONE & NOSE CONE		25.4				
41							
42							
43	SPEED BRAKES - STRUCTURE						
44	- SUPPORTS						
45							
46							
47							
48							
49							
50							
51							
52							
53							
54							
55	TOTALS		457.4				
56	TOTALS - SECONDARY STRUCTURE						
57	TOTAL (TO BE BROUGHT FORWARD)						457.4

*From main distribution point to actuating unit.

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BODY GROUP
SECONDARY STRUCTURE
(DOORS, PANELS & MISCELLANEOUS)

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MODEL
REPORT

1 / 2 / 3	* Location	** Type Power	Area Sq. Ft.	Structure	Operating Mechanism: Mechanism & Controls	Power Trans.	Actuator	Lock Mech.	Emerg.
4 CODE NO.									
5 DOORS & FRAMES									
6 - LANDING	NOSE		12.1	20.0	13.3				
7 - ENG INTAKE				50.0	20.1				
8 - ENG EXHAUST				130.0	49.60				
9									
10 - BOMB									
11									
12									
13 - GUN									
14									
15 - AMMUNITION									
16									
17 - ROCKET									
18									
19 - LIFE RAFT									
20									
21 - ESCAPE									
22									
23									
24 - WATER TIGHT									
25									
26 - COMPARTMENT									
27									
28 - ENTRANCE									
29									
30									
31 - ACCESS	FUEL			30.2					
32									
33									
34									
35 - ENGINE									
36									
37 - CAMERA									
38									
39 PANELS - (NON STRUCTURAL)									
40 - COCKPIT				30.0					
41 - FUEL CELLS				19.8					
42									
43									
44									
45									
46									
47									
48 MISC				120.0					
49 WALKWAYS, STEPS, GRIPS									
50									
51 FAIRING & FILLETS									
52 EXTERIOR FINISH									
53									
54 TOTALS				400.0	83.0				
55 TOTAL - SECONDARY STRUCTURE (DOORS, PANELS, MISC.)									483
56 TOTALS FROM PGS. 8 & 9									1367
57 TOTAL - BODY GROUP									1850

*Indicate location for major doors by B - Booms, F or H for Fuselage or Hull.

**H - Hydraulic, E - Electrical, P - Pneumatic; power transmission from main distribution point to actuating unit.

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ALIGHTING GEAR GROUP

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MODEL

REPORT

1											
2	TYPE:						*LOCATION	MAIN	NOSE		
3							CODE NO.				
4	*LOCATION										
5		No.	Size	No.	Size	No.	Size				
6	WHEELS							72	11		
7	TIRES							54	11		
8	TUBES							8			
9	AIR							2			
10	BRAKES							48			
11	NO. & TYPE										
12	ENERGY CAP. **										
13	ANTI-SKID DEVICE										
14											
15	FLOATS - BULKHEADS										
16	- FRAMES										
17	- COVERING										
18	- COVERING STIFFENERS (LONGITUDINAL)										
19	- KEELSONS										
20	- KEEL										
21	- LONGITUDINAL PARTITIONS										
22	- CHINE & SPRAY STRIP										
23	- STEP ASSEMBLY										
24	- POST ASSEMBLY										
25	- NOSE BUMPER										
26	INSPECTION DOORS										
27	WALKWAYS										
28	EXTERIOR FINISH										
29	SKIDS OR BUMPERS										
30	SKIIS										
31											
32	TOTALS - RUNNING GEAR							(184)	(22)	( )	( )
33											
34	STRUTS - DRAG								12		
35	- SIDE							24			
36	- FLOAT										
37	PYLON										
38	SHOCK STRUT - STRUT (INCL. LBS. OIL)							242	56		
39	- FORK										
40	- AXLE										
41	- TORQUE ARMS										
42	- TRUNNIONS										
43	SHIMMY DAMPER OR SNUBBER										
44											
45	FITTINGS - MAIN ATTACH. - WING										
46	- TAIL										
47	- BODY										
48	- NACELLE										
49											
50	FAIRING										
51											
52											
53											
54											
55	PINS, BOLTS, NUTS, ETC.										
56	TOTALS - STRUCTURE							(266)	(68)	( )	( )
57	TOTALS (LINES 32 & 56) (TO BE BROUGHT FORWARD)							450	90		

*Descriptive location - Nose, Tail, Main, Outrigger, Bumper, etc.

**Ft. lbs./brake

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ALIGHTING GEAR GROUP
CONTROLS

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MODEL
REPORT

1 / 2 **LOCATION	MAIN			NOSE							
3 / 4	Retract	Brake Oper.	Emerg. Ext.	Steering	Retract	Brake Oper.	Emerg. Ext.	Retract	Emerg. Ext.	Retract	Emerg. Ext.
5 CODE NO.											
6 MECHANICAL OPER. MECH.											
7 CONTROLS	4	2	8								
8 ACTUATORS											
9											
10											
11											
12 ELECTRICAL OPER. MECH.											
13 CONTROLS											
*14 WIRING, CONDUIT, ETC.											
15 OPERATING MOTORS											
16 MECHANISM	4										
17											
18											
19											
20 HYDRAULIC OPER. MECH.											
21 CONTROLS				2							
*22 PLUMBING & FLUID	12	10		2	5						
23 PUMPS											
24 RESERVOIRS		2									
25 ACCUMULATORS											
26 ACTUATORS	12	4		8	4						
27 MECHANISM											
28											
29											
30											
31 PNEUMATIC OPER. MECH.											
32 CONTROLS											
*33 PLUMBING											
34 PUMPS											
35 BOTTLES (AIR)											
36 ACTUATORS											
37 MECHANISM											
38											
39											
40											
41 LOCKING MECHANISM	6										
42 BRACES											
43 LINKS											
44 PARKING BRAKE CONTROL											
45 POSITION INDICATING MECH.											
46											
47											
48 SUPP'TS, GUIDES, ETC. - WING											
49 - TAIL											
50 - BODY											
51 - NACELLE											
52											
53 TOTALS	38	18	8	12	9						
54 TOTALS - CONTROLS	64			21							
55 TOTALS FROM PG. 11	450			90							
56 TOTALS	514			111							
57 TOTAL - ALIGHTING GEAR GROUP											625

*From main distribution point to actuating unit.

**Descriptive location - Nose, Tail, Main, Outrigger, Bumper, etc.

SURFACE CONTROLS GROUP
COCKPIT & AUTOPILOT

		Cockpit Controls	~~Autopilot~~
1			
2			
3	CODE NO.		
4	CONTROL COLUMNS	24	
5	CONTROL STICK OR COLUMN (PILOT)		
6	CONTROL STICK OR COLUMN (ASSIST. PILOT)		
7	CONNECTING MEMBERS FOR ABOVE		
8	SUPPORTS		
9			
10			
11			
12			
13	RUDDER PEDALS	45	
14	RUDDER PEDALS, INCLUDING BRAKE TREADLE (PILOT)		
15	RUDDER PEDALS, INCLUDING BRAKE TREADLE (ASSIST. PILOT)		
16	CONNECTING MEMBERS FOR ABOVE		
17	SUPPORTS		
18			
19			
20			
21			
22			
23	INTEGRAL PARKING LOCK		
24	CONTROL STICK		
25	RUDDER PEDALS		
26	SURFACES		
27			
28			
29			
30			
31			
32			
33	AUTOPILOT OR AUTO. FLIGHT CONTROL (TYPE:)		
34	CONTROLLER		
35	TRANSMITTER		
36	SERVO AMPLIFIER		
37	SERVO MOTORS		
38	GYROS		
39			
40	STABILITY AUGMENTATION		40
41			
42			
43			
44			
45			
46	SUPPORTS & BRACKETS		
47			
*48	PLUMBING & FLUID		
*49	ELECTRIC PANELS, BOXES, SWITCHES, RELAYS, WIRING		
50	PULLEYS, SPROCKETS, CHAINS, CABLES		
51			
52			
53			
54			
55			
56	TOTALS - COCKPIT CONTROLS & AUTOPILOT	69	40
57	TOTAL (TO BE BROUGHT FORWARD)		109

*From main distribution point to actuating units.

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SURFACE CONTROLS GROUP
SYSTEM CONTROLS
(INCLUDING V.T.O. CONTROLS)

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MODEL
REPORT

		All.	HOR TAIL ~~Elev.~~	Rud.	Wing Sweep	Wing Incid.	L.E. Flaps or Slats	T.E. Flaps	Spoilers	Speed Brakes	Stab. Adj.	V.T.O.
1												
2												
3												
4	CODE NO.											
5	MECHANICAL OPER. MECH.											
6	CONTROLS											
7	TENSION REGULATORS											
8	ACTUATORS											
9	TRIM CONTROLS											
10												
11	ELECTRICAL OPER. MECH.											
**12	TYPE											
13	CONTROLS											
*14	WIRINGS, SWITCHES, ETC.											
15	OPERATING MOTORS											
16	MECHANISM											
17	TRIM CONTROLS											
18												
19	HYDRAULIC OPER. MECH.											
**20	TYPE											
21	CONTROLS											
*22	PLUMBING & FLUID											
23	PUMPS											
24	RESERVOIRS											
25	ACCUMULATORS											
26	ACTUATORS											
27	MECHANISM											
28	TRIM CONTROLS											
29												
30	PNEUMATIC OPER. MECH.											
**31	TYPE											
32	CONTROLS											
*33	PLUMBING											
34	PUMPS											
35	BOTTLES (AIR)											
36	ACTUATORS											
37	MECHANISM											
38	TRIM CONTROLS											
39												
40	ARTIFICIAL FEEL											
41	BUNGEE											
42	BOB WEIGHT											
43												
44												
45												
46												
47												
48	SUPPORTS, GUIDES, ETC.											
49	WING											
50	TAIL											
51	BODY											
52	NACELLE											
53												
54	TOTALS	94	82	43				48	46			254
55	TOTALS - SYSTEM CONTROLS											567
56	TOTAL (FROM PG. 13)											109
57	TOTAL - SURFACE CONTROLS GROUP + V.T.O. CONTROLS											676

*From main distribution point to actuating units.

**Type - add (P) for "Powered Controls"
- or (B) for "Boost Controls"

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ENGINE SECTION OR NACELLE GROUP

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MODEL
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		L/C	LIFT Center	Outboard	
1					
2					
3	CODE NO.				
4	ENGINE MOUNT	16	48		
5					
6	SUPPORT BAY				
7	VIBRATION ABSORPTION DEVICES				
8					
9					
10	NACELLE STRUCTURE	212			
11	BULKHEADS AND FRAMES				
12	COVERING & STIFFENERS				
13	FITTINGS				
14	LONGERONS				
15	ATTACHING ANGLES, ETC.				
16					
17					
18					
19	PYLONS & STRUTS				
20					
21					
22					
*23	FIREWALL				
24					
25	SHROUDS FOR FIRE PROTECTION				
26					
27	COWLING				
28	ENGINE COWL				
29					
30					
31					
32					
33					
34					
35	BAFFLES				
36	ACCESSORY COWL OR SKIRT				
37	COWL FLAPS				
38	COWL FLAP CONTROLS & OPERATING MECH.				
39					
40					
41					
42					
43					
44					
45	FAIRING - NACELLE TO WING OR PYLON				
46	STEPS & GRIPS				
47	WORKING PLATFORM (BUILT IN)				
48	INTERNAL WALKWAYS				
49					
50					
51	INSTALLATION HARDWARE				
52					
53					
54					
55					
56	TOTALS - SECTIONS OR NACELLES	228	48		
57	TOTAL (TO BE BROUGHT FORWARD)				276

*If in nacelle, or non-structural in wing or body.

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NACELLE GROUP
DOORS, PANELS & MISCELLANEOUS

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		* Location	** Type Power	Area Sq. Ft.	Structure	Operating Mechanism: Mechanism & Controls	Power Trans.	Actuator	Lock Mech.	Emerg.
1										
2										
3										
4	CODE NO.									
5	DOORS & FRAMES									
6	- LANDING									
7										
8										
9										
10	- BOMB									
11										
12										
13	- ACCESS									
14										
15	- ENGINE									
16	NOTE:									
17	ENGINE INLET &									
18	EXHAUST DOORS									
19	CODED TO BODY									
20										
21										
22										
23										
24										
25										
26	PANELS - (NON STRUCTURAL)									
27										
28										
29										
30										
31										
32										
33										
34										
35										
36										
37										
38										
39										
40										
41										
42										
43										
44										
45										
46										
47										
48										
49										
50										
51										
52	EXTERIOR FINISH									
53										
54	TOTALS									
55	TOTAL - DOORS, PANELS & MISC.									0
56	TOTAL FROM PG 15									276
57	TOTAL - ENGINE SECTION OR NACELLE GROUP									276

*Indicate location for major doors by inb'd, center, outb'd.

**H-Hydraulic, E-Electrical, P-Pneumatic; power transmission from main distribution point to actuating unit.

1		~~Auxiliary~~ L/C	~~Motor~~	LIFT
2	CODE NO.			
3	ENGINE INSTALLATION	974		2322
4	ENGINE (AS INSTALLED)	774	2322	
5	ENGINE & AFTERBURNER (AS INSTALLED)			
6	REDUCTION GEAR BOX			
7	EXTENSION DRIVE SHAFT			
8	DIVERTER VALVES	186		
9	DIVERTER VALVE INSULATION	14		
10	AFTERBURNERS (IF FURNISHED SEPARATELY)			
11	ACCESSORY GEAR BOXES & DRIVES			
12	SUPERCHARGER - COMPLETE (FOR TURBOS)			
13	- LUBRICATING SYSTEM			
14	- SUPPORTS			
15	- CONTROLS			
16	- PIPING (EXHAUST TO SUPER.)			
17				
18				
19	AIR INDUCTION SYSTEM			30
20	INTERCOOLERS AND SUPPORTS			
21	AIR DUCTING ~~AND SURROUNDING~~		30	
22	INTAKE DOORS & OPERATING CONTROLS			
23	AIR FILTERS			
24	SCREENS & CONTROLS			
25				
26				
27				
28				
29				
30				
31	EXHAUST SYSTEM	344		404
32	EXHAUST STACKS			
33	EXHAUST COLLECTORS			
34	COLLECTOR OR ENGINE SHROUD			
35	TAIL PIPE	186	72	
36	TAIL PIPE SHROUD AND INSULATION	66	56	
37	TAIL CONE			
38	SILENCING DEVICES			
39	SUPPORTS, BRACKETS, ETC.			
40	SPHERICAL NOZZLES	92	276	
41				
42				
43				
44	COOLING SYSTEM			
45	RADIATOR AND SUPPORTS			
46	SHUTTERS, SCOOP & DUCTS			
47	EXPANSION TANK & SUPPORTS			
48	LIQUID IN SYSTEM(GALS.)			
49	PIPING, VENTS, CLAMPS ETC.			
50				
51				
52	FANS			
53	CONTRAVANES			
54	FAN DRIVES			
55	CONTROLS & OPERATING MECH.			
56				
57	TOTALS (TO BE BROUGHT FORWARD)	1318		2756

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PROPULSION GROUP
LUBRICATING & FUEL SYSTEMS

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MODEL
REPORT

						Auxiliary		Main	
1									
2						Lubricating	Fuel	Lubricating	Fuel
3					CODE NO.				
4	TANKS	Type	Location	No.	Vol. Gp. (Gals.)				
5		BLADDER							92
6									
7									
8									
9									
10									
11									
12									
13									
14									
15									
16									
17									
18									
19									
20	INTEGRAL TANK SEALS & SEALANT								
21	BACKING BOARD								4
22	TANK SUPPORTS & PADDING								
23	TANK BAY SEALING								
24									
25	TANK RELEASE & CONTROLS								
26	OIL COOLING INSTALLATION								
27	COOLERS & SUPPORTS (SIZE) (NO.)								
28	DUCTS & SHUTTERS								
29	AUTOMATIC OIL TEMP. CONTROL VALVE								
30	SHUTTER CONTROLS								
31									
32	FUEL VAPOR RECOVERY								
33									
34	OIL DILUTION SYSTEM								
35									
36	FUEL VAPOR INERTION SYSTEM - CYL. & SUPPORTS								
37	- GENERATOR								
38	- CONTROLS, ETC.								
39	PUMP INSTALLATION	No.	Type						
40	ENGINE DRIVEN								
41	BOOSTER								28
42	HAND (INCL. CONTROLS)								
43	TRANSFER								
44									
45									
46	FILLING SYSTEM - GROUND								10
47	- IN FLIGHT								
48									
49	DISTRIBUTION SYSTEM								156
50	TRANSFER SYSTEM								
51	VENT SYSTEM								10
52	PRESSURIZATION SYSTEM								
53	DUMP SYSTEM								
54									
55									
56	TOTALS - LUBRICATING & FUEL SYSTEMS								
57	TOTALS (TO BE BROUGHT FORWARD)								300

PROPULSION GROUP

1					~~Auxiliary~~ L/C		~~Main~~ LIFT
2				CODE NO.			
3	WATER INJECTION SYSTEM						
4	TANKS (NO.) (GALS/TANK)						
5	PUMP						
6	METERING UNIT						
7	VALVES & PLUMBING						
8	CONTROLS						
9							
10							
11	ENGINE CONTROLS				20		60
12	IGNITION						
13	THROTTLE						
14	MIXTURE						
15	SUPERCHARGER (SUP. INTEG. WITH ENG.)						
16	AFTERBURNER						
17							
18							
19							
20	STARTING SYSTEM (AIR IMPINGMENT)				16		48
21	STARTER POWER UNIT (TYPE:)						
22	STARTER (TYPE:)						
23	STARTER CONTROLS						
24	CRANK & EXTENSION						
25	PRIMER & PIPING						
26	MESHING SOLENOID						
27	SWITCHES, WIRING & CONDUIT						
28							
29							
30							
31							
32	PROPELLER INSTALLATION (DIA.) (NO.)						
33	PROPELLER						
34	CUFFS						
35	SPINNER						
36	CONTROLS	Type	*Aux.	*Main			
37	SPEED						
38	PITCH						
39	FEATHERING						
40	REVERSING						
41							
42							
43							
44							
45							
46							
**47	OIL (GALS)						
**48	TANK & PLUMBING						
49							
50							
51							
52							
53							
54							
55	TOTALS				36		108
56	TOTALS FROM PGS. 17 & 18				1318		3056
57	TOTAL - PROPULSION GROUP						4518

*GFP Weight

**When separate oil system used.

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AUXILIARY POWER PLANT GROUP

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		CODE NO.
1		
2	ENGINE OR POWER UNIT (MODEL)	
3	ENGINE SUPPORTS	
4		
5		
6	AIR INDUCTION SYSTEM	
7		
8		
9	EXHAUST SYSTEM	
10	STACKS	
11	COLLECTOR	
12		
13		
14		
15	COOLING SYSTEM	
16	RADIATORS	
17		
18		
19		
20	LUBRICATING SYSTEM	
21	TANKS (VOL. GALS.)	
22	PLUMBING	
23		
24		
25		
26	FUEL SYSTEM	
27	TANKS (VOL. GALS.)	
28	PLUMBING	
29		
30		
31	CONTROLS	
32		
33		
34		
35	STARTING SYSTEM	
36	STARTER	
37		
38		
39		
40	SUPPORTS	
41		
42		
43		
44		
45		
46		
47		
48		
49		
50		
51		
52		
53		
54		
55		
56		
57	TOTAL - AUXILIARY POWER PLANT GROUP	0

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INSTRUMENT & NAVIGATIONAL EQUIPMENT GROUP
INSTRUMENTS

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* FUNCTIONAL GROUPS & ITEMS	CODE NO.	Number	Indicator	Transmitter & Amplifier	Installation	Total
FLIGHT INSTRUMENTS						
AIRSPEED & MACH METER		2	3.6	8.0	5.8	17.4
ATTITUDE/VERTICAL VELOCITY		2	4.0			4.0
ATTITUDE DIRECTOR		2	4.0			4.0
HORIZONTAL SITUATION		2	3.0			3.0
STANDBY COMPASS		2	1.4		0.4	1.8
STANDBY ATTITUDE		1	2.0			2.0
STANDBY AIRSPEED		1	2.0			2.0
STANDBY ALTIMETER		1	1.4			1.4
STANDBY TURN & SLIP		1	1.2			1.2
THRUST VECTOR POSITION		2	3.0	5.0	3.0	11.0
SIDE SLIP		2	2.4	4.0	3.0	9.4
ENGINE INSTRUMENTS						
FUEL QUANTITY		2	2.0	8.0	8.0	18.0
HYDRAULIC PRESSURE		4	1.2	0.5	1.0	2.7
ENGINE CONDITION		2	10.0	15.0	4.0	29.0
LIFT ENGINE OPERATION		2	3.0	6.0	3.0	12.0
MISCELLANEOUS						
FLAP POSITION		2	1.0	0.4	0.4	1.8
CLOCK		2	1.0		0.4	1.4
STABILITY AUGMENTATION SYST. O.K.		2	8.0	11.0	3.0	22.0
CAUTION ANNUNCIATOR		2	3.0		0.5	3.5
			57.2	57.9	32.5	
INSTRUMENT POWER SYSTEM (TYPE						
TOTAL - INSTRUMENTS (TO BE BROUGHT FORWARD)						147.6

*List items by functional groups (Flight, Engine & Misc.). List sub-groups by crew stations; add supp. pg. 21A if necessary.

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INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP
NAVIGATIONAL

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		Installation	Navig. Equip.
1			
2	CODE NO.		
3	NAVIGATIONAL INSTALLATION		
4	LOCKERS & SUPPORTS		
5	TABLES & SUPPORTS		
6	SHELVES & SUPPORTS		
7	MAP CASES & REPORT HOLDERS	0.4	2.0
8	ASTRODOME		
9	CHARTBOARD		
10	STOWAGES & SUPPORTS - CHARTBOARD		
11	- SPOTLIGHT		
12	- DRIFTSIGHT		
13	- ASTROCOMPASS		
14			
15			
16			
17			
18			
19			
20			
21			
22			
23			
24			
25	NAVIGATIONAL EQUIPMENT		
26	PLOTTER		
27	PARALLEL RULER		
28	CHARTS		
29	DRIFT SIGHT		
30	SEXTANT & CASE		
31	BINOCULARS		
32	NAVIGATING WATCHES & BOXES		
33	COMPUTERS		
34	ASTROCOMPASS & ADAPTERS		
35			
36			
37			
38			
39			
40			
41			
42			
43			
44			
45			
46			
47			
48			
49			
50			
51			
52			
53			
54	TOTALS	0.4	2.0
55	TOTAL - NAVIGATIONAL		2.4
56	TOTAL - INSTRUMENTS FROM PG. 21		147.6
57	TOTAL - INSTRUMENT & NAVIGATIONAL EQUIP. GROUP		150.0

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*HYDRAULIC & PNEUMATIC GROUP

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				Hydraulic			Pneumatic		
1				Utility	Emergency		Utility	Emergency	
2									
3			CODE NO.						
4 PUMPS & COMPRESSORS		Model		28					
5									
6									
7									
8									
9									
10									
11									
12 REMOTE PUMP DRIVES									
13	No.	Model	Capacity						
14 RESERVOIRS				20					
15									
16 AIR BOTTLES									
17 ACCUMULATORS									
18									
19									
20									
21 FILTERS				12					
22 PRESSURE REGULATORS				2					
23									
24 VALVES-									
25									
26									
27									
28									
29									
30 CONTROLS									
31									
32									
33									
34									
35 PLUMBING				74					
36									
37									
38									
39									
40 FLUID IN SYSTEM (TYPE)(GALS.)				18					
41									
42									
43 SUPPORTS - WING									
44 - TAIL									
45 - BODY									
46 - NACELLE									
47 TOTALS				154					
48 TOTAL - HYDRAULIC & PNEUMATIC GROUP									154

49 FURNISHES POWER FOR - (ITEMS)	AILERONS	STAB. AUGMENT.
50	STABILATOR	
51	RUDDER	
52	WHEEL BRAKES	
53	LANDING GEAR DOORS	
54	ENGINE DOORS	
55	DIVERTER VALVES	
56	REACTION JETS	
57 SYSTEM PRESSURE (PSI)	ENGINE EXHAUST VECTOR	

*Includes system from sources of power to main distribution points.

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*ELECTRICAL GROUP

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MODEL

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						AC		DC	
1									
2									
3				CODE NO.					
4	POWER SUPPLY EQUIPMENT	Driven By	KVA	Amp.	No.		134		34
5	GENERATORS								
6									
7									
8									
9									
10	REMOTE GENERATOR DRIVES								
11	BATTERY (AN) (NO.)								
12	BATTERY CONTAINER, OVERFLOW INST. & SUPPORTS								
13									
14	POWER CONVERSION EQUIPMENT		Model		No.		25		23
15	INVERTER (DC TO AC)								
16	CONVERTER (AC TO DC)								
17	TRANSFORMER								
18	RECTIFIER								
19	MOTOR-GENERATOR								
20	PHASE ADAPTER								
21	FREQUENCY CONVERTER								
22									
23									
24	POWER DISTRIBUTION & CONTROL						170		16
25	GENERATOR CONTROL BOXES								
26	CUTOUTS & VOLTAGE REGULATORS								
27	AMMETERS & VOLTMETERS								
28	SWITCHES, RHEOSTATS, SWITCH PANELS OR BOXES								
29	CIRCUIT BREAKERS & FUSES								
30	JUNCTION, FUSE, DISTRIBUTION BOXES & PANELS								
31	RECEPTACLES & CONNECTOR PLUGS								
32	RELAYS								
33	WIRING								
34	CONDUIT								
35									
36	LIGHTS & SIGNAL DEVICES						3		
37	LIGHTS - INTERIOR								
38	- EXTERIOR								
39	- LANDING (INCL. RETRACT MECH.)								
40									
41	SIGNAL DEVICES - LIGHTS								
42	- HORNS								
43	- BELLS								
44									
45									
46	EQUIPMENT SUPPORTS - WING								
47	- TAIL								
48	- BODY								
49	- NACELLE								
50	TOTALS						332		73
51	TOTAL - ELECTRICAL GROUP								405
52	FURNISHES POWER FOR - (ITEMS)								
53									
54									
55									
56									
57									

*Includes system from sources of power to main distribution points.

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ELECTRONICS GROUP

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	*FUNCTIONAL GROUP	*EQUIPMENT COMPONENTS & PART NUMBERS OR IDENTIFICATION	CODE NO.	Equipment GFP	Equipment CFE	Installation
4	VHF	RADIO VHF		(30.7)		(7.6)
5						
6		TRANSCEIVER 618M-1A		19.0		
7		CONTROL 313-N3		4.0		
8		ANTENNA 137X-1		6.0		
9		SHOCKMOUNT 390Y-2		1.7		
10		WIRE & MISC.				7.6
11						
12						
13	ATC	ATC TRANSPONDER		(30.5)		(7.5)
14						
15		TRANSPONDER 621A-3		25.2		
16		CONTROL		3.0		
17		SHOCKMOUNT 350E-3D		2.1		
18		L-BAND ANTENNA		.2		
19		WIRE & MISC.				7.5
20						
21						
22	NAVIGATION	RADIO RANGE-VOR/ILS		(20.9)		(5.2)
23						
24		RECEIVER 51RV-1		18.5		
25		SHOCKMOUNT 390Y-1		1.7		
26		ANTENNA 37P-4		.7		
27		WIRE & MISC.				5.2
29		MARKER BEACON		(4.2)		(1.0)
30						
31		RECEIVER 51Z-4		3.3		
32		ANTENNA 37X-2		.3		
33		SHOCKMOUNT 390R-1		.6		
34		WIRE & MISC.				1.0
35						
36		TACAN		(37.7)		(9.3)
37						
38		TRANSCEIVER 860E-2		36.0		
39		ANTENNA 237Z-1		.2		
40		SHOCKMOUNT 350E-3D		1.5		
41		WIRE & MISC.				9.3
42						
43						
44						
45						
46						
47	ELECTRONIC INSTALLATION					
48	TABLES					
49	RACK, SHELVES & SUPPORTS					
50	LOCKERS					
51						
52						
53						
55	SUBTOTALS - EQUIPMENT GFP & CFE - (TO BE BROUGHT FORWARD)			124.0		30.6
56	TOTALS				154.6	
57	TOTAL - ELECTRONIC GROUP					

*List components (incl. Radomes, Mts., Antennae, Switches, Relays, Filters, etc.) from main distribution point to unit operated, by functional groups (e.g., Comm., VHF, Search, Navig., Intercomm., etc.). Add supplementary pg. 25A if necessary.

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ELECTRONICS GROUP

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1 *FUNCTIONAL GROUP	*EQUIPMENT COMPONENTS & PART NUMBERS OR IDENTIFICATION	Equipment GFP	Equipment CFE	Installation
2				
3	CODE NO.			
4				
5 NAVIGATION	PERIPHERAL COMMAND SYSTEM	(10.2)		(2.5)
6				
7	INDICATOR	5.0		
8	AMPLIFIER	4.0		
9	SHOCK MOUNT	1.2		
10				
11	WIRE & MISC.			2.5
12				
13				
14	COMPASS SYSTEM	(14.2)		(3.5)
15				
16	GYRO COMPASS & AMPL. C704130	9.2		
17	FLUX VALVE DT-173/AJN	3.0		
18	CONTROL	2.0		
19				
20	WIRE & MISC.			3.5
21				
22				
23				
24				
25				
26				
27				
28				
29				
30				
31				
32				
33				
34				
35				
36				
37				
38				
39				
40				
41				
42				
43				
44				
45				
46				
47 ELECTRONIC INSTALLATION				
48 TABLES				
49 RACK, SHELVES & SUPPORTS				
50 LOCKERS				
51				
52				
53				
54 TOTALS FROM PAGE 25		124.0		30.6
55 SUBTOTALS - EQUIPMENT GFP & CFE		148.4		36.6
56 TOTALS				
57 TOTAL - ELECTRONIC GROUP				185.0

*List components (incl. Radomes, Mts., Antennae, Switches, Relays, Filters, etc.) from main distribution point to unit operated, by functional groups (e.g., Comm., VHF, Search, Navig., Intercomm., etc.). Add supplementary pg. 25A if necessary.

1	TURRET OR GUN LOCATIONS						
	NO. OF GUNS						
	CALIBER OF GUNS						
4	CODE NO.						
5	EQUIPMENT						
6	AMMUNITION - FEED CHUTES						
7	- BOXES						
8	- BOX SUPPORTS						
9	CASE EJECTION CHUTES						
10	LINK EJECTION CHUTES						
11	CASE & LINK RECEPTACLES						
12	MOUNTS						
13	MOUNT SUPPORTS						
14	BLAST TUNNELS (INTEGRAL)						
15							
16	AMMUNITION BOOSTERS						
17							
18	GUN CHARGERS INSTALLATION						
19							
20							
21							
22							
23	HEATING						
24							
25	FIRE INTERRUPTERS						
26							
*27	TURRET SYSTEMS (TYPE:)						
29							
30							
31							
32							
33							
34							
35							
36							
37							
38							
39							
40							
41							
42							
43							
44							
45							
46							
47							
48							
49							
50							
51							
52							
53							
55							
56	TOTALS						
57	TOTAL (TO BE BROUGHT FORWARD)						0

*List components by packaged groups (turret structure, accessories, sighting, computation, gun accessories, elec. cabling, etc.). Add supplementary pg. 26A if necessary.

ARMAMENT GROUP
ACCOMMODATIONS FOR PERSONNEL

		CODE NO.	
1			
2	ROCKET INSTALLATION - CONTROLS		
	- RELEASE MECHANISM		
*4	- WIRING, ETC.		
5			
6			
7			
8			
9			
10			
11	BOMB, MINE & TORPEDO INSTALLATION		
12	BOMBER'S CONTROL PANEL		
*13	RELEASE - ELECTRICAL		
14	- MANUAL		
15	- MANUAL EMERGENCY		
16	INTERVALOMETER		
17			
18	ARMING & SAFE MECHANISM		
19			
20	BOMB HOIST MECHANISM (IF BUILT IN)		
21			
**22	BOMB RACKS (TYPE) (NO.)		
23			
24	BOMB RAILS OR TRACKS		
25			
26			
27			
29			
30	GUNFIRE PROTECTION (PASSIVE DEFENSE)		
31	BULLET RESISTANT GLASS		
32	ARMOR PLATE (NON STRUCTURAL) - OIL COOLER		
33	- OIL TANK		
34	- ENGINE SECTION		
35	- CREW - PILOT		
36	-		
37	-		
38	-		
39			
40			
41			
42	FLAK CURTAINS		
43			
44			
45	SONOBUOY - STOWAGE		
46	- DISPENSERS		
47			
48			
49			
50			
51			
52			
53	SMOKE TANK PROVISIONS		
55	TOTAL		
56	TOTAL FROM PG. 26		
57	TOTAL - ARMAMENT GROUP		0

*From main distribution point to actuating unit.

**If not specified as useful load.

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FURNISHINGS & EQUIPMENT GROUP
ACCOMMODATIONS FOR PERSONNEL

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1	CODE NO.								
2	CREW SEATS & PASSENGER CHAIRS								
3 4	Location	No.	Cushions	Seat PARACHUTE	Safety Belt	Harness & Inertia Reel	Adj. Mech.	Catapult or Elect. Mech.	Tracks & Supports	
5	PILOT	1	12	79	3	4	5	20	19	142
6	ASST. PILOT	1	12	79	3	4	5	20	19	142
7										
8										
9										
10										
11										
12										
13										
14										
15										
16										
17	HEAD REST (IF NOT INTEGRAL WITH SEAT)									
18	BUNKS (NO.) & SUPPORTS									
19										
20	LITTER SUPPORTS									
21	BOMBERS & GUNNERS KNEELING PADS (NO.)									
22	PARACHUTE STOWAGE PROVISIONS									
23	TOILETS & RELIEF TUBES									
24	WASH BASIN & SHOWERS									
25	WATER TANKS & PIPING									
26	DRINKING WATER CONTAINERS & SUPPORTS									
27	LOCKERS FOR - FOOD									
28	- PERSONAL EFFECTS									
29										
30	GALLEY STOVES & HOTPLATES									
31	REFRIGERATOR									
32										
33										
34										
35										
36										
37	ANTI-G SUIT PROVISIONS									
38										
39	OXYGEN INSTALLATION									20
*40	BOTTLES - INCL. CHARGE (TYPE) (SIZE) (NO.)									
41										
42	CONVERTOR									
*43	REGULATORS (TYPE) (NO.)									
44	SUPPORTS - BOTTLES & REGULATORS									
45	PLUMBING, ETC.									
46										
47										
48										
49										
50										
51										
52										
53										
54										
55										
56										
57	TOTAL - ACCOMMODATIONS FOR PERSONNEL (TO BE BROUGHT FORWARD)									304

*If not specified as useful load or special equipment.

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FURNISHINGS & EQUIPMENT GROUP
MISC. EQUIPMENT & FURNISHINGS

PAGE 29
MODEL [redacted]
REPORT

		Misc. Equip.	Furnishings
1			
2			
3	CODE NO.		
4	MISCELLANEOUS EQUIPMENT		
* 5	PORTABLE PLATFORMS & LADDERS		
6			
7	DATA CASES & REPORT OR FORM HOLDERS	1	
8			
* 9	MANUALS - FLIGHT & MAINTENANCE-		
*10	BALANCE COMPUTER & SUPPORT		
11			
12	TOOL LOCKERS		
13			
14	WINDSHIELD WIPER & WASHER INSTALLATION		
15	RELEASE MECHANISM & FITTINGS - TARGET & GLIDER TOW		
16			
17	BILGE SYSTEM		
18	STALL WARNING DEVICES		
19	REAR VIEW MIRROR		
20			
21	AUXILIARY FLOORING		
22	INSTRUMENT BOARDS	12	
23	CONSOLES	12	
24	CONTROL STANDS		
25			
*26	CARGO HANDLING EQUIPMENT		
27	RAMPS		
28	HOISTS & BOOMS		
29	MONORAILS		
30	MONORAIL MOTORS		
31	TIE DOWN FITTINGS		
32			
33			
34			
35	PYROTECHNIC INSTALLATION		
36	SIGNAL PISTOL HOLDER		
37	SIGNAL AMMUNITION HOLDER (CAP.)		
38	PARACHUTE FLARE - CONTAINERS (NO.)		
39	- RACKS (CAPACITY)		
40	- RELEASE MECHANISM		
41			
42	SMOKE CANDLE (GRENADE) HANDLE		
43			
44	FLOAT LIGHT RACK & RELEASE MECH. (CAP.)		
45			
46	FURNISHINGS		
47	FLOOR COVERING, RUGS, ETC.		
48	SOUNDPROOFING & THERMAL INSULATION		
49	TRIM		
50	CURTAINS & SCREENS		
51	CRASH PADDING		
52	PARTITIONS (NON-STRUCTURAL)		
53			
54			
55			
56	TOTALS - MISC. EQUIP. & FURNISHINGS	25	
57	TOTAL (TO BE BROUGHT FORWARD)		25

*If not specified as special equipment.

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FURNISHINGS & EQUIPMENT GROUP
EMERGENCY EQUIPMENT

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MODEL
REPORT

1		CODE NO.							
2			Compartments						
3	FIRE EXTINGUISHERS		Engine	Baggage	Fuel				
4	BOTTLES	Type							
5		Size							
6		No.							
7		Weight							
8	CONTROLS								
9	PLUMBING								
10	BOTTLE SUPPORTS								
11									
12									
13	TOTAL - COMPT.								
14	PORTABLE (TYPE) (SIZE) (NO.)								
15									
16									
17									
18	PORTABLE EXTINGUISHER SUPPORTS								
19									
20	FIRE DETECTION SYSTEM (INCLUDING WET TUBE FIRE EXTINGUISHING)								23
21									
22	FIRE RESISTANT PAINT								
23	FIRE CURTAINS								
24									
25	FIRST AID KITS (NO.) & STOWAGE								
26									
27	FLASHLIGHTS (NO.)								
28									
29	STOWAGE - EMERGENCY RATIONS & WATER								
30									
*31	LIFE RAFTS - (TYPE) (NO.)								
32									
33									
34									
35									
36	- SUPPORTS OR CRADLES								
37									
38	DITCHING STATION EQUIPMENT								
39									
40									
41									
42									
43									
44									
45									
46									
47									
48									
49									
50									
51									
52									
53									
54									
55	TOTAL - EMERGENCY EQUIPMENT								23
56	TOTALS FROM PGS. 28 & 29								329
57	TOTAL - FURNISHINGS & EQUIPMENT GROUP								352

*If not specified as useful load or special equipment.

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AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP
AIR CONDITIONING

PAGE 31
MODEL
REPORT

	Pressurization System	Ventilating System	Heating System	Cooling System	CABIN	
1						
2						
3 CODE NO.						
* 4 HEAT EXCHANGERS (NO.)						
5						
* 6 HEATERS (BTU CAPACITY) (NO.)						
7						
8 HEATING FLUID (GALS.)						
9 AIR CONDITIONING UNIT					56	
10 COMPRESSORS OR SUPERCHARGERS						
11						
12 MOTORS						
13 TURBINES						
14 FANS					3	
15						
16						
17 TANKS						
18 WATER SEPARATOR						
19 REGULATOR					4	
20 PLENUM CHAMBERS					4	
21						
22 SCOOPS						
23 DUCTING					60	
24 SHROUDS						
25						
26 VALVES					9	
27 PLUMBING						
28						
29						
30 BOMB BAY HEATING						
31						
32						
33						
34						
35						
36 CONTROLS - MANUAL					3	
37						
38 - ELECTRICAL						
39						
40 - HYDRAULIC						
41						
42 - PNEUMATIC						
43						
44						
45						
46						
47						
48 SUPPORTS & BRACKETS - WING						
49 - TAIL						
50 - BODY						
51 - NACELLE						
52						
53 PRESSURIZATION SEALING						
54						
55						
56 TOTALS					139	
57 TOTAL - AIR CONDITIONING (TO BE BROUGHT FORWARD)						139

*If not specified as special equipment.

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AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP
ANTI-ICING

PAGE 32
MODEL
REPORT

			Wing	Tail	Air Induction	Propeller	Canopy & Windshield	Fuel System	ENGINE
1									
2									
3		CODE NO.							
* 4 HEATERS	No.	BTU Capacity							
5									
6									
7									
8									
9									
10									
*11 HEAT EXCHANGERS (NO.)									
12									
13									
14									
15 DUCTING									
16 SHROUDING									
17									
18									
*19 BOOTS									
20									
*21 ATTACHING STRIPS									
22									
23 OIL SEPARATORS									
24									
25 AIR PUMPS									
26									
27 AIR LINES & HOSES									
28									
29 TANKS									
30									
*31 FLUID (GALS.)									
32									
33									
34									
35 PLUMBING									
36									
37									
38 DISTRIBUTOR - VALVE									
39 - CONTROLS									
40									
41									
42 CONTROLS - MANUAL									
43 - ELECTRICAL									
44 - HYDRAULIC									
45 - PNEUMATIC									
46									
**47 WIRING, SWITCHES, RELAYS									4
48									
49 SUPPORTS & BRACKETS - WING									
50 - TAIL									
51 - BODY									
52 - NACELLE									
53									
54 TOTALS									4
55 TOTAL - ANTI-ICING									4
56 TOTAL FROM PG. 31									139
57 TOTAL - AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP									143

*If not specified as special equipment.

**From main distribution point to actuating unit.

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PHOTOGRAPHIC GROUP

PAGE 33
MODEL
REPORT

			Photo Equipment	Installation Provision
1				
2				
3		CODE NO.		
* 4	EQUIPMENT ITEM	Model		
5	CAMERAS			
6				
7				
8				
9				
10				
11				
12				
13				
14				
15				
16				
17				
18				
19				
20				
21	MOUNTS & ADAPTERS			
22				
23				
24				
25				
26				
27				
28	INSTALLATION PROVISIONS			
29	TRIP & SYNCHRONIZATION UNITS			
30				
31				
32				
33	PHOTO-ELECTRIC SYSTEM			
34				
35				
36				
37	FLASHLIGHT BOMB INST.			
38				
39				
40				
41	VACUUM SYSTEM - PUMP			
42	- PLUMBING			
43	- GAGE			
44				
45				
46	PRESSURE SYSTEM - PUMP			
47	- PLUMBING			
48	- GAGE			
49				
50	CONTROLS			
51	SUPPORTS			
52				
*53	WIRING, RELAYS, SWITCHES			
54				
55				
56	TOTALS			
57	TOTAL - PHOTOGRAPHIC GROUP		0	

*If not specified as useful load or special equipment.

**From main distribution point to actuating unit.

NAME

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AUXILIARY GEAR GROUP

MODEL

REPORT

		Handling	Arrest.	Catapult	ATO
1					
2	CODE NO.				
3	HANDLING GEAR				
4	ANCHOR				
5	ANCHOR LINE				
6	PENDANT & CLAMP FITTING				
7	LIZARD				
8	SHEAVES				
9	WINCH - COMPLETE				
10	WINCH CRANK				
11	ANCHOR HANDLING RIG OR DAVIT				
12	WINCH ENGINE OR MOTOR				
13					
*14	HOISTING SLING				
15	WING HANDLING LINES				
16	WATER RUDDER				
17	FITTINGS - RECOVERY HOOK				
18	- BEACHING GEAR ATTACHMENT				
19	- TIEDOWN				
20	- JACKING				
21	- TOWING				
22	- MOORING & SNUBBING				
23	- ANCHORAGE				
24	- LEVELING				
25	- HOISTING				
26					
27	ARRESTING OR DECELERATION GEAR				
28	TRAILING HOOK				
29	HOOK POINT (TYPE)				
30	EXTENSION GEAR				
31	RETRIEVING GEAR				
32	BUMPER				
33	SHOCK ABSORBER				
34	ATTACHMENT FITTINGS				
35					
36	BARRIER CRASH FITTINGS				
37					
38	DECELERATION - PARACHUTE				
39	- CONTAINER & FITTINGS				
40	- CONTROLS				
41					
42					
43	CATAPULTING GEAR				
44	CATAPULT FITTINGS				
45	CATAPULT HOOKS				
46	HOLDBACK FITTINGS				
47					
48	ASSISTED TAKE OFF				
49	HOOKS				
50					
51	CONTROLS - FIRING				
52	- BOTTLE RELEASE				
53	BOTTLE STOWAGE PROV. (NO. BOTTLES)				
54					
55					
56	TOTALS				
57	TOTAL - AUXILIARY GEAR GROUP				0

*If not specified as special equipment.

D. CENTER-OF-GRAVITY TRAVEL ENVELOPE

Figure V-1 presents the center-of-gravity envelope for the Concept C design with eight YJ85-GE-19 engines. The travel range is based on normal fuel consumption under the following conditions:

- Flight with either or both pilots (The single pilot is assumed to be in the front cockpit)
- Flight with either 500 pounds or 800 pounds of payload

Ballast was not considered in calculating the envelope.

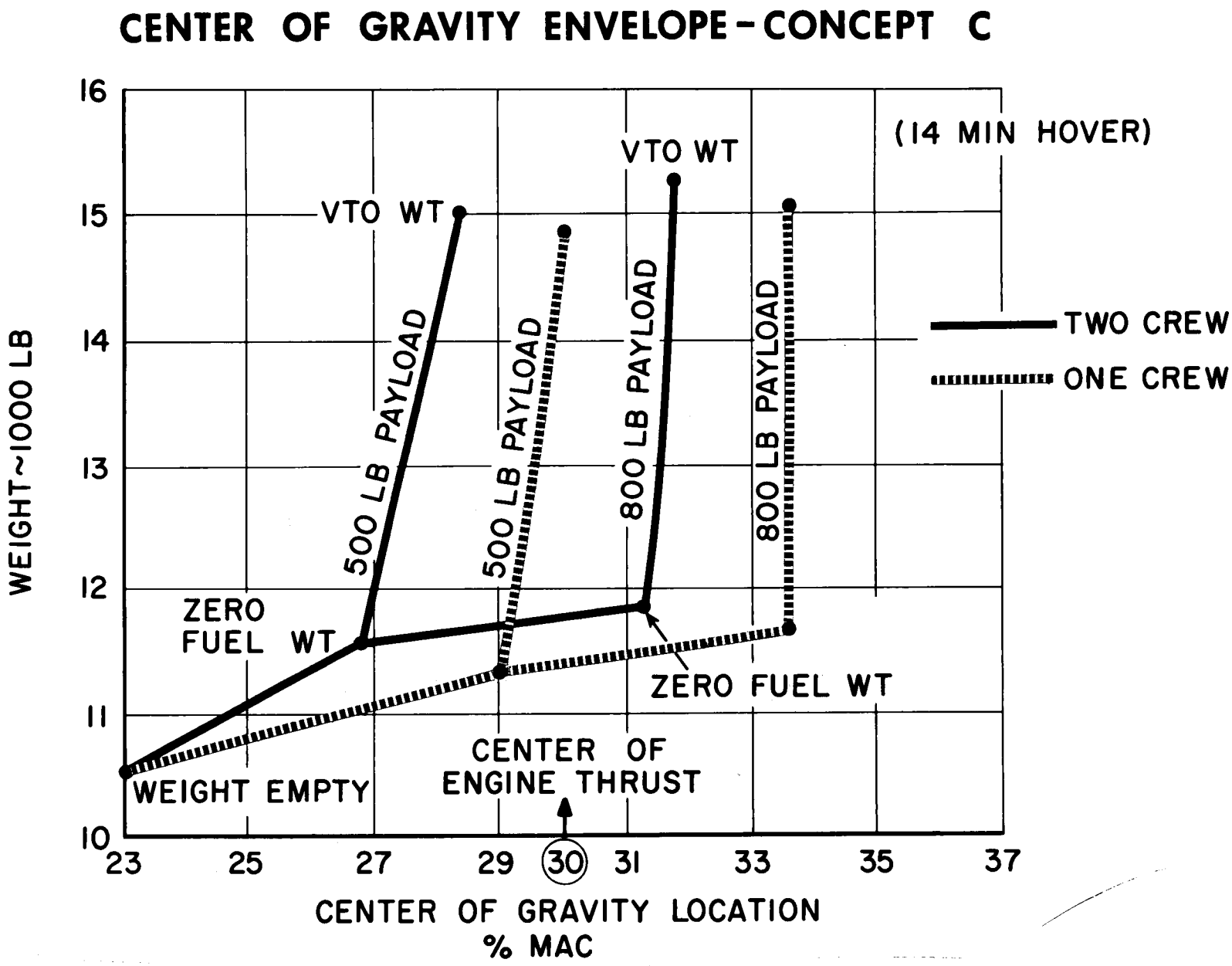
CENTER OF GRAVITY ENVELOPE - CONCEPT C
(14 MIN HOVER)
TWO CREW
ONE CREW
VTO WT
VTO WT
500 LB PAYLOAD
500 LB PAYLOAD
800 LB PAYLOAD
800 LB PAYLOAD
ZERO FUEL WT
ZERO FUEL WT
WEIGHT EMPTY
CENTER OF ENGINE THRUST
WEIGHT~1000 LB
16
15
14
13
12
11
10
23
25
27
29
30
31
33
35
37
CENTER OF GRAVITY LOCATION % MAC

Figure V-1

E. WEIGHT GROWTH

1. Effect of Weight Growth on Hover Endurance

Figure V-2 shows the effect of operating weight empty growth on hover time for a "frozen" Concept C configuration with a VTO weight held constant at 15,300 pounds, and then held constant again at 16,000 pounds. For the first case, the solid line indicates the reduction in hover time as the weight empty increases, and the increase in hover time as the weight empty decreases for instances where more fuel can be added. For the second case, the broken lines indicate the effect of permitting a gradual change in VTO weight (holding fuel quantity constant) up to a maximum of 16,000 pounds and then offloading fuel for further increases in weight empty. The latter case corresponds to a reduction of 3% in the required thrust-to-weight ratio for the maximum hover time design point.

2. Effect of Weight Growth at Constant Hover Endurance

Figure V-3 shows how the Concept C VTO gross weight (for constant hover time) is affected when changes in operating weight empty occur. For example, if this weight empty were to increase by 10% (1100 pounds), the take-off gross weight would increase approximately 2200 pounds. The operating weight empty is defined as the VTO gross weight less fuel and 800 pounds of payload.

CONCEPT C
EFFECT OF WEIGHT GROWTH ON HOVER TIME
8 J85 ENGINES

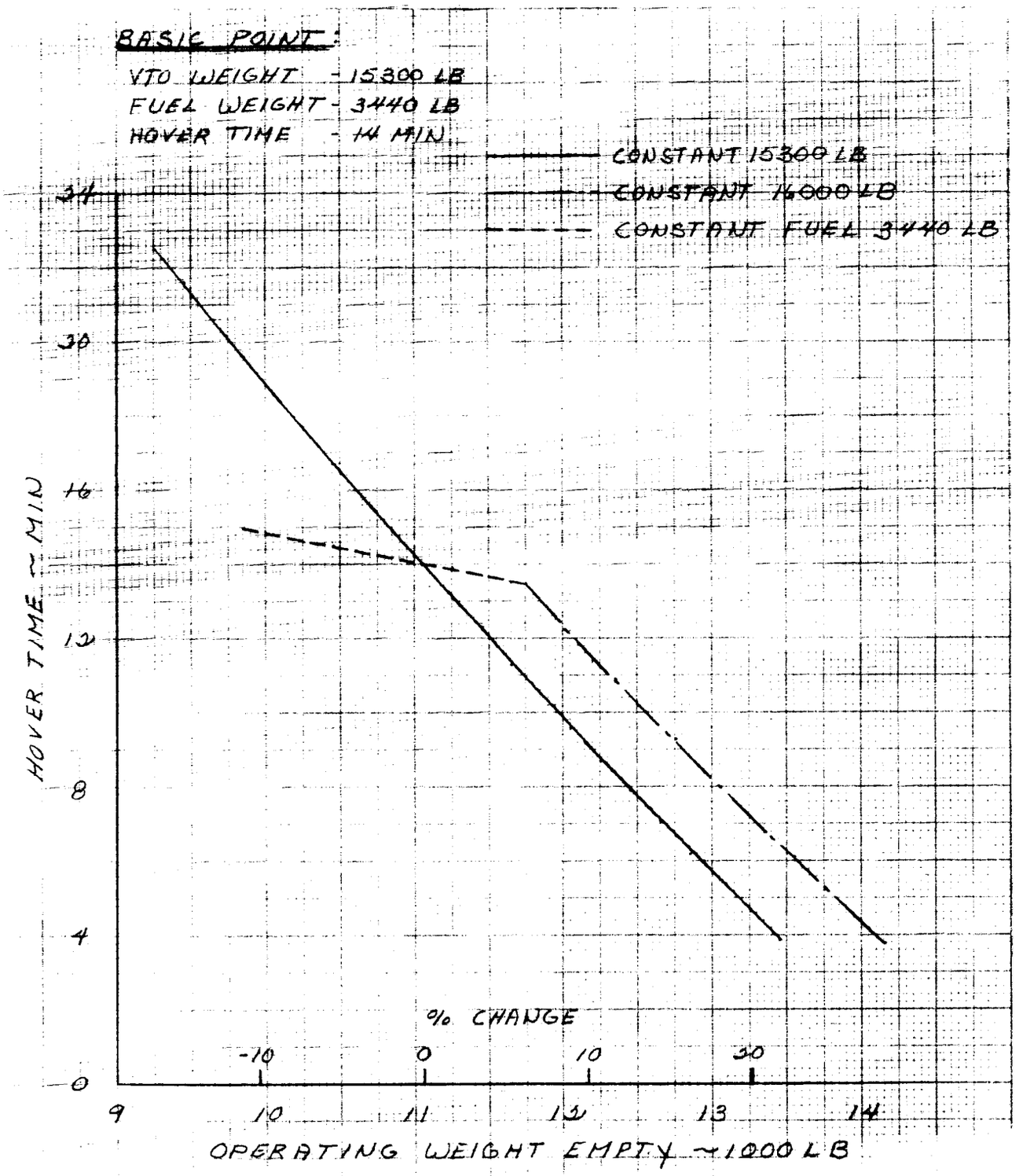

Figure V-2

CONCEPT C
CONSTANT PERFORMANCE WEIGHT GROWTH

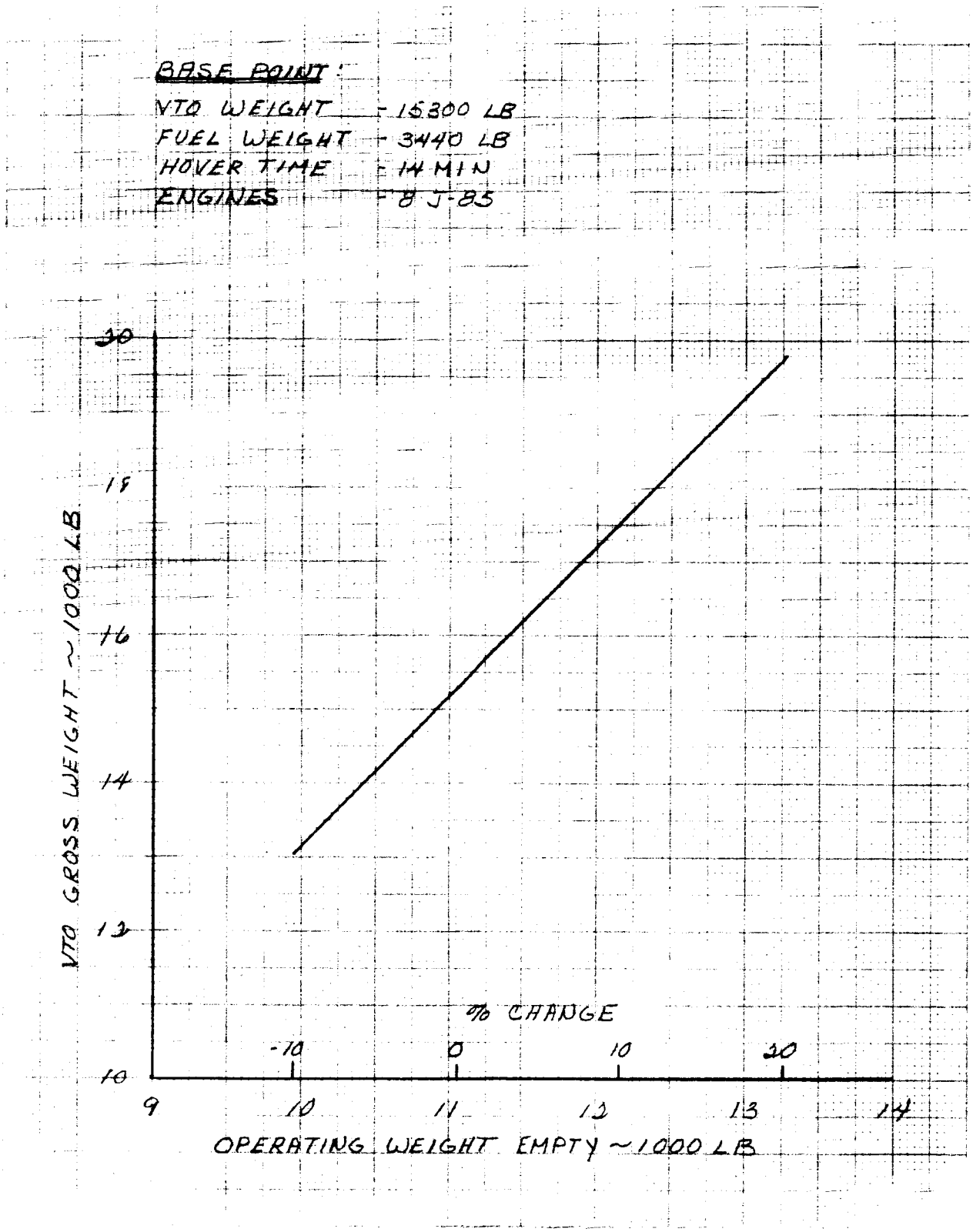

Figure V-3

CONCEPT J

A. WEIGHT, BALANCE, AND MOMENTS OF INERTIA

The following presents the results of a preliminary weights analysis of the selected Concept J design. Group weight statements, mass moments of inertia, and center of gravity diagrams are included.

1. Summary Group Weight Statement

The Summary Group Weight Statement is given in **Table** V-2.

2. Group Weight Statement

The Group Statement is presented below.

3. Center-of-Gravity Travel Envelope

Figure V-4 shows the center-of-gravity travel of several typical loading conditions for a Concept J design configuration with ten YJ85-GE-19 engines. The center-of-gravity envelope maintains reasonable symmetry with reference to the center of combined engine thrusts, which indicates the feasibility of normal fuel sequencing.

Loading conditions presented are:

- In flight with either one or two crewmen
- In flight with 800 pounds down to 500 pounds of ballast

The envelope was calculated for a weight range from full fuel to zero fuel without considering ballast.

4. Moments of Inertia

Table V-3 presents the moments of inertia for the VTO and fuel empty weight conditions.

D. CENTER-OF-GRAVITY TRAVEL ENVELOPE

Figure V-1 presents the center-of-gravity envelope for the Concept C design with eight YJ85-GE-19 engines. The travel range is based on normal fuel consumption under the following conditions:

- Flight with either or both pilots (The single pilot is assumed to be in the front cockpit)
- Flight with either 500 pounds or 800 pounds of payload

Ballast was not considered in calculating the envelope.

TABLE V-2
CONCEPT J
SUMMARY GROUP WEIGHTS
10 J-85 ENGINES ~ 12 MINUTES HOVER

	POUNDS
STRUCTURE	(5907)
WING	1978
TAIL	297
BODY	2740
LANDING GEAR	892
PROPULSION & NACELLE	(5887)
ENGINES	4090
AIR INDUCTION	170
EXHAUST	695
FUEL SYSTEM	308
ENGINE CONTROLS	80
STARTING	70
NACELLE/ENGINE SECTION	474
POWER SYSTEM	(1573)
SURFACE CONTROLS	582
VTO CONTROLS	420
HYDRAULIC & PNEUMATIC	157
ELECTRICAL	414
EQUIPMENT GROUPS	(827)
INSTRUMENTS & NAVIGATION	150
ELECTRONICS	155
FURNISHINGS	348
AIR CONDITIONING	144
CONTINGENCY	(297)
WEIGHT EMPTY	14491
CREW	400
OIL, OXYGEN & TRAPPED FUEL	180
PAYLOAD	800
HOVER FUEL	3886
VTO GROSS WEIGHT	19757
RAMP FUEL	324
RAMP GROSS WEIGHT	20081

CONCEPT J
CENTER OF GRAVITY ENVELOPE
(12 MIN. HOVER)

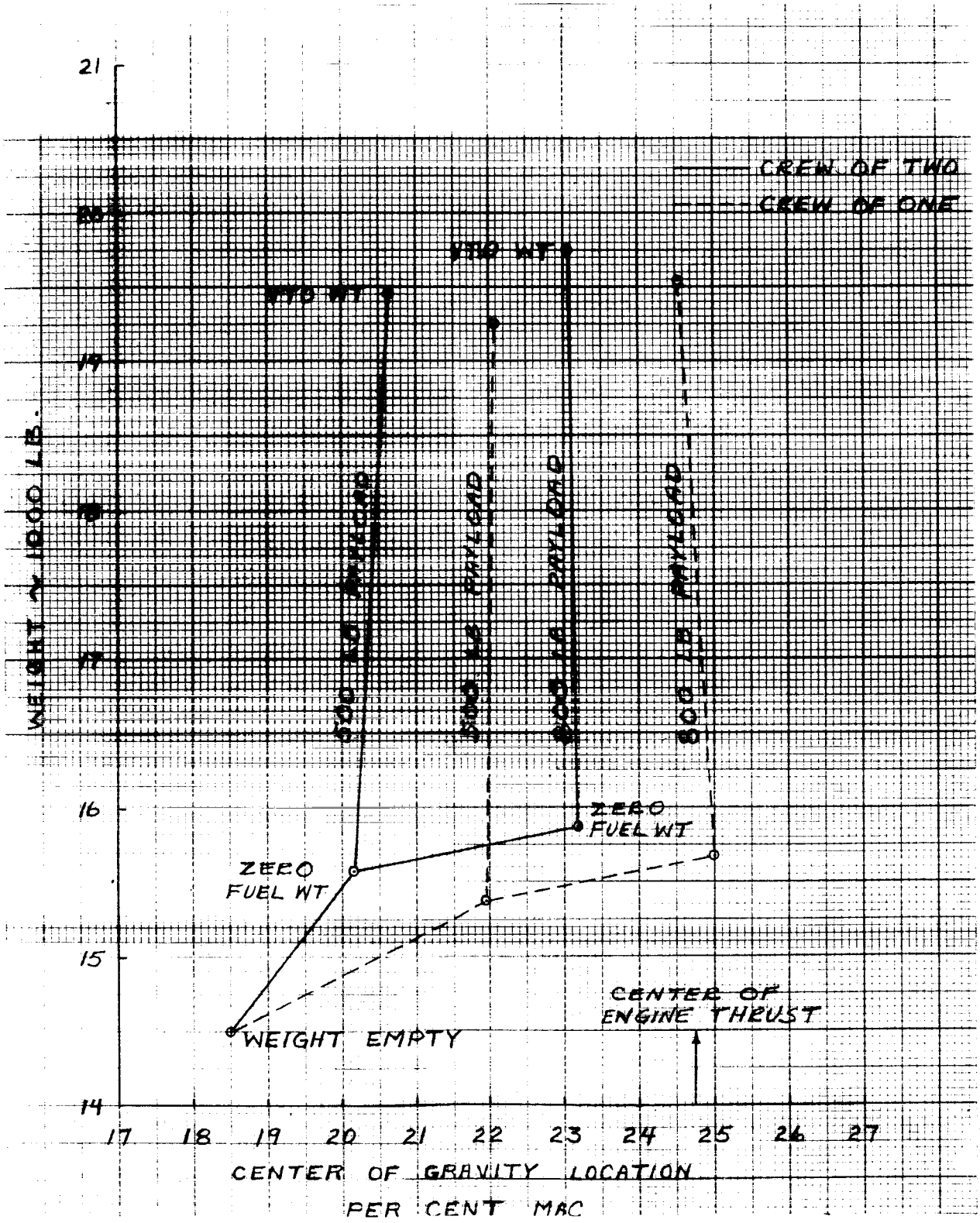

Figure V-4

TABLE V-3

CONCEPT J

MOMENTS OF INERTIA

	V.T.O Condition	Zero Fuel Condition
Weight	19,757 lbs	15,871 lbs
Center-of-Gravity	23.1 % M.A.C.	23.2 % M.A.C.
I_x (Roll)	15,956 Slug-ft^2	15,099 Slug-ft^2
I_y (Pitch)	47,951 Slug-ft^2	39,269 Slug-ft^2
I_z (Yaw)	58,825 Slug-ft^2	50,407 Slug-ft^2

B. GROWTH ANALYSIS

Figure V-5 shows the effect of operating weight empty growth on hover time for the VTO weight held constant at 19,760 pounds. The broken lines indicate the effect of permitting a gradual growth in VTO weight (as described in Section F. 1. f. 1) to a maximum of 20,000 pounds.

C. WEIGHT SUBSTANTIATION

1. T-39 Breakdown

A side-by-side comparison of the weight breakdown between an existing T-39A aircraft and the new Concept J is given in Table V-4. A weight breakdown for an existing T-39A stripped of unusable systems is also included.

2. VTO Controls Weight Substantiation

A preliminary weight analysis of the VTO control system for Concept J was done on the same basis is for Concept C. The dimensional data and weight summary for a stainless steel design configuration are presented in Tables V-5 and V-6. The weight of the system for Concept J is considerably higher than for Concept C which is due primarily to the larger size requirements. Concept J has a longer fuselage, a greater wing span, and ten engines all of which contribute to additional control force requirements resulting in larger duct areas.

CONCEPT J
EFFECT OF WEIGHT GROWTH ON HOVER TIME
10 J85 ENGINES

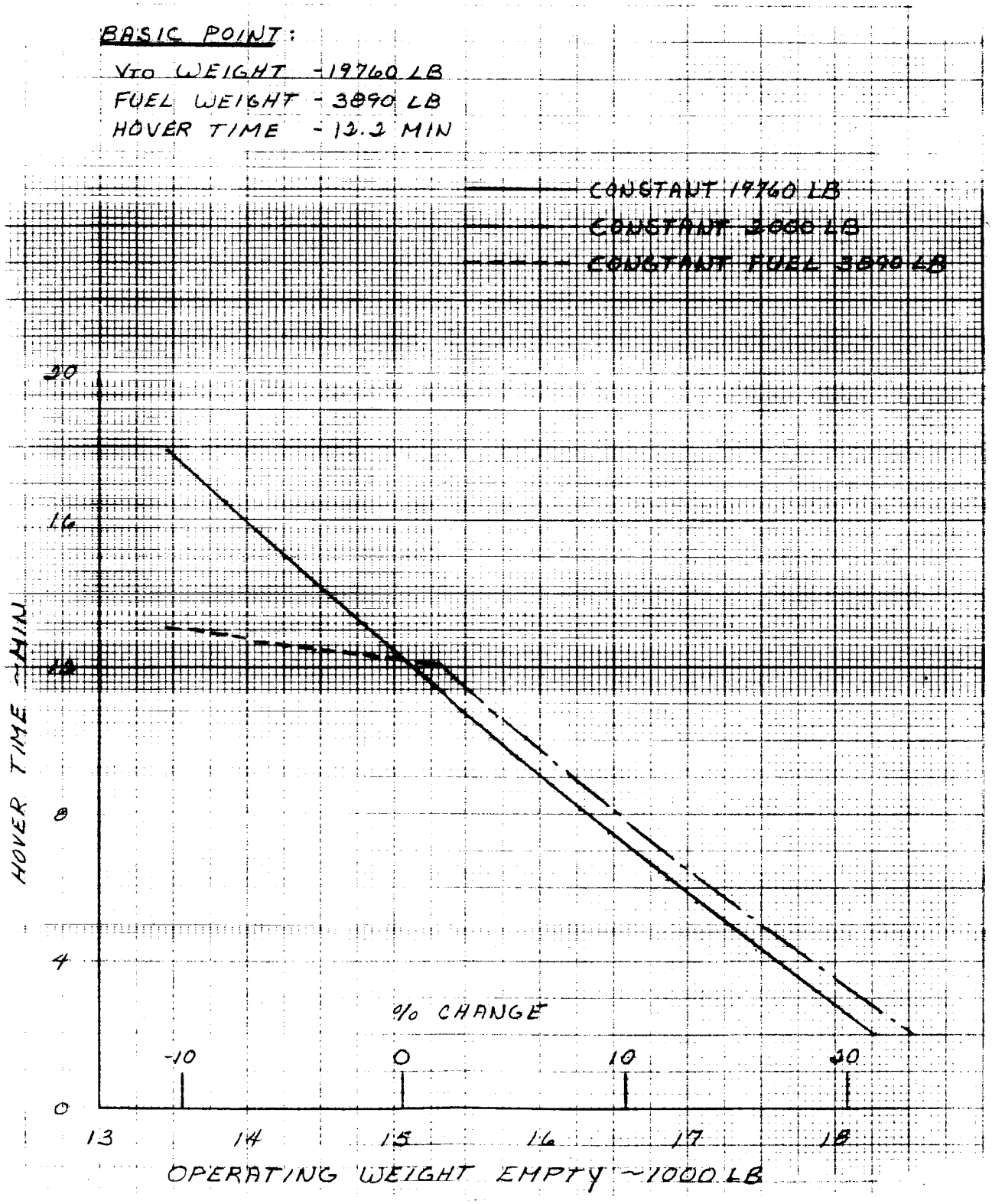

Figure V-5

TABLE V-4

SIDE-BY-SIDE WEIGHT COMPARISON
T-39A vs Concept J

Item	Existing T-39A	Stripped T-39A	Concept J 10 J85 Engs
Wing	1753	1753	1978
Tail	297	297	297
Fuselage	2014	1445	2740
Landing Gear	728	277	892
Surface Controls	344	344	582
VTO Controls	0	0	420
Engine Section/Nacelles	315	0	474
Propulsion Group	(1301)	(0)	(5413)
Eng. Instl. (2)	959	0	4090
Air Induction	12	0	170
Exhaust System	30	0	695
Cooling-Bleed & Drain Prov.	40	0	0
Lub. System-Plumbing	5	0	0
Fuel System	190	0	308
Engine Controls	34	0	80
Starting System	31	0	70
Instruments & Nav. Equip.	122	0	150
Hydraulics & Pneu. Group	116	0	157
Electrical Group	720	0	414
Electronics	407	0	185
Furnishings & Equipment	857	0	348
Air Conditioning & Anti-Icing	333	0	144
Contingency	0	0	297
Weight Empty	9307	4116	14,491
Crew (2)	340		400
Unusable Fuel	65		89
Oil	24		75
Oxygen	0		16
Payload	1160		800

TABLE V-4 (Cont'd)

SIDE-BY-SIDE WEIGHT COMPARISON
T-39A vs Concept J

Item	Existing T-39A	Stripped T-39A	Concept J 10 J85 Engs
Operational Weight Empty	10,896		15,871
Internal Fuel	5,805		3,886
VTO - Gross Weight	16,701		19,757
Ramp Fuel	-		324
Ramp - Gross Weight	-		20,081

TABLE V-5

VTO CONTROLS WEIGHT SUMMARY

Component	Stainless Steel Ducting Mat: 18-8 Type 347
Reaction Control Nozzles	(81)*
Pitch and Yaw	41
Pitch	18
Roll	22
Ducting - Fuselage & Wing	189
Clamps and Supports	56
Flanges	16
Bellows	28
Engine Manifolds & Check Valves	50
Total Weight Per Airplane	420

* Valve parts of the reaction nozzles to be made of titanium.

TABLE V-6

VTO CONTROLS - CONCEPT J

WEIGHT SUBSTANTIATION

Components - Dimensional Data	Weight (lb) (Steel)
Reaction Control Nozzles	(81)
Pitch and Yaw (Aft Fuselage)	41
Pitch Valve Side Plates (2) Mat: Titanium $\bar{t}$= .10	
Valve Floor (1) Mat: Titanium $\bar{t}$= .125	
Ducting, Flanges & Hardware-Steel	
Incr. Nozzle wt. of Concept C:(12.0)(1.344)=16.1 #	
Yaw Valve Side Plates (2) Mat: Titanium $\bar{t}$= .10	
Valve Floors (2) Mat: Titanium $\bar{t}$= .125	
Ducting, Flanges & Hardware-Steel	
Incr. Nozzle wt. of Concept C:(18.4)(1.344)=24.7 #	
Pitch Control (Forward Fuselage)	18
Incr. Nozzle wt. of Concept C:(12.0)(1.471)=17.7 #	
Roll Control Nozzle (2) (Wing)	22
Incr. Nozzle wt. of Concept C:(8.0)(2)(1.392) =22.3 #	
Ducting - Stainless Steel, t = .015, ρ = .288	(189)
Fuselage: Dia. = 9.75 in. L = 274 in. - one duct	36
Dia. = 10.5 in. L = 306 in. - one duct	44
Wing: Dia. = 5.5 in. L = 366 in. - two ducts	55
Dia. = 3.75 in. L = 266 in. - four ducts	54
Clamps and Supports	(56)
Average Spacing Assumed = 24 inches	
Clamps: Marman Type - wt. varies with Diameter	
Supports: $\bar{t}$= .025 Ave. Unit wt. = .017 # (+ 5% contingency)	
Fuselage: (24) (clamp wt. = 4.3 #; support wt. = 14.0 #)	18
Wing: (75) (clamp wt. = 24.5; support wt. = 13.4 #)	38

TABLE V-6 (Cont'd)

VTO CONTROLS - CONCEPT J

WEIGHT SUBSTANTIATION

Components - Dimensional Data	Weight (lb) (Steel)
Flanges	(16)
Spacing = 24 in. $\bar{t}$ = .025 (+ 5% contingency)	
Fuselage: Wt. = 6.9 # Wing: Wt. = 9.4 #	
Bellows	(28)
Average Spacing = 48 in. $\bar{t}$ = .020	
Fuselage: Wt. = 11.7 # Wing: Wt. = 16.1 #	
Engine Manifolds & Check Valves	(50)
Check Valve Unit Wt. = 0.8 #	
Manifold Lines Between Engines (Excluding Starting Sys.)	
Steel - Ave. Diameter = 3.5 in. Wt. Per Eng. = 4.2 #	
Total Wt. = (0.8 + 4.2) (10) = 50.0 #	

GROUP WEIGHT STATEMENT

ESTIMATED [redacted]

(Cross out those not applicable)

CONTRACT NO. NASA V/STOL - Concept J

AIRPLANE, GOVERNMENT NO. Modified T-39A

AIRPLANE, CONTRACTOR NO. 12 Minutes Hover

MANUFACTURED BY Republic Div. Fairchild Hiller

		LIFT/CRUISE	LIFT
ENGINE	MANUFACTURED BY	General Electric	General Electric
	MODEL	J85-GE-19	J85-GE-19
	NO.	2	8
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

AN-9103-D
NAME
DATE

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 2
MODEL Concept J
REPORT -A30J

1	WING GROUP		1978
2	CENTER SECTION - BASIC STRUCTURE		
3	INTERMEDIATE PANEL - BASIC STRUCTURE		
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS LBS.)	1627	
5			
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)	61	
7	AILERONS (INCL. BALANCE WEIGHT 15 LBS.)	69	
8	FLAPS - TRAILING EDGE	104	
9	- LEADING EDGE		
10	SLATS	117	
11	SPOILERS		
12	SPEED BRAKES		
13			
14			
15	TAIL GROUP		297
16	STABILIZER - BASIC STRUCTURE	113	
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.)	70	
18	SECONDARY STRUCTURE (STAB. & FINS)	9	
19	ELEVATOR (INCL. BALANCE WEIGHT 19 LBS.)	68	
20	RUDDERS (INCL. BALANCE WEIGHT 8 LBS.)	37	
21			
22			
23	BODY GROUP		2740
24	FUSELAGE OR HULL - BASIC STRUCTURE		
25	BOOMS - BASIC STRUCTURE		
26	SECONDARY STRUCTURE - FUSELAGE OR HULL		
27	- BOOMS		
28	- SPEEDBRAKES		
29	- DOORS, PANELS & MISC.		
30			
31	ALIGHTING GEAR GROUP - LAND (TYPE:)		892

32 / 33	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS	
34	Wing - Main Gear	214	339	133	686
35	Fwd. Fus. - Nose Gear	33	104	44	181
36	Wing - Skids		25		25
37					
38					
39					

40	ALIGHTING GEAR GROUP - WATER				
41	LOCATION	FLOATS	STRUTS	CONTROLS	
42					
43					
44					
45					

46	SURFACE CONTROLS GROUP		1002
47	COCKPIT CONTROLS	73	
48	AUTOMATIC ~~PILOT~~ (Stability Augmentation)	40	
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS 198 LBS.)	469	
50	VTO Controls (Ducting and Nozzles)	420	
51	ENGINE SECTION OR NACELLE GROUP		474
52	INBOARD	374	
53	CENTER		
54	OUTBOARD		
55	DOORS, PANELS & MISC. (Mlg Bay)	100	
56			
57	TOTAL (TO BE BROUGHT FORWARD)		7383

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GROUP WEIGHT STATEMENT
WEIGHT EMPTY

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1	PROPULSION GROUP				5413
2		LIFT (8)		LIFT/CRUISE (2)	
3	ENGINE INSTALLATION J85-GE-19		3096	994	
4	AFTERBURNERS (IF FURN. SEPARATELY)				
5	ACCESSORY GEAR BOXES & DRIVES				
6	SUPERCHARGERS (FOR TURBO TYPES)				
7	AIR INDUCTION SYSTEM		120	50	
8	EXHAUST SYSTEM		496	199	
9	COOLING SYSTEM				
10	LUBRICATING SYSTEM				
11	TANKS				
12	COOLING INSTALLATION				
13	DUCTS, PLUMBING, ETC.				
14	FUEL SYSTEM			308	
15	TANKS - PROTECTED				
16	- UNPROTECTED				
17	PLUMBING, ETC.				
18	WATER INJECTION SYSTEM				
19	ENGINE CONTROLS		64	16	
20	STARTING SYSTEM		56	14	
21	PROPELLER INSTALLATION				
22					
23					
24	AUXILIARY POWER PLANT GROUP				
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP				150
26	HYDRAULIC & PNEUMATIC GROUP				157
27					
28					
29	ELECTRICAL GROUP				414
30					
31					
32	ELECTRONICS GROUP				185
33	EQUIPMENT			145	
34	INSTALLATION			40	
35					
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)				
37	FURNISHINGS & EQUIPMENT GROUP				348
38	ACCOMMODATIONS FOR PERSONNEL			248	
39	MISCELLANEOUS EQUIPMENT				
40	FURNISHINGS			38	
41	EMERGENCY EQUIPMENT			62	
42					
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP				144
44	AIR CONDITIONING			140	
45	ANTI-ICING			4	
46					
47	PHOTOGRAPHIC GROUP				
48	AUXILIARY GEAR GROUP				
49	HANDLING GEAR				
50	ARRESTING GEAR				
51	CATAPULTING GEAR				
52	ATO GEAR				
53					
54	Contingency				297
55	MANUFACTURING VARIATION				
56	TOTAL FROM PG. 2				7383
57	WEIGHT EMPTY				14491

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GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

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1	LOAD CONDITION			VTO		RAMP	
2							
3	CREW (NO. 2)			400		400	
4	PASSENGERS (NO.)						
5	FUEL	Type	Gals.				
6	UNUSABLE	JP-4	16	89		89	
7	INTERNAL	JP-4	598 + 50	3886		4210	
8							
9							
10	EXTERNAL						
11							
12	BOMB BAY						
13							
14	OIL			75		75	
15	TRAPPED						
16	ENGINE						
17							
18	FUEL TANKS (LOCATION)						
19	WATER INJECTION FLUID (GALS)						
20							
21	BAGGAGE						
22	CARGO						
23							
24	ARMAMENT						
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.			
26							
27							
28							
29							
30							
31							
32	AMMUNITION						
33							
34							
35							
36							
37							
38							
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)						
*40	BOMB OR TORPEDO RACKS						
41							
42							
43	Oxygen			16		16	
44							
45							
46	EQUIPMENT			800		800	
47	PYROTECHNICS						
48	PHOTOGRAPHIC						
49							
*50	OXYGEN						
51							
52	MISCELLANEOUS						
53							
54							
55	USEFUL LOAD						
56	WEIGHT EMPTY			14491		14491	
57	GROSS WEIGHT			19757		20081	

*If not specified as weight empty.

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GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

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1	LENGTH - OVERALL (FT.) 46.91			HEIGHT - OVERALL - STATIC (FT.)				
2		Main Floats	Aux. Floats	Booms	Fuse or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
3	LENGTH - MAX. (FT.)				46.91			
4	DEPTH - MAX. (FT.)				6.60			
5	WIDTH - MAX. (FT.)				5.75			
6	WETTED AREA (SQ. FT.)				677.50			
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							
8	FUSELAGE VOLUME (CU. FT.)		PRESSURIZED	0		TOTAL		
9						Wing	H. Tail	V. Tail
10	GROSS AREA (SQ. FT.)					342.1	77.0	41.6
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)					5.78	2.42	2.65
12	SPAN (FT.) AR = 5.8					44.64	17.55	7.66
13	FOLDED SPAN (FT.)							
14								
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES)					28.6	30.0	30.0
16	- AT % CHORD LINE (DEGREES)							
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES) $t/C_R = 11.3\%$					139.92	81.22	94.54
18	- MAX. THICKNESS (INCHES)					15.81	8.11	9.44
***19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)							
20	- MAX. THICKNESS (INCHES)							
***21	THEORETICAL TIP CHORD - LENGTH (INCHES) $t/C_T = 9.4\%$					44.91	24.37	28.32
22	- MAX. THICKNESS (INCHES)					4.20	2.43	2.83
23	DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)							
24	TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.)							
25	AREAS (SQ. FT.) Flaps	L.E.		T.E. 40.3				
26	Lateral Controls	Slots 36.3		Spoilers		Ailerons 16.4		
27	Speed Brakes	Wing		Fuse. 7.6				
28								
29								
30	ALIGHTING GEAR (LOCATION)				Nose	Main		
31	LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)							
32	OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)							
33	FLOAT OR SKI STRUT LENGTH (INCHES)							
34	ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)							
35	HYDRAULIC SYSTEM CAPACITY (GALS.)							
36	FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected		No. Tanks	****Gals. Unprotected	
37	Fuel - Internal	Wing						
38		Fuse.				2	660	
39	- External							
40	- Bomb Bay							
41								
42	Oil	Engine					10	
43								
44								
45	STRUCTURAL DATA - CONDITION			Fuel in Wings (Lbs.)		Stress Gross Weight		Ult. L.F.
46	FLIGHT			0		19757		4.5
47	LANDING							
48								
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL							
50	CATAPULTING							
51	MIN. FLYING WEIGHT							
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)							
53	WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)							
54	STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)							
55	PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)							
56								
57	AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)							

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

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