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CONCEPTUAL POINT DESIGN STUDY OF A NEW CTOL SETOLS CAS AIRCRAFT--ETC(U)
JUN 79 P D SORENSEN, R L BAYLESS, E F NOEL N62269-79-C-0438
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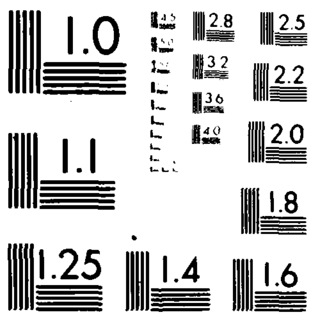

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FINAL REPORT

CONCEPTUAL POINT DESIGN STUDY OF A NEW CTOL SETOLS CAS AIRCRAFT FOR 1995 IOC

Report No. NADC-78155-20

SANDAIRE

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San Diego Aircraft Engineering, Inc.
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FINAL REPORT

CONCEPTUAL POINT DESIGN STUDY OF A NEW CTOL SETOLS CAS AIRCRAFT FOR 1995 IOC

REPORT NO. NADC-78155-20

June 20, 1979

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Contract N62269-79-C-0438
Dated December 29, 1978

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A conceptual design of a CTOL SETOLS CAS aircraft has been developed. The significant design feature is the use of a Surface Effect Takeoff and Landing System (SETOLS) in lieu of a normal landing gear. The SETOLS is an integral part of the point design, thereby achieving full design compatibility. This feature provides a takeoff and landing capability, on the inflated rubber fabric type trunk installed on the bottom of the fuselage, from any unobstructed area such as water, grass, soil, etc. Effective operation is thereby (over)

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achieved with a Conventional Takeoff and Landing (CTOL) aircraft design in the Close Air Support (CAS) mission from forward bases where time prevents construction and repair of long, hard surface runways. Advanced state-of-the-art design, appropriate for 1995 IOC, has been incorporated. One advanced technology Pratt & Whitney STF 529 turbofan is used for propulsion and trunk pressurization.

FOREWORD

This document is the Final Report required by Contract N62269-79-C-0438 dated December 29, 1978. Two oral progress reports were given on March 6, 1979 and May 10, 1979 to designated Navy personnel, at the Contractor's plant, as required by contract.

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INTRODUCTION

The Navy currently is defining a series of V/STOL (Vertical/ Short Takeoff and Landing) aircraft that could satisfy several Navy Missions, including Close Air Support (CAS), in the post-1990 time frame. The CAS mission requires operation from forward bases with minimum facilities. Prepared hard surface runways will not be available, and the mandatory military requirements of fast base establishment and relocation will not allow time to prepare, maintain, and protect such runways.

Utilization of a Surface Effect Takeoff and Landing System (SETOLS) will provide an operational capability from unobstructed areas, that require only minimum preparation, and the resulting aircraft should be attractive when compared to VTOL (Vertical Takeoff and Landing) and STO/VL (Short Takeoff/Vertical Landing) aircraft that have a forward base capability.

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SUMMARY

A conceptual design of a CTOL SETOLS CAS Aircraft has been developed. The significant design feature is the use of a Surface Effect Takeoff and Landing System (SETOLS) in lieu of a normal landing gear. The SETOLS is an integral part of the point design, thereby achieving full design compatibility compared to an add-on which is typical of flight test work to date. This feature provides a takeoff and landing capability on the inflated rubber fabric type trunk installed on the bottom of the fuselage from any unobstructed area, such as a river, lake, swamp, grass, soil, etc. Effective operation is thereby achieved in the Close Air Support (CAS) mission from forward bases where time prevents preparation and maintenance of conventional runways. The design is, therefore, a Conventional Takeoff and Landing (CTOL) aircraft with all the inherent design advantages of low weight and low cost compared to V/STOL (Vertical/Short Takeoff and Landing) aircraft.

Advanced state-of-the-art design appropriate for 1995 IOC has been incorporated. This consists of use of composite structure to reduce weight, 10% for the wing, 25% for the tail, and 15% for the fuselage. Advanced NASA airfoil technology is in the wing design to allow the use of thicker wing sections to save weight without sacrifice of a high performance capability.

One advanced technology Pratt & Whitney STF 529 turbofan, with 13202 pounds thrust, a thrust to weight ratio of 8.2, and other favorable characteristics, powers the aircraft. A P&W designed peripheral fan bleed is used to inflate and pressurize the trunk. This engine, if funded, will meet the MQT (Military Qualifications Test) requirements for availability by fiscal year 1985.

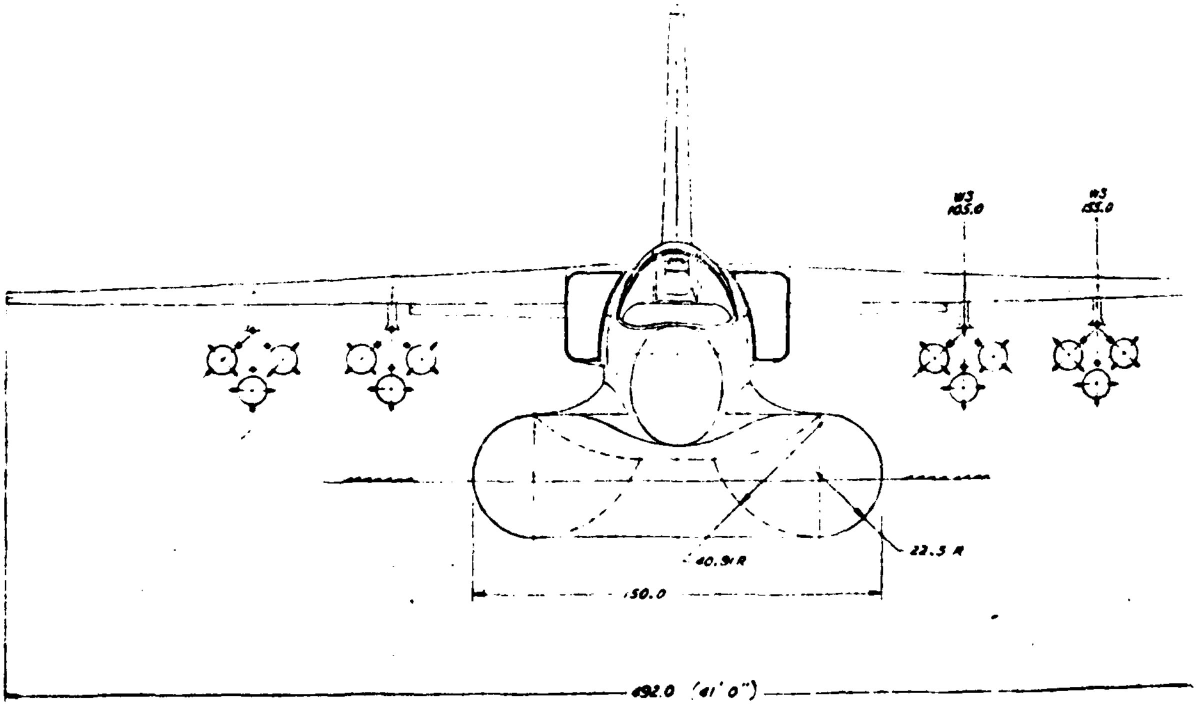
The aircraft has a high wing in order to carry twelve Mk-82 Snakeye bombs (6840 lb droppable weight) below the wing in the specified CAS mission radius of 160 N.M. Sufficient internal fuel capacity is provided for the specified 2500 N.M. ferry mission. The high speed is M = 0.89 at 35000 feet, and a maximum sustained maneuver load factor of 4g is achieved at M = .70 at 5000 ft. Gross weight is 24,300 lb; wing area 280 sq ft; takeoff speed 138 knots; and takeoff run 2945 ft at sea level, 89.8°F (Navy Tropical Day). Mission Profiles are on Pages 14 and 16.

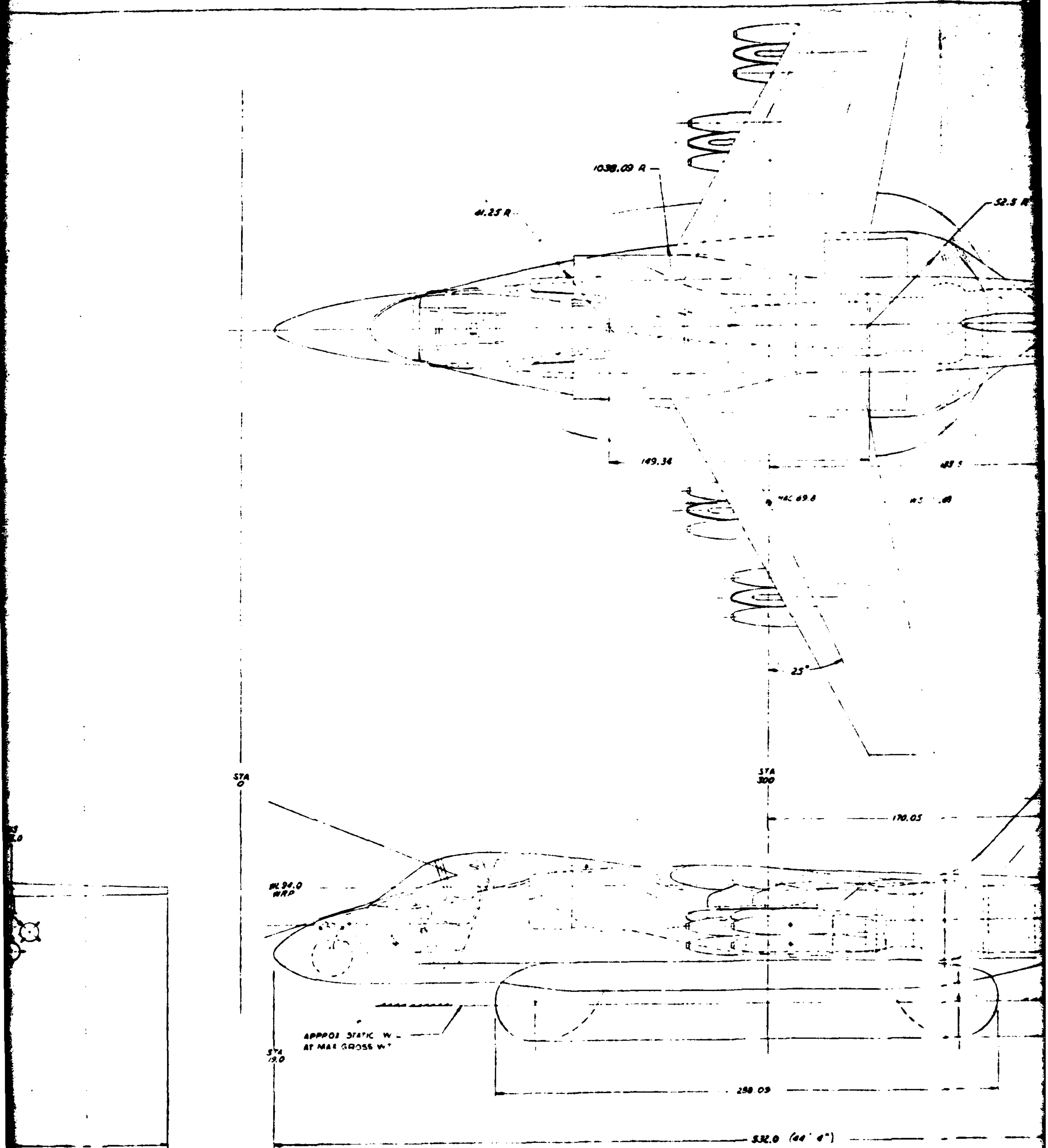
The design is shown by the following three-view and inboard profile drawings. Design features, weight and performance results are summarized briefly in the following Design Analysis Section. Supporting data and calculations are presented in six attached appendices.

Preliminary work summarized in Appendix E shows the design should have one engine versus two engines, engine fan bleed instead of an auxiliary power unit to pressurize the trunk, and the advanced P&W STF 529 turbofan as the best of the candidate engines.

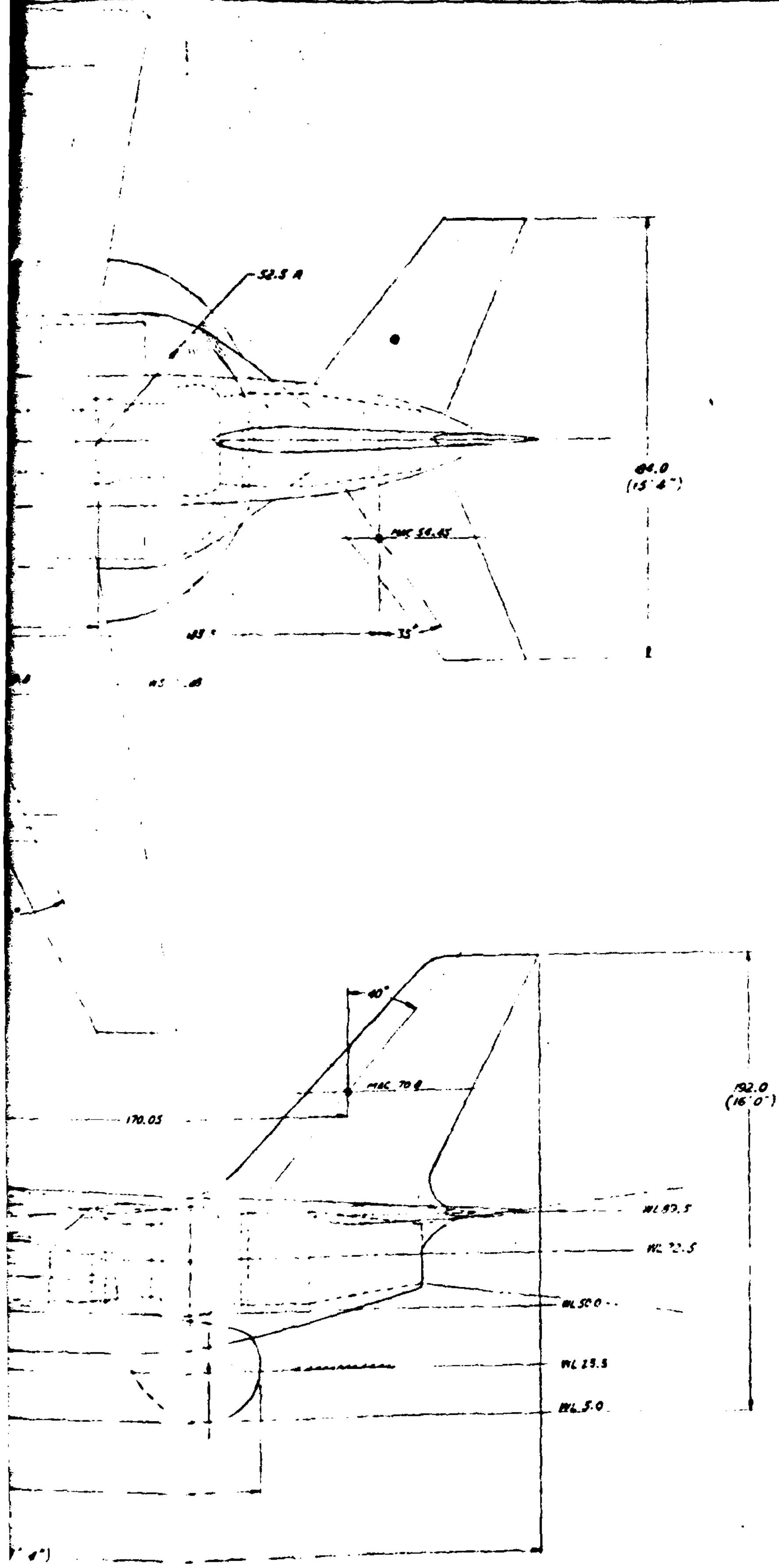
CHARACTERISTICS	WING	H. TAIL	V. TAIL
AREA (SQ FT)	[illegible]	[illegible]	47.5
ASPECT RATIO	6.0	3.5	1.5
SPAN (FT)	41	15.33	8.44
ROOT CHORD (IN)	125.08	[illegible]	[illegible]
TIP CHORD (IN)	37.5	35.0	45.0
TAPER RATIO	.3	.5	.5
MAC (IN)	89.8	54.45	[illegible]
SWEEP (DEG)	25	35	[illegible]
T/C ROOT (%)	12	12	12
T/C TIP (%)	10	10	10

GROSS WT	(LBS)	24300
CUSHION PERIMETER	(FT)	49.528
CUSHION AREA	(SQ FT)	150
CUSHION PRESS	(LBS/SQ FT)	162
TRUNK PRESS	(LBS/SQ FT)	360
AIR FLOW	(LBS/SEC)	39




1038.09 R
41.25 R
52.5 R
149.34
MAC 69.8
25°
STA 0
STA 300
170.05
WL 94.0 WRP
APPROX STATIC WL AT MAX GROSS WT
STA 19.0
298.09
332.0 (44' 4")

2

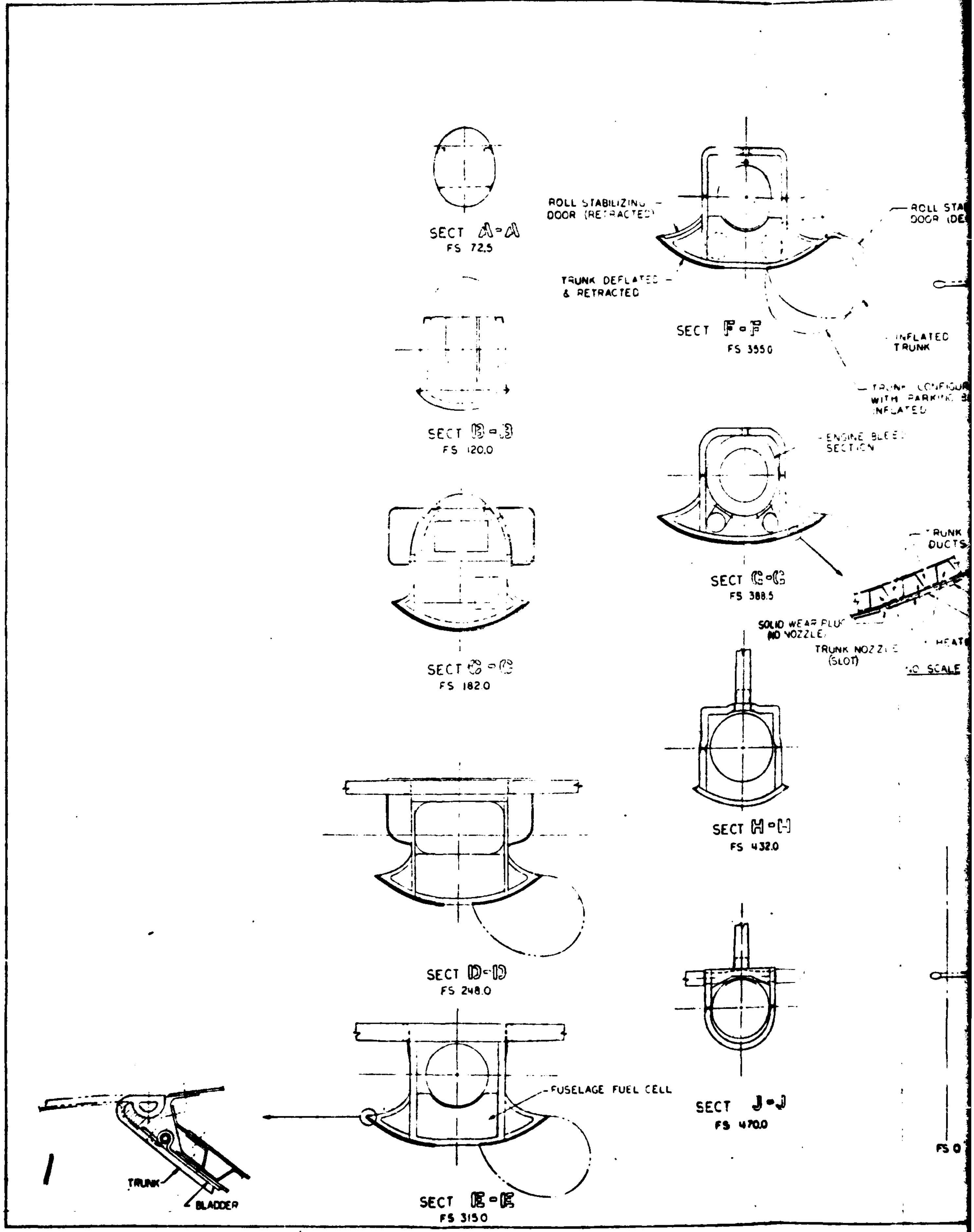


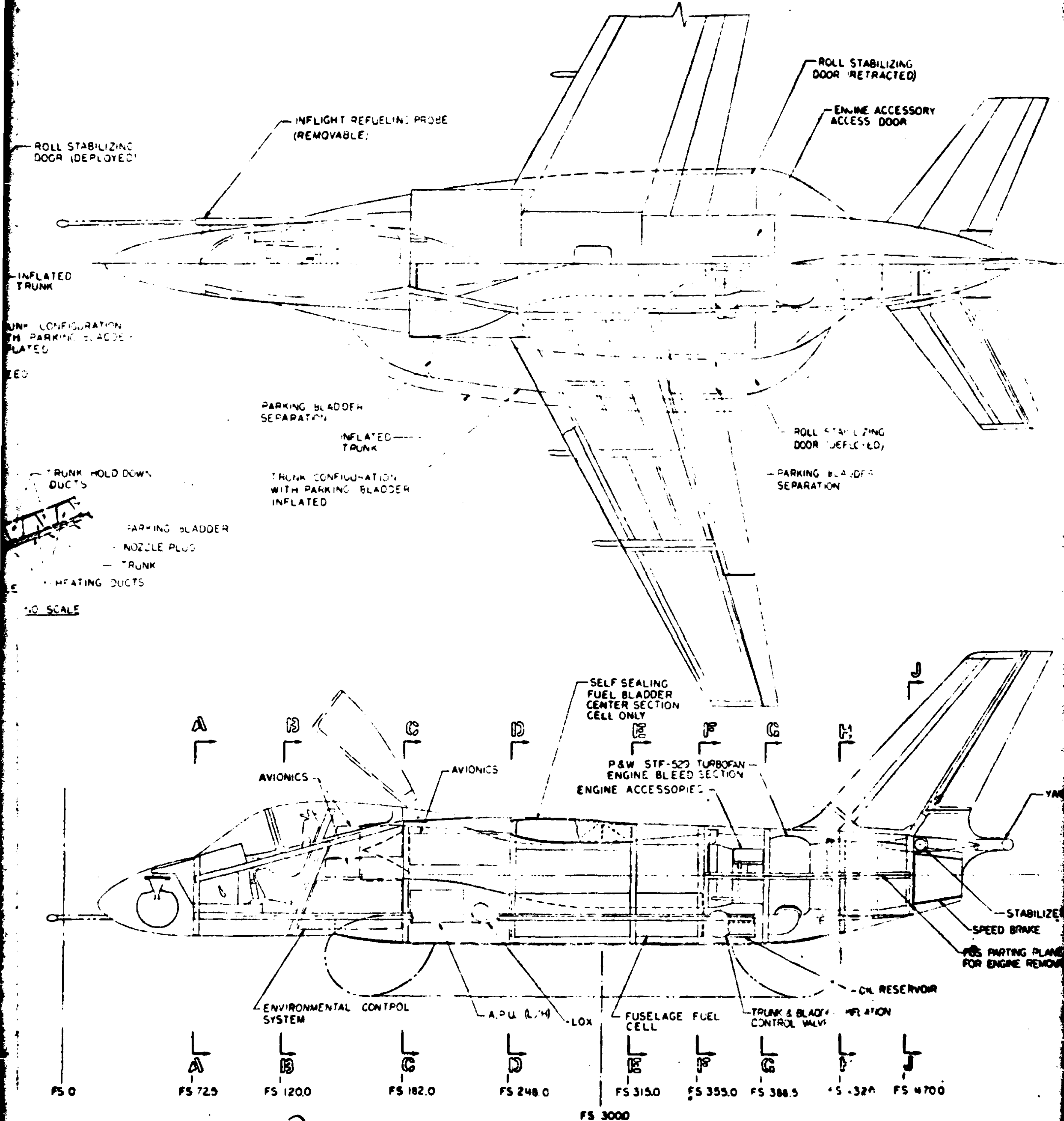
SAN DIEGO AIRCRAFT ENGINEERING, INC.

GENERAL ARRANGEMENT - CAS SETOLS AIRCRAFT DESIGN CONCEPT

25727

SAE-79-007


SECT A-A
FS 72.5
SECT B-B
FS 120.0
SECT C-C
FS 182.0
SECT D-D
FS 248.0
SECT E-E
FS 315.0
FUSELAGE FUEL CELL
TRUNK
BLADDER
ROLL STABILIZING DOOR (RETRACTED)
TRUNK DEFLATED & RETRACTED
SECT F-F
FS 355.0
INFLATED TRUNK
ENGINE BLEED SECTION
SECT G-G
FS 388.5
SOLID WEAR PLUG (NO NOZZLE)
TRUNK NOZZLE (SLOT)
SECT H-H
FS 432.0
SECT J-J
FS 470.0



ROLL STABILIZING DOOR (DEPLOYED)
INFLIGHT REFUELING PROBE (REMOVABLE)
ROLL STABILIZING DOOR (RETRACTED)
ENGINE ACCESSORY ACCESS DOOR
INFLATED TRUNK
PARKING BLADDER SEPARATION
INFLATED TRUNK
TRUNK CONFIGURATION WITH PARKING BLADDER INFLATED
ROLL STABILIZING DOOR (DEFLECTED)
PARKING BLADDER SEPARATION
TRUNK HOLD DOWN DUCTS
PARKING BLADDER
NOZZLE PLUG
TRUNK
HEATING DUCTS
NO SCALE
SELF SEALING FUEL BLADDER CENTER SECTION CELL ONLY
AVIONICS
AVIONICS
P&W STF-529 TURBOFAN ENGINE BLEED SECTION
ENGINE ACCESSORIES
STABILIZER
SPEED BRAKE
FUS PARTING PLANE FOR ENGINE REMOVAL
OIL RESERVOIR
ENVIRONMENTAL CONTROL SYSTEM
A.P.U. (L/H)
LOX
FUSELAGE FUEL CELL
TRUNK & BLADDER INFLATION CONTROL VALVE
A
B
C
D
E
F
G
H
J
FS 0
FS 72.5
FS 120.0
FS 182.0
FS 248.0
FS 300.0
FS 315.0
FS 355.0
FS 388.5
FS 470.0

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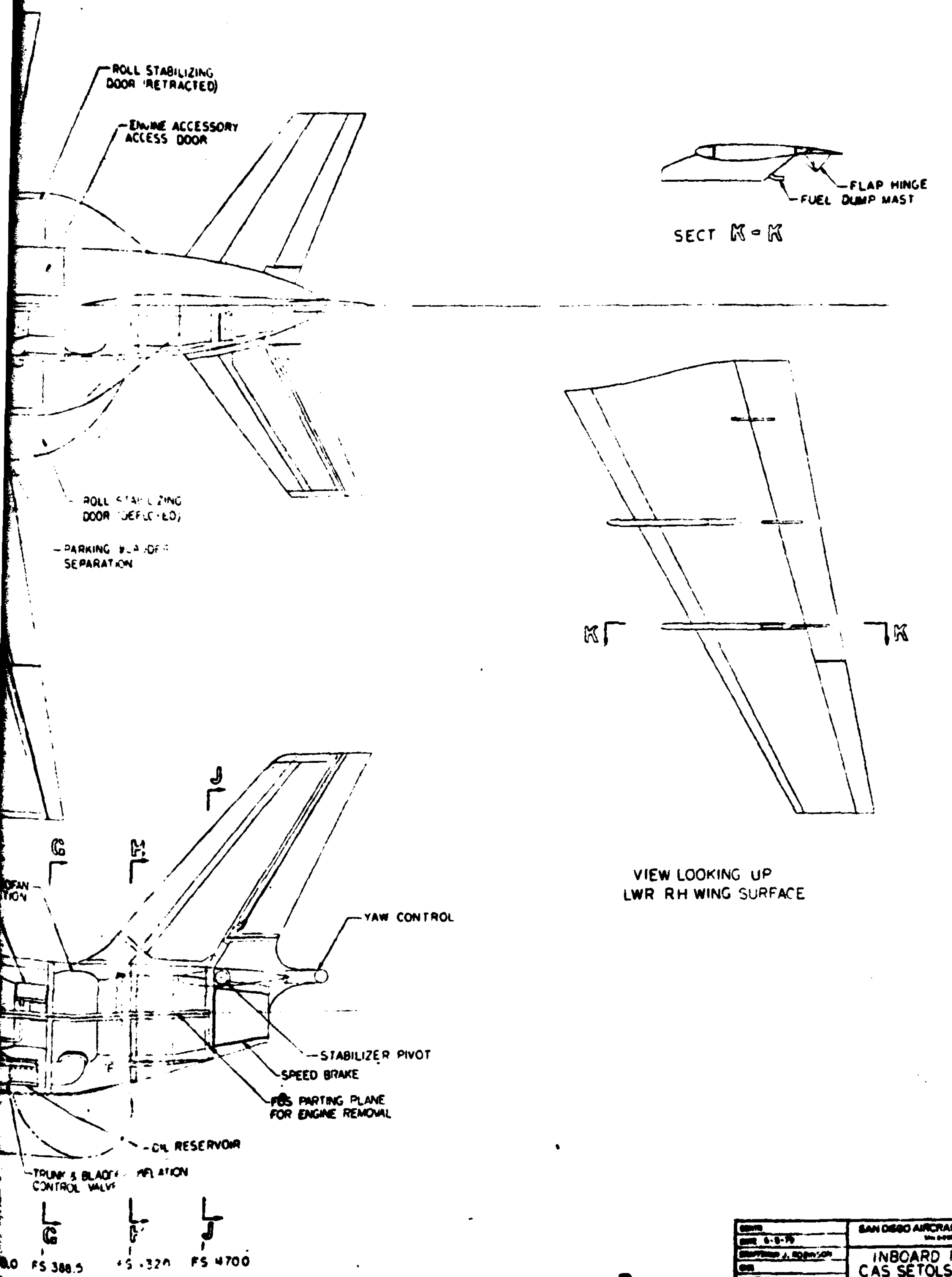

SAN DIEGO AIRCRAFT ENGINEERING, INC

INBOARD PROFILE
CAS SETOLS AIRCRAFT
DESIGN CONCEPT

23737 SAE-79-008

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DESIGN ANALYSIS

(1) PRELIMINARY SIZING STUDIES

Preliminary work to establish the size and configuration using parametric and tradeoff studies is summarized in Appendix E and shows the following:

(a) A two-engine configuration has substantially greater gross weight and size compared to a one engine configuration.

(b) Pressurization of the SETOLS trunk is best accomplished by engine fan bleed compared to use of an auxiliary power unit.

(c) The Pratt & Whitney STF 529 Turbofan is a conceptual engine but is the best of the candidate engines.

(d) A one engine configuration with a high wing and the bottom of the fuselage shaped to support the trunk, both in the pressurized and in the collapsed and stowed conditions, results in the best overall design for weight, simplicity, and design risk.

(2) FINAL CONFIGURATION

The final configuration is shown on the preceding drawings, SAE 79-007 and SAE 79-008, and uses the P&W STF 529 turbofan (scale 1.0) for propulsion and trunk pressurization. P&W quotes this engine as having a high probability of meeting the Military Qualifications Test (MQT), if funded, and be available by FY-1985. There is sufficient internal fuel capacity to accomplish the ferry mission without the use of external tanks. Only the required stores for the CAS mission are carried externally below the wing on four pylons. Rationale and calculations are presented in the following design analysis and supporting appendices to substantiate the results of this conceptual design of a CTOL SETOLS CAS aircraft.

(3) DESIGN REQUIREMENT CHANGES

Changes to the design requirements were made early in the study as follows:

(a) Decrease of the 8,000 foot takeoff run to between 2,000 feet and 4,000 feet which resulted in selection of 3,000 feet for design.

(b) Elimination of the provisions for a 25 mm gun installation.

(c) The high speed requirement of M = 0.91 at 35,000 feet is compromised to M = 0.89 favoring other design goals such as weight, size and minimized propulsion system.

(4) WING SIZING

The selected wing loading was based on the desire to hold the takeoff speed between 135-140 knots in the interest of some conservatism with respect to the fairly new SETOLS state of the art.

For the specified 1995 IOC, use of an advanced airfoil section is appropriate even though only meager data are available. An increase in the Mach No. for drag rise of the order of $\Delta M = .07$ has been incorporated in the drag polars based on mostly qualitative information about these airfoils. The typically blunt airfoil nose shape should preclude the necessity of wing leading edge flaps or slats, to obtain satisfactory stall characteristics, and none are incorporated. Appropriate selection of outboard camber and twist, when airfoil data are available, should produce satisfactory flight characteristics.

The wing sweep of $\Lambda_{c/4} = 0.25^{\circ}$ was selected because of its adequate stall characteristics without the use of wing leading edge devices. The wing configuration resulting from this sweep permits reasonable C.G. control through the placement of stores and inherent fuel management while providing sufficient stores clearance, especially during rotation at take-off and landing. Wing thickness selection then results from the need to meet the high speed requirement at 35,000 feet. The selected wing thickness, $t/c = .12-.10$ root to tip, produces the drag rise shown in Appendix B and results in a small compromise of this requirement to $M = .89$.

Sufficient iteration and design refinement were done in the preliminary work to establish a gross weight of 24,300 pounds for design. Early calculations, prior to receipt of the P&W STF 529 engine data, indicated a considerably higher weight; however, the high engine thrust to weight ratio of 8.2 and other favorable characteristics of the STF 529 gave a substantial reduction in weight down to the 24,300 pounds.

Considering, then, the above 135-140 knot takeoff speed objective, a wing area of 280 square feet gave a takeoff speed of 138 knots at sea level, 89.8°F. This is based on takeoff after 4.5 minutes of fuel has been used of the 5 minutes at maximum thrust specified, as shown in Appendix B.

No extensive numerical analyses were attempted for the selection of wing aspect ratio and taper ratio. Factors bearing on the selection of $AR = 6$, $\lambda = 0.30$ are:

(a) Low span for low weight.
(b) Adequate span to carry the required stores.
(c) High taper for low weight as limited by satisfactory stall characteristics with the selected span.
(d) Admittedly, comparison with previous aircraft also has a significant effect in the selections.
(e) Subsequent weight and mission calculations and the configuration design layouts have qualitatively verified that the selected wing geometry is appropriate for this aircraft design.

The selected high wing configuration is mandatory for the stores to ground/water clearance in takeoff and landing. A wing incidence of 3° is used which works well with respect to fuselage attitude for takeoff, landing, and cruise.

A summary of the wing characteristics is included with the following tail data.

(5) TAIL SIZING

From the configuration design layout work, a conventional tail became appropriate with the horizontal mounted on the fuselage. An all-movable horizontal (no elevator) was considered; however, it was not used pending an in-depth control system analysis which is outside the scope of this study.

Selection of the tail geometry considered the usual factors of:

(a) Displacement of $(\bar{c}/4)_H$ and $(\bar{c}/4)_V$ to prevent adding peak pressures with resultant adverse Mach No. effects. The displacement used, 13.4 inches, is considered a minimum.

(b) Sweep and thickness combination to give a higher critical Mach No. for the tail (for lift) than developed by the wing. Thus, tail effectiveness will be retained after excessive speed warning (buffeting) occurs due to the normal lift deterioration with Mach No. on the wing.

(c) Low span and high taper for low weight as limited by tail effectiveness and past practice.

The following table summarizes the wing and tail characteristics. Calculations are in Appendix B.

		Wing	H. Tail	V. Tail
S	Sq Ft	280	67.1	47.5
AR		6	3.5	1.5
b	Ft	41	15.33	8.44
C_R	In	126.1	70.0	90.0
C_T	In	37.8	35.0	45.0
λ		.30	.50	.50
$\bar{c}$	In	89.8	54.5	70.0
$\Lambda_{c/4}$	Deg	25	35	40
t/c (Root-Tip)		.12-.10	.12-.10	.12-.10
l_t $(\bar{c}/4)_{Wing} - (\bar{c}/4)_{Tail}$	In		183.5	170.1 (Pg. 3)

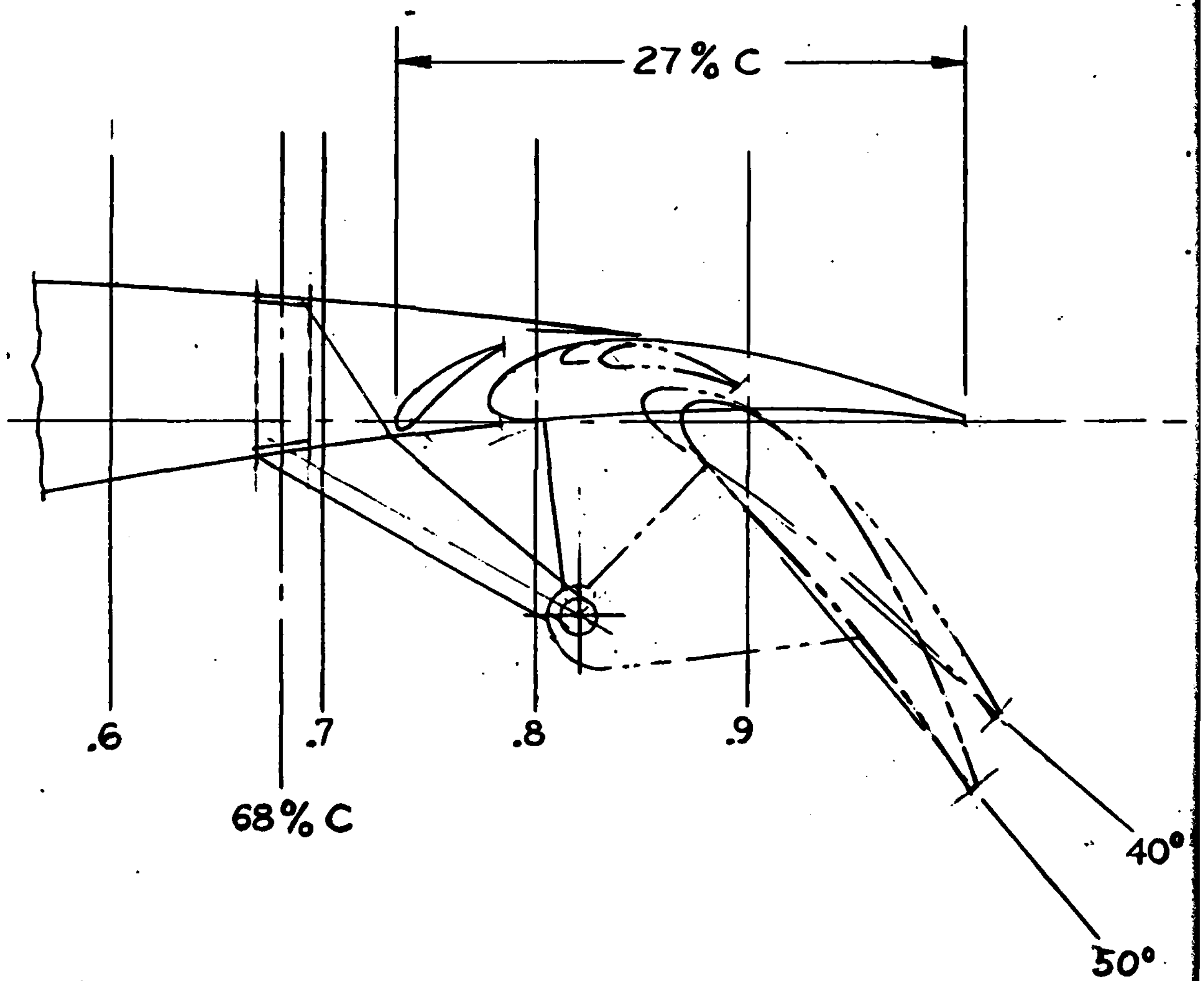
(6) ENGINE FAN BLEED REQUIREMENT

Appendix A presents the SETOLS trunk analysis. Briefly this analysis shows:

(a)	Mean trunk to ground effective clearance in takeoff (daylight gap), *estimated from other data*	0.28 in
(b)	Trunk pressure, design	360 lb/sq ft
(c)	Trunk orifice area exhausting to atmosphere, design	20%
(d)	Trunk shape, design	Pages 3, 28 & 29
(e)	Trunk contact centerline perimeter, design	49.5 ft
(f)	Trunk contact centerline area (cushion area), design	150 sq ft
(g)	Required cushion pressure for the design gross weight of 24,300 lb	162 lb/sq ft
(h)	Required engine fan bleed to pressurize the trunk	39 lb/sec

(7) FLAP CHARACTERISTICS

Due to the reduction in the required takeoff run from 8000 feet to 3000 feet, Page 5, Item (3a), a large flap setting is used to give this shorter run; however, it may not be the best for takeoff over an obstacle. Fixed vane, double-slotted flaps are selected with external hinges. Flaps extend from the fuselage to 70% semispan. The rear wing spar is at 68% chord, which allows use of a 27% chord flap as shown diagrammatically by the following sketch.



Data for flap application to advanced airfoil sections, Item (4) above, are not available. However, flap characteristics are estimated from available data for other flapped airfoil sections as shown in Appendix B.

(8) TAKEOFF PERFORMANCE (89.8°F at Sea Level)

Takeoff ground run is calculated from basic relations and the flap characteristics in Appendix B.

Loading	CAS mission
Takeoff Weight	23686 lb after 4.5 min. fuel has been used as explained on page 7
Flaps	40 Deg
Takeoff Speed	138 knots (page 7)

Engine data are shown in Appendix D. P&W provides a 6% throttle advance for 90° F takeoff at sea level to minimize the adverse effect of a hot day. This overcomes the normal thrust deterioration with this temperature.

The installed thrust is reduced 18.7% (Appendix B) to account for the 39 lb/sec fan bleed to pressurize the trunk.

The major portion of the takeoff run is with fuselage level (trunk level) and $\mu = .05$ (the coefficient of takeoff surface friction) is a representative value based on available data. A factor of 1.3 is applied to give an average $\mu = .065$ to account for greater values of μ at the start of the run and at pull up. See analysis in Appendix A.

The calculated ground run, at 89.8°F, sea level, is 2945 ft.

The air distance over a 50 ft obstacle is calculated from an empirical method that checks well with test data. It includes transition from the level takeoff run to the climb path. The distance is 1272 ft making a total distance over 50 feet of 4217 ft (89.8°F at sea level). Only 2.4 deg rotation from the level takeoff run is required to lift off which should favor smooth operation with the SETOLS.

(9) ENGINE SIZING

The design takeoff run is established as 3000 ft on pg 5, item (3a). Preliminary work indicated that a scale 1.0 P&W STF 529 engine was needed to achieve this distance. The final calculation, App. B, shows 2945 ft using a scale 1.0 engine. Therefore since the preliminary work has indicated that takeoff is the critical requirement for sizing the engine, scale 1.0 is now established as the final engine size. The small compromise of the M = 0.91 high speed requirement discussed on page 5, item (3) and page 6, item (4), would not be significantly improved by any reasonable increase in engine size unless the combination of wing sweep and wing thickness was changed to delay the drag rise with Mach No. This is not considered a justifiable change as previously discussed on pages 5 and 6.

(10) LANDING PERFORMANCE

Landing is calculated in Appendix B for the maximum landing design gross weight, (MLDGW) = gross weight minus 60% CAS mission fuel of 4552 lb for standard day at sea level.

$$MLDGW = 24300 - .60 \times 4552 = 21569 \text{ lb}$$

Flaps are 50°; trunk is pressurized; approach at 1.2 V_s; and landing at 1.1 V_s.

The glide angle is 11.3 degrees, which requires 250 ft to clear a 50 ft obstacle and, as in takeoff, the rotation or flare angle is small, 2 degrees, thus increasing the fuselage angle of attack to 4.5 degrees at touch down. The transition distance needed to slow from approach to landing speed is 548 ft. The ground run is based on developing an average ratio of braking force to aircraft weight of .27 and is equal to 2200 ft. Therefore, the total landing distance required to clear a 50 ft obstacle is 2998 ft (SLS).

P&W has calculated that the minimum throttle setting, with the required 39 lb/sec fan bleed to pressurize the trunk, gives 2500 lb thrust which is dissipated by turning vanes in the tail pipe.

(11) WEIGHT

Some of the weight equations are empirical. They are reasonably accurate and are based on comparisons with various aircraft; most have been used in preliminary design study work before. The development of these equations and the weight calculations are shown in Appendix C.

Anticipated advances in technology are incorporated as follows; this must be recognized when making comparisons with other weight data:

(a) Use of composite materials
Wing group weight reduced 10%
Tail group weight reduced 25%
Fuselage basic weight reduced 15%

(b) Flight controls weight reduced 10%
due to the fly-by-wire system

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Calculated weight is summarized as follows. A Group Weight Statement is in Appendix C where the weight allocation has been changed to conform to the Group Weight Statement Form. The weight total is of course the same.

Wing Group	2334 lb
Horiz. Tail Group	307
Vertical Tail Group	206
Fuselage basic	2088
Canopy	260
Speed brakes	104
Engine	1618
Tail pipe extension	43
Engine section	146
Inlet ducts	160
Engine controls starting, lub., and oil (including unusable oil)	94
Flight controls	567
Fuel tanks - wing integral	107
- fuselage integral	96
Unusable fuel	46
Fuel system	170
Instruments	85
Electrical	350
Anti-icing	130
Air conditioning	160
Furnishings, incl. ejection seat	330
APU	120
Armament provision	200
Equipment (incl. oxygen & survival)	175
SETOLS	756
Protection	
fuel cells in wing	67
armor - pilot (allowance)	300
other (allowance)	100
CAS mission loading (specified 8574 lb)	
installed avionics	770
crew	180
4 - TERs	384
12 - Mk 82 (droppable)	6840
4 - pylons	400
CAS mission fuel	4552
Unassigned	55
Gross Weight	24300 lb

(12) BALANCE

Balance calculations are in Appendix C and are summarized below. All conditions are considered satisfactory.

Loading Condition	CAS Mission Weight Lb	CAS Mission C.G. % $\bar{c}$
Weight empty for balance (gross weight less pilot, 12 - Mk 82 droppable stores, and fuel)	12728	25.4
Operating weight (plus 180 lb pilot)	12908	22.7
Zero fuel weight (plus 12 - Mk 82 droppable stores, 6840 lb)	19748	23.6
Gross weight (plus CAS mission fuel, 4552 lb)	24300	23.0

The C.G. locations shown above leave margin for the inevitable aft drift of the C.G. when the aircraft is built, see tail sizing in Appendix B.

(13) DRAG

Preliminary drag estimates were made in support of the early work and, since then, drag changes were incorporated corresponding to the changes in the configuration as the design work progressed. The final drag estimate shown in Appendix B is very close to the preliminary estimate as modified for configuration changes. Therefore, conclusions made based on the preliminary work are considered reliable, and no recycling of the preliminary work is needed.

(14) MISSION PERFORMANCE AND PROFILES

The majority of the performance was done with computer programs developed for this study as explained in Appendix F using engine data from Appendix D and a fuel weight of 6.8 pounds/gallon. CAS and Ferry Mission Profiles, along with additional performance data, follow.

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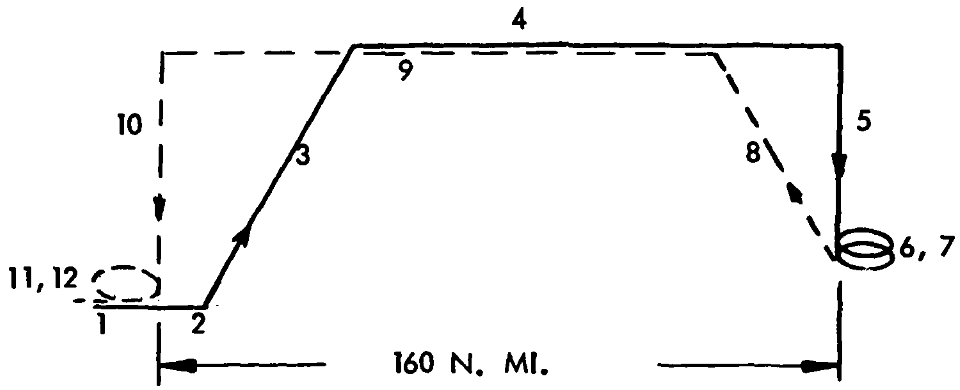
CAS MISSION



OPERATION

1.	Initial	
2.	Warmup and Takeoff	5 minutes at maximum power
3.	Climb Out	Best speed to 36089 feet
4.	Cruise Out	At 36089 feet
5.	Descend	To 5000 Feet (No time, fuel, or dist.)
6.	Loiter	1 hour at best speed
7.	Drop Stores	Retain TERs and pylons
8.	Climb Back*	Best speed to 36089 feet
9.	Cruise Back*	At 36089 feet
10.	Descend	To sea level (no time, fuel, or dist.)
11.	Loiter	10 minutes at best speed
12.	Land and Reserve	5% initial fuel

For specific data on each operation, see following page.

*Minimum fuel to return. Cruise back at best cruise altitude requires more fuel because of additional climb fuel required.

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CAS MISSION

Configuration: Clean + 4 pylons and TERs + 12 Mk-82
S_W = 280 sq ft
Engine = P&W STF 529

G.W. = 24300 lbs
W_{Fuel} = 4552 lbs
Drop Wt = 6840 lbs

	Operation	Δ Fuel Lbs	WAEOO* Lbs	Alt. Ft.	Mach No.	Δ Dist N Mi	Σ Dist N Mi	Δ Time Min	Σ Time Min
1.	Initial	-	24300	SL	-	-	-	-	-
2.	WU & TO	651	23649	SL	-	0	0	5.0	5
3.	Climb Out	810	22839	36089	.6	79	79	13.2	18.2
4.	Cruise Out	415	22424	36089	.69	81	160	12.3	30.5
5.	Descent	0	22424	5000	-	0	160	0	30.5
6.	Loiter	1635	20789	5000	.31	0	160	60.0	90.5
7.	Drop Stores	0	13949	5000	.31	0	160	0	90.5
8.	Climb Back	260	13689	36089	.7	26	186	3.6	94.1
9.	Cruise Back	375	13314	36089	.595	134	320	23.6	117.7
10.	Descent	0	13314	SL	-	0	320	0	117.7
11.	Loiter	178	13136	SL	.235	0	320	10	127.7
12.	Land & Res.	228	12908	-	-	-	320	-	127.7

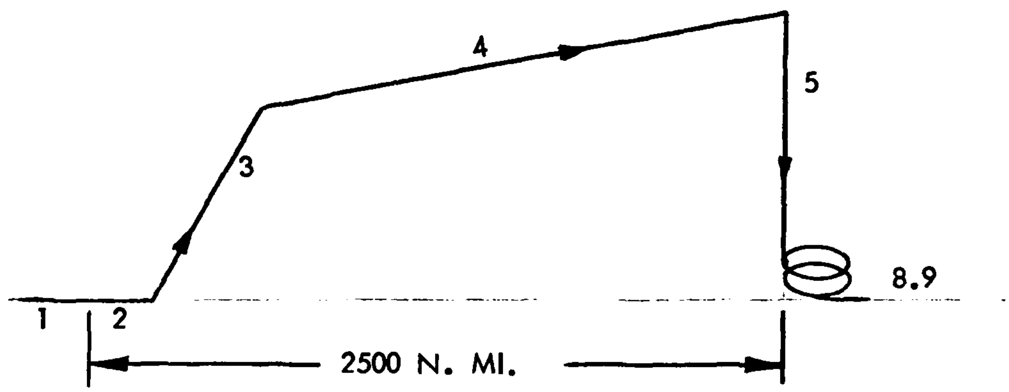
Radius: 160 N. Mi.

Δ Fuel: 4552 Lbs.

Δ Time: 2.128 Hrs.

*Weight at End of Operation.

FERRY MISSION



OPERATION

1.	Initial	
2.	Warmup and Takeoff	5 minutes at maximum power
3.	Climb Out	Best speed to cruise altitude
4.	Cruise Out	Best altitude and speed
5.	Descend	To S.L. (No time, fuel, or dist.)
6.	Loiter	10 minutes at best speed
7.	Land and Reserve	5% initial fuel

For specific data on each operation, see following page.

FERRY MISSION

Configuration: Clean + 4 pylons
S_W = 280 square feet
Engine = P&W STF 595

G.W. = 20243 lbs
W_{Fuel} = 7687 lbs

	Operation	Δ Fuel Lbs	WAEOO* Lbs	Alt. Ft.	Mach No.	Δ Dist N Mi	ΣDist N Mi	Δ Time Min	ΣTime Min
1.	Initial	-	20243	SL	-	-	-	-	-
2.	WU & TO	651	19592	SL	-	0	0	5.0	5
3.	Climb Out	583	19009	45800	.7	81	81	11.7	16.7
4.	Cruise Out	5899	13110	54200	.775	2419	2500	119.5	343.3
5.	Descend	0	13110	SL	-	0	2500	119.5	343.3
6.	Loiter	170	12940	SL	.23	0	2500	10.0	353.3
7.	Land & Res.	384	12556	-	-	-	2500	-	353.3

Range: 2500 N Mi.

Δ Fuel: 7687 Lbs.

Δ Time: 5.888 Hrs.

*Weight at End of Operation.

(d) Additional Performance Data

Performance Item		Attained
Takeoff ground run (CAS mission loading)	W =	24300 lb - 4.5 min at T_{Max}
S.L. Std.		2813 ft
S.L. 89.8°F		2945 ft
Takeoff over 50 ft obstacle (CAS mission loading)	W =	24300 lb - 4.5 min at T_{Max}
S.L. Std		4045 ft
S.L. 89.8°F		4217 ft
Landing over 50 ft obstacle (CAS mission loading)	MLDGW =	21569 lb
S.L. Std		2998 ft
S.L. 89.8°F		3158 ft
Best cruise Mach No. and altitude for clean configuration (ferry)	M = Alt. =	.775 45800 to 52500 ft
Max range vs cruise Mach No. for clean configuration	See Appendix F, *Sections (2) and (3)*	
Best endurance Mach. No. and altitude, Hr	For CAS mission, 1.0 hr at M = .31 at 5000 ft. (a gain of 2 to 3% in endurance may be possible at 10,000 ft based on questionable extrapolated engine data - see Appendix F)	
Service ceiling (300 ft/min R/C), CAS mission loading, W at start of climb = 24300 - 5 min at T_{Max} (fuel consumed in climb)	36800 ft	

Max rate of climb (ft/min) CAS mission loading, for S.L., 5000 ft and 15000 ft for Std day and Navy tropical day (89.8°F) W at start of climb = 24300 - 5 min. at T_{Max} (fuel consumed in climb)		Std.	89.8°F
	S.L.	9880	Eng. data not available; see App. F
	5000 ft	8250	
	15000 ft	5000	

SANDAIRE

Performance Item	Attained Std.	Attained 89.8°F
Max rate of climb (ft/min) at S.L., CAS loading, W = 24300 lb - 5 min at T_{Max}		
SETOLS deployed, takeoff speed, flaps 40°	2597	2719
SETOLS retracted, takeoff speed, flaps 40°	7367	7441
Max rate of climb (ft/min) at S.L., CAS loading less 60% fuel (MLDGW) W = 24300 - .60 x 4552 = 21569 lb		
SETOLS deployed, Approach speed, flaps 50°	2838	2963
SETOLS retracted, Approach speed, flaps 50°	4486	4659

	g	S.L.	5000 ft
Max sustained maneuver load factor vs Mach No. at S.L. and 5000 ft, CAS mission payload less 40% fuel W = 24300 - .40 x 4552 = 22479 lb	2	.765	.775
	3	.755	.755
	4	.725	.695

	g	S.L.	5000 ft
Max sustained maneuver load factor vs Mach No. at S.L. and 5000 ft, CAS mission less 40% fuel and less 12 Mk 82 dropped W = 22479 - 6840 = 15693 lb	2	.850	.865
	3	.845	.855
	4	.835	.845

	Std	Navy Trop. 89.8°F
Stall speed at S.L., SETOLS deployed - knots		
Des. G.W. 24300 lb	112	115
MLDGW 21569 lb	105	108
Flaps 50°		

	Std	Navy Trop. 89.8°F
Landing approach speed at S.L. SETOLS deployed - knots		
MLDGW 21569 lb	126	130
Flaps 50°		

Aircraft range with the ferry mission payload	No external fuel tanks
GW = 20243 lb	
Fuel = 7687 lb	2500 N.M.

(15) DESIGN

(a) Conformity With Design Requirements

Conformity with the design requirements is essentially achieved except for the M = .91 high speed requirement which is compromised to M = 0.89 favoring other design options as discussed in preceding sections.

Structural design load factors from the Statement of Work are

+7.0, - 3.0

and are applied to the Basic Flight Design Gross Weight (BFDGW) which is defined for this study as Design Gross Weight less 40% of the CAS mission initial fuel:

$$24300 - .40 \times 4552 = 22479 \text{ lb}$$

The design limit load on the wing is then 7 x 22479/280 = 562 lb/sq ft which is used in the calculation of the wing weight. Ultimate load is 1.5 times the design limit load.

A specific list of design requirements is presented below with comments on the conformity of this design study.

Design Requirements	Status
Aircraft shall have a CTOL capability and shall include a SETOLS	Aircraft is conventional in design except for the SETOLS which has been conservatively designed. The cushion pressure is 162 lb/sq ft (1.125 psi) which is in line with current state-of-the-art.
Aircraft shall be land based with capability of takeoff and landing from flat terrain such as fields, marshes, lakes and rivers.	The design of the SETOLS has considered trunk stability and aircraft control in takeoff and landing at low speed. The design as presented should have acceptable characteristics, see Appendix A.

SANDAIRE

Design Requirements	Status
Specified maximum military load Installed avionics 770 lb 4 - TERs 384 12 - Mk 82 6840 4 - pylons 400 Total 8394	This load carried for CAS mission as specified.
Propulsion system shall utilize one or more turbojet or turbofan engines	One P&W STF 529 study turbofan engine used
JP-5 fuel at 6.8 lb/gal shall be used	Fuel weight 6.8 lb/gal
Fuel dumping capability shall be provided	Fuel dumping provided at outboard pylons and through bottom of fuselage for ferry tank
Aircraft shall have self starting capability	APU provided for self starting
Aircraft shall have an environmental control and life support system	Air conditioning is provided and oxygen and survival weight is included
Speed brakes shall be provided	Speed brakes on the aft fuselage are provided
An ejection seat escape system shall be provided	Ejection seat provided
Aircraft shall be single seat	Design has one crew member

SANDAIRE

Design Requirements	Status
Single point type ground pressure fueling shall be provided	Provided and located on left hand side of fuselage below wing center section
Provisions for aerial refueling shall be provided	Forward boom type to connect with refueling aircraft drogue. Boom not included in weight
The avionic system shall provide the following functional capabilities: weapon delivery stores management mission computer control and display communication/navigation/identification flight system electronic warfare	The specified 770 lb installed avionics is incorporated with 35 cu ft space allowed and distributed in three locations in the fuselage. No work was done with respect to the avionics system capability.
External carriage shall be provided for 4 - TERs 12 - Mk 82 (3 per TER) 4 - pylons 4 - 300 gal ferry fuel tanks	4 - TERs, 12- Mk 82 and 4 pylons carried below the wing (see pg 3). No external fuel carriage provided as internal fuel capacity is sufficient for ferry.
MLDGW (Maximum Landing Design Gross Weight) = TOGW for CAS mission minus 60% of maximum internal fuel	MLDGW = Design Gross Weight (24300 lb) - (60% of CAS mission fuel of 4552 lb) = 21569 lb

(b) SETOLS

The SETOLS (Surface Effects Takeoff and Landing System) is another name for an air cushion landing system, and to date very little development work has been performed. The present, limited R&D efforts indicate the potential value of the system for use in landing aircraft on various surfaces other than runways. These surfaces could be swamps, rivers, lakes, protected bays, beaches, unimproved fields (even if moderately battlescarred), snow, ice, etc., as long as they are generally free of large surface disruptions throughout the distances required for landing and takeoff.

The system consists of a large, elastic air container (trunk) attached to the bottom of the fuselage. Inflating it with low pressure air creates a large, doughnut-type pad whose planform area (cushion area) is essential to the support of the vehicle. Discharging the air through controlled orifices in the bottom of the trunk produces a positive pressure against this area sufficient to suspend the vehicle a small distance above a surface. This is an oversimplification of the system, but many papers are available giving detailed information on the system principles; therefore more is not warranted here.

The system uses controlled engine fan bleed air through a sonic orifice to provide a constant air flow of 39 pounds/second (approximately 510 cu.ft./second) at 360 pounds/sq. ft. pressure to inflate the trunk and support the vehicle during takeoff and landing. Four bags or parking bladders are installed within the trunk and, when inflated to 275 pounds/sq.ft., are used to support the vehicle when it is not in operation. Brake pads installed in the bottom of the trunk and actuated pneumatically are used to stop and hold the vehicle. A suction system, integral with the trunk support structure, is used to hold the trunk, when deflated, tightly against the fuselage bottom to prevent it from fluttering in flight.

Trunk

The trunk size and shape are the result of many iterations of trunk planforms and pressures to minimize the effect on the vehicle's size, configuration and performance. The trunk configuration chosen is shown on the general arrangement drawing, Page 3, and on Pages 28 and 29. This configuration was selected because of its continuously curving planform which will help prevent flagellation in flight and trunk side flutter in hover while permitting aerodynamic shaping of the sponson. The planform width was a compromise between maximum sponson extension and vehicle roll stability. Thus the length was established by the cushion area required. This shape contains a cushion area of 150 sq. ft. which requires a cushion pressure of 162 pounds/sq. ft. to support the vehicle at its maximum gross weight of 24300 pounds. Using a trunk pressure of 360 pounds/sq. ft., the cushion to trunk pressure ratio P_c/P_T varies from .45 at maximum gross weight to approximately .27 at minimum gross weight (Figure 5A, App. A).

The trunk material elongation characteristics used for this design are shown in Fig 2A App.A. Development will be required to obtain these characteristics but should not be difficult, as they are similar to existing materials. The girth elongation ratio of $\ell/\ell_o = 3$ was chosen as the working design point, which corresponds to a material tension (T) of 56.25 pounds/inch. Using this tension, the girth outer radius (r), shown on Page 28, for a maximum gross weight vehicle in hover, is 22.5 inches and the inner radius (R) is 40.91 inches. Therefore, the cushion area and inner radius, R, decrease as the vehicle approaches empty weight.

Jet nozzles or discharge ports are installed on the bottom periphery (49.528 feet) of the trunk to help provide an air cushion supporting the vehicle off the surface. The trunk planform and cross-section, Pages 28 and 29, change with weight or P_c/P_T ratio and only at the design point, maximum gross weight, is the jet nozzle height (h) constant around the periphery. At this point, h is approximately .28 inches but increases at the ends as P_c/P_T decreases. This maximum gap is referred to as ΔH. As ΔH increases, h decreases on the sides to maintain proper cushion pressure and approaches .09 inches for the empty weight vehicle; see Fig 13A App.A. This gap is caused by the underside of the fuselage being a different radius than the sponsons, resulting in the elongation, and thus tensions, being different except at the design point.

The trunk discharge area is 1.4603 sq. ft., which is also the equivalent area that will permit trunk pressure to remain at 360 pounds/sq. ft. when the vehicle is out of ground effect. Only a portion (80%) of this area (1.1682 sq. ft.) is used to help maintain a cushion pressure while the remainder (.2921 sq. ft.) is outboard of the ground tangent line and is used for air lubrication. Because a total of 1.5343 sq. ft. is required to maintain the 162 pounds/sq. ft. cushion pressure, an additional .3661 sq. ft. is provided through control valves between the trunk and cushion area. At minimum cushion pressure of 96.5 pounds/sq. ft., the valve area needed is approximately one-half that required at the design point. This value is controlled by pressure sensors to maintain the trunk pressure constant; see Page 30 for the system schematic. The areas and flows are established using a discharge coefficient of .66 and a cushion to atmosphere discharge coefficient of 1.0 during a hot day operation.

Slits installed in the bottom of the trunk will become openings or jet nozzles as the trunk is inflated. They will be sized and distributed, about the ground tangent line, to obtain the discharge area previously mentioned. The size and distribution has not been determined, as the trunk material characteristics will affect the configuration of the nozzle opening during trunk expansion. This determination will have to be made after some testing. Replaceable nozzle plugs, containing pads to help prevent trunk abrasion, will be used in conjunction with the slits. Additional abrasive pads will be required on either side of the area assigned to the discharge nozzles to help alleviate this condition.

It was decided to use pneumatically operated braking pads on the bottom of the trunk in place of suction braking because of the additional weight, volume and complexity required. The pump or fan needed for this system would also have to withstand contaminated water that could possibly be ingested during water landings. The braking pads are actuated by pressurizing a bag located above the pads with cooled engine bleed fan air at the command of the pilot. This action forces the pads against the surface, causing the trunk locally to deflect upwards, venting the remainder of the cushion pressure. This reduction in cushion lift, forces the trunk to flatten over the surface creating additional braking from the abrasive pads. Differential braking, in conjunction with a yaw control nozzle located aft of the vertical stabilizer, provides directional control at low speeds.

The center of pressure of the cushion is located 6.8 inches forward of the vehicle C.G. and the centerline of thrust is approximately 7 inches below the vehicle C.G. Thus the cushion lift and the engine thrust both produce a positive pitching moment. This moment is balanced by trunk lift which is approximately 91*inches aft of the C.G., plus trunk to surface friction if the vehicle is moving. For a design gross weight of 24300 lbs., max takeoff thrust of 10172 lbs., cushion lift of 22549 lbs. and a trunk lift of 1751 lbs. the resulting friction force is 875 lbs. assuming a friction coefficient of .5. This corresponds to an effective friction coefficient for the vehicle of .036 which is about the same as rolling friction.

The inner and outer periphery of the trunk contains indexed holes for attachment to the vehicle structure by threaded fasteners to react the trunk loads and create a seal. The index holes will thus provide the proper pretension which is approximately 77% at the forward and aft ends and 6% on the side. Because of this low pretension, a suction system is used to prevent flagellation during flight.

Roll stabilizing doors, that retract into the upper surface of the sponsons, are installed on each side of the vehicle. These doors help to prevent roll perturbations and vehicle "lean" during turns and crosswinds. They are coupled to the roll axis of the autopilot system and decoupled after vehicle liftoff.

<u>Parking Bladders</u>

Four elastic bags, one at each end and on both sides, are installed within the trunk and, after inflation, are used to support the vehicle when it is not in use. The bags, or parking bladders, are used so that any one air leak would still leave the vehicle partially supported. The parking bladders, when inflated, force the trunk to expand to its design point of $\ell/\ell_o = 3$, or approximately 590 cu. ft. of volume. At this point, the bladders are

*Distance from aircraft c.g. to centroid of aft trunk area flattened against the ground. (See Figure 14-A, Page A-22).

pressure relieved at 275 pounds/sq. ft. Inflation is initiated by switching the engine fan bleed air from the trunk to the bladders. The bleed air rate of 39 pounds/second (approximately 510 cu. ft./second), fills the bladders as fast as the bladders can force the air out of the trunk and the vehicle will settle approximately 3 inches below its hover height. The trunk is inflated, from the parking position, by engine fan bleed air while simultaneously venting the bladder to the trunk, causing the bladder to deflate to its stowed position.

Engine Operation

Engine fan bleed air, used in operating the system, is captured by a scroll added to the STF-529 engine and then distributed through two exit nozzles. Sonic orifices, culminating in three-way valves, are installed between the nozzles and the trunk system for control. The valves control the air flow to the trunk, bladder and a shut-off position. The orifices are sized to permit a constant air flow of 39 pounds/second at a pressure to maintain 360 pounds/sq. ft. in the trunk; the resulting pressure drop reduces the engine air from approximately 270°F to 128°F. It is possible, through further design, that the orifices could be incorporated into a jet pump system, thus reducing the required flow from the engine, and consequently the thrust or conversely maintaining the same thrust level and increasing the flow capability.

To maintain 39 pounds/sec engine fan bleed, the minimum throttle setting required results in an engine thrust of approximately 2500 pounds, thus nullifying some of the braking action if not diverted. This force must be controlled and/or preferably diverted into reverse thrust at landing or during braking, but also some thrust must remain for taxiing. Taxiing might be accomplished by reducing throttle, and consequently air flow to the trunk, causing the trunk to partially collapse, thus increasing trunk drag to balance the thrust. This technique would have a slow cycle response, increase trunk abrasion, and put an undue hardship on the pilot to constantly rebalance the engine because of surface variance. Insufficient data are available to do such an analysis. A full thrust reverser would be heavy and could cause some problems through reingestion while operating on unimproved surfaces or water. Although not shown on the inboard profile drawing, it is planned that diverters be employed for landing which would redirect the 2500 pounds of thrust up and outward resulting in no net forward thrust.

A proposed method is shown on Page 31. This would help prevent reingestion and impingement of the exhaust on adjacent aircraft or personnel during taxiing or parking. Taxiing would also be made simpler by manipulating the diverters for the required thrust.

SANDAIRE

The near frictionless contact between the vehicle and the surface makes it mandatory to use a means for directional control at slow speeds other than aerodynamic controls. Therefore, a yaw control nozzle using engine exhaust gases is installed at the aft end of the fuselage below the vertical stabilizer. The gases are directed sideward by eyelid-type diverters to produce a yawing moment. When not in use, the gases exhaust rearward for thrust.

Comments and Recommendations

Some time after the present shape and planform were established, it was found that other forms may be advantageous but time did not permit a change.

During the trunk analysis, it was discovered that the running tension load at the front of the trunk around the inner radius is over 500 pounds/inch. A larger inner radius is required to reduce this load. Also, the pretension in the side trunk is low (approximately 6%) and could be increased by making the bottom radius of the sponsons larger. In fact, it would be ideal if this radius was the same, or nearly so, over the complete trunk area; however this would result in extension of the sponsons. These are but two areas where improvements could possibly be achieved, and it appears a different planform, such as shown on Page 32, would be better.

If the SETOLS is to be a serious contender for future operations, comprehensive design and testing should be instituted, especially as a system integrated with an acceptable aerodynamic vehicle configuration. Much needed data could be gleaned from a structural mockup or model containing a trunk system. The mockup could be self-propelled and used for all ground testing of braking systems, jet nozzle configurations, abrasion protection, trunk dynamics including forwardly propelled drop tests, etc. Also, airborne testing for temperature effects, trunk flutter, inflation loading, high speed effects, etc., could be done using the mockup or model affixed to the underside of a suitable aircraft or a ground track sled vehicle.

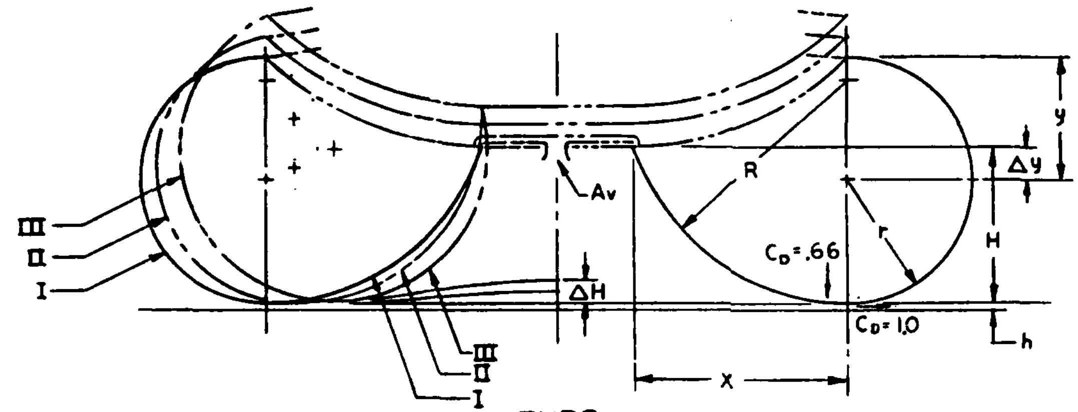
It is unknown, and even questionable, that an elastic trunk system could be developed for use on the exterior of a high Mach Number aircraft. Therefore, work should be done on an inelastic system that could be mechanically retracted into a protected compartment. An inelastic trunk system would have some new and different problems than an elastic system, but conversely could alleviate some.

I GROSS WT $P_c/P_t = .45$

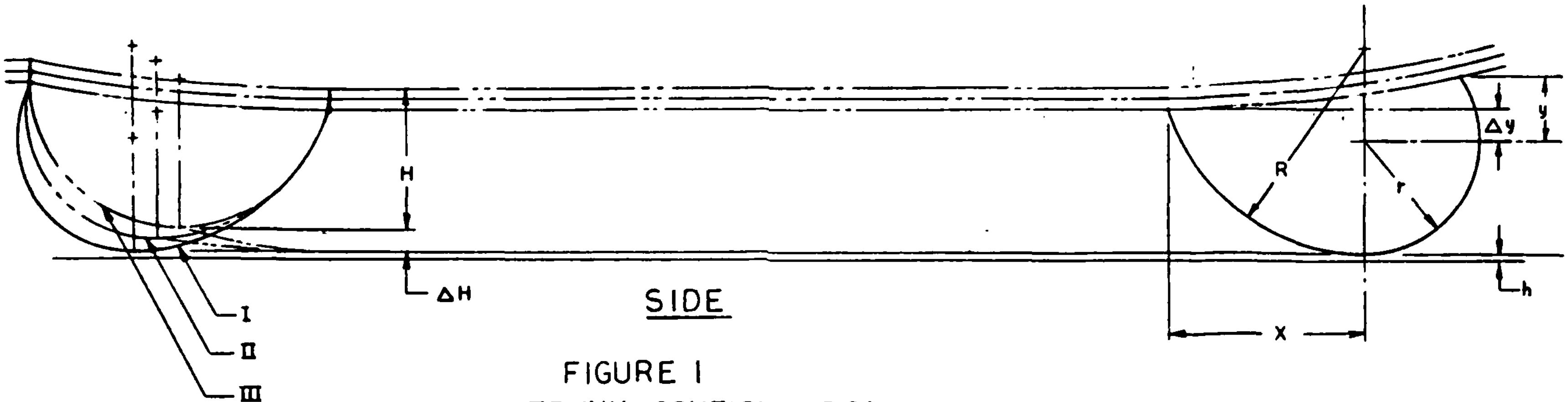
II EMPTY WT $P_c/P_t = .268$

III FREE AIR $P_c/P_t = 0$

		X	r	R	Δy	ΔH	H
SIDE	I	38.98	22.5	40.91	6	0	28.5
	II	33.82	24.76	33.83	7.7	2.06	32.46
	III	26.48	27.74	27.74	8.26	4.3	36.0
ENDS	I	38.98	22.5	40.91	6	0	28.5
	II	34.74	25.68	35.08	4.74	2.06	30.42
	III	29.58	29.65	29.65	2.05	4.3	31.7



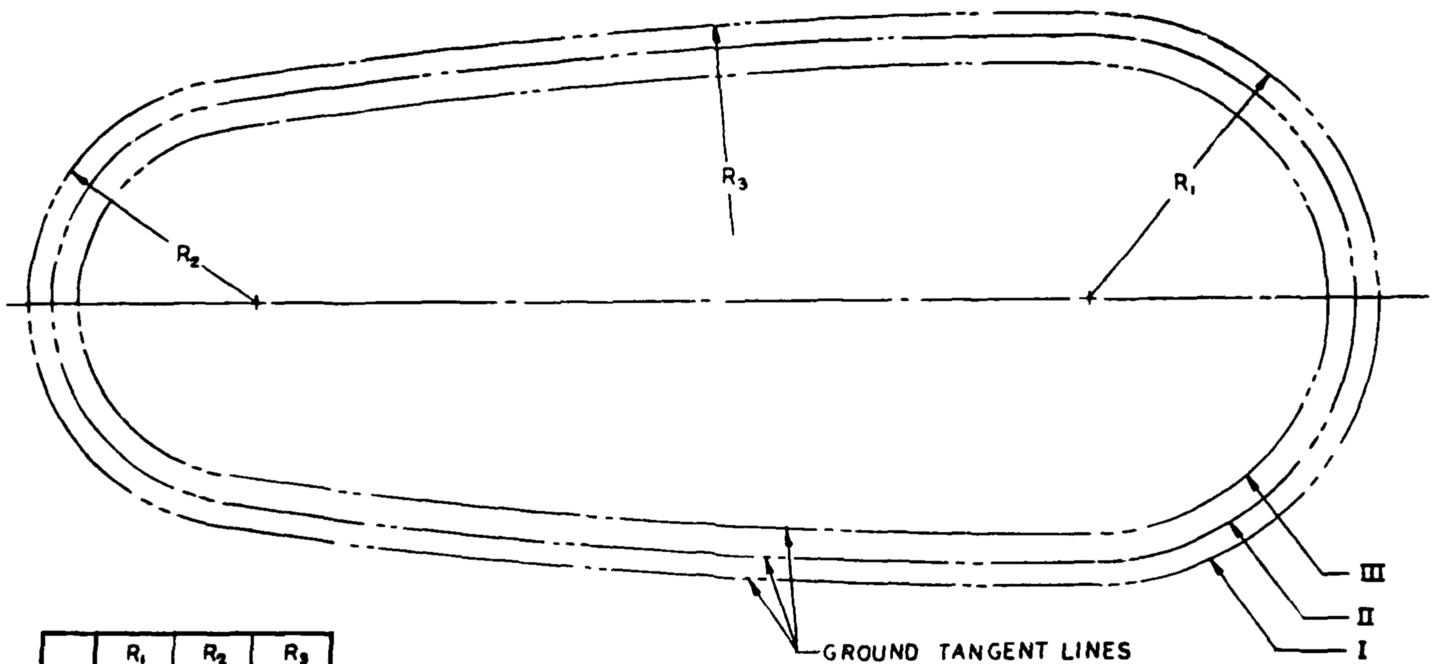
ENDS



SIDE

FIGURE I
TRUNK CONFIGURATION

I GROSS WEIGHT
II EMPTY WEIGHT
III FREE AIR



	R_1	R_2	R_3
I	52.5	41.25	1038.09
II	48.25	37.09	1033.84
III	43.09	31.84	1028.68

FIGURE 2
TRUNK SYSTEM PLANFORM

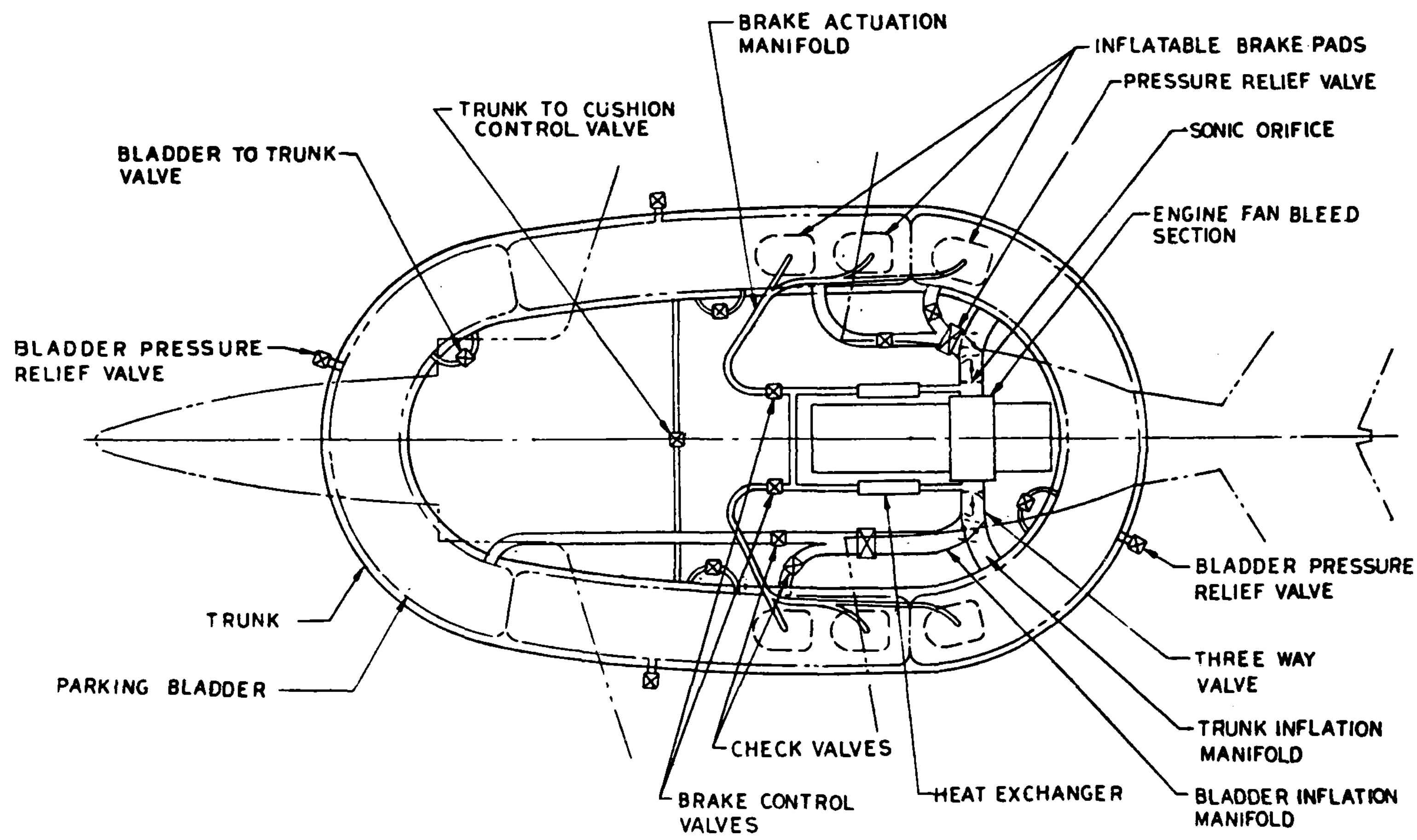


FIGURE 3

SCHEMATIC - TRUNK AIR SYSTEM

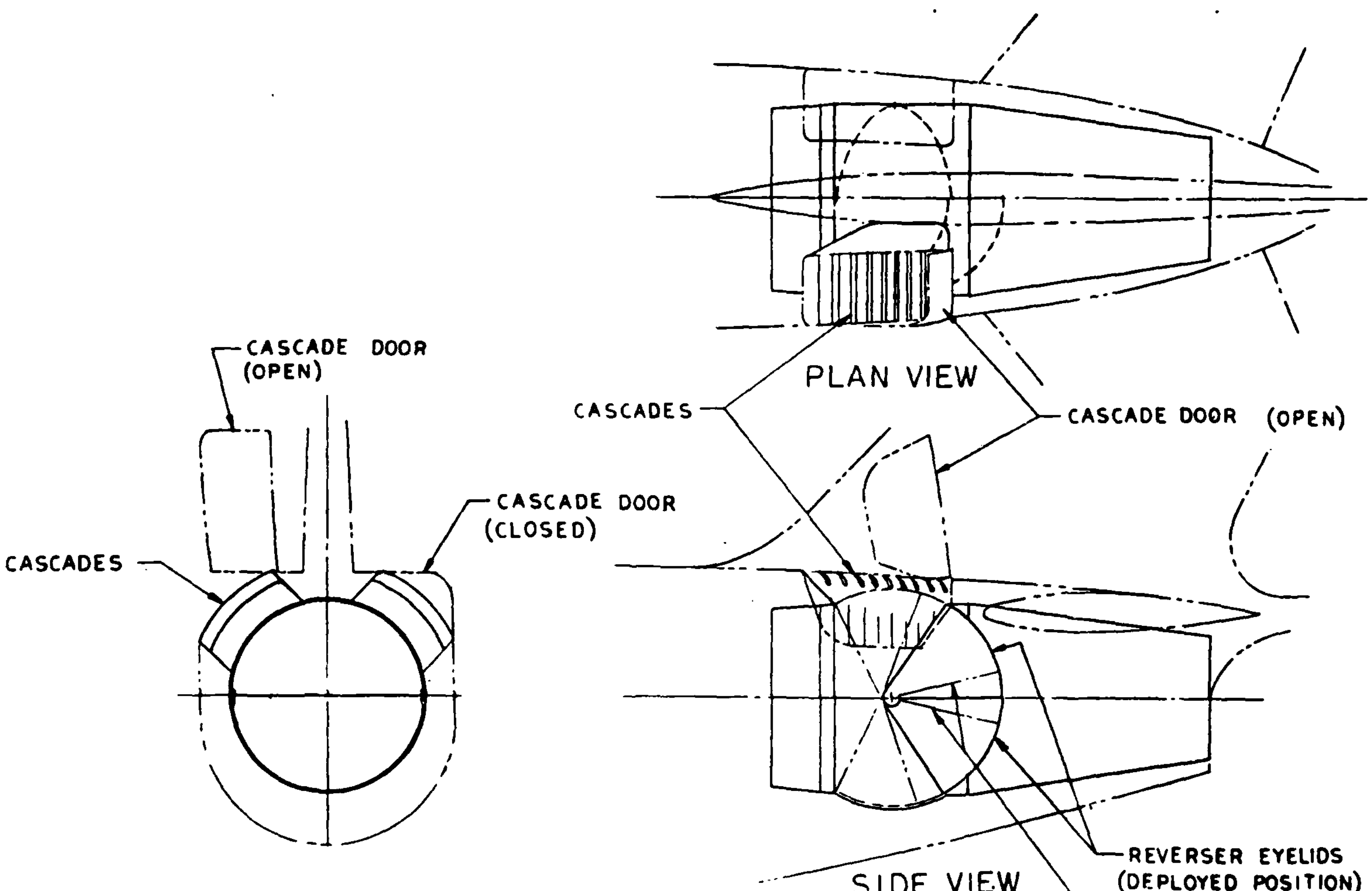


FIGURE 4
CONCEPT-THRUST REVERSER

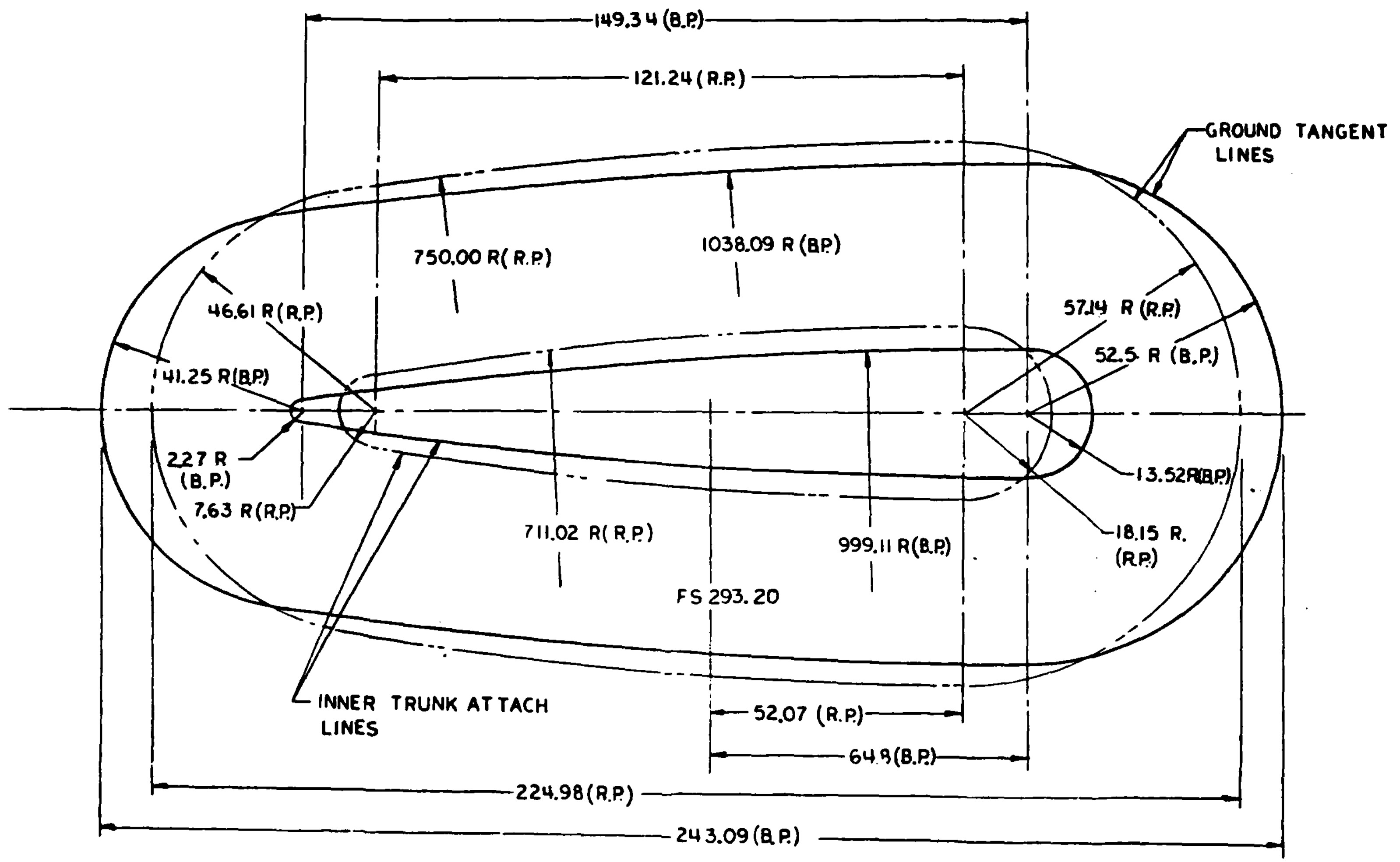


FIGURE 5
BASIC AND PROPOSED REVISED TRUNK PLANFORMS

B.P. = BASIC PLANFORM
R.P. = REVISED PLANFORM

(c) Subsystems

Flight Control - Flight is controlled by a fly-by-wire totally electric subsystem. Pilot flight commands are inputs to a central computer from which outputs are commands to electro-mechanical actuators at the control surfaces. By incorporating the V/n diagram as part of the central computer program, flight commands could never cause the vehicles to exceed the safe structural limitations. An override provision must be permitted, in an emergency at the pilot's discretion, so that the V/n diagram would be expanded to the ultimate structural limitations. Additional inputs from sensors, such as altitude, attitude, velocities, accelerations, heading, etc., would be required for the computer to make proper judgments and consequential outputs. All computer inputs and outputs are relayed via fiber optics to and from miniprocessors located at the various termini.

In the 1995 time frame, electro-mechanical actuators should be weight to power competitive with any other mode of actuation. Thus, the R&M would be improved if for no other reason than the elimination of additional subsystems for the source of power. A high reliability should be obtainable through redundancy of major components in the system.

Electrical - The electrical subsystem is basically an AC/DC system deriving its energy from engine-driven alternators (prime), an APU (standby) or battery/inverter (emergency). AC is used for all power systems, while the DC is provided to maintain the battery charge and for some discrete commands. All power transmission is by hard wire controlled with discrete commands relayed via fiber optics to and from miniprocessors. This should provide the lightest weight system even though "battle damaged" hardened by redundancy.

Avionics - The avionics system was specified at 770 pounds and 28 cubic feet installed. It is intended for the major components to be installed immediately forward and aft of STA 188.0 (approximately the aft end of the canopy) forward of which is environmentally controlled. The usable volume between the pilot's seat and STA 188.0 is approximately 20 cubic feet and will contain those components needing environmental control. The useful volume immediately aft of STA 188.0 is approximately 15 cubic feet and will contain the remainder of the components. Servicing of this section will be through a door in the top and through the open canopy area for those components forward. The armor protection required in the pilot's compartment will thus provide some degree of protection for those components installed in the cockpit area.

Life Support - The cockpit, pilot's pressure suit and selected avionics equipment will be supplied with conditioned air from the ECS (environmental control system). The ECS will use high temperature, high pressure engine bleed air and consist of air-to-air heat exchangers, an expansion device and pressure/temperature regulating, mixing and control valves. It will be used to select and control cockpit temperatures and pressures, pressure suit operation including G-system, necessary avionics cooling and pressurization, defogging and emergency ram air operation. In addition, a LOX system will supply oxygen to the pressure suit in emergencies.

An "any attitude zero velocity" ejection seat is installed for pilot escape. Canopy ejection will be mandatory because of the intended use of high strength, bullet-resistant glass.

(d) Propulsion System

Engine Installation - Installation information for the STF-529 engine is limited because it has not been built, but it is assumed installation can be made using standard practices. Provisions are made to remove the empennage and part of the aft fuselage as a unit to permit the installation and removal of the engine. A door in the top of the fuselage will allow access and inspection of the accessory section which is located at the top front of the engine. Additional access doors are provided at other special points of interest.

The fuselage is extended to provide tailpipe protection during water landings and to attempt to reduce the IR signature. The basic IR energy will be reduced below that for comparable present-day engines because of the lower exhaust temperatures resulting from mixing the fan and core gases prior to expulsion from the tailpipe.

The air inlets are quite long because of the engine position required in the fuselage for proper balance and the need to have the inlets high on the fuselage to help prevent water ingestion. Additional inlet area will be required for high engine thrust at no, or low, vehicle velocities and will be provided through pressure-balanced "suck-in" doors. As greater knowledge is obtained of the engine characteristics, it may be possible to use an inflatable bulb on the inlet lip in place of "suck-in" doors.

The engine starter is a hot gas turbine-type deriving its energy from the APU which also is used as standby electrical power source. The vehicle battery is used to start the APU.

Fuel System - The entire mission fuel is contained within the wing, with the center section fuel protected in a self-sealing fuel bladder (1150-pound capacity) sized for 50 CAL projectiles. The wing fuel tanks extend to 63% of semi-span (WS155) and can accommodate a total of 4600 pounds of fuel plus 7% ullage. An additional tank with 3200 pounds capacity is housed in the bottom of the fuselage and is intended for use during ferry operations. (Space is available if a greater range is desired).

All fuel is transferred to the center section tank, from which the engine is supplied, via boost pumps located in a negative G sump. A single point refueling valve with associated control valves, for wing tanks or ferry tank, is located on the left hand side of the fuselage below the wing center section. Hand filling can be done through receptacles mounted on the upper surface of the wing for each outboard tank. Because of the fuel volume available, only provisions for an inflight refueling boom is provided on the right hand side of the fuselage forward of the sponsons. Wing fuel dump provisions are provided in both outboard pylons (WS155) and through the bottom of the fuselage for the ferry tank.

Lubrication - The engine oil is stored in a tank located on the lower left hand side of the fuselage. Access for replenishment is made when the trunk stability doors are extended. Skin radiation is used for cooling but, if greater heat transfer is needed, the alternate hat sections of the trunk support structure could be used as a radiator.

(e) Structures

The structural weight was assumed to be less than that determined from empirical equations for present day conventional construction. This reduction was predicated on the use of improved materials, composite construction techniques and other technical advancements existing in the 1990 decade. Even so, it is intended that metal monocoque construction be used in most primary load paths.

The major structural difference between a conventional aircraft and one equipped with SETOLS is in the trunk support area. This results from the pressure required in the trunk and cushion area to support the aircraft. It is envisioned that this area would be constructed of double skins separated by corrugations or a continuous hat section with each alternate section spanning a row of perforations (holes or slits) in the outer skin. By manifolding these alternate sections to a suction source, a positive pressure can be maintained against the external surface of the trunk. This external force would help the trunk pretension to hold it firmly against the bottom when deflated. This, or a similar method, will be mandatory to prevent trunk flagellation at high speeds or during maneuvers. Warm air could also be circulated through each remaining hat section to help maintain trunk flexibility during exposure to extreme cold.

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The fuselage size was dictated by the maximum diameter of the engine and the area necessary to support the trunk system. The latter results in a sponson-type structure on the bottom of the fuselage which contributes to the vehicle drag. However, this increased volume does have some benefits, such as providing sufficient buoyancy in the event of a deflated trunk and/or parking bladder, providing enough room for additional fuel tankage (sufficient fuel to perform the 2500 NM ferry mission without external tanks), permitting the installation of the valving and plumbing necessary to control trunk and bladder inflation and also room for additional miscellaneous equipment. The entire fuselage in the area of the sponsons (WL60) down to the bottom is sealed against water entry. All maintenance and access doors would be above this area.

It appears that from 16 to 20 inches could be removed from the length of the fuselage and approximately six inches in depth and still have adequate volume for all the necessary equipment. Because of time constraints, this was not analyzed for its effect on performance; therefore, it remains in this design for potential growth.

Dual spars are used in the wing at 13% and 68% of the chord to provide volume for the basic mission fuel. Close out ribs are located at the intersection with the fuselage (BL 27) and at the outboard pylon (WS155) which is also the extremity of the fuel tank area. From this station, outer panels would finish out the span of the wing and contain the ailerons.

Flaps, located from the side of the fuselage to 70% of the semi-span, comprise the last 27% of the wing chord. The leading edge contains deicing provisions, and all electrical wiring and plumbing is aft of the rear spar. The integration of the trunk system could be improved if the wing could be blended into the bottom of the fuselage, but it would then be difficult to provide clearance for the external stores.

The entire horizontal tail is movable for trim with full span elevators compromising the trailing 25% chord. Spars are located at 30% and 70% of chord, with the forward spar reacting all bending loads. Installation or removal can be made after removing the vertical tail trailing edge fairing.

The vertical tail contains two spars at 25% and 70% of chord, with a rudder comprising the trailing 25% chord. The rear spar coincides with the fuselage frame that supports the horizontal tail.

The rear end of the fuselage is split for removal to provide for engine installation. For normal maintenance, a semi-structural access door is provided in the top of the fuselage over the engine accessory section.

(f) Aerospace Ground Equipment (including assessment of peculiar ground support equipment - PSGE)

The vehicle readiness is determined through its onboard instrumentation and automatic checkout, which is part of the central computer program, making it self-sufficient except for flight expendables. This method provides detection and identification of any LRU (line replaceable unit) that needs replacement. Using this concept will reduce the standard support equipment necessary for maintaining flight readiness in remote areas. Unfortunately, this concept does not alleviate the requirement for some peculiar equipment for this vehicle.

Some special equipment and potential solutions are needed for:

(1) Elevating and moving the vehicle resulting from trunk and/or bladder failure.

Potential solutions: (a) aircraft wrecker and a mobile cart, (b) bolt-on jacks with built in wheels, or (c) built-in jacks (which would increase vehicle weight).

(2) Vehicle support during trunk or bladder replacement.

Potential solutions: (a) standard aircraft jacks, (b) same as (b) above, or (c) support stands.

(3) Pretensioning the trunk during replacement.

Potential solutions: (a) clamps around the periphery of the trunk, which are bolted to the structure, to stretch the trunk through camming action, or (b) reduce pretension to a level permitting installation by hand.

(4) Trunk or bladder hole repair.

Potential solutions: (a) tire and tube type repair kit, or (b) develop special plugs.

(5) Empennage and engine removal.

Potential solution: (a) special slings and mobile hoist.

(6) Empennage and engine support.

Potential solutions: (a) standard "Air Log" type mobile stands with special support adapters, (b) built-up stands, (c) modified shipping containers.

AREAS THAT WARRANT FURTHER STUDY

(1) Parking bladder failure or puncture could result in the vehicle resting on its understructure. Also, a large portion of the understructure is occupied by the trunk system. Therefore, little area remains for antenna placement to prevent cross coupling, proper pattern coverage or breakage.

(2) Wave action could be responsible for the penetration of a wing tip during landing or takeoff on water, resulting in damage to the vehicle or injury to the pilot. The wave height or sea state limitations should be established or buoyancy requirements for the wing tips determined. Whatever is used for wing tip buoyancy should contain a skid for ground protection in the event of a bladder or roll stabilization system failure.

(3) An elastic trunk system was used for this study because some data and test results were available. Very little or, in some areas, no data exists on inelastic systems, but they appear to offer a better solution to flagellation at high speed flight if a reasonable retraction system, into a protected compartment, could be developed. Further work in this area may be justifiable.

(4) The minimum engine RPM or thrust level is high (approximately 2500 pounds of thrust) to maintain the required air flow to the trunk. In order to take full advantage of the braking system during landing, a means of eliminating or reversing this thrust is required. A full thrust reverser could produce undesirable reingestion, especially when used on unimproved or water surfaces. The partial employment of a full thrust reverser to control the minimum thrust would produce an undesirable conical dispersion on adjacent parked aircraft or ground personnel. Therefore, a diverter to control only the 2500 pounds should be lighter and give better control of the exhaust gases. It is unique because it will be required to reverse or nullify a minimum thrust after landing yet permit enough thrust for taxiing.

(5) The trunk dynamics and their effect on the vehicle at speeds below aerodynamic control velocities have not been studied. The worst of these effects results in pitch, roll and heave perturbations that could possibly become unstable under certain conditions. These actions have appeared on at least one full scale test vehicle. Using roll stabilizers controlled by the roll channel of the autopilot is only a partial solution and needs further analysis. But solutions for pitch oscillations and vertical perturbations are not as readily available. Just the simple things, such as a changing friction coefficient on the dragging trunk surface, can excite these two conditions and, when coupled with varying terrain and maneuvers, it becomes worse. A possible solution could be roll and pitch reaction nozzles, thereby eliminating the need for roll stabilizing doors, and couple the nozzles to the pitch and roll axis of the autopilot. This would not eliminate heave, as heave is caused by cushion pressure variations. But this should result in low amplitude, low frequency perturbations and may not be too disturbing to the pilot. One solution is always available, and that is to stop the forward motion of the vehicle. A true simulation would require a complex computer program. Further study is needed in this area.

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APPENDIX A

TRUNK ANALYSIS

The trunk planform used for this study was originally sized for the following design conditions:

(GW)	- Maximum Design Weight	=	27000 Lb
(P_C/P_T)	- Cushion/Trunk Pressure Ratio	=	0.5
P_C	- Cushion Pressure	=	180 PSFG
P_T	- Trunk Pressure	=	360 PSFG
w	- Maximum Cushion Width	=	8.75 Ft (Approx 20% Wing Span)
A_C	- Cushion Area	=	150 Sq Ft
S_W	- Wing Area	=	315 Sq Ft

In addition to the above design conditions, it was decided that the trunk planform would be "egg" shape to improve trunk stability, maximize trunk width, and minimize the ground tangent circumference, for a given cushion area. The selected trunk shape will also maximize the trunk footprint at landing impact.

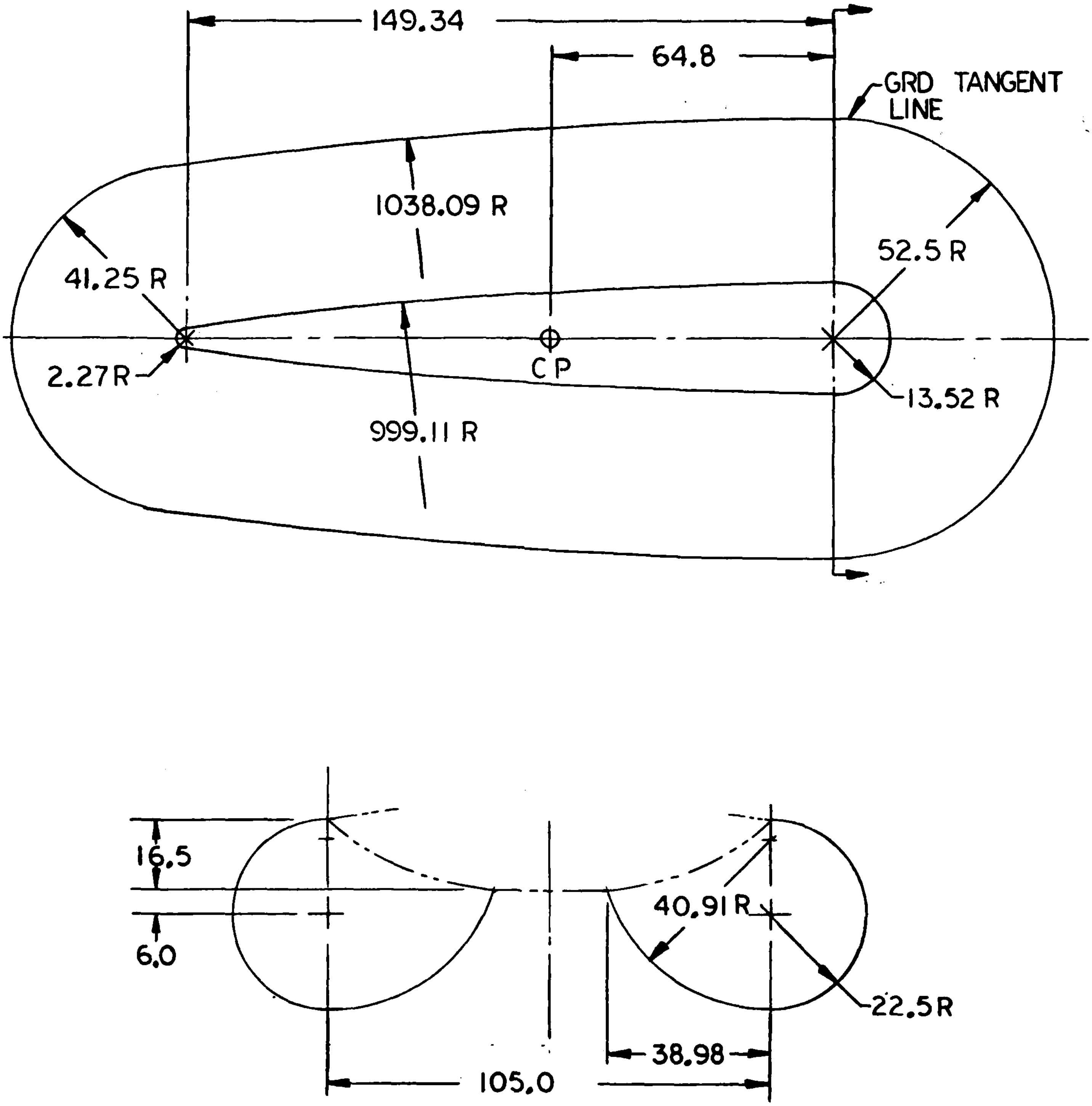
The outer trunk radius (r), at the design hover condition, was sized by static hover over water. That is, the aircraft would not be allowed to sink to a depth greater than the outer trunk radius (r).

The above design constraints are all reasonable, when compared with previous studies, with the possible exception of the trunk width. The trunk width was a compromise between roll stiffness requirements during ground operations and the size of the external fairing (sponsons) required to mount the trunk system. A low wing configuration would be preferable and would permit the installation of a more optimum trunk planform; however, the design requirement for the aircraft to carry 12-MK-82 stores under the wing seems to dictate a high wing configuration. To improve the roll stiffness of this configuration, roll stabilizers were added.

To meet the static flotation requirements, a mimimum outer trunk radius (r) of 22.5 inches is required. This resulted in an inner trunk radius (R) of 45 inches for the design cushion/trunk pressure ratio of 0.5. These trunk radii did not allow sufficient space to attach the inner trunk to the bottom of the aircraft. The design cushion/trunk pressure ratio was reduced to 0.45, which resulted in a trunk inner radius (R) of 40.909 inches and permitted sufficient space (minimum) to attach the inner trunk to the bottom of the aircraft. The trunk pressure was raised to 400 PSFG to obtain the 0.45 pressure ratio.

The above design constraints resulted in the cushion footprint and the trunk cross-section shown in Figure 1A.

The aircraft was then resized using new engine data (P&W STF-529) and the trunk configuration described above. The aircraft gross weight was reduced to



CUSHION FOOTPRINT AND
TRUNK CROSS SECTION

FIGURE 1 A

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W_{TO} = 24300 pounds, and the wing area was reduced to 280 square feet. The final cushion planform and trunk cross-section used for this study (Figure 1A) are based on the following design conditions:

W_{TO}	=	24300 Lb	
P_C/P_T	=	.45	
P_C	=	162 PSFG	
P_T	=	360 PSFG	
A_C	=	150 Ft^2	
S_W	=	280 Ft^2	
w	=	8.75 Ft	(Approx. 21.3% Wing Span)

If time had permitted, the cushion width would have been increased and the cushion length reduced to increase the radius of the inner attach line at the forward torus (as shown on Page 34). This would reduce the stress on the trunk materials and improve roll stiffness.

The requirement for the elastic trunk to hug the bottom of the fuselage when it is deflated requires the use of two-way stretch material with a programmed memory. For this study, the trunk material was assumed to be similar to that used for the XC-8A program. The trunk is constructed of nylon tire chord wound around a natural rubber core. This is sandwiched between natural rubber sheets and molded into a homogenous sheet. The attachment holes and nozzles are also molded into the trunk material. By varying the number of coils per inch, the individual tapes can be programmed to have the desired elongation characteristics. The material stretch characteristics assumed for this study are shown in Figure 2A. These characteristics represent the average conditions.

For the design hover conditions, the ratio of relaxed length to stretched length (ℓ/ℓ_o) of the trunk material is 3.0 with a tension of 56.25 lb/in (at the ground tangent line). When deflated, the trunk elongation ratio, prestretch, varies from 1.066 on the sides to 1.77 at each end. The bottom surface of the aircraft is curved to minimize trunk flagellation during flight. The radius of curvature on the bottom of the aircraft varies from 300 inches at the front and rear to 54.3 inches on the sides. This results in the trunk tension, when deflated, to be 3 pounds per inch and 19 pounds per inch on the sides and ends, respectively. If the tension in the trunk material is:

$$T = \Delta p \times R \quad \text{Lb/in}$$

Then $\Delta p = T/R$

Where: Δp = the differential pressure (psi) across the trunk material required to pull the material away from the fuselage.

R = the radius of curvature (in)

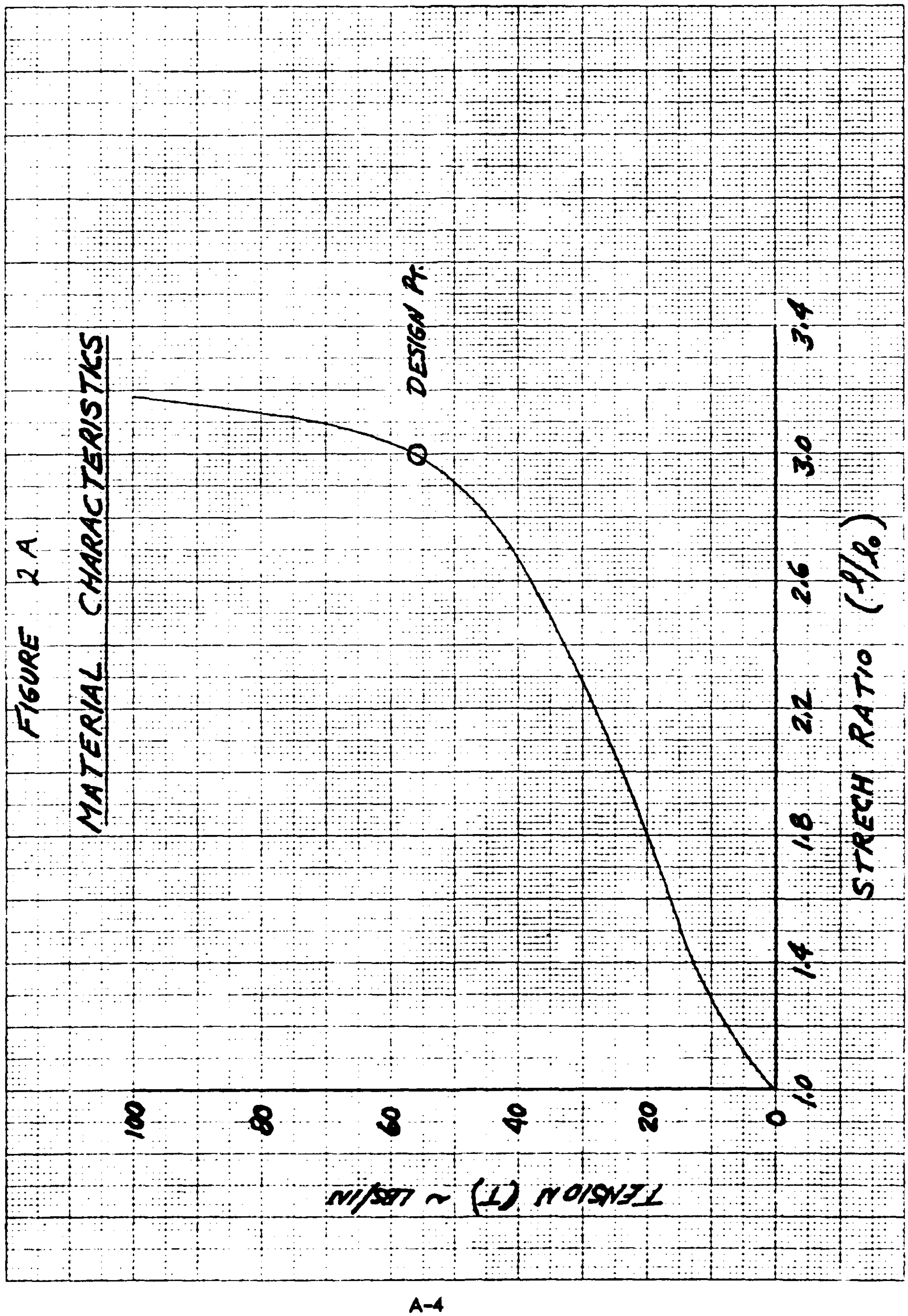


FIGURE 2A
MATERIAL CHARACTERISTICS
DESIGN PT.
100
80
60
40
20
0
1.0
1.4
1.8
2.2
2.6
3.0
3.4
STRECH RATIO (l/l₀)
TENSION (T) ~ LBS/IN

Trunk Ends Δp = 19/300 = .0633 psi

Trunk Sides Δp = 3/54.3 = .05525 psi

It is anticipated that in high speed flight at low altitudes, the differential pressure across the trunk material might be as high as 1.0 psi. It was therefore concluded that prestretch of the trunk material by itself would not be sufficient to hold the trunk material against the bottom of the fuselage in the trunk deflated configuration. Therefore, a suction system has been utilized to hold the deflated trunk in place during flight (see Page 25).

The equations used to determine the tension in the trunk material at the design hover condition are defined on Figure 3A. The calculated trunk tensions are summarized in the table shown on Figure 4A.

The effect of cushion/trunk pressure ratio on various trunk parameters during hover was calculated. These data are presented in Figures 5A through 9A. The relation between cushion lift and pressure ratio is shown in Figure 10A.

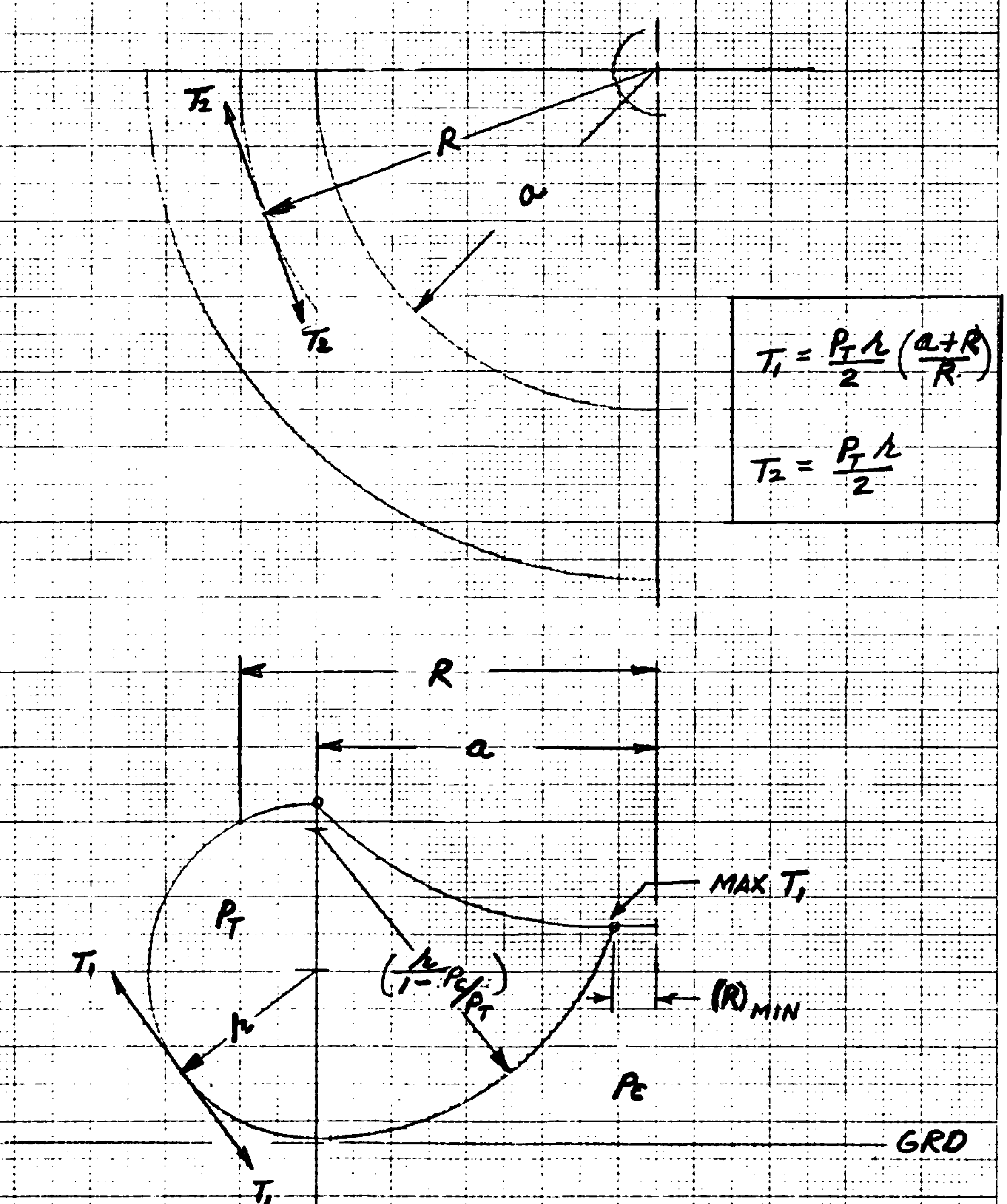
Parking

To park the aircraft, four bladders inside the trunk are inflated to 275 PSFG. The trunk pressure was selected such that maximum stretch ratio (fore and aft trunk torus) was 3.0, and the cushion pressure was zero ($P_C/P_T = 0$). This resulted in a trunk radius of 29.4 inches for the forward and aft torus, and a trunk radius of 26.76 inches at the sides, when no load is on the trunk. However, due to the trunk attachment locations, the distance from fuselage hard structure to the bottom of the trunk (H) is 33.6 inches and 30 inches for the sides and ends of the trunk, respectively. Calculations were made to determine the distance from the ground to the bottom of the aircraft (H) as a function of aircraft gross weight when setting on the inflated parking bladders. The results of these calculations are presented in Figure 11A.

With the parking bladders inflated, it is possible to park the aircraft in water. At maximum gross weight (24300 pounds), the aircraft would float with the hard structure approximately six inches out of the water.

FIGURE 3 A

TRUNK TENSION



T_1 ~ GIRTH TENSION
T_2 ~ PERIPHERAL TENSION

a, r, R ~ INCHES
P_C, P_T ~ PSIg
T_1, T_2 ~ LBS / IN

FIGURE 4A

SUMMARY TRUNK TENSION
(DESIGN HOVER CONDITION)

	P_C PSIG	P_T PSIG	a (in)	r (in)	R (in)	T_1 (Lbs/in)	T_2 (Lbs/in)
Front Torus at Fuselage C.L.							
Inboard Attach	1.125	2.5	41.25	22.5	2.2684	539.57*	28.125
Ground Tangent	"	"	"	"	41.25	56.25	"
Maximum R	"	"	"	"	63.75	46.32	"
Outboard Attach	"	"	"	"	60.4684	47.31	"
Side Trunk							
Inboard Attach	1.125	2.5	1038.094	22.5	999.113	57.347	28.125
Ground Tangent	"	"	"	"	1038.094	56.25	"
Maximum R	"	"	"	"	1060.594	55.653	"
Outboard Attach	"	"	"	"	1038.094	56.25	"
Aft Torus at Fuselage C.L.							
Inboard Attach	1.125	2.5	52.5	22.5	13.5184	137.351	28.125
Ground Tangent	"	"	"	"	52.5	56.25	"
Maximum R	"	"	"	"	75.0	47.813	"
Outboard Attach	"	"	"	"	71.7184	48.713	"

*By increasing minimum R from 2.2684 inches to 7.627 inches and (a) to 46.6116 inches, T_1 would be reduced from 539.57 Lbs/in to 200 Lbs/in (ΔR_{min} = 5.3586 in).

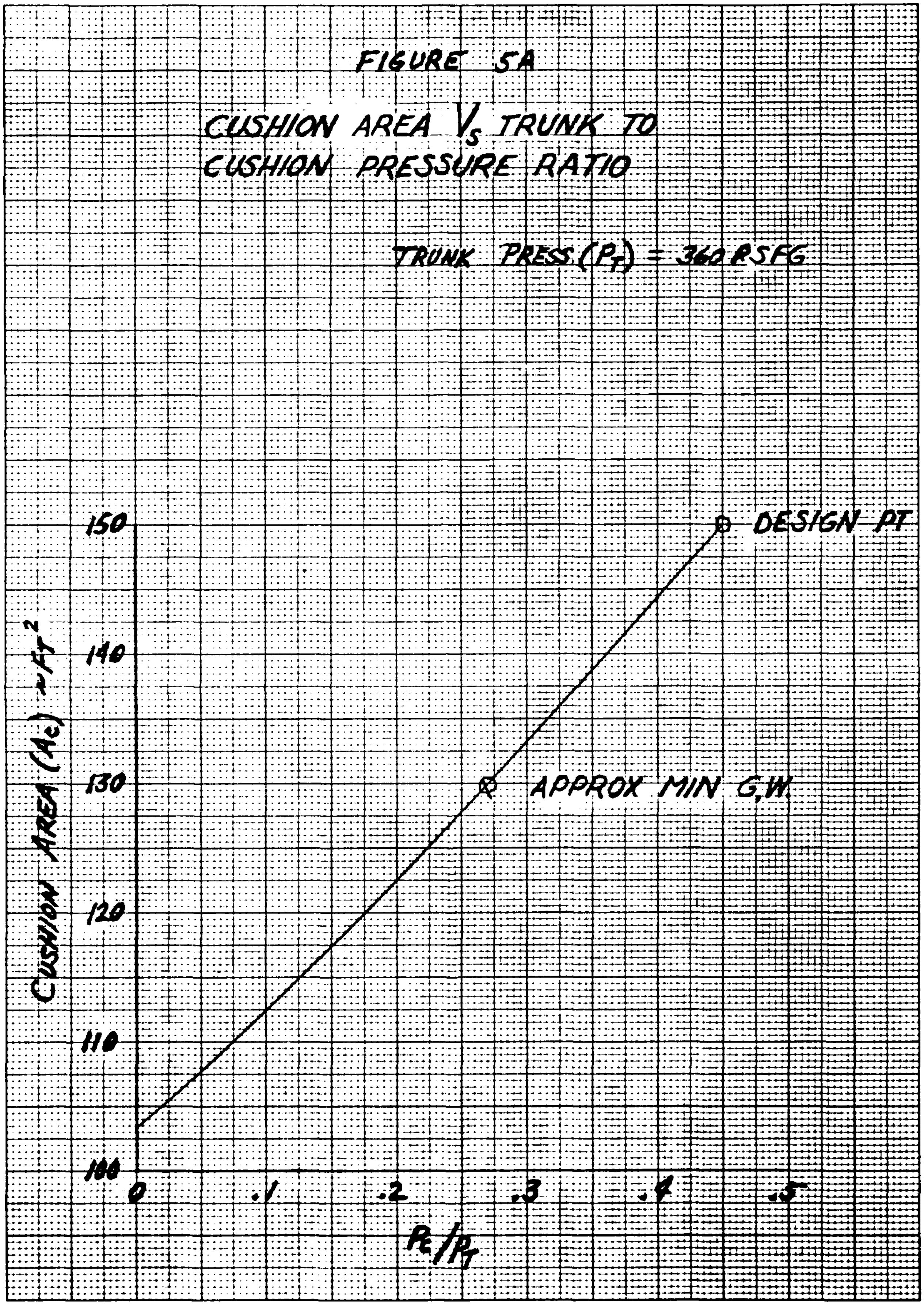


FIGURE 5A
CUSHION AREA Vs TRUNK TO CUSHION PRESSURE RATIO
TRUNK PRESS (PT) = 360 PSFG
DESIGN PT
APPROX MIN G.W.
150
140
130
120
110
100
CUSHION AREA (Ac) ~ FT2
0
.1
.2
.3
.4
.5
Pc/PT

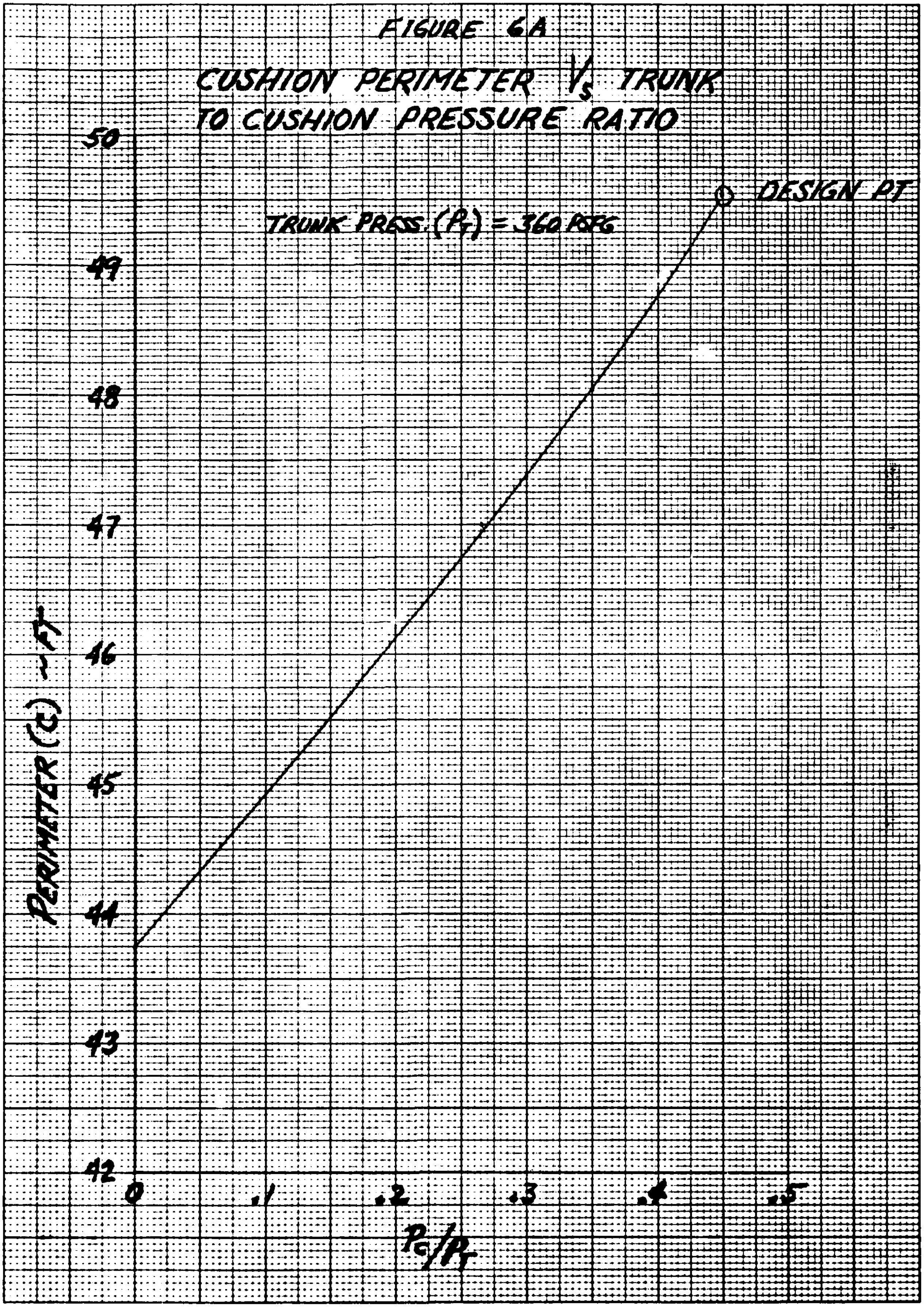


FIGURE 6A
CUSHION PERIMETER Vs TRUNK TO CUSHION PRESSURE RATIO
TRUNK PRESS. (P_T) = 360 PSFG
DESIGN PT
PERIMETER (ℓ) ~ FT
50
49
48
47
46
45
44
43
42
0
.1
.2
.3
.4
.5
P_C/P_T

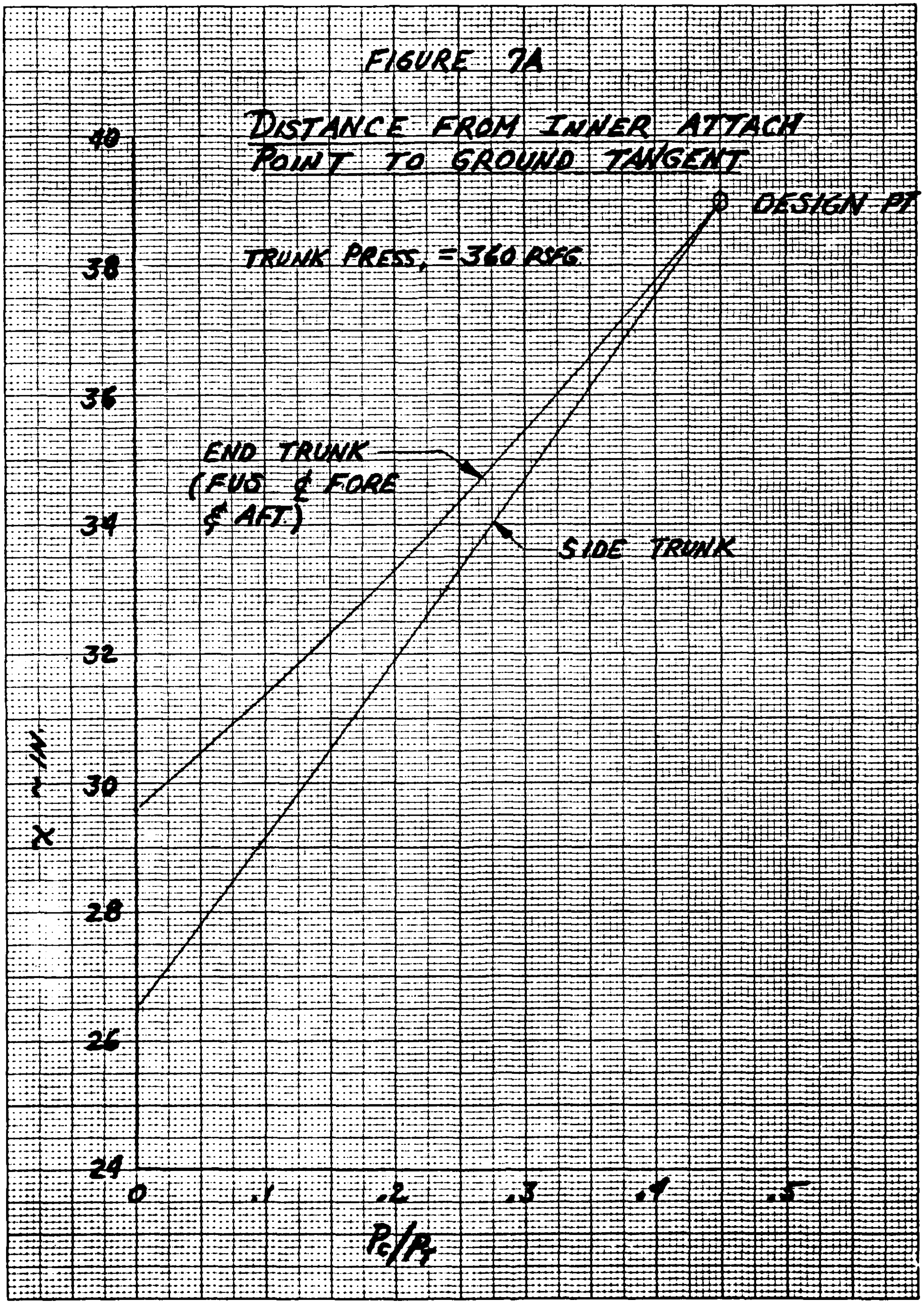
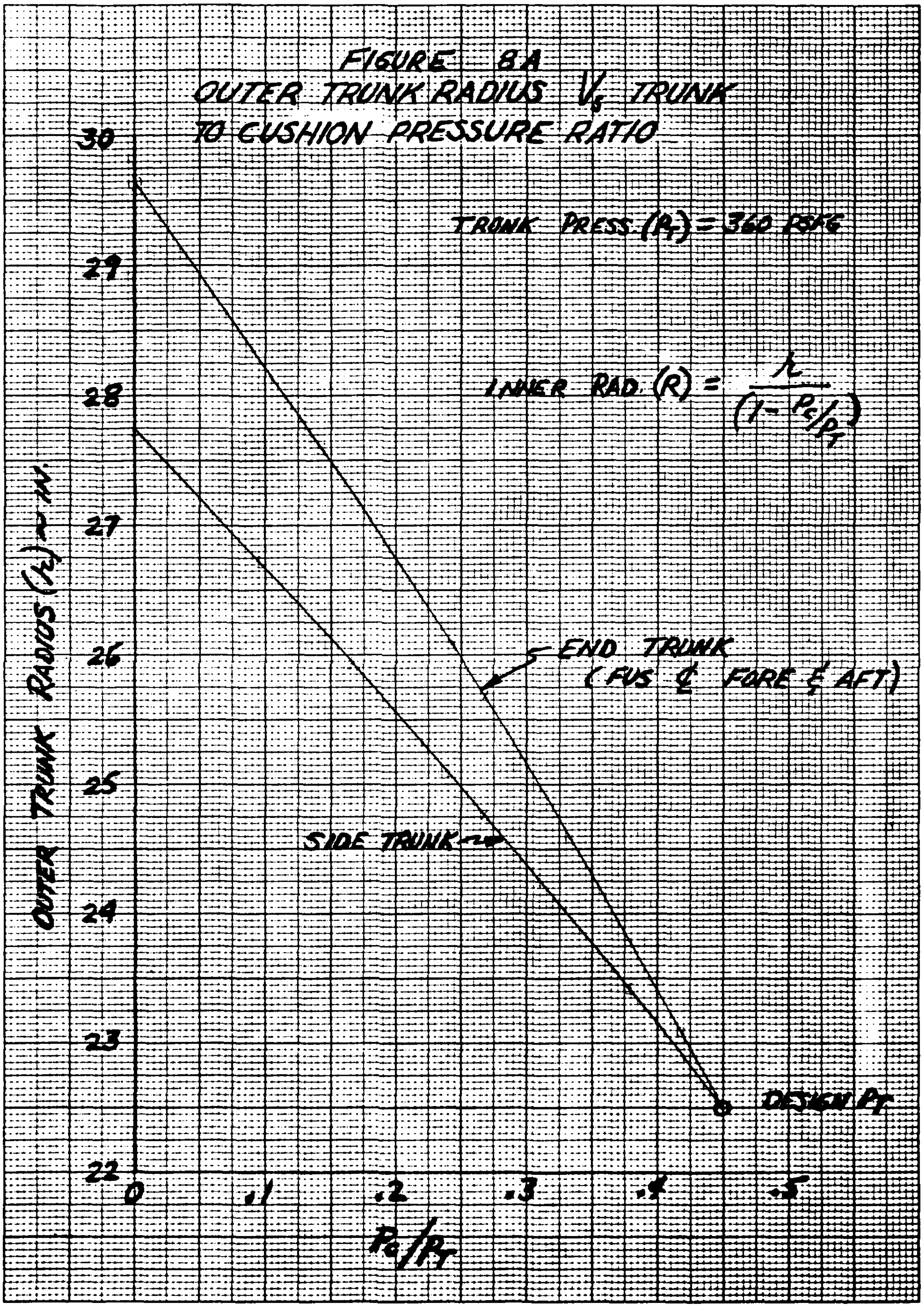


FIGURE 7A
DISTANCE FROM INNER ATTACH POINT TO GROUND TANGENT
DESIGN PT
TRUNK PRESS. = 360 PSFG
END TRUNK (FUS ¢ FORE ¢ AFT)
SIDE TRUNK
40
38
36
34
32
30
28
26
24
X, IN.
0
.1
.2
.3
.4
.5
Pc/PT



K•E 10 X 10 TO ½ INCH 7 X 10 INCHES KEUFFEL & ESSER CO. MADE IN U.S.A. 46 1327

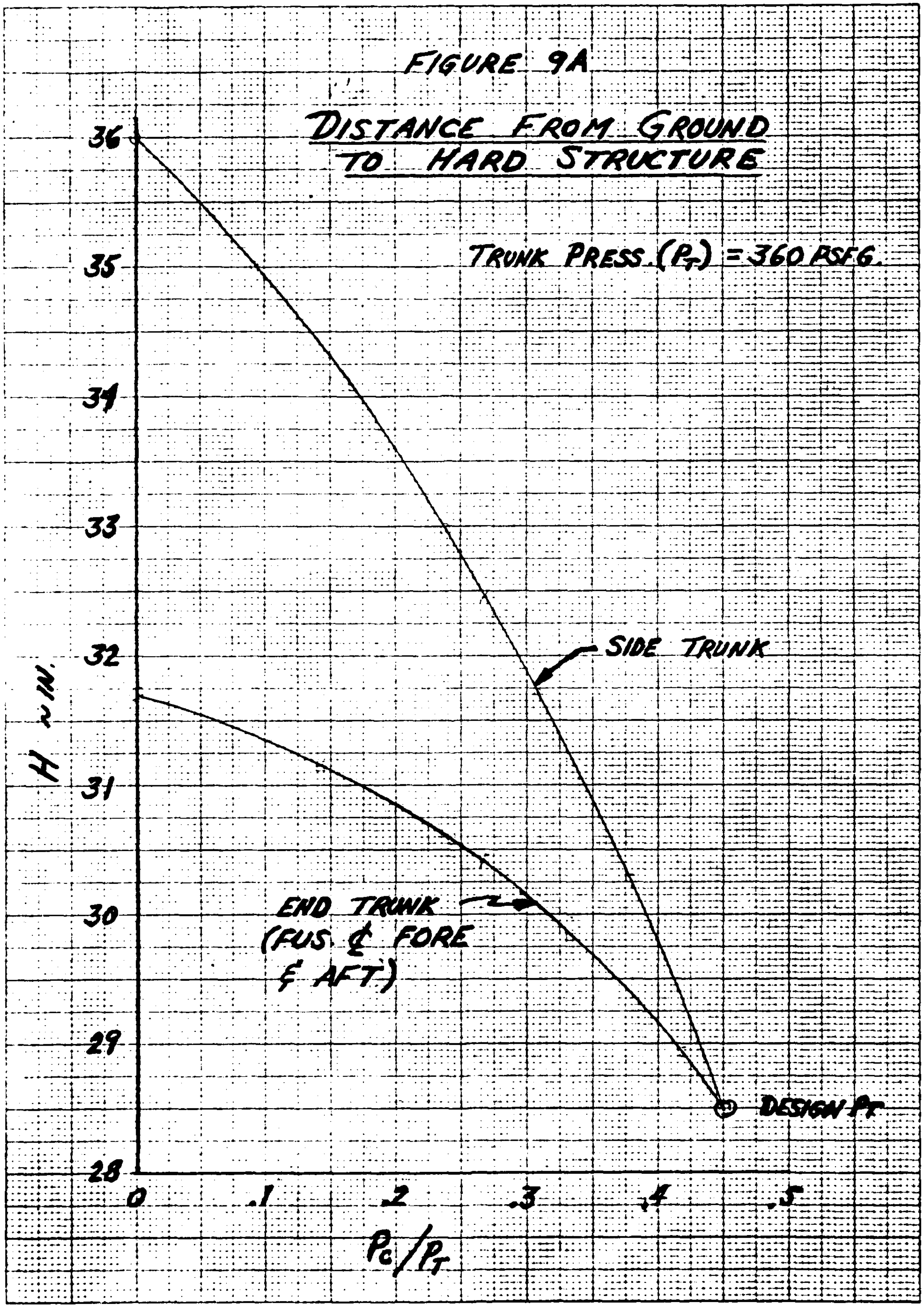


FIGURE 9A
DISTANCE FROM GROUND TO HARD STRUCTURE
TRUNK PRESS. (P_T) = 360 PSFG.
SIDE TRUNK
END TRUNK (FUS ℄ FORE ℄ AFT)
DESIGN PT
H ~ IN.
P_C/P_T
36
35
34
33
32
31
30
29
28
0
.1
.2
.3
.4
.5

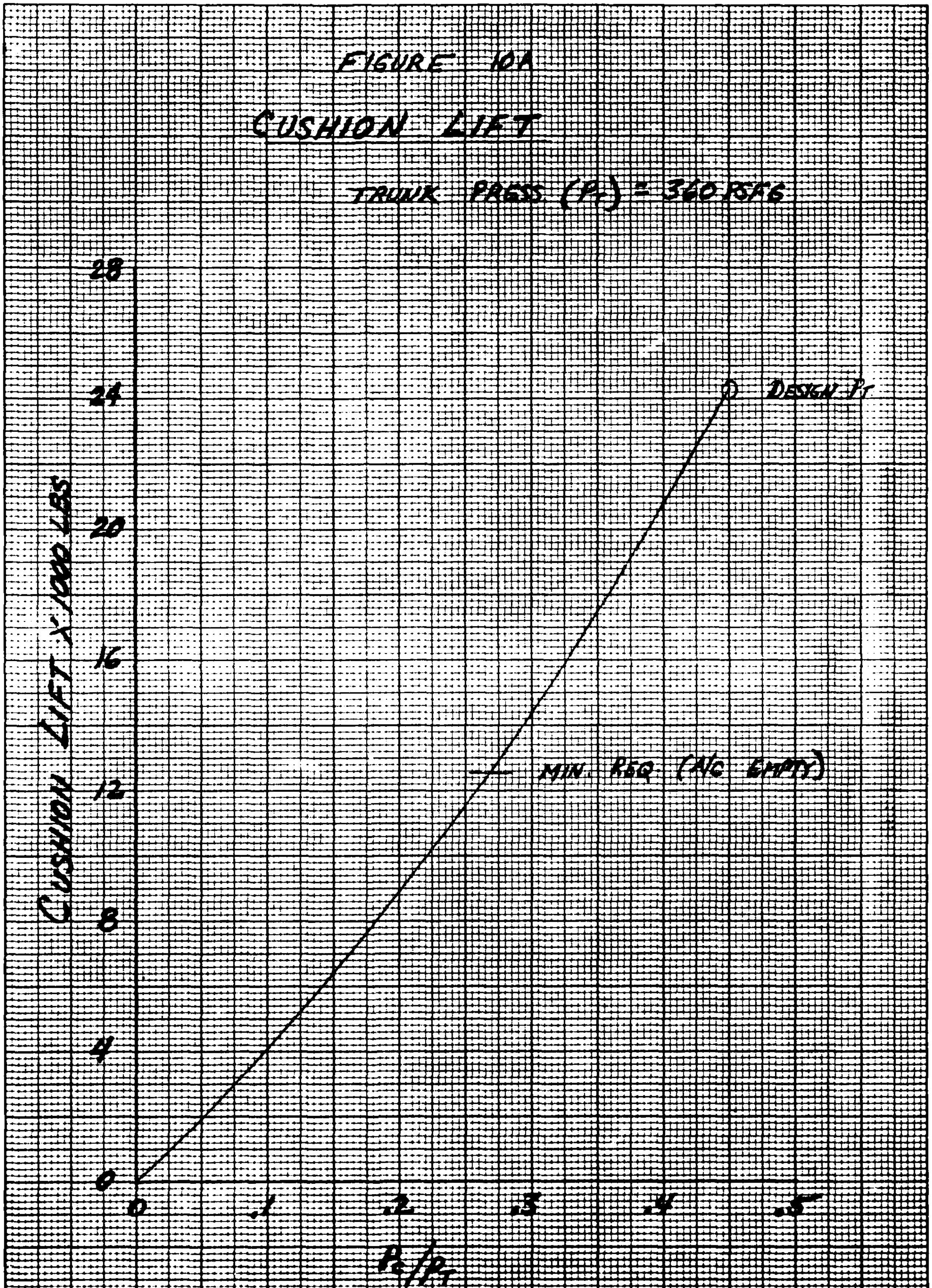


FIGURE 10A
CUSHION LIFT
TRUNK PRESS (P_T) = 360 PSFG
DESIGN P_T
MIN. REQ (A/C EMPTY)
CUSHION LIFT X 1000 LBS
0
4
8
12
16
20
24
28
0
.1
.2
.3
.4
.5
P_C/P_T

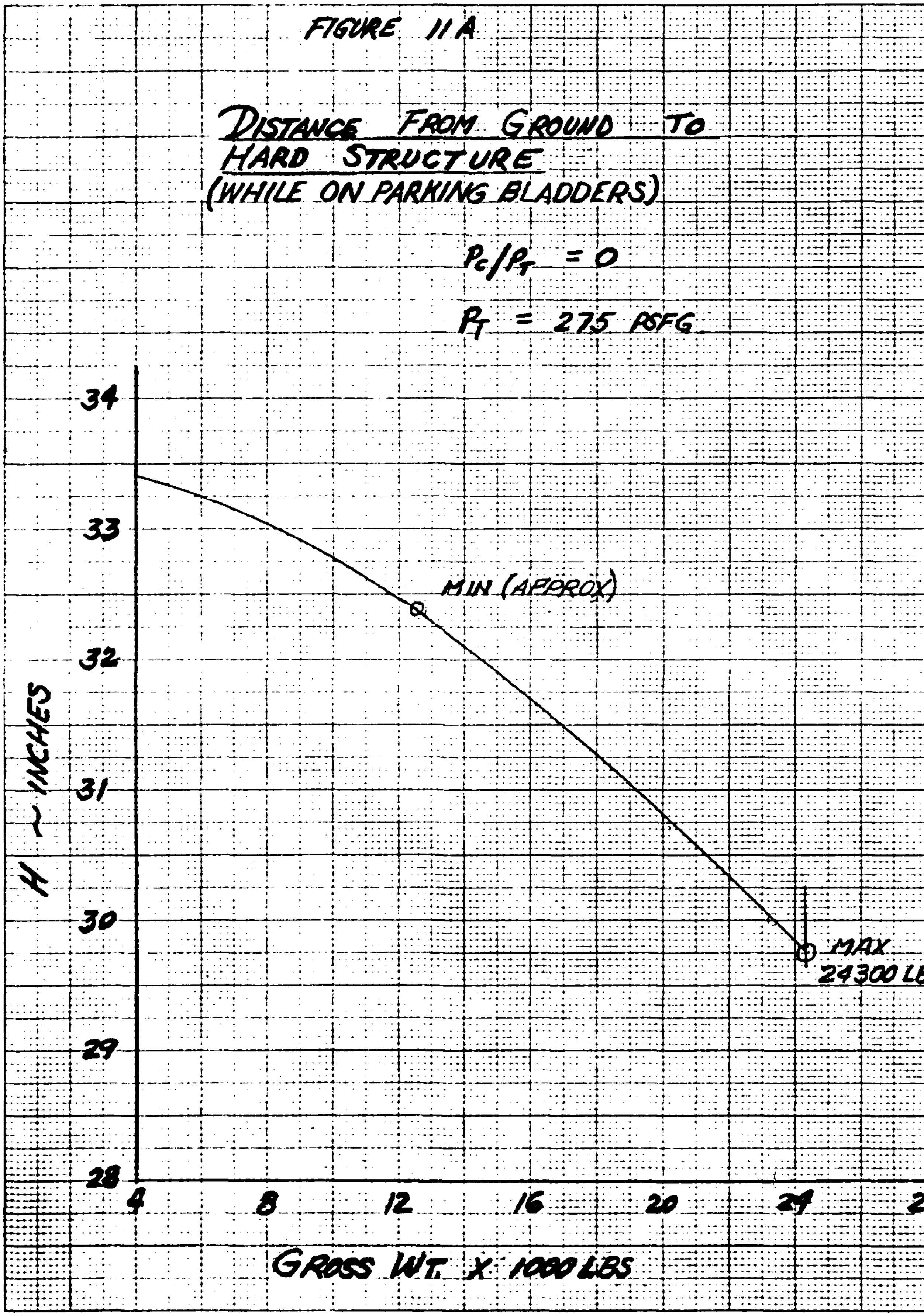


FIGURE 11A
DISTANCE FROM GROUND TO HARD STRUCTURE
(WHILE ON PARKING BLADDERS)
$P_C/P_T = 0$
P_T = 275 PSFG
MIN (APPROX)
MAX
24300 LB
H ~ INCHES
34
33
32
31
30
29
28
4
8
12
16
20
24
28
GROSS WT. X 1000 LBS

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Air Flow

Design Conditions

1. Cushion Area (A_C) = 150 Ft^2
2. Cushion Perimeter (C) = 49.528 Ft
3. Cushion Press (P_C) = 162 PSFG
4. T.O. Weight (W_{TO}) = 24300 Lb
5. Trunk Pressure (P_T) = 360 PSFG
6. P_C/P_T = .45
7. Engine Bleed Air Press (P_E) = 36.7 PSIA
8. Engine Bleed Air Temp (T_E) = 730° R

Assumptions

1. Twenty percent (20%) of the air flow from the trunk will be discharged to atmosphere (outside the ground tangent line of the trunk) to provide an air bearing when the trunk is flattened against the ground.

2. Bleed air from the engine will pass through a sonic orifice to provide a constant air flow.

3. Trunk pressure will be maintained at a constant pressure (P_T = 360 PSFG). Variable orifices (controlled by trunk pressure) will vary the trunk flow to the cushion to maintain a constant trunk pressure.

4. Total Air Flow $(W_a)_{Tot}$ = 39.00 lb/sec

   Note: This air flow was calculated earlier, based on a mean air gap ($\bar{h}$) = .25 in. The engine was sized for takeoff at the air flow. The mean air gap ($\bar{h}$) is corrected, herein, based on the updated bleed air temperature and pressure shown above.

5. Discharge Coefficient (C_d)

   (a) C_d = .66 (from trunk)

   (b) C_d = 1.0 (from cushion)

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Basic Equations

1. Air flow from trunk to atmosphere (hover)

$$(W_a)_1 = \rho g \; C_d \; (A_J)_1 \; [2 g \; P_T / \rho g]^{0.5}$$

$(A_J)_1$ = Area trunk orifices to atmos (Ft^2)

P_T = 360 PSFG

ρg = .070623 Lb/Ft^3

C_d = .66

$$(W_a)_1 = 26.7061 \; (A_J)_1$$

2. Air flow from trunk to cushion (hover)

$$(W_a)_2 = \rho g \; C_d \; [(A_J)_2 + (A_v)] \; [2g \; (P_T - P_C)/\rho g]^{0.5}$$

$(A_J)_2$ = Area trunk orifices to cushion (Ft^2)

A_v = Area variable trunk vents to cushion (Ft^2)

P_T = 360 PSFG

P_C = 162 PSFG

ρg = .074452 Lbs/Ft^3

C_d = .66

$$(W_a)_2 = 20.3347 \; [(A_J)_2 + A_v]$$

3. Air flow from cushion to atmosphere (hover)

$$(W_a)_3 = \rho g \; (C_d)_3 \; C \; \bar{h} \; [2g \; P_c / \rho g]^{0.5}$$

C = Cushion perimeter (Ft)

$\bar{h}$ = Mean air gap (Ft)

P_C = 162 PSFG

ρg = .070623 (Lb/Ft^3)

C_d = 1.0

$$(W_a)_3 = 1344.3870 \; \bar{h}$$

Determine Mean Air Gap ($\bar{h}$)

$$(W_a)_3 = (W_a)_2$$

$$\therefore [(A_J)_2 + A_v] = 66.1129\ \bar{h}$$

$$(W_a)_{Tot} = (W_a)_1 + (W_a)_2$$

$$\text{But } (W_a)_1 = .2\ (W_a)_{Tot}$$

$$\therefore (W_a)_{Tot} = (W_a)_2 / .8 = 1680.4838\ \bar{h} = 39.00$$

Then $\bar{h}$ = .023208 Ft (.2785 in)

Determine Area of Orifices in Trunk

1. Area of trunk orifices to atmosphere (hover)

$$(A_J)_1 = .2\ (W_a)_{Tot} / 26.7061 = \underline{\underline{.29207}}\ Ft^2$$

2. Area of trunk orifices to cushion in free air

   Variable vents closed and P_T = 360 PSFG and ρg = .070623 lbs/ft^3

$$(W_a)_t = 39.0 = 26.7061\ [(A_J)_1 + (A_J)_2]$$

$$\therefore (A_J)_2 = (39/26.7061) - (A_J)_1$$

$$\text{Then } (A_J)_2 = \underline{\underline{1.16827}}\ Ft^2$$

3. Total orifice area in trunk

$$(A_J)_t = (A_J)_1 + (A_J)_2$$

$$\text{Then } (A_J)_t = \underline{\underline{1.46034}}\ Ft^2$$

$$\frac{(A_J)_1}{(A_J)_t} = .20$$

Determine Maximum Required Variable Vent Area from Trunk to Cushion (To Maintain P_T = 360 PSFG)

From Above

$$(A_J)_2 + A_v = 66.1129\,\bar{h} \quad Ft^2$$

$$\bar{h} = .023208 \quad Ft^2$$

$$(A_J)_2 = 1.16827 \quad Ft^2$$

Total flow area to cushion at design hover

$$A_v + (A_J)_2 = \underline{\underline{1.5343}} \quad Ft^2$$

$$\text{Then } A_v = \underline{\underline{.36605}} \quad Ft^2$$

Summary of Data at Design Hover Condition

Weight Takeoff	(W_{TO})	=	24300 Lb	
Cushion Area	(A_C)	=	150 Ft^2	
Cushion Pressure	(P_C)	=	162 PSFG	
Trunk Pressure	(P_T)	=	360 PSFG	
Cushion Perimeter	(C)	=	49.528 Ft	
Air Flow (Total)	(W_{a_t})	=	39.00 Lb/Sec	
Trunk Orifice Area	(A_{J_t})	=	1.4603 Ft^2	(20% of this is outboard of ground tangent)
Variable Vent Area	(A_v)	=	.36605 Ft^{2*}	(Trunk to cushion)
Mean Air Gap	$(\bar{h})$	=	.023208 Ft**	
Air Temperature in trunk		=	128.8°F	
Air Temperature in Cushion		=	115°F	
Air Temp Discharged to Atmosphere		=	103°F	

* Varies with P_C/P_T. Equal to zero in free air.

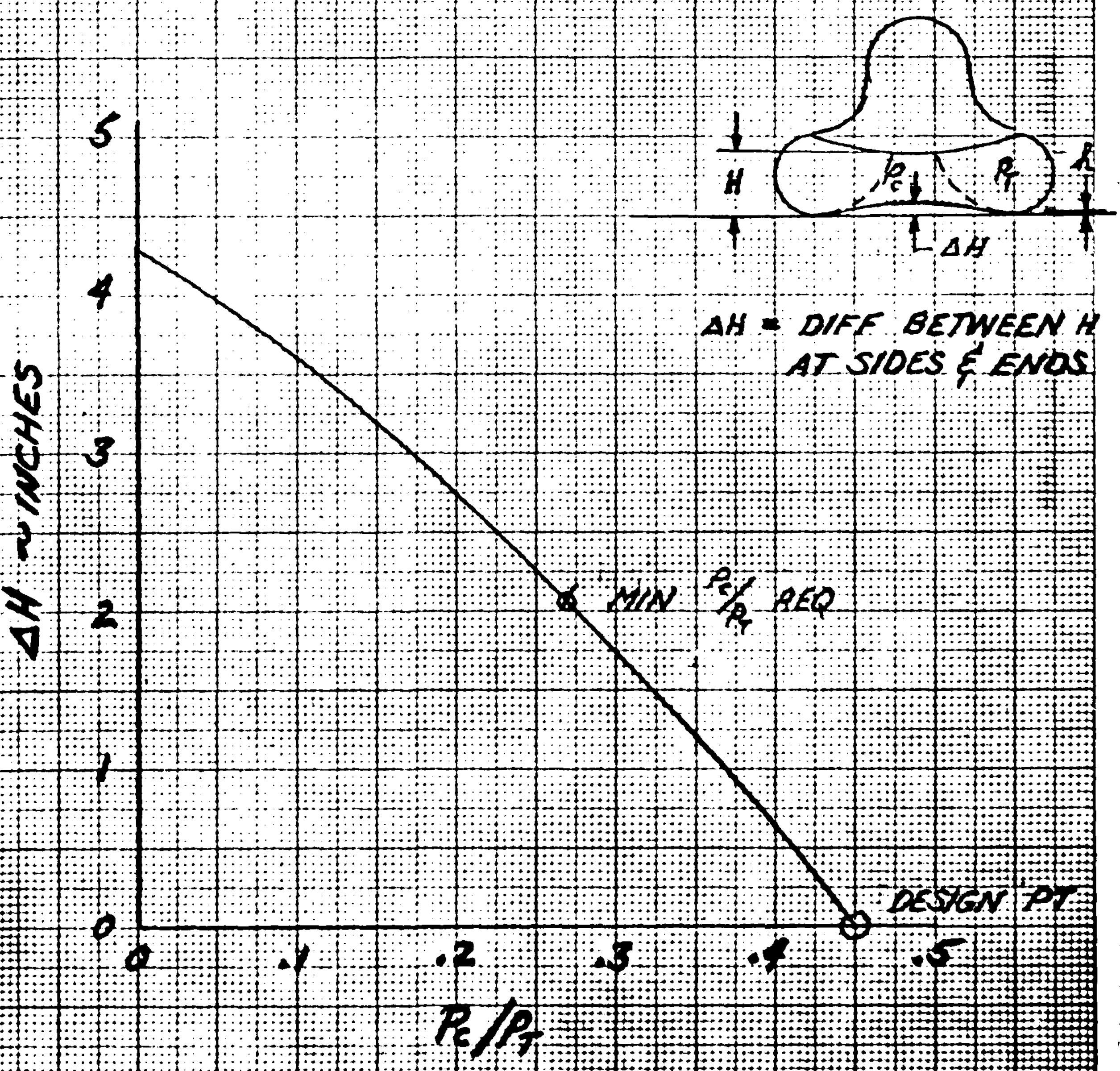
**The effect of P_C/P_T on $\bar{h}$ is shown on Figures 12A and 13A.

FIGURE 12 A

FORWARD & AFT GAP HEIGHT

(HOVER)

P_T = 360 PSFG
$\dot{W}_a$ = 39 LBS/SEC



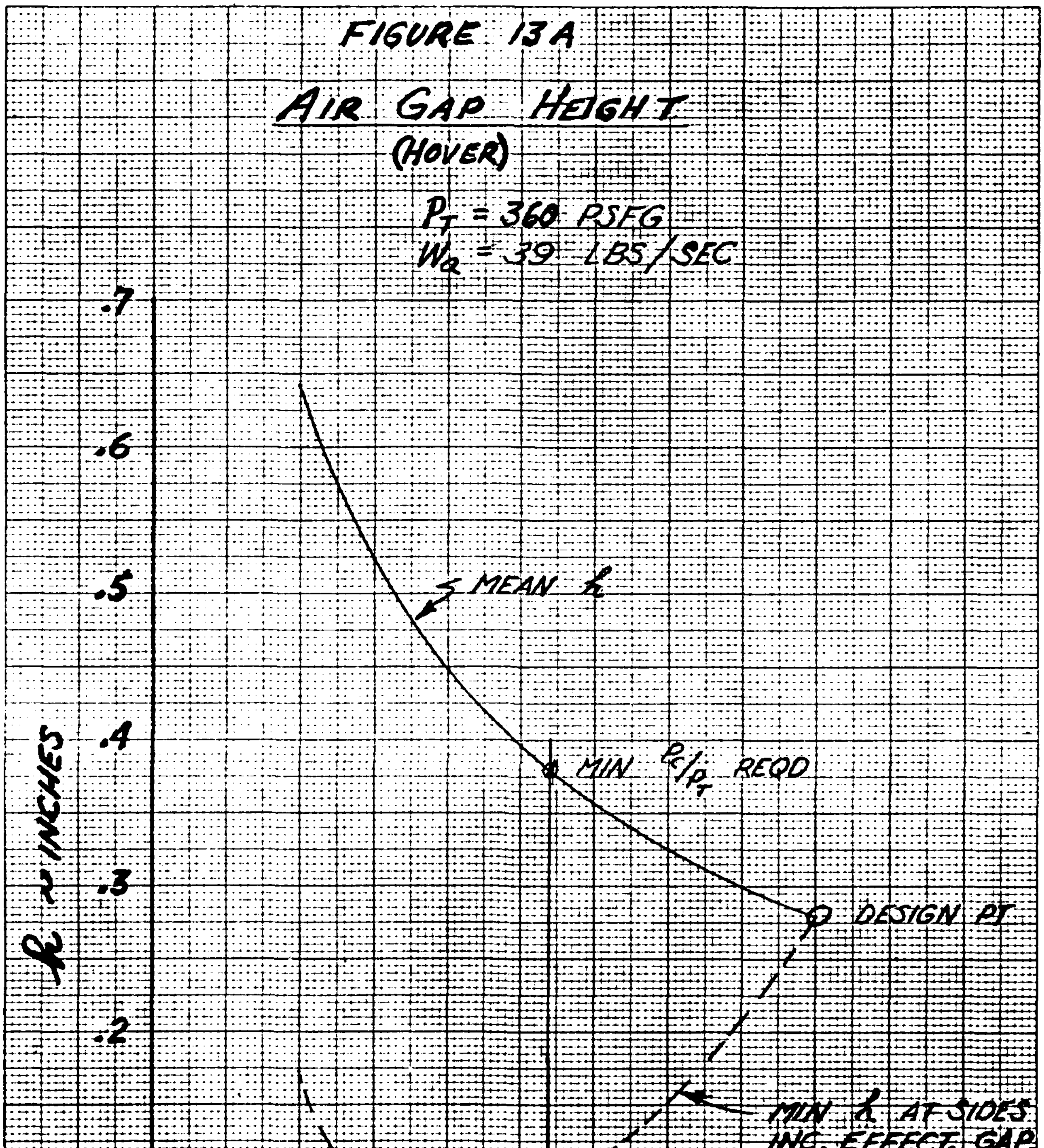


FIGURE 13A
AIR GAP HEIGHT
(HOVER)

Summary of Data at Design Hover Condition (cont'd)

Air Density - Trunk	=	.0790	Lb/Ft^3
Air Density - Cushion	=	.07445	Lb/Ft^3
Air Density - Discharged to Atmosphere	=	.07062	Lb/Ft^3

Trunk Drag (Takeoff)

The center of pressure of the air cushion is located 6.8 inches ahead of the aircraft c.g. During taxi operations, this will cause some of the aircraft weight to be supported by the aft end of the trunk. The drag force due to flattening part of the trunk against the ground will tend to stabilize the aircraft directionally. A force diagram is shown in Figure 14A.

$$L_T = \frac{(T-D)b + aW}{(a + c + \mu d)} \quad = \text{Lift on Trunk (Lbs)}$$

$$F = \mu L_T \quad = \text{Drag on Trunk (Lbs)}$$

At Takeoff (Low Speed)

a	=	6.8 in	
b	=	7.0 in	
c*	=	91.0 in	
d	=	74.5 in	
(T-D)	=	10172 lbs	(D = 0 at low speed)
W	=	24300 lbs	(maximum G.W.)

μ	L_T	F	L_C	$(\mu)_{eff}$
.4	1853	741	22447	.0305
.5	1751	875	22549	.0360
.6	1659	996	22641	.0410

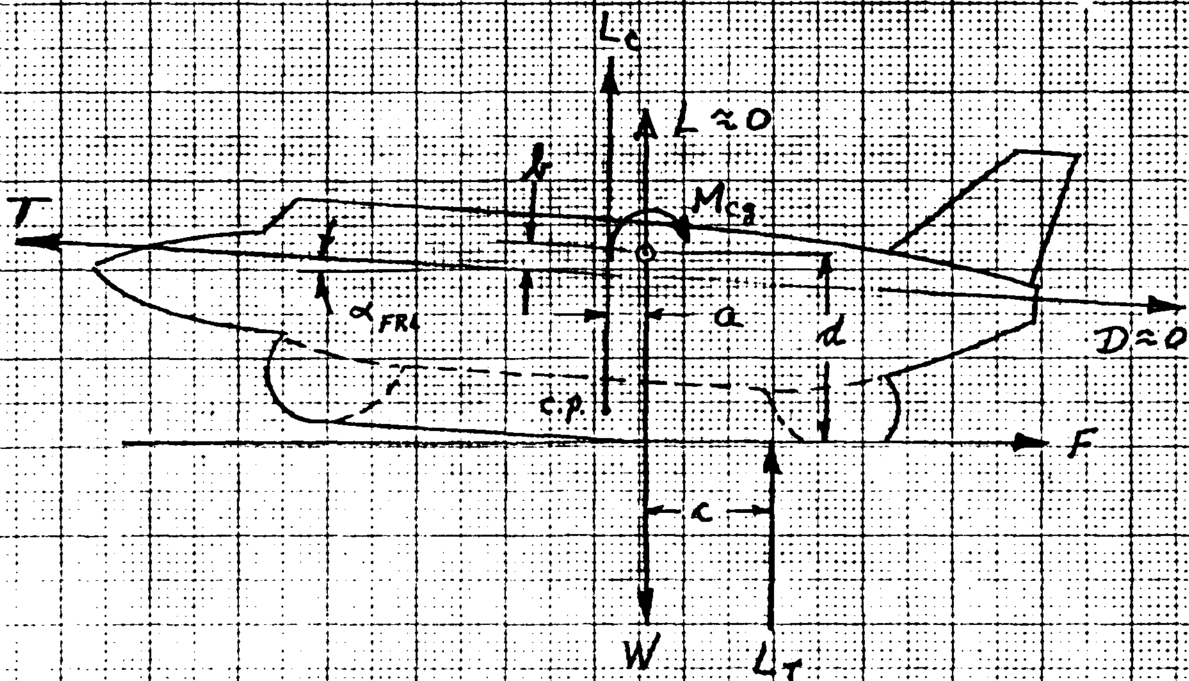
It is apparent from the calculations shown above that the maximum trunk drag during takeoff is approximately 740 to 1000 pounds. This is roughly equivalent to the rolling coefficient of conventional landing gear.

*c is distance from aircraft c.g. to centroid of aft trunk area flattened against the ground.

FIGURE 14A

FORCE DIAGRAM ~ TAKE-OFF

(LOW SPEED; AIRCRAFT LIFT & DRAG ≈ 0)



$$M_{CG} = 0 = (T-D)b + (L_c)a - (L_T)c - (F)d$$

$$F = \mu L_T$$

$$W = L_c + L_T$$

$$\mu_{EFF} = \frac{F}{W}$$

$$L_T = \frac{(T-D)b + (W)a}{(a + c + \mu d)}$$

Trunk Drag (Braking)

The force diagram for the braking condition is shown in Figure 15A. To check the nose down attitude of the aircraft during braking (plough-in), the following assumptions were made:

W	=	21569 Lb	(Maximum landing weight)
P_C/P_T	=	.061	(After brake applied and cushion vented)
P_C	=	22 PSFG	
P_T	=	360 PSFG	
$(A_C)_{eff}$	=	$108.8-27.1 = 81.7\ Ft^2$	($\Delta A = 27.1$ due to flattening trunk on ground)
$\mu_t = k\ \mu$	=	$.25\ \mu$	(Air lube on trunk)
$(T-D)$	=	0	
a	=	6.8 in	(Reference Figure 15A)
b	=	7.0 in	" " "
c	=	29 in	" " "
d	=	79.5 in	" " "
e	=	75 in	" " "

$$L_C = 81.7 \times 22 = 1798 \quad \text{(Lb)}$$

$$L_t = W - L_C - L_b = 19771 - L_b \quad \text{(Lb)}$$

$$F = \mu\,(L_b + k\ L_t) = \mu\,[L_b + k\,(19771 - L_b)] = \mu\,(.75\ L_b + 4943)$$

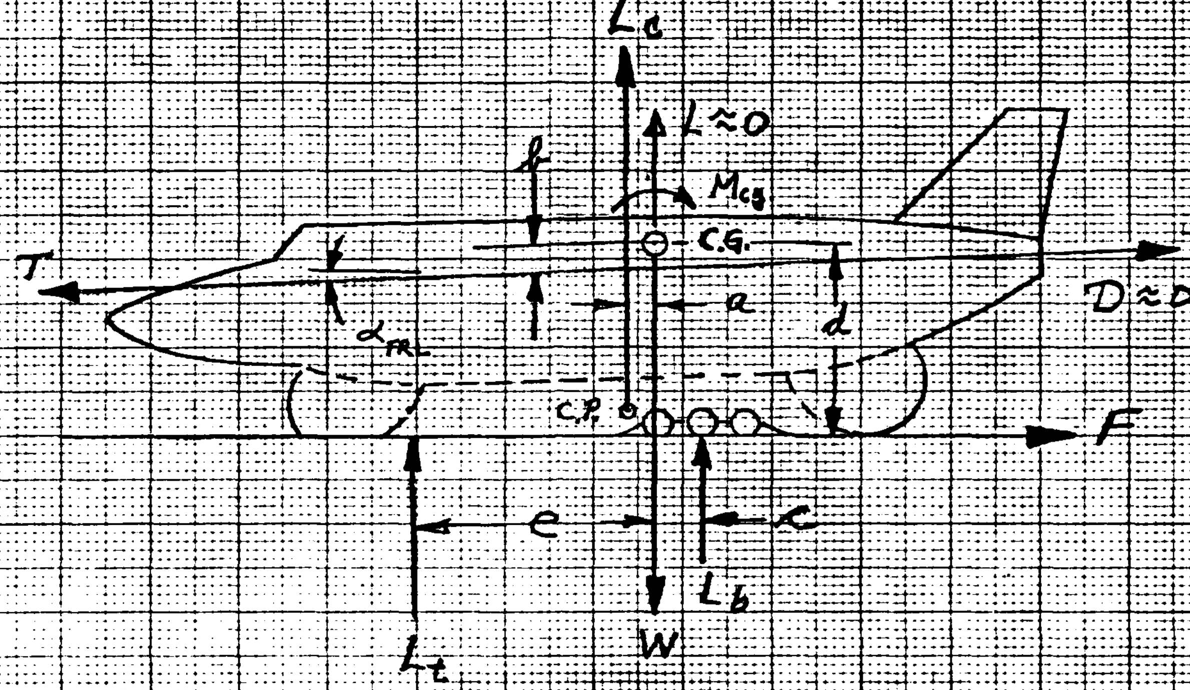
$$L_b = \frac{(T-D)\,b + (L_C)\,a + 19771\,e - 4943\,d\mu}{(c + e + .75\,d\mu)}$$

Using data from Figure 16A and substituting into the equation for L_b, the effective braking coefficient and plough-in angle were calculated as a function of μ.

FIGURE 15A

FORCE DIAGRAM ~ BRAKING

(LOW SPEED ~ AIRCRAFT LIFT & DRAG ≈ 0)



$$M_{cg} = 0 = (T-D)b + (L_c)a + (L_t)e - (L_b)c - (F)d$$

$$L_T = L_t + L_b \quad \text{(LIFT ON TRUNK + LIFT ON BRAKES)}$$

$$F = \mu(L_b + k L_t) \quad \text{WHERE: } k = \mu_t/\mu$$

$$W = L_c + L_T = L_c + L_b + L_t$$

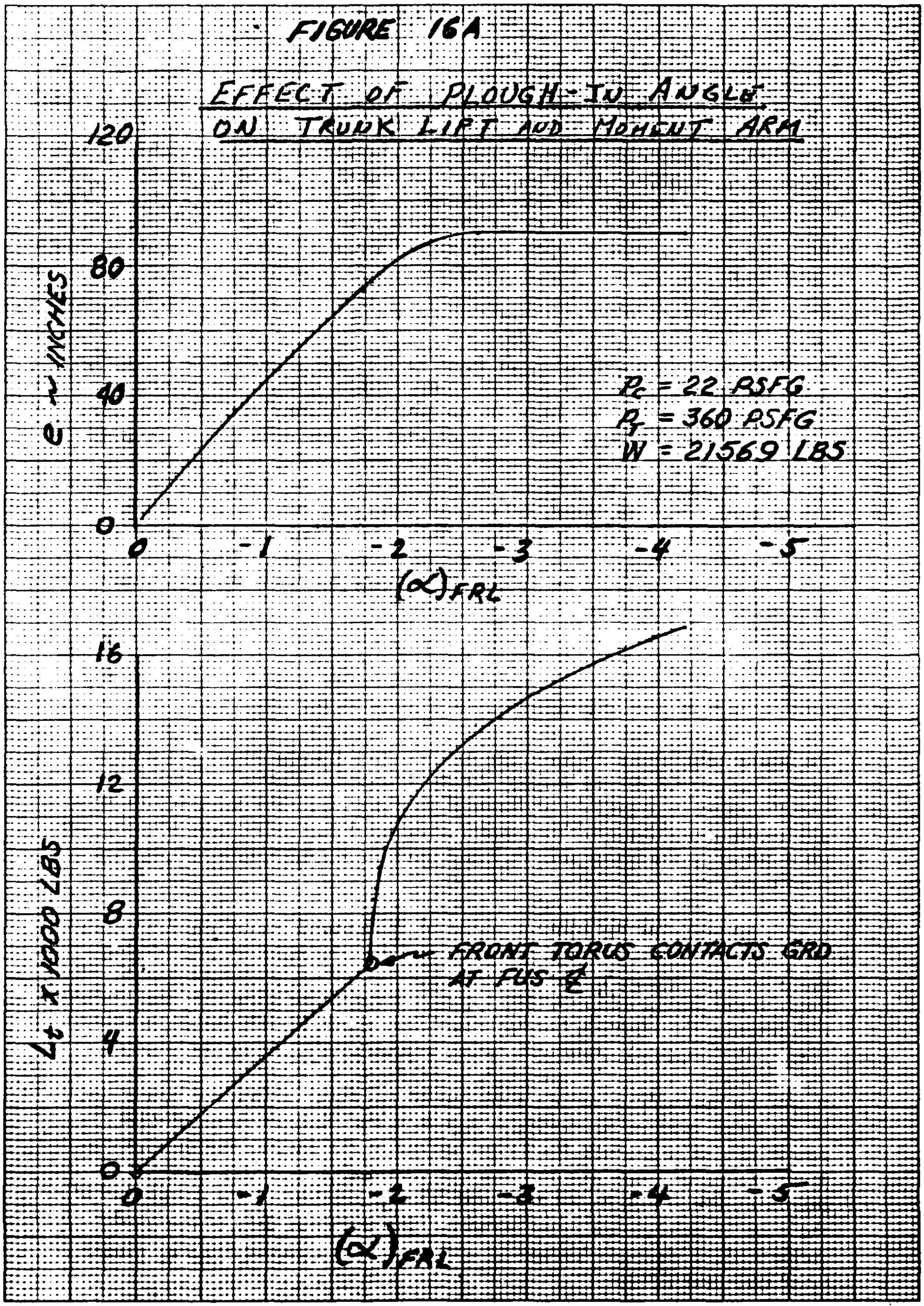


FIGURE 16A
EFFECT OF PLOUGH-IN ANGLE ON TRUNK LIFT AND MOMENT ARM
e ~ INCHES
120
80
40
0
-1
-2
-3
-4
-5
(α)FRL
Pc = 22 PSFG
PT = 360 PSFG
W = 21569 LBS
LT x 1000 LBS
16
12
8
4
FRONT TORUS CONTACTS GRD AT FUS ℄

SANDAIRE

μ	e (in)	L_b (lbs)	L_t (lbs)	L_T (lbs)	F (lbs)	$(\mu)_{eff}$	FRL (deg)	W (lbs)
.3	75.0	11333	8438	19771	4033	.1870	-1.81	21569
.35	76.0	10977	8794	19771	4612	.2138	-1.83	21569
.40	77.0	10643	9128	19771	5170	.2397	-1.85	21569
.45	78.0	10328	9443	19771	5710	.2647	-1.87	21569
.50	79.0	10067	9704	19771	6247	.2896	-1.89	21569
.55	80.0	9717	10054	19771	6727	.3119	-1.91	21569
.60	81.0	9452	10319	19771	7219	.3347	-1.94	21569

The effective braking coefficient is equal to F/W and is plotted as a function of the braking coefficient on the brake pads (Figure 17A). The landing calculations for this report are based on an effective braking coefficient of .27. This will require a braking coefficient of .465 on the brake pads. Selected design value of μ = .465 is reasonable for normal landing surfaces; in fact, values up to .8 might be attained on dry concrete. For other landing surfaces, such as grass, sod, ice, snow, etc., very little data exist to predict the braking coefficient for an air cushion system with pillow brakes. Other systems, such as suction braking, might be used with better results. One possible problem with suction braking could be foreign object ingestion when operated over unimproved or wet surfaces.

The plough-in angles associated with the braking system presented herein should not exceed -2 degrees on hard surfaces. This is low enough to preclude any problems such as pilot discomfort and/or damage to the aircraft.

The above calculations to determine the effective braking coefficient during landing ground run do not account for the favorable effect of deflected thrust (see discussion on Page 28 and Figure 4 on Page 33). The deflected thrust will create a positive pitching moment around the c.g. This will increase the lift required in the brake pads (L_b) and result in a higher effective braking coefficient.

If it is assumed that the minimum thrust of 2500 pounds is deflected 45 degrees from vertical, with a turning efficiency of 70%, the resultant vertical force vector will be approximately 1250 pounds. The effective moment arm of this thrust vector is approximately 140 inches. Calculations show that this moment will increase the effective braking coefficient about 12-14% (shown on Figure 17A). The plough-in angle will reduce approximately .01 to .02 degrees.

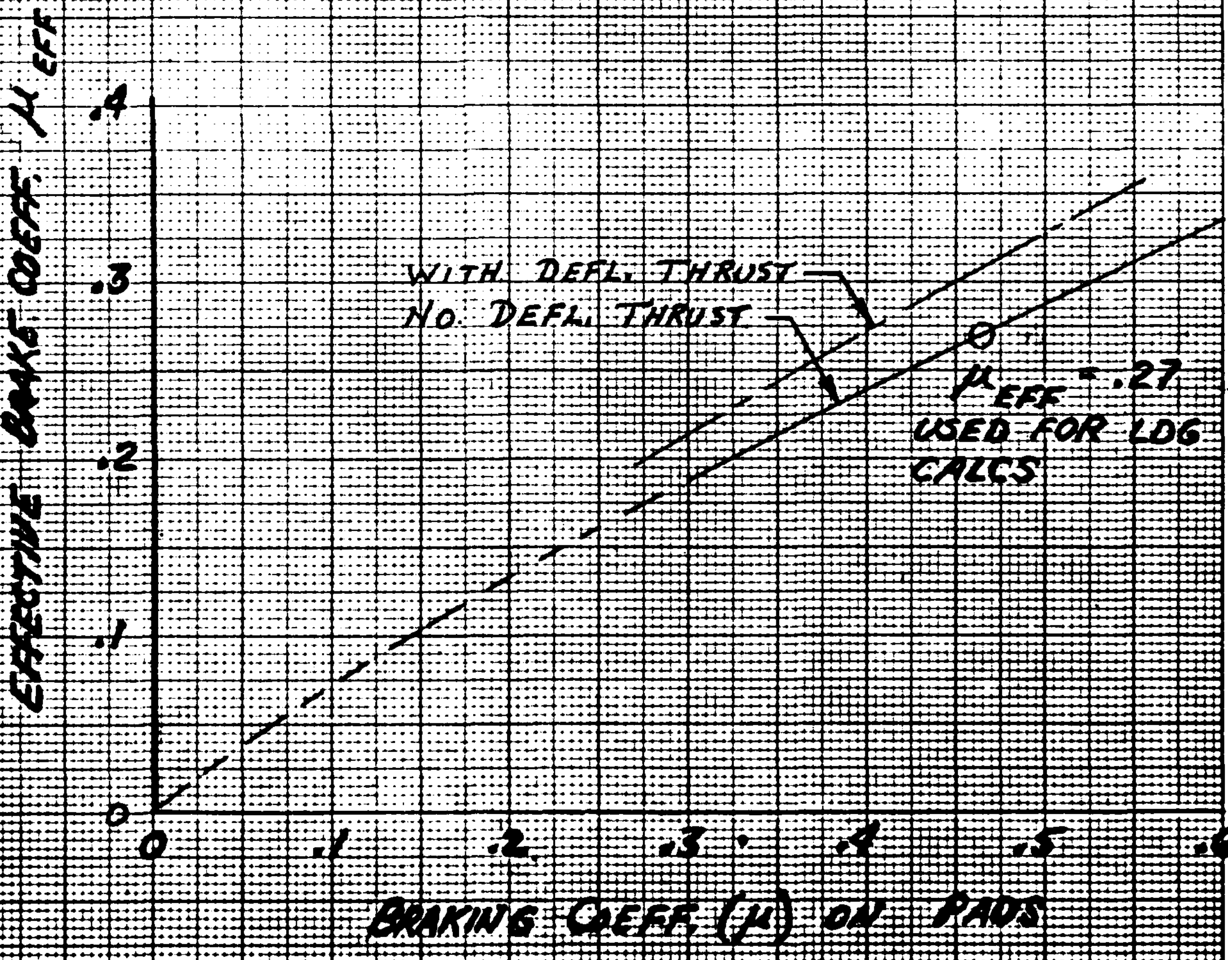
It will be necessary to deflect a portion of the thrust during taxi operations, as the thrust required for taxiing is less than the minimum thrust of 2500 pounds needed to produce the design airflow to the trunk. The moment due to the deflected thrust will increase the drag during taxi approximately 50 percent. This is due to the increased lift (L_t) required on the aft portion of the trunk to balance the thrust moment. The increased drag of the trunk, during taxi, is desirable because it will improve the ground handling characteristics (yaw and drift).

FIGURE 17A

EFFECTIVE BRAKING COEFFICIENT

$$\mu_{EFF} = \frac{F}{W}$$

W = 21569 LBS (MAX LDG WT)



END

SANDAIRE

APPENDIX B

AERODYNAMICS

This Appendix consists of the following sections:

(1) Minimum Low Speed Drag
(2) Transonic Drag Rise and Drag-Due-To-Lift
(3) Stores Drag
(4) Total Drag
(5) Lift and Drag with Flap Deflection
(6) Takeoff Performance
(7) Landing Performance
(8) Tail Sizing

Most of the following data are for input to the computer programs that compute all of the performance except for Takeoff and Landing which is shown in sections (6) and (7). Computed data are in Appendix F.

Drag estimates for the early preliminary work, summarized in Appendix E, were done with approximations that were updated as the design development progressed. Block coefficients were used to estimate wetted area and some anticipation of the final fuselage size and wing geometry was incorporated. Final drag data check reasonably well with these earlier approximations when configuration development changes are considered. Therefore conclusions based on the preliminary work are considered reliable and no recycling of the earlier configurations to incorporate final data is needed.

Reference is made to the text of the report for dimensions, particularly the three view drawing on page 3 and the tabulation on page 8 . In addition, other data are used as follows:

$b_{exposed\ wing}$	36.5 ft
$S_{exposed\ wing}$	234.4 sq ft
geometric washout	3 deg
C_{l_i} (camber)	0.20
L.E. radius/chord	.02 for t/c = .12
$\Lambda_{c/2}$ wing	20.6 deg
ℓ_f, length fuselage	42 ft
$(S_{WET})_F$, wetted area fuselage from section cuts plotted page B-3	834.7 sq ft

Retracted trunk area, effectively the cushion area — 150 sq ft

$(S_{\pi_F})_{Max}$, Max. cross section area fuselage from section cuts plotted page B-4 — 34.2 sq ft

$(\ell/d)_{FUSE}$ fineness ratio fuselage $= 42/(4 \times 34.2/\pi)^{0.5}$ = — 6.365

$\left[(S_{\pi F}) + (S_{\pi\, wing})\right]_{Max} - (S_{\pi\, \text{inlet ducts capture}})$ — 40.0 sq ft (page B-4)

Total fineness ratio $=42/(4 \times 40/\pi)^{0.5}$ = — 5.885

$S_{\pi c})_{Max}$, Max cross section area canopy from section cuts plotted page B-4 — 4.3 sq ft

Reynolds No. (Representative), — RN $= 2 \times 10^6$/ft

Method - reference is made to the data of General Dynamics, Convair Division, TN-70-AM-01 dated 23 March 1970 by R. E. Craig, which has been distributed by Convair, on request, to the Navy, Airforce and others. Craig's first reference is to Linden and O'Brinski's paper "Some Procedures for use in Performance Prediction of Proposed Aircraft Designs" presented to the Society of Automotive Engineers Oct. 1965, Pub., 650800. He lists many other references.

(1) Minimum Low Speed Drag

From the plotted data in the above referenced method, equations were developed to fit the plots for computer input. The following shows this development.

Δf (equivalent drag area - sq ft) for the wing

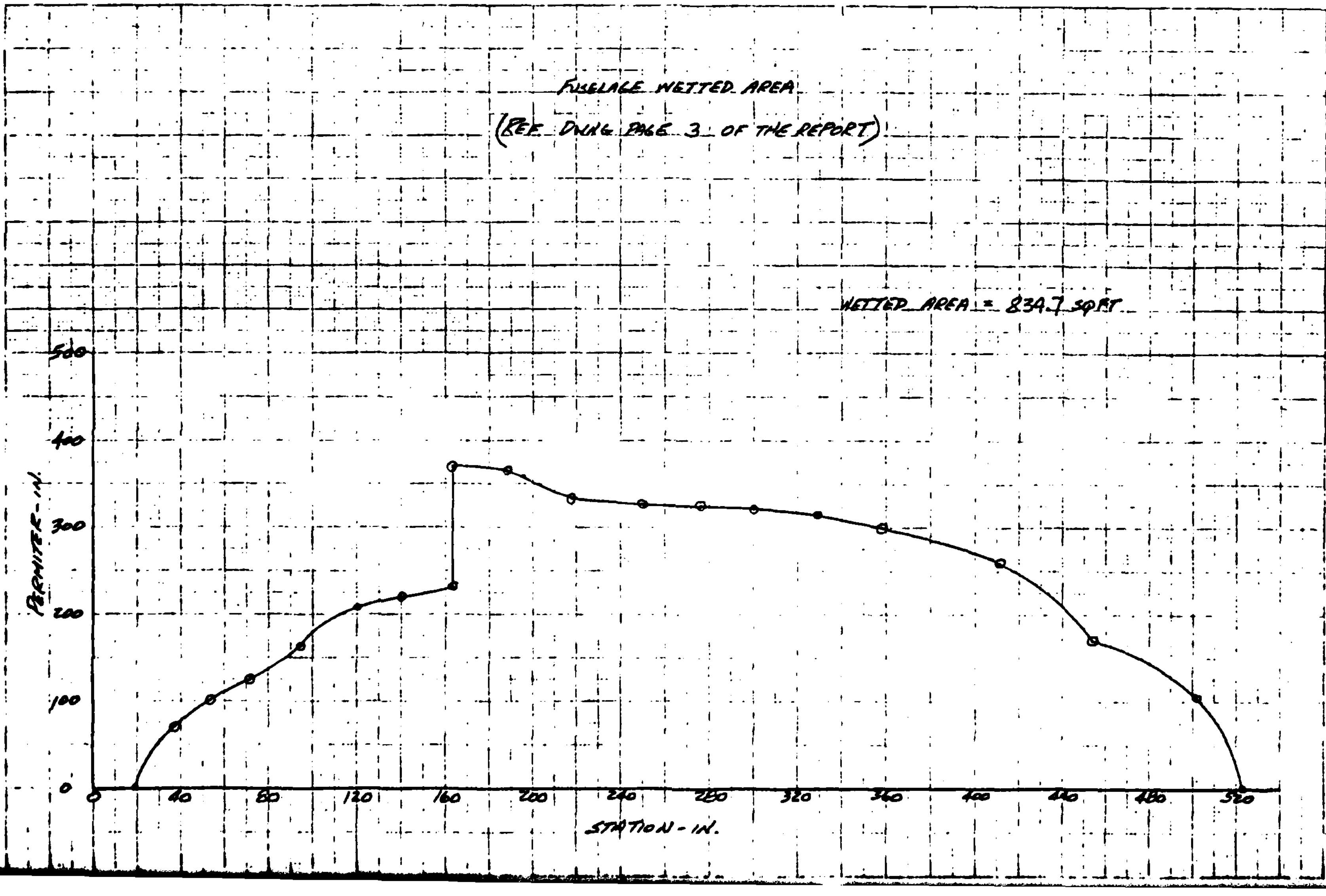
$$= \left[C_{D_S} \left(\frac{9 \times 10^6}{RN}\right)^{0.11} + \Delta C_{D_S}\right] \times S_{wing}$$

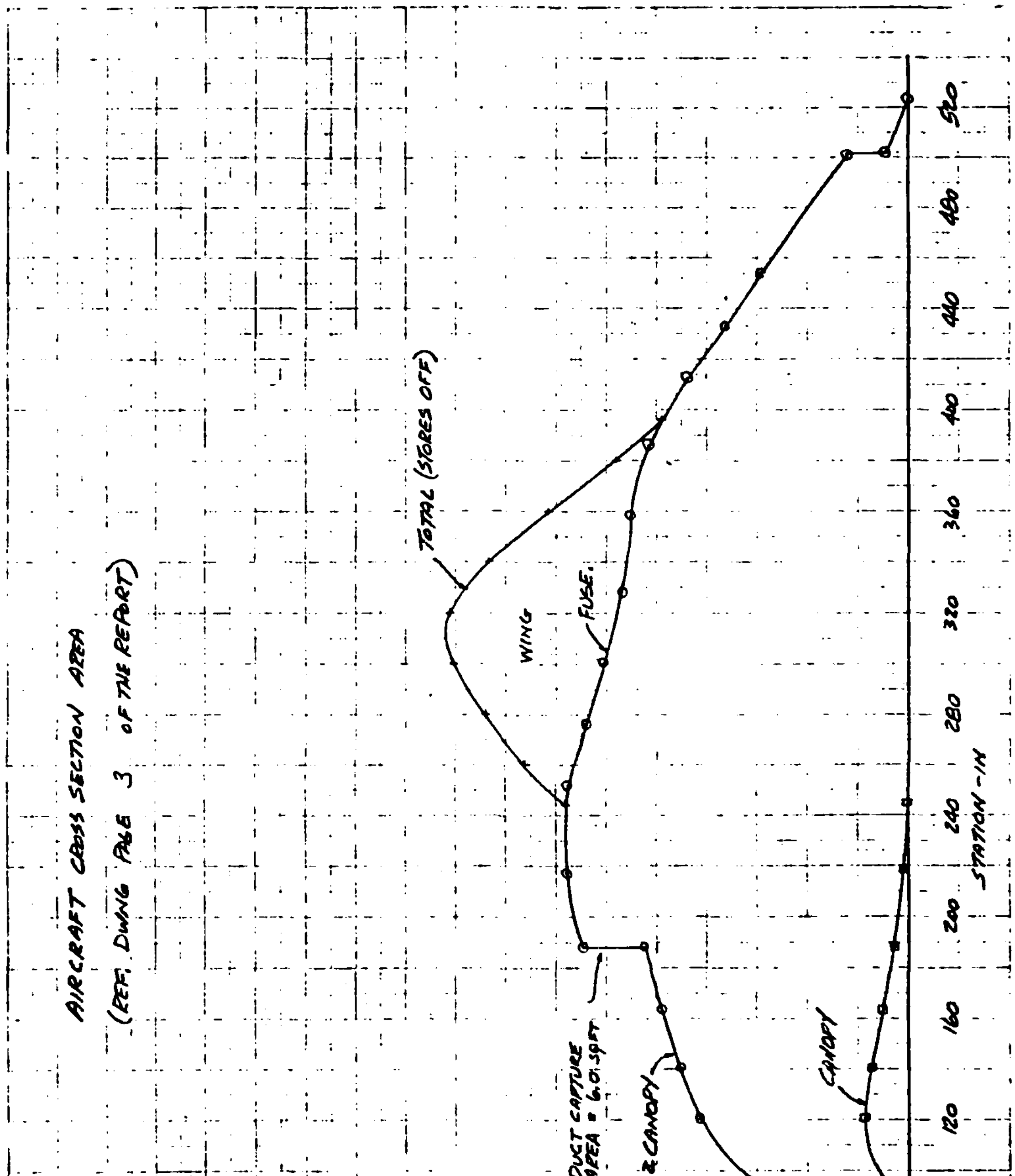
C_{D_S} = Drag coeff. $= .0052 + .018\ (^t/c)_{Max}$

$= .0052 + .018 \times 0.12 = .00736$ where $^t/c$ at root is used as effective value

RN $= 2 \times 10^6 \times (89.8/12) = 15 \times 10^6$ where $\bar{c}$ is used as effective length

ΔC_{D_S} = drag addition due to leading edge irregularity caused by leading edge lift devices. No such devices are used but the addition is retained to account for leading edge roughness.


FUSELAGE WETTED AREA
(REF DWNG PAGE 3 OF THE REPORT)
WETTED AREA = 834.7 SQ FT
PERIMITER-IN.
0
100
200
300
400
500
0
40
80
120
160
200
240
280
320
360
400
440
480
520
STATION-IN.



AIRCRAFT CROSS SECTION AREA
(REF. DWNG PAGE 3 OF THE REPORT)
TOTAL (STORES OFF)
WING
FUSE.
DUCT CAPTURE AREA = 6.0 SQFT
FUSE. & CANOPY
CANOPY
S - SQFT
STATION - IN
0
10
20
30
40
50
0
40
80
120
160
200
240
280
320
360
400
440
480
520

$$\Delta C_{D_s} = .0007 \times \left(\frac{b_{exp}}{b}\right)^{0.85} = .000634$$

$$\Delta f_{wing} = \left[.00736\ (9/15)^{0.11} + .000634\right] \times 280^{*} = 2.126$$

$$\Delta f_{vert} = 1.35\ C_{D_s} \left(\frac{9 \times 10^6}{R_N}\right)^{0.11} S_{V\ exposed}$$

$$C_{D_s} = .0040 + .018\ (t/c)_{Root} = .00616$$

$$R_N = 2 \times 10^6 \times (70/12) = 11.7 \times 10^6$$

$$S_{V_{Exp.}} = S_V = 47.5 \text{ sq ft}$$

$$\Delta f_{vert} = 1.35 \times .00616 \times (9/11.7)^{0.11} \times 47.5 = 0.384 \text{ sq ft}$$

Δf_{Horiz} = Same basis

$$C_{D_S} = .00616$$

$$R_N = 2 \times 10^6 \times (54.5/12) = 9.1 \times 10^6$$

$$S_{H_{Exp}} = S_H$$

$$\Delta f_{Horiz} = 1.35 \times .00616\ (9/9.1)^{0.11} \times 67.1^{**} = 0.557 \text{ sq ft}$$

$$\Delta f_{Fuse} = C_{f_{FP}} \times \left(\frac{C_f}{C_{f_{FP}}}\right) S_{WET}$$

$\left(\frac{C_f}{C_{f_{FP}}}\right)$ = Overspeed correction factor to account for shape, roughness, leakage etc.

$$= 1 + \frac{60}{(\ell/d)^3_{Fuse}} + .0025\ (\ell/d)_{Fuse}$$

$$= 1 + 60/(6.365)^3 + .0025 \times 6.365$$

$$= 1.2486$$

$C_{f_{FP}}$ = Turbulent flat plate skin friction coefficient including compressibility effects at the start of zero lift drag rise Mach No., M_R = 0.7525 from section (2).

$$C_{f_{FP}} = \left[\frac{1.697}{\ln\ (2 \times 10^6 \times k \times \ell_f)}\right]^{2.58} \Big/ (1 + 0.125\ M_R^2)^{0.653}$$

k = 0.25 to account for the typical termination of $C_{f_{FP}}$ decrease with Reynolds No. due to roughness.

$$C_{f_{FP}} = \left[\frac{1.697}{\ln(2 \times 10^6 \times .25 \times 42)}\right]^{2.58} \Big/ (1 + 0.125 \times 0.7525^2)^{0.653}$$

$$= .00256$$

---

* By using this area, instead of S_{Exp}, wing-fuse. interference is accounted for.
** By using this area, instead of $S_{H_{Exp}}$, tail-fuse interference is accounted for.

S_{WET} (effective to account for the drag of the retracted trunk)

$= S_{WET} +$ (cushion area) $\times 0.30$ where it is assumed the retracted trunk increases the skin friction drag 30%

$= 834.7 + 0.30 \times 150 = 879.7$ sq ft

$\Delta f_{FUSE} = .00256 \times 1.2486 \times 879.7 = 2.812$ sq ft

$\Delta f_{Canopy.} = .05 \times (S_{\pi c})_{Max}$, the .05 being an assumed drag coefficient.

$= .05 \times 4.3 = 0.215$ sq ft

$\Delta f_{camber\ \&\ twist} = 0.7\, C_{L_k}^2 \times S_{Exp.\ wing}$

C_{L_k} = lift coefficient for minimum drag

$= 0.15 \left(C_{l_i} \times \frac{S_{Exp}}{S}\right) + C_{L_{k\,twist}}$

$C_{L_{k\,twist}} = .002312/\text{deg}$ at $RN = 15 \times 10^6$

$= .002312 \times 3 = .006936$

$C_{L_k} = 0.15(0.20 \times 234.5/280) + .006936$

$= .03206$

$\Delta f_{camber\ \&\ twist} = 0.7 \times .03206^2 \times 234.5 = 0.169$

$\Delta f_{4\ pylons\ \&\ misc.} = 0.90$

$\Delta f_{tot} = 2.126 + 0.384 + 0.557 + 2.812$
$+ 0.215 + 0.169 + 0.90 = 7.163$

$C_{D_{min}}$, Low Speed $= 7.163/280 = .0256$

Computer calculation of the above minimum drag, with the same inputs, gave essentially the same value as shown by the following printout copy.

2.126034051	FW
.3833408937	FV
.5577646139	FH
2.809684437	FF
0.	FN
0.215	FC
.1687678707	FT
0.9	FM
7.160591866	F
.0255735424	CDM

(2) Transonic Drag Rise and Drag-Due-To-Lift

As for minimum drag, in section (1) above, equations were developed from the referenced method plotted data to fit the plots for computer input. Mach. No. for the start of drag rise is:

$$M_R = M_{DD} - 0.12$$

Where M_{DD} = drag divergence Mach No. and the variation of drag coefficient with Mach No. is 0.10.

$$M_{DD_o} = 0.92 + .07 - \left\{ \left[1/(AR \times t/c \times \cos\Lambda_{c/4})_{wing} \right] - 4.5 \right\}^2 / 75$$

Where M_{DD_o} is the drag divergence Mach No. at zero lift and the .07 factor is added based on qualitative information about the NASA advanced airfoil sections in delaying the drag rise.

$$M_{DD_o} = 0.92 + .07 - \left\{ \left[1/(6 \times 0.12 \times \cos 25^\circ) \right] - 4.5 \right\}^2 / 75$$
$$= 0.8725$$

$M_R = 0.8725 - 0.12 = 0.7525$ at zero lift

The drag divergence Mach No. decrease with lift is

$$\Delta M_{DD} = - C_L \left[.05 + k_\Lambda \left({}^t/c - .04 \right) \right]$$

k_Λ is empirical and

$$= \left\{ \left[(\Lambda_{c/4})_{wing} - 5 \right]^{2.62} / 12363 \right\} + 0.5$$

$$= \left[(25 - 5)^{2.62} / 12363 \right] + 0.5 = .707$$

$$\Delta M_{DD} = -C_L\left[.05 + .707\ (.12 - .04)\right]$$
$$= -0.1066C_L$$

The drag rise with Mach No., ΔC_{DM}, is given by two equations.

For $(M-M_{DD})$ less than .015

$$\Delta C_{DM} = .0080\left\{1 - \sqrt{1 - \left[\frac{(M-M_{DD})-(-0.12)}{0.14}\right]^2}\right\} \times \frac{S_{ExpWHV}}{S_{Wing}}$$

M = free stream Mach No.

$M_{DD} = M_{DD_o} - \Delta M_{DD}$

S_{ExpWHV} = exposed area wing and tail
= 234.4 + 67.1 + 47.5 = 349 sq ft

S_{Wing} = 280 sq ft

For $(M - M_{DD})$ greater than .015

$$\Delta C_{DM} = \left[2.7817\ (M - M_{DD}) + 0.3163\right]^5 \times \frac{S_{ExpWHV}}{S_{Wing}}$$

For computer input, ΔC_{DM} was developed through the transonic range to M = 1.2, even though this aircraft will not fly at these speeds. This development is omitted here.

Drag-due-to-lift,

$$\Delta C_{DL} = (C_L - C_{Lk})^2 / (\pi \times AR \times e)$$

Where C_{Lk} is the lift coefficient for minimum drag and is computed in section (1) above as C_{Lk} = .03206 which is constant to the start of drag rise. For higher Mach No.,

$$(C_{Lk})_{M > M_R} = .03206 - \left(\frac{.03206}{1-M_R^*}\right)(M - M_R)$$
$$= .03206 - \frac{.03206}{1-.7525}(M - M_R)$$
$$= .03206 - 0.1295\ (M - M_R)$$

---

* M_R for zero lift

$$e_{Max} = \text{function of } (AR, \lambda, \Lambda c/4, \ t/c\ )$$

$$= 0.842 \text{ (from page B-10)}$$

e_{Max} decreases with M greater than M_R

$$e_M = (e_{Max}) - (e_{Max} - e_{M=1.2}) \times \left[\frac{M-M_R}{1.2-M_R{}^*}\right]^2$$

$$e_{M=1.2} = 1\Big/\left[\pi \times AR\ (\Delta C_{D_L}/\Delta C_L^2\ )_{M=1.2}\right]$$

$$(\Delta C_{D_L}/\Delta C_L^2\ )_{M=1.2} = \left[\frac{1}{(C_{L_\alpha})_{Rad}}\right]_{M=1.2}$$

$$= \frac{1}{57.3 \cos(\Lambda_{c/2}/2)\ \left[0.11-.0001(10-AR)^3\right]}$$

$$= \frac{1}{57.3 \cos 10.3^\circ \left[0.11 - .0001\ (10-6)^3\right]} = 0.171$$

$$e_{M=1.2} = 1\big/(\pi \times 6 \times 0.171) = 0.310$$

e_M decreases with lift

$$(e)_{C_L > 0.4} = e_M \left[1 - (C_L - 0.4)^2\big/1.08\right] \quad \text{for } AR = 6$$

(3) Stores Drag

Stores drag was obtained from several sources as shown on page B-11. A portion of the plot was estimated.

(4) Total Drag

$$C_{D_{Tot}} = (C_{DMin})_{Low\ Spd} + (\Delta C_{DM})_{C_L} + \Delta C_{D_L} + \Delta C_{DStores}$$

Where basic factors have been developed above and $(\Delta C_{DM})_{C_L}$ is the ΔC_{DM} in section (2) with correction for the variation of M_{DD} with C_L (given above as $\Delta M_{DD} = -0.1066\ C_L$)

As an example, the drag is hand calculated below for the CAS mission loading, with and without droppable stores, and for the clean configuration, for a range of lift coefficient and Mach No.

---

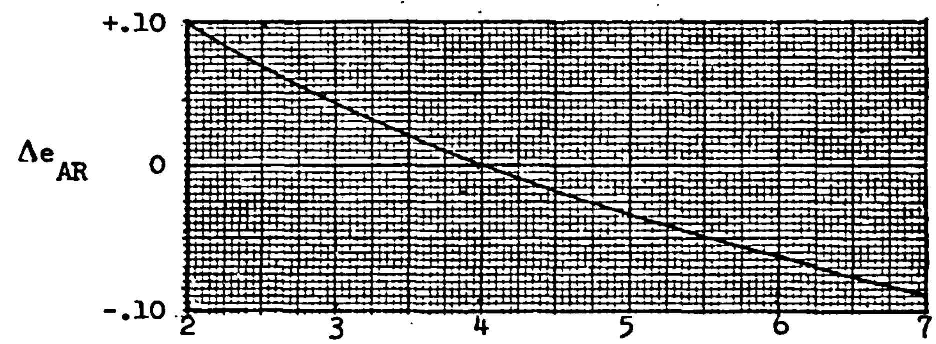
* M_R for zero lift.

EFFECTS OF VARIATIONS IN WING ASPECT RATIO, TAPER RATIO, SWEEPBACK AND THICKNESS ON AIRPLANE EFFICIENCY FACTOR, e
(SUBSONIC SPEEDS)

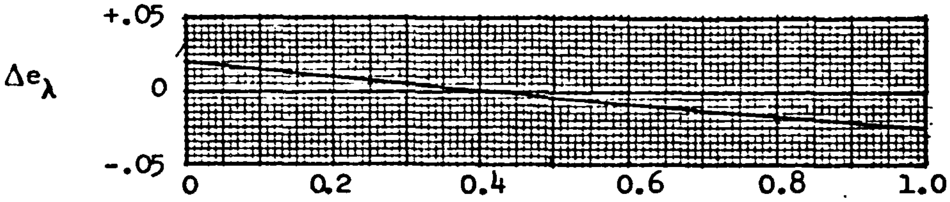
$$e_{MAX} = e_{BASIC} + \Delta e_{AR} + \Delta e_{\lambda} + \Delta e_{\Lambda} + \Delta e_{t/c}$$

$$e_{BASIC} = 0.8, \quad AR = 4.0, \quad \lambda = 0.4, \quad \Lambda_{c/4} = 40^{\circ}, \quad t/c = .068$$

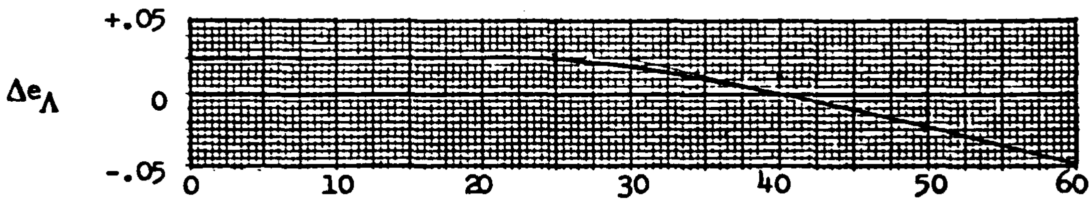
Reproduced from Craig's report referenced above



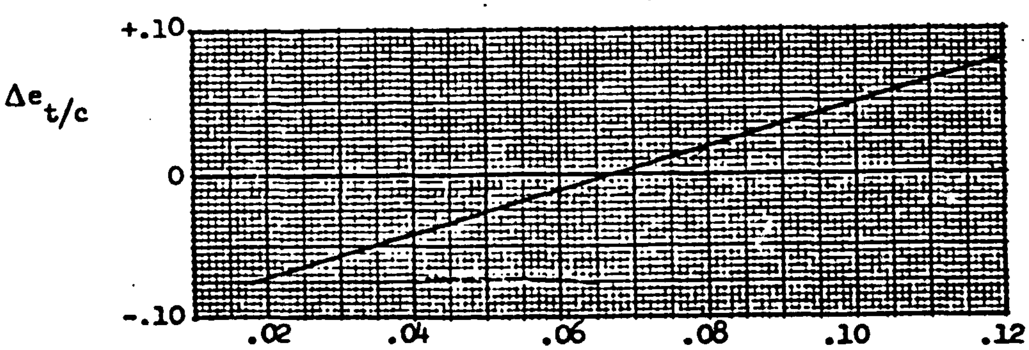
ASPECT RATIO (AR)



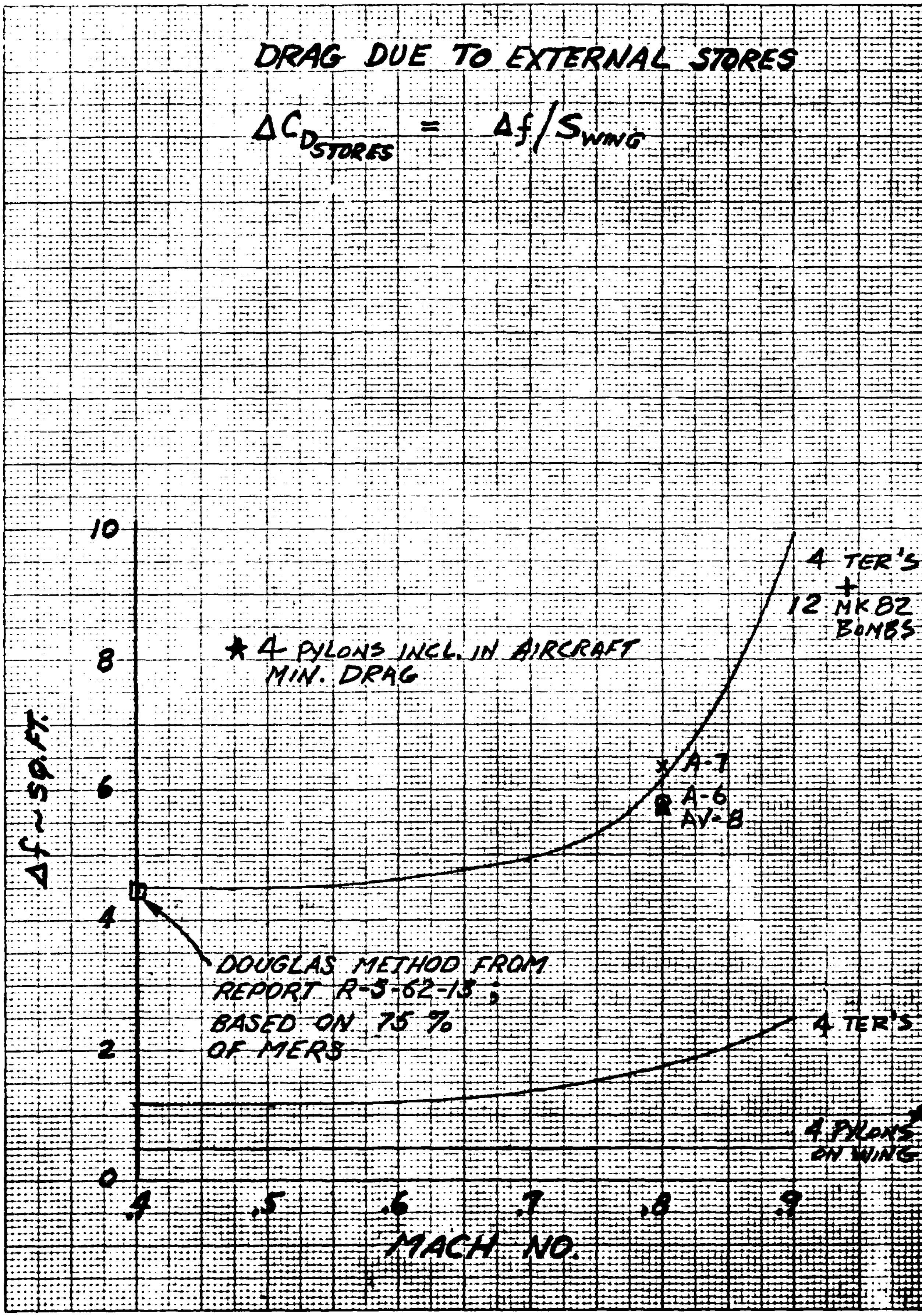
TAPER RATIO (λ)



SWEEPBACK ($\Lambda_{c/4}$), DEG.



WING THICKNESS (t/c)


DRAG DUE TO EXTERNAL STORES
ΔC_D STORES = Δf/S WING
★ 4 PYLONS INCL. IN AIRCRAFT MIN. DRAG
4 TER'S + 12 MK 82 BOMBS
A-7
A-6
AV-8
DOUGLAS METHOD FROM REPORT R-3-62-13 ; BASED ON 75 % OF MERS
4 TER'S
4 PYLONS ON WING
Δf ~ SQ.FT.
MACH NO.
10
8
6
4
2
0
.4
.5
.6
.7
.8
.9

SANDAIRE

For M Less Than 0.50 (Start of Stores Drag Increase with M), C_{L_k} = .032, $C_{D_{Min}}$ = .0256, $\Delta C_{D_{stores}}$ = .0161, $\Delta C_{D_{stores}}$ (Mk 82 Dropped) = .0043

C_L	e	$\frac{(C_L - C_{L_k})^2}{6 \pi e}$	C_D Stores on	C_D Droppable Stores off	C_D Clean	L/D Clean
.032	.842	0	.0417	.0299	.0256	1.25
.10	"	.0003	.0420	.0302	.0259	3.86
.20	"	.0018	.0435	.0317	.0274	7.30
.40	"	.0085	.0502	.0384	.0341	11.73
.60	.811	.0211	.0628	.0510	.0467	12.85
.80	.717	.0436	.0853	.0735	.0692	11.56

Clean L/D vs C_L is plotted on page B-13

For Higher Mach. No.

		C_L =		0		.20		.40		.60	
$M-M_R$	$M-M_{DD}$	ΔC_{D_M}	C_{L_k}	M	e	M	e	M	e	M	e
0	-.12	0	.032	.7526	.842	.731	.842	.710	.842	.689	.811
.04	-.08	.0004	.027	.793	.838	.771	.838	.750	.838	.729	.807
.08	-.04	.0018	.022	.833	.825	.811	.825	.790	.825	.769	.794
.12	0	.0048	.017	.8726	.804	.851	.804	.830	.804	.809	.774
.135	.015	.0073	.015	.888	.794	.866	.794	.845	.794	.824	.765
.16	.04	.0178	.011	.913	.774	.891	.774	.870	.774	.849	.745
.18	.06	.0328	.009	.933	.756	.911	.756	.890	.756	.869	.728

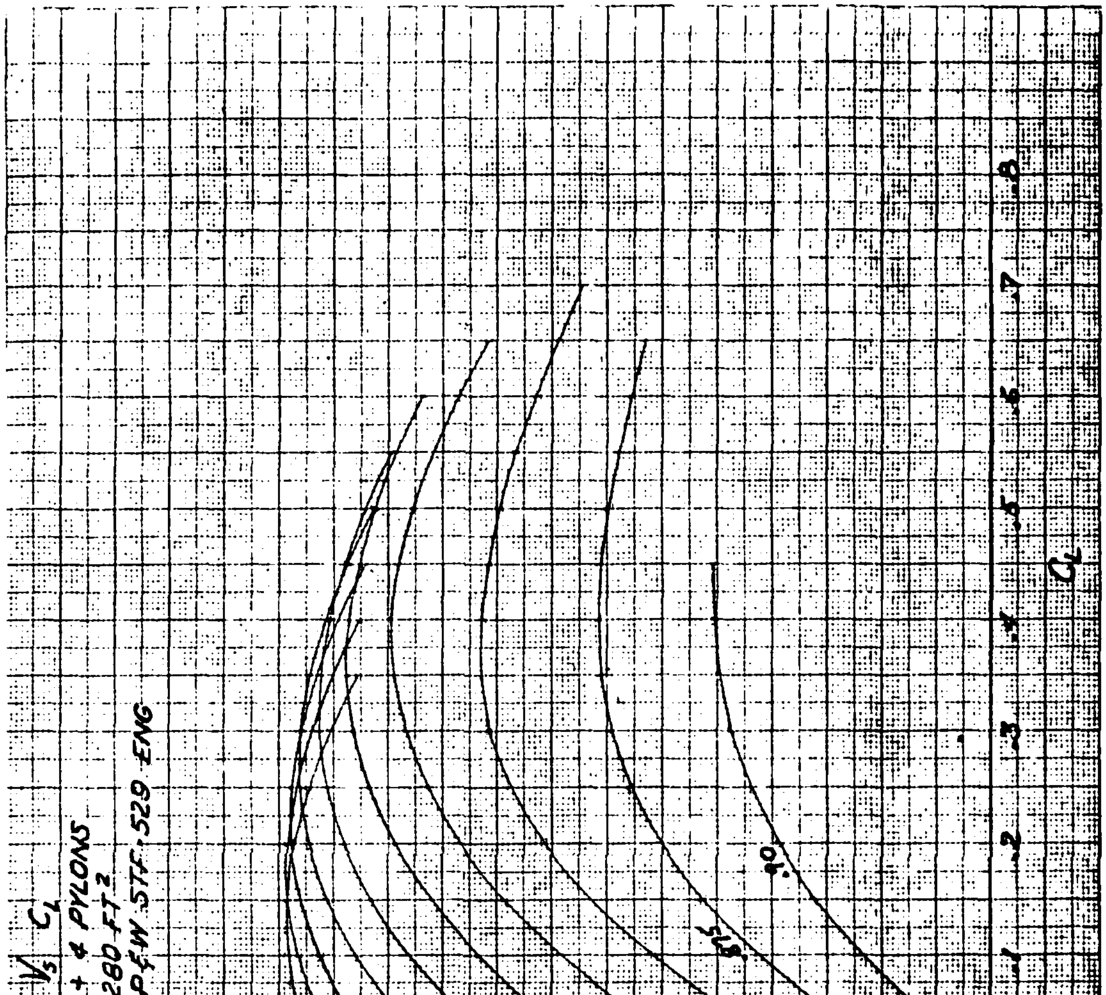
For Example at

C_L = .40, M = .830, Stores on

$$C_D = .0256 + .0048 + \frac{(.40-.017)^2}{\pi \times 6 \times .804} + \frac{6.96}{280}$$

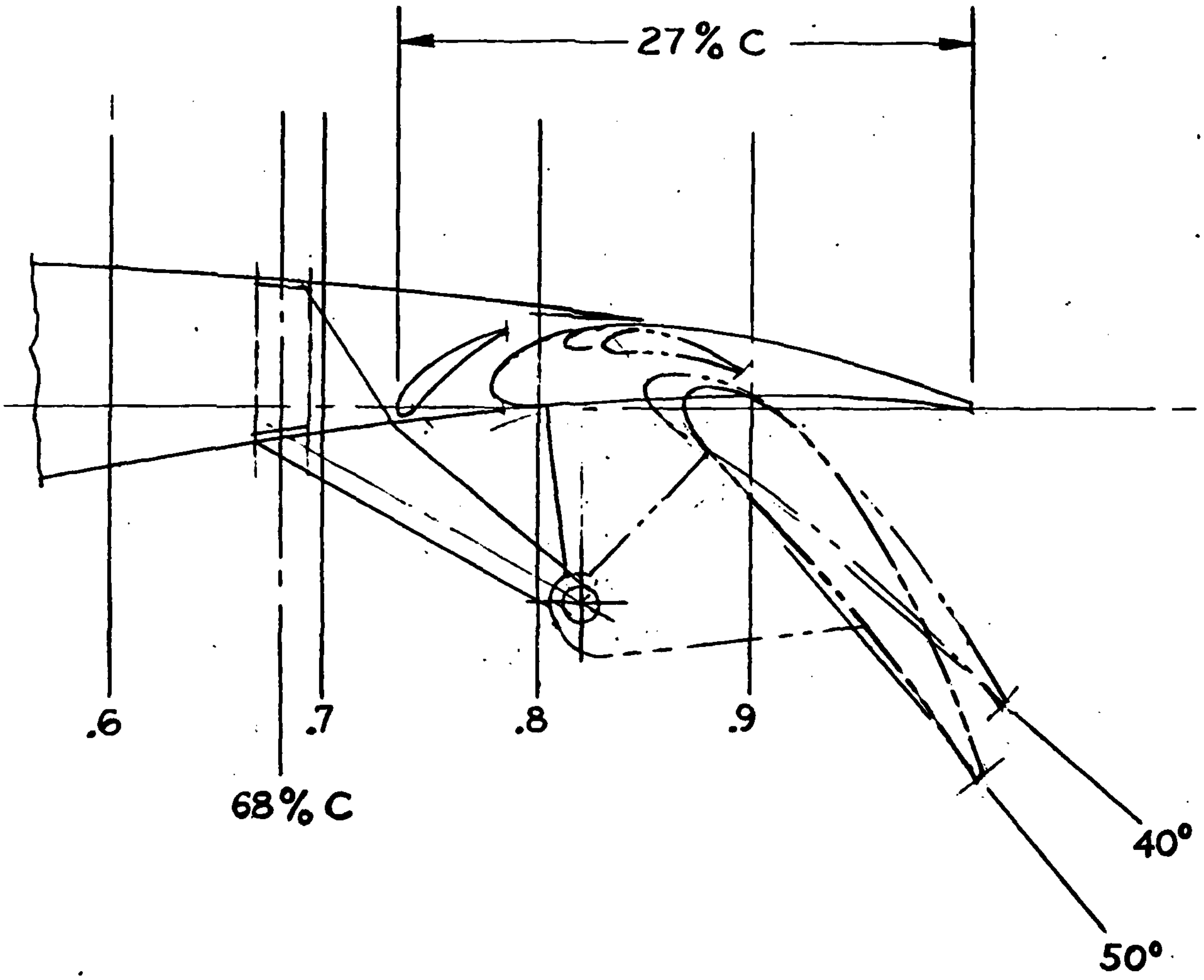
$$= .0649$$

These data would be plotted if performance calculations were done by hand. The computer program avoids this work.


L/D Vs C_L
CLEAN + 4 PYLONS
S_W = 280 FT²
100% PFW STF 529 ENG
M = 0 - .65
.70
.725
.75
.775
.80
.825
.85
.875
.90
L/D
C_L

(5) Lift and Drag With Flap Deflection

Due to the reduction in the required takeoff run from 8000 ft to 3000 ft, page 5 Item (3a), of the report, a large flap setting is used to favor this shorter run. Fixed vane double slotted flaps are selected with external hinges. Flaps extend from the fuselage to 70% semispan. The rear wing spar is at 68% chord which allows use of a 27% chord flap as shown diagrammatically by the following sketch.



SANDAIRE

Data for flap application to advanced airfoil sections are not available. However, flap characteristics are estimated from available data for other flapped airfoil sections.

Ground effect is estimated to give an effective aspect ratio of 8.6 based on the wing height from the ground in the takeoff and landing run (7.0 ft from Dwng SAE-79-007 in the report). This increases the lift curve slope from C_{L_α} =.076/Deg, see following section (8), to

$$C_{L_\alpha} \text{ (in ground effect)}$$
$$= \frac{2\pi}{57.3} \times \left(\frac{8.6 \cos 25}{8.6 + 2 \cos 25}\right) = .082/\text{Deg}$$

Since the angle of attack equivalence to flap deflection at constant lift coefficient, $(\alpha_{\delta_f})_{c_\ell}$, is independent of aspect ratio, section data can be used directly after correction for flap span. From available NASA section data that are fairly representative of this type flap, but 30% chord,

Deg	C_ℓ Flaps 0°	C_{d_o} Flaps 0°	C_ℓ Flaps 40°	C_{d_o} Flaps 40°	C_ℓ Flaps 50°	C_{d_o} Flaps 50°
-4	-.30	.0061	2.00	.072	2.30	.115
0	.14	.0060	2.41	.090	2.64	.132
4	.55	.0063	2.73	.120	2.93	.163
8	.98	.0076	2.98	.161	3.12	.210
9	1.08	.0084	3.06	.175	3.16	.230
10	1.15	.0092	3.08	(Max)	3.18	(Max)
12	1.38					
18	1.78	(Max)				

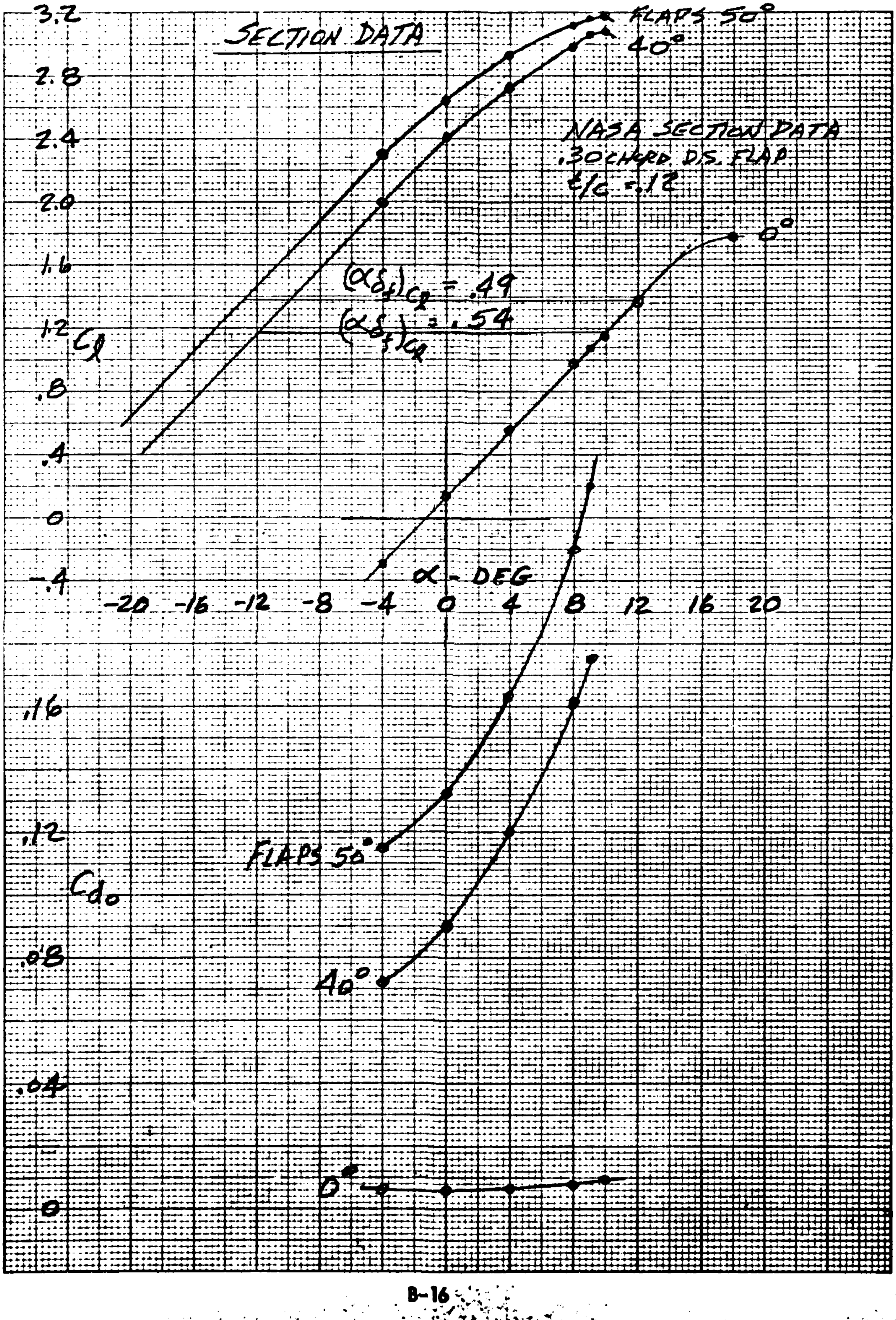
These section data are plotted on page B-16 from which $(\alpha_{\delta_f})_{c_\ell}$ is read. For 70% span, 27% chord, the correction factor for lift is

$\left(.70 \times \frac{.27}{.30} \times .85\right)$ where the .85 factor accounts for end loss.

For the ground run, flaps 40°, in ground effect, with fuselage level (cushion level), wing incidence = 3°

$$C_{L_{Run}} = .082\left(.70 \times \frac{.27}{.30} \times .85 \times .54 \times 40 + 3\right)$$
$$= 1.19$$

Where it is assumed that any negative $\alpha_{C_L = 0}$ (flaps 0°) due to camber is offset by the lift reduction due to the fuselage.



SECTION DATA
FLAPS 50°
40°
NASA SECTION DATA
.30 CHORD D.S. FLAP
t/c = .12
0°
$(\alpha_{\delta_f})_{C_\ell} = .49$
$(\alpha_{\delta_f})_{C_\ell} = .54$
C_ℓ
3.2
2.8
2.4
2.0
1.6
1.2
.8
.4
0
-.4
α - DEG
-20 -16 -12 -8 -4 0 4 8 12 16 20
.16
.12
.08
.04
0
C_{d_o}
FLAPS 50°
40°
0°

$C_{D_{Run}}$ is built up from,

<u>Flaps 0°</u>

C_L (out of ground effect),

fuselage level, $i_w = 3^\circ$

$= 3 \times .076 = 0.228$

From the minimum drag estimate, section (1) above, $C_D = .0256$,

and the added profile drag (at $\alpha_w = 3^\circ$) $= \dfrac{(0.228-.032)^2}{6\pi} \cdot \left(\dfrac{1}{.842} - 1\right)$

$= .0004.$

Available data indicate that the pressurized trunk will about double the aircraft drag and .0260 is added for fuselage level. External stores (CAS loading) add $\Delta C_D = .0161$, section (1).

From the above section data for $\alpha = 3^\circ$,

$\Delta C_{d_o} = C_{d_o}(\text{FL. } 40^\circ) - C_{d_o}(\text{FL. } 0^\circ) = (.111-.006) = .105$

With the correction for flap span and chord

$\Delta C_{D_o} = .105 \times .70 \times \dfrac{.27}{.30} = .066$, Flaps 40°

For drag, no end loss factor is applied because the addition of $C_L^2/(\pi \times AR)$ accounts for end loss.

$C_{D_{Run}}$, $\alpha = 3^\circ$ in ground effect (CAS stores on),

$$= .0260 + .0260 + .0161 + .066$$

$$+ 1.19^2/8.6\pi = 0.1865$$

Note that "e" effect is included in the .0260 and .066 factors.
The $C_{L_{TO}}$ is at 1.2 V_s and

$C_{L_{Max}}$, flaps 40°, is estimated from the above section data as

$$C_{L_{Max}} = (3.08 - 1.78) \times .70 \times \frac{.27}{.30} \times .85$$

$+ 1.30 = 2.00$ where the 1.30 is the estimated

$C_{L_{Max}}$ flaps 0°.

$$C_{L_{TO}} = 2.00/(1.2)^2 = 1.39$$

A major reason for the selected conditions above is that only (1.39-1.19)/.082 = 2.4 deg. rotation from the takeoff run is required to lift-off which will favor smooth operation with the SETOLS.

Similar calculations are made to obtain the following plotted data, pages 19, 20 and 21.

(6) Takeoff Performance

(a) Takeoff ground run is calculated from basic relations and the flap characteristics of section (5) above, for 89.8°F at sea level.

$$\frac{dv}{dt} = a, \quad \frac{ds}{dt} = v, \qquad s = \int_0^{V_{TO}} \frac{v}{a}\, dv$$

Warm-up, taxi, and takeoff fuel is specified as 5 min. at maximum thrust. It is assumed that 4.5 min. of this is used prior to the takeoff; therefore for the CAS loading, 89.8°F at S.L., 8190 lb/hr Fuel Flow (Appendix D)

$$W_{TO} = 24300 - \frac{4.5}{60} \times 8190 = 23686 \text{ lb}$$

$C_{L_{TO}}$ = 1.39, Flaps 40°, (Pg B-17)

C_s (speed sound, 89.8°F at S.L.)

$$= 1117 \times \left(\frac{549.8}{519}\right)^{0.5} = 1150 \text{ ft/sec}$$

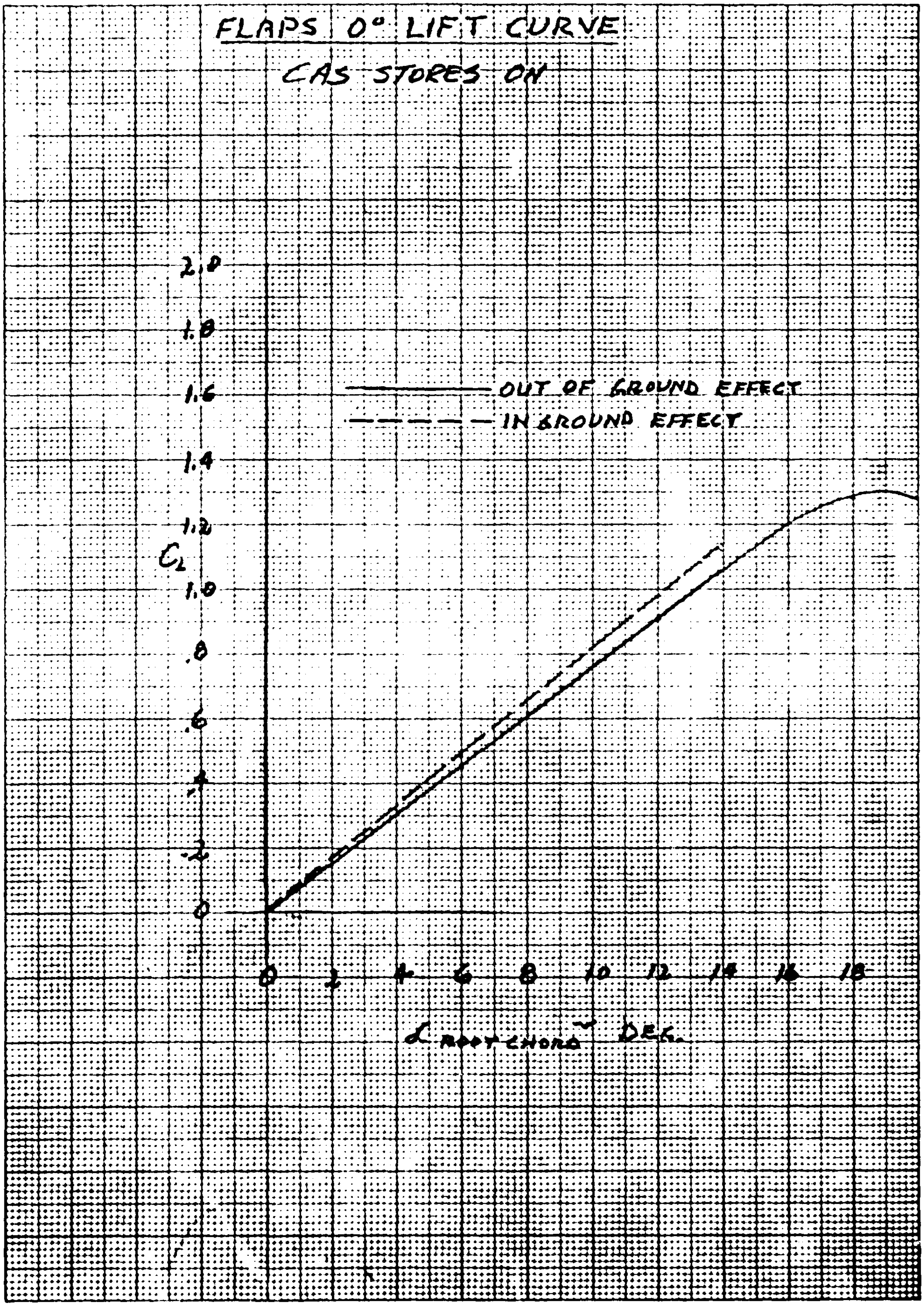
$$\rho = .002378 \times \left(\frac{519}{549.8}\right) = .002245$$

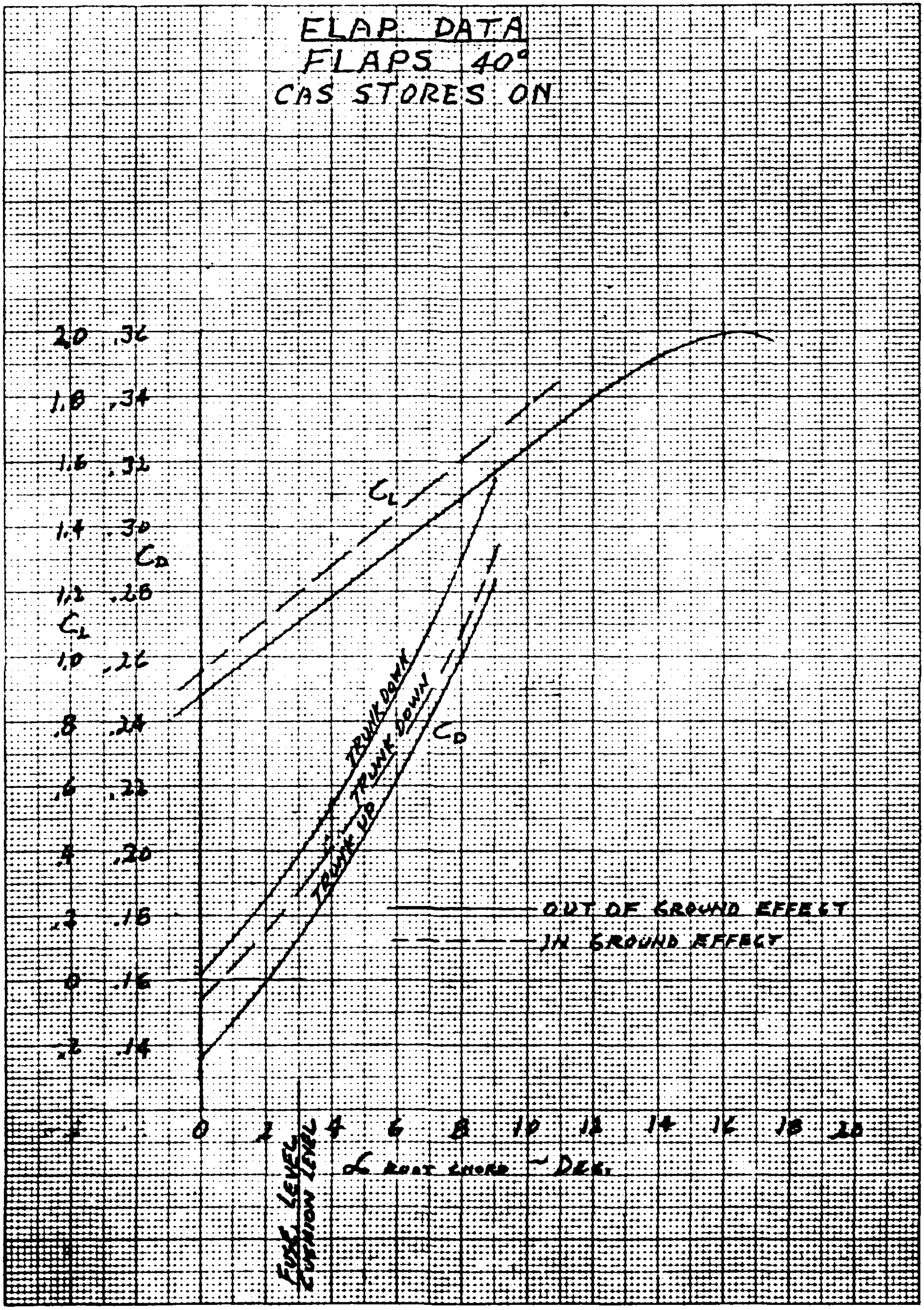
$$V_{TO} = \left(\frac{23686 \times 2}{.002245 \times 280 \times 1.39}\right)^{0.5} = 233 \text{ ft/sec}$$

$$= 138 \text{ kn}$$

Engine data are in Appendix D. P&W provides a 6% throttle advance for 90°F takeoff at sea level to minimize the adverse effect of a hot day. This overcomes the normal thrust deterioration at 90°F compared to standard temperature.

The required engine fan bleed to pressurize the trunk is, from Appendix A, 39 lb/sec. To calculate the corresponding engine thrust loss due to this bleed, P&W computer printout for the engine with afterburner installed (only data available) is used. For 90°F at S.L., M=0, and


FLAPS 0° LIFT CURVE
CAS STORES ON
OUT OF GROUND EFFECT
IN GROUND EFFECT
C_L
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0
0
2
4
6
8
10
12
14
16
18
α ROOT CHORD ~ DEG.



FLAP DATA
FLAPS 40°
CAS STORES ON
C_L
C_D
TRUNK DOWN
TRUNK DOWN
TRUNK UP
OUT OF GROUND EFFECT
IN GROUND EFFECT
FUSE. LEVEL
CUSHION LEVEL
α ROOT CHORD ~ DEG.

AD-A088 351 SANDAIRE SAN DIEGO CA F/G 1/2

CONCEPTUAL POINT DESIGN STUDY OF A NEW CTOL SETOLS CAS AIRCRAFT--ETC(U)

JUN 79 P D SORENSEN, R L BAYLESS, E F NOEL N62269-79-C-0438

UNCLASSIFIED NADC-78155-20 NL

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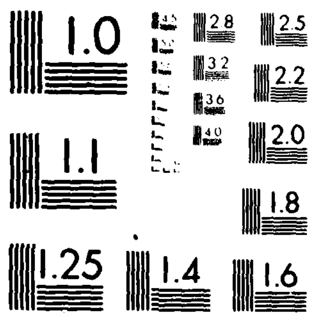
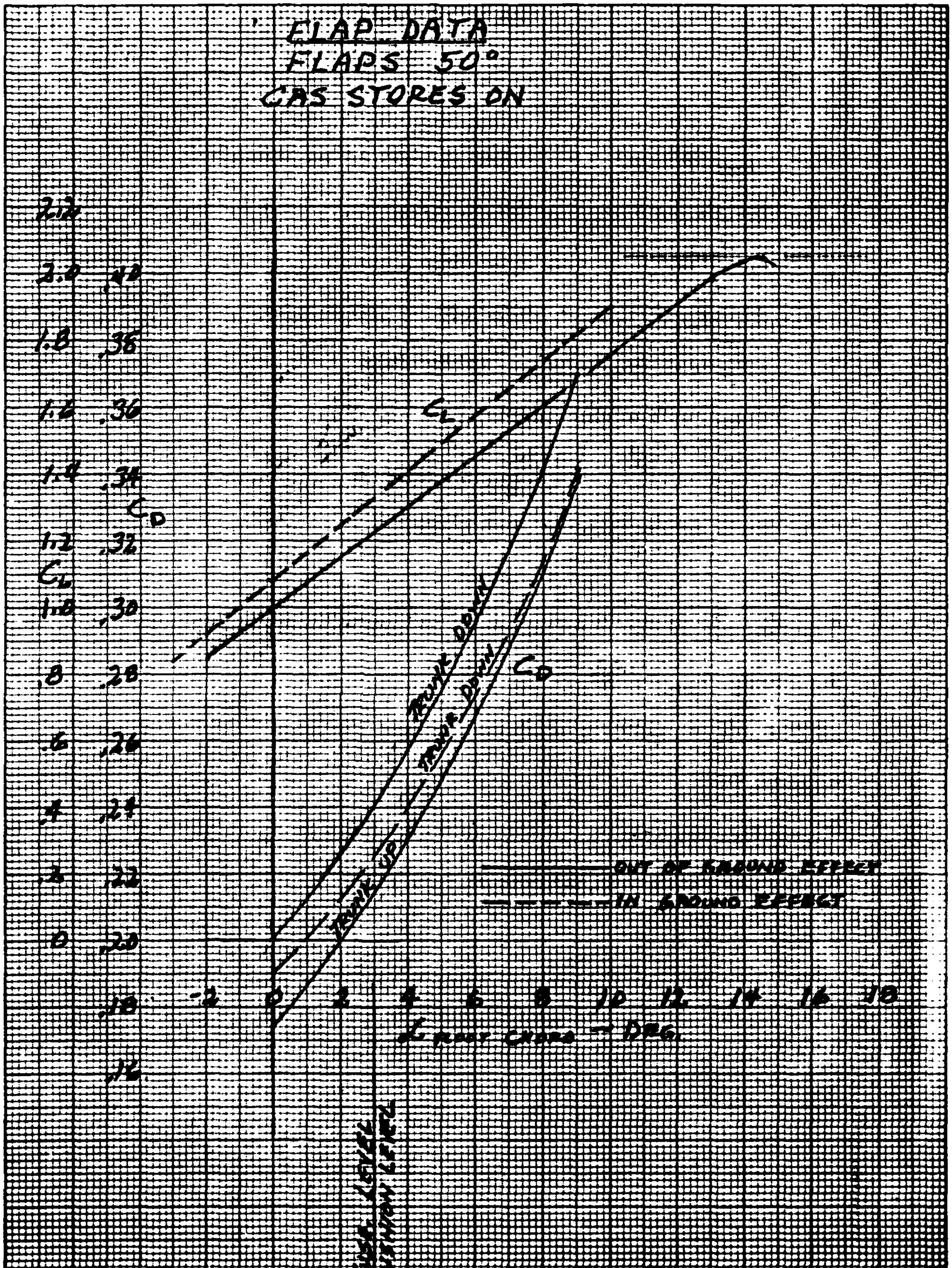


MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS 1963-A


FLAP DATA
FLAPS 50°
GAS STORES ON
C_L
C_D
TRUNK DOWN
TRUNK UP
OUT OF GROUND EFFECT
IN GROUND EFFECT
α ROOT CHORD ~ DEG.

T_{Max} (Mil Pwr) = 13046 lb (with AB but uninstalled)

Nozzle gas flow 221.56 lb/sec

Fuel flow 8182 lb/hr

Airflow = 221.56 - 8182/3600 = 219.3 lb/sec

Afterburner loss factor = 12990/13202

T_{Max} with subsonic nozzle

$$= 13046 \times \frac{13202}{12990} = 13259 \text{ lb}$$

Ratio airflow/T_{Max} = 219.3/13259

= .01654

Thrust loss due 39 lb/sec fan bleed = 39/.01654 = 2358 lb

The installed T_{Max}, 89.8°F at S. L., M = 0, is

13259 x .95 = 12596 lb which may be read also from Appendix D.

T_{TO} (M = 0) = 12596 - 2358 = 10238 lb

For Std Day and M = 0, see note on Page B-26.

SANDAIRE

For flaps 40°, takeoff run in ground effect, 89.8°F at sea level.

V Ft/Sec	Lift-Lb $C_{L_{Run}}$=1.19 Pg B-15	W_{TO}- Lift Lb	μ(W-L) μ=.065 (Pg 10 of the report)	Drag Lb $C_{D_{Run}}$= .1865 Pg B-17	Tot Resisting Force - Lb
0	0	23686			
50	935	22751	1479	147	1626
100	3740	19946	1296	586	1882
150	8415	15271	993	1319	2312
200	14960	8726	567	2345	2912
233	20304	3382	220	3182	3402

V Ft/Sec	M	T_{Inst} Lb App. D	T_{TO} Lb Bleed Loss Deducted	T_{Excess} Lb	V/a Sec	ΔS Ft
0					0	
50	.043	12260	9902	8276	4.44	111
100	.087	11960	9602	7720	9.53	349
150	.130	11730	9372	7060	15.63	629
200	.174	11560	9202	6290	23.39	976
233	.203	11480	9122	5720	29.96	880
				Total		2945

ground run, 89.8°F at sea level

M = V/1150 for 89.8°F @ S.L.

$$V/a = \frac{V \times W}{g \times T_{Excess}} = \frac{23686\ V}{32.2\ T_{Excess}}$$

with V Ft/Sec

$$\text{Lift} = C_{L_{Run}}\, qS\, V^2$$

$$= 1.19 \times (.002245/2) \times 280 \times V^2$$

$$\text{Drag} = (C_{D_{Run}} / C_{L_{Run}}) \times \text{Lift}$$

Takeoff run may be calculated also from the following equation

$$S_G\ (\text{Ft}) = \frac{13.05\ W_{TO}}{C_{L_{TO}} \times S \times \sigma \left[\frac{T_{0.7V_{TO}}}{W_{TO}} - \frac{C_{D_{Run}}}{2\ C_{L_{TO}}} - \left(1 - \frac{C_{L_{Run}}}{2\ C_{L_{TO}}}\right)\mu \right]}$$

Where all factors have been explained except

$T_{0.7V_{TO}} = T_{TO}$ at $0.7V_{TO}$

M at $0.7V_{TO}$ = 0.7 x 233/1150 = .142

T_{TO} = 11680 - 2358 = 9322 Lb

$$S_G = \frac{13.05 \times 23686}{1.39 \times 280 \times .944 \left[\frac{9322}{23686} - \frac{.1865}{2 \times 1.39} - \left(1 - \frac{1.19}{2 \times 1.39}\right) \times .065 \right]}$$

= 2908 ft This is within 1.3% of the above calculation of 2945 ft.

(b) The air distance over a 50 ft. obstacle is calculated from an empirical method.

For S.L., 89.8°F

$$K = \left[1.9 - \left(\frac{C_D}{C_L}\right)^2_{TO} \right]^{0.5} \times 11.28/V^2_{TO_{knots}}$$

$$K_1 = \frac{K \times \text{(obstacle height)}}{(T/W)_{TO} - \left(\frac{C_D}{C_L}\right)_{TO}}$$

(The distance over an obstacle divided by the Takeoff speed squared, $(S_{Air})/(V_{TO_{knots}})^2$, is plotted as a function of K_1 for various values of $(T/W)_{TO}$ on Pg B-25.

Substituting values out of ground effect, Pg B-20 , and assuming bleed off and trunk retraction for transition and climb over 50 ft obstacle.

$C_{L_{TO}}$ = 1.39, $C_{D_{TO}}$ = .237

$$K = \left[1.9 - \left(\frac{.237}{1.39}\right)^2\right]^{0.5} \times 11.28/138^2 = .000810$$

$$K_1 = \frac{.000810 \times 50}{(11480/23686) - \left(\frac{.237}{1.39}\right)} = 0.129$$

$(T/W)_{TO}$ = (11480/23686) = 0.485

$S_{Air}/138^2$ = .0668, Pg B-25

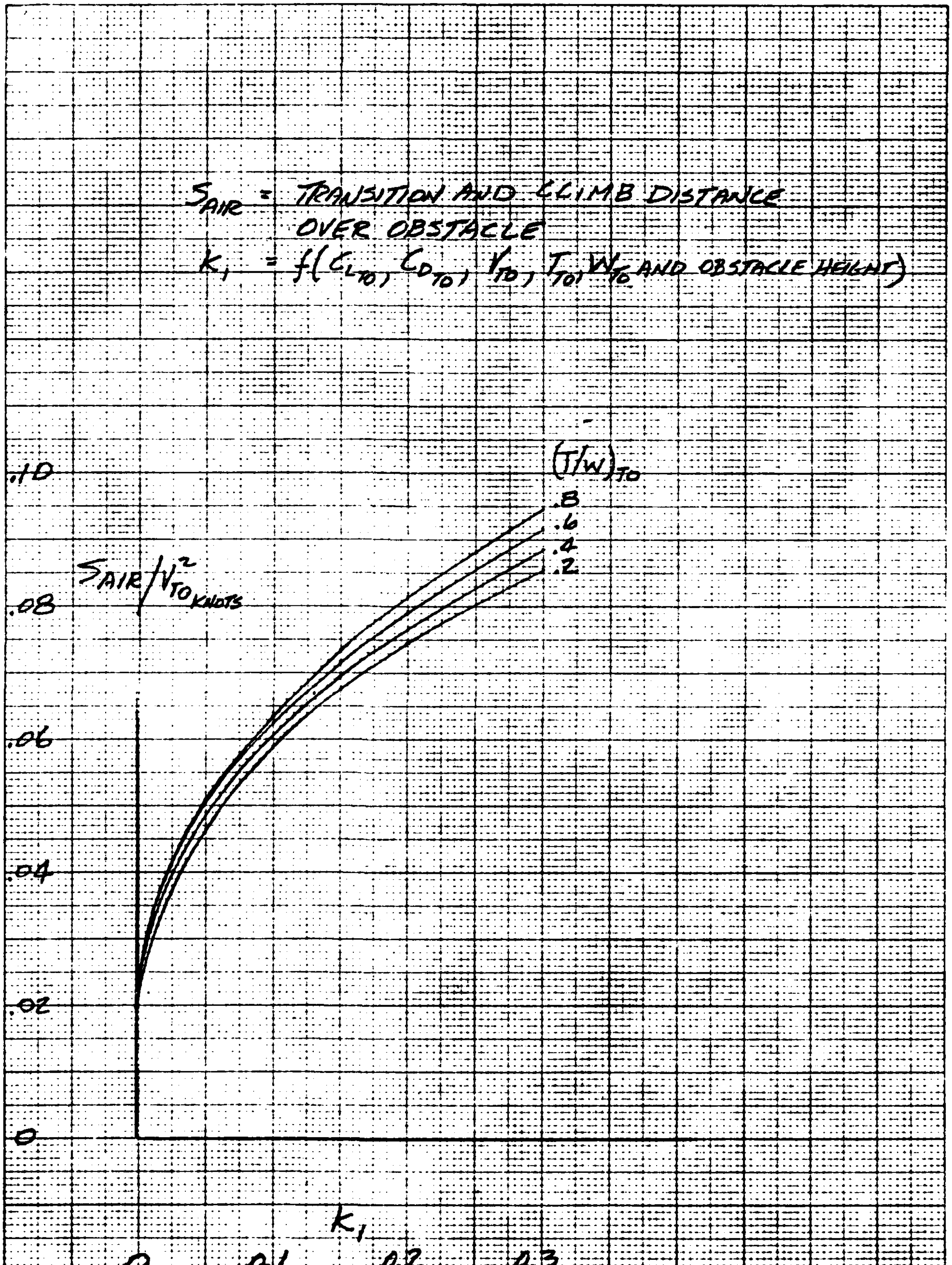
S_{Air} = 1272 ft (over 50 ft obstacle)

(c) The total distance is

S_{TO} over 50' obs. = 1272 + 2945 = 4217 ft (S.L. 89.8°F)

(d) Takeoff for the requested standard day at sea level follows the same method and is

Ground run	2813 ft.
Total distance over 50 ft obstacle	4045 ft.



SAIR = TRANSITION AND CLIMB DISTANCE OVER OBSTACLE
K1 = f(CLTO, CDTO, VTO, TTO, WTO AND OBSTACLE HEIGHT)
(T/W)TO
.8
.6
.4
.2
SAIR/VTO KNOTS ²
.10
.08
.06
.04
.02
0
K1
0
0.1
0.2
0.3

(e) Rate of climb at V_{TO} is calculated for several conditions as requested, at sea level, out of ground effect.

	89.8°F Trunk	Std Down	89.8°F Trunk	Std Up
W_{TO} - Lb	23636*	23714*	23686*	23714*
C_L (FL 40°)	1.39	1.39	1.39	1.39
C_D	.265	.265	.237	.237
Drag-lb	4516	4521	4039	4043
V - ft/sec	233	226.5	233	226.5
M	.203	.203	.203	.203
T_{Max} - lb	11480	11410	11480	11410
ΔT_{bleed} -lb	2358	2358***	0	0
T_{climb} -lb	9122	9052	11480	11410
T_{Excess} -lb	4606	4531	7441	7367
R/C - ft/min**	2719	2597	4392	4222

* W_{TO} = 24,300 - 4.5 Min. @ T_{Max}

** R/C - ft/min = T_{Excess} x V x 60/W

*** It is assumed that the ΔT_{bleed} is the same standard and 89.8°F even though there is probably a small reduction for standard temperature. A rough approximation is

$$\text{Bleed}_{Std} = 39 \times \left(\frac{.002378}{.002245}\right)^{0.5} = 40.14 \text{ lb/sec}$$

From P&W printout for standard, S.L., using Pg B-22 procedure

T_{Max} with AB (AB not lit) 12990 lb

Gas flow 227.91 lb/sec

Fuel flow 7811 lb/hr

Air flow 225.74 lb/sec

T_{Max} subsonic nozzle

$$= 12990 \times \frac{13202}{12990} = 13202 \text{ lb}$$

Ratio 225.74/13202 = .01710

ΔT_{bleed} = 40.14/.01710 = 2347

The item in question is the 40.14 lb/sec which is only an approximation.

(7) Landing Performance

(a) Landing is calculated for the Maximum Landing Design Gross Weight (MLDGW) = gross weight minus 60% CAS mission fuel of 4552 lb, for standard day at S.L.; therefore CAS mission stores are on.

$$W_{Lnd} = 24300 - .60 \times 4552 = 21569 \text{ lb}$$

Flaps are 50°, trunk is down, approach at $1.2V_s$ and landing at $1.1V_s$.

$$C_{L_{Max}} = 2.05, \text{ Pg B-21}$$
$$C_{L_{Appch}} = 2.05/1.2^2 = 1.42$$
$$C_{L_{Lnd}} = 2.05/1.1^2 = 1.69$$

As in takeoff, the rotation is small and only 2 deg. approach to landing; flaps are dumped and the nose is dropped 4.5 deg. to fuselage level (cushion level) for the landing run.

P&W has calculated that the minimum throttle setting, with the required 39 lb/sec fan bleed, gives 2500 lb thrust which is dissipated by turning vanes in the tail pipe to eject the exhaust 90°.

C_D in approach, trunk down, out of ground effect is .284 (Pg B-21), $L/D = 5.0$

Glide angle is $\text{TAN}^{-1} = 1/5.0 = 11.3^{\circ}$

Distance over 50 ft obstacle

$$S_{50} = 50/\text{TAN } 11.3^{\circ} = 250 \text{ ft}$$

(b) The transition distance to slow from approach to landing speed is given by the average (V/a) multiplied by the speed change.

$$(V/a)_{AVE} \times (V_{Appch} - V_{Lnd})$$

Cond	C_L	C_D Pg B-21 out of ground effect, trunk down	V ft/sec	V knots	Drag lb	V/a sec
Appch	1.42	.284	214	127	4330	33.11
Land	1.69	.370	196	116	4732	27.75

$$V/a = \frac{21569 \times V_{FPS}}{32.2 \times \text{Drag}}$$

$$S_{Trans} = (33.11 + 27.75)/2 \times (214-196)$$
$$= 548 \text{ ft}$$

The ground run is based on developing an average ratio of braking force to aircraft weight of 0.27. The lift and drag coefficients for fuselage level in ground effect, flaps 0 ($\alpha = 3^{\circ}, \alpha_{L=0} = 0^{\circ}$) trunk down

$$C_L = .082 \times 3 = .246$$
$$C_D = 2 \times \left[.0256 + \frac{(3 \times .076 - .032)^2}{6\pi}\left(\frac{1}{.842} - 1\right)\right] + .0161 + .246^2/(8.6\pi) = .0703$$

V Ft/Sec	Lift Lb	W-L Lb	.27(W-L) Lb	Drag Lb	Tot Force Lb
196	3146	18423	4974	899	5873
150	1843	19726	5326	527	5853
100	819	20750	5603	234	5837
50	305	21364	5768	59	5827
0	0	21569			

V Ft/Sec	V/a Sec	ΔS Ft
196	22.35	
150	17.17	909
100	11.48	716
50	5.75	431
0		144
	Tot	2200

$$V/a = \frac{V \times 21569}{32.2 \times (\text{Tot. Force})}$$

(c) Tot. landing distance over 50 ft obstacle

S_{50}	250
S_{Trans}	548
S_{Run}	2200
Tot.	2998 Ft (Std, S.L.)

(d) Landing distance for the requested 89.8°F at sea level follows the same method and is

S_{50}	250
S_{Trans}	579
S_{Run}	2329
Tot.	3158 Ft (89.8°F, S.L.)

(e) Rate of climb at V_{Appch} is calculated for several conditions as requested, all at sea level, W = 21569 lb, out of ground effect

	89.8°F	Std	89.8°F	Std
	Trunk Down		Trunk Up	
$C_{L_{Appch}}$ (flaps 50°)	1.42	1.42	1.42	1.42
C_D (Pg B-21)	.284	.284	.257	.257
Drag - lb	4314	4314	3904	3904
V - ft /sec	220.2	214	220.2	214
M	.192	.192	.192	.192
T_{Max} - lb	11510	11440	11510	11440
ΔT_{bleed} - lb	2358	2358*	0	0
T_{climb} - lb	9152	9082	11510	11440
T_{Excess} - lb	4838	4768	7606	7536
R/C - ft/min	2963	2838	4659	4486

(8) Tail Sizing

From the configuration design layout work, a conventional tail became appropriate with the horizontal mounted on the fuselage. An all-movable horizontal (no elevator) was considered; however it was not used pending an in-depth control system analysis which is outside the scope of this study.

Selection of the tail geometry considered the usual factors of

(a) Displacement of $(\bar{c}/4)_H$ and $(\bar{c}/4)_V$ to prevent adding peak pressures with resultant adverse Mach No. effects; displacement used of 13.4 ins is considered a minimum.

(b) Sweep and thickness combination to give a higher critical Mach for the tail (for lift) than developed by the wing. Thus tail effectiveness will be retained after excessive speed warning occurs due to the normal lift deterioration with Mach No. on the wing.

(c) Low span and high taper for low weight as limited by tail effectiveness and past practice.

* See note bottom page B-26 marked (***)

SANDAIRE

The following table summarizes the wing and tail characteristics, calculations follow to derive the tail areas.

		Wing	H. Tail	V. Tail
S	- sq ft	280	67.1	47.5
AR		6	3.5	1.5
b	- ft	41	15.33	8.44
C_R	- in	126.1	70.0	90.0
C_T	- in	37.8	35.0	45.0
λ		.30	.50	.50
$\bar{c}$	- in	89.8	54.5	70.0
$\Lambda_{c/4}$	- deg	25	35	40
t/c (root-tip)		.12-.10	.12-.10	.12-.10
l_t $(\bar{c}/4)_{wing} - (\bar{c}/4)_{tail}$ -ins (From Dwng SAE-79-007, Page 3)			183.5	170.1
Airfoil		Advanced	Sym	Sym
C_{l_α} rad	Sect	2π	2π	2π
C_{l_α} application factor (k)		1.0	0.90	0.95
(k) basis		hi wing	fuselage intersection	above horiz.
C_{l_α} deg	(see below)	.076	.055	.052

$$C_{L_\alpha} \text{ deg} = \frac{2\pi K}{57.3} \left(\frac{AR \cos \Lambda_{c/4}}{AR + 2 \cos \Lambda_{c/4}} \right)$$

Effective AR for the vertical tail is 2.8 to account for the horizontal end plate effect.

q_t/q (Est.)	0.95
Design Aft C.G. aircraft	$.30\ \bar{c}$
The actual aft C.G. is forward of $.30\ \bar{c}$; however this provides the added margin that is always needed when aircraft are built and the C.G. inevitably drifts aft.	
Design a.c. aircraft	$.35\ \bar{c}$
a.c. Wing (Est.)	$.26\ \bar{c}$
a.c. Wing & Fuse. (Est.)	$.19\ \bar{c}$
Downwash factor (DATCOM)	0.547
C_{N_β} Design (minimum without artificial means)	.0005/deg
C_{N_β} Wing & Fuse. (Est.)	-.0024/deg
Tail C_{N_β} Req'd	.0029/deg
a.c. shift due to tail required	$.16\ \bar{c}$

$$S_H/S = \frac{(.16) \times C_{L_{\alpha\,wing}} \times \bar{c}_{wing} \times 1.15}{(C_{L_\alpha})_H \times (\text{downwash factor}) \times \ell_{t_H} \times (q_t/q)}$$

Where the 1.15 factor is added margin for control.

$$S_H = \frac{.16 \times .076 \times (89.8/12) \times 1.15 \times 280}{.055 \times .547 \times (183.5/12) \times .95} = 67.1 \text{ sq ft}$$

$$S_V/S = (.0029 \times b_{wing}) / \left[(C_{L_\alpha})_V \times \ell_{t_V} \times (q_t/q) \right]$$

$$S_V = 280 \times (.0029 \times 41) / \left[.052 \times (170.1/12) \times .95 \right] = 47.5 \text{ sq ft}$$

APPENDIX C

WEIGHT AND BALANCE

(1) This appendix includes a Group Weight Statement, but the allocation of weights may be different than shown in the weight section (11) of the report. However, the total weight is, of course, the same.

(2) Also included here are Pages C-17 to C-27, inclusive, that compare the weights, calculated for this study, with fighter aircraft whose ultimate load factors range from 9 to 11. As always, insufficient information is available to verify that weight allocation is comparable, which may obscure the comparisons in some cases. In making any weight comparison, one must realize that this study incorporates weight reduction for the use of composite materials in some cases, and the P&W engine includes advanced technology that gives a high thrust to weight ratio.

(3) Another purpose of this appendix is to provide the basis in some detail for the weight calculations used for this study and to document the balance calculation.

(4) Weight Calculation Bases

Several of the weight equations are empirical. They are reasonably accurate and are based on comparisons with available aircraft data. Most have been used in preliminary design study work before. Dimensions and weights used are shown in the report.

(a) Wing Group Weight - Wing weight is a function of many factors; however the usual complicated empirical equation is reduced to a relatively simple form with the empirical constants chosen to be appropriate for this type aircraft. The equation uses the design limit wing loading and the structural span to depth ratio as the principal factors. It is expressed as wing group weight, $W_W = 500 \times K_1 \times 10^{-6}(W_D n/S)^{0.75} \times (b_s/t_r)^{0.75} \times S^{1.5} \times K_2$ pounds.

The (500×10^{-6}) and the exponents are empirical; K_1 is 0.90 to account for the use of composite materials in the wing for this study. W_D is the basic flight design gross weight (BFDGW) defined as design gross weight less 40% CAS mission fuel

$$= 24300 - .40 \times 4552$$
$$= 22479 \text{ lb}$$

Weight

n = specified design limit load factor
= 7 (ultimate = 1.5 x limit)
S = wing area = 280 sq ft
$W_D n/S = 22479 \times 7/280 = 562$ lb/sq ft

$b_s = (b/2)/\cos \Lambda_{c/4}$

b_s = wing structural half span
b = wing span = 41 ft
$\Lambda_{c/4}$ = wing quarter chord sweepback = 25 deg

t_R = equivalent wing root depth
$= C_R \times (t/c)_R$

C_R = equivalent wing root chord by extending the leading and trailing edges to the center line = 126.1 inches.
$(t/c)_R$ = wing thickness/wing chord ratio at C_R = 0.12

$$b_s/t_R = \frac{(41/2)/\cos 25^\circ}{(126.1/12) \times 0.12} = 17.94$$

K_2 = 1.10, a factor to account for the flap installation, Sect. (7) of the report, compared to a simple flap.

$$W_w = 500 \times 0.90 \times 10^{-6}(562)^{0.75} \times (17.94)^{0.75}$$
$$\times 280^{1.5} \times 1.10 =$$ 2334

(b) Tail Group Weight - This empirical equation is similar to the wing equation except the design dynamic pressure (q_{DES}) lb/sq ft, replaces the design wing loading. This is more representative for the tail, as maximum tail loads are developed by surface deflection at high (q) in contrast to (g) loads on the wing at pull up.

Design (q) is selected as M = 0.90 at 5000', q_{DES} = 999 lb/sq ft

A factor of 0.75 is applied to account for the use of composite materials.

Horizontal

S_H = 67.1 sq ft
$b_s = (15.33/2)/\cos 35^\circ = 9.36$ ft
$t_R = (70/12) \times .12 = 0.70$ ft
$b_s/t_R = 9.36/0.70 = 13.37$

Weight

$$W_H = 600 \times .75 \times 10^{-6} \times 999^{0.75} \times 13.37^{0.75} \times 67.1^{1.5} =$$ 307

<u>Vertical</u>

$S_V = 47.5$ sq ft

$b_s = 8.44/\cos 40^\circ = 11.02$ ft

$t_R = (90/12) \times .12 = 0.90$ ft

$b_s/t_R = 11.02/0.90 = 12.24$

$$W_V = 600 \times .75 \times 10^{-6} \times 999^{0.75} \times 12.24^{0.75} \times 47.5^{1.5} \times 1.2 =$$ 206

Where the 1.2 factor is for high speed maneuver

(c) <u>Fuselage Group Weight</u> - Fuselage weight includes the effects of bending load, arising from wing and balancing tail load; crank load, due to wing sweep; dynamic pressure (q_{DES}); and landing and takeoff loads. These effects are combined in the empirical equation

$$W_F/S_F = 238 \times 0.85 \times 10^{-4} (W_D n)^{0.25} \times (q_{DES})^{0.25} / \cos \Lambda_{c/4} \text{ - lb/sq ft}$$

The (238×10^{-4}) and the exponents are empirical. The 0.85 factor is to account for the use of composite materials.

$W_D n = 22479 \times 7$ as for the wing

S_F = measured fuselage shell area = 835 sq ft

$$W_F/S_F = 238 \times 0.85 \times 10^{-4} (22479 \times 7)^{0.25} \times 999^{0.25} / \cos 25^\circ = 2.50 \text{ lb/sq ft}$$

$$W_F = 2.50 \times 835$$ 2088

Effective canopy cutout area is 29.5 sq ft.

$W_t = 29.5 \times 3 \times 2.50/.85 =$ 260

Where the 0.85 factor removes the composite material.

	Weight
Speed brakes are aft on the fuselage; est. weight is	104
(d) Propulsion System - Scale 1.0, P&W STF-529 turbofan with fan bleed, subsonic nozzle, no reverser	1618
P&W quote is 1618 lb with thrust reverser. No reverser is planned; however vanes are required to dissipate the 2500 lb thrust produced by the lowest throttle setting that will allow 39 lb fan bleed to pressurize the trunk for landing, as stated on page D-1 . Also some variable exhaust deflection means may be needed for steering if asymmetrical braking is inadequate. Therefore the reverser weight is retained to allow for these items.	
Tail pipe extension 26.5" lgth x 38.5" diameter	
$\text{Wt} = \frac{26.5}{12} \times \frac{38.5\,\pi}{12} \times .04/12$ (gage) $\times$ 500 (Wt/cu ft) $\times$ 1.15 (installation) =	43
Engine section 9% of engine weight = .09 x 1618 =	146
Inlet ducts Forward twin duct effective length = 71.5", aft single duct effective length = 100" engine inlet 35" dia.	
Fwd duct equiv. dia - $(35^2/2)^{0.5} = 24.7"$	
Duct surface area = $(24.7\,\pi \times 71.5 \times 2 + 35\,\pi \times 100)/144 = 153$ sq ft	
For AL., .032" gage $\text{Wt} = .032 \times 153 \times 144 \times 0.10$ (lb/cu in) = 71 lb	
Est. two inlets plus duct installation 89 lb, total =	160
Est. engine controls .015 x Wt Eng.	
Est engine starting .020 x Wt Eng.	
Est engine lub .015 x Wt Eng.	

	Weight
Est. engine oil (Incl. unusable) $.008 \times$ Wt Eng.	
Total $.058 \times 1618 =$	94
(e) Flight Controls - This is an empirical equation	
$$W_{FC} = 0.9 \times 0.17 \left[(L_F + b/\cos\Lambda_{c/4}) \times W_{DES} \times n \right]^{0.5}$$	
L_F = Fuse. Lgth = 42 ft	
W_{DES} = BFDGW = 22479 lb	
b = Wing Span = 41 ft	
$\Lambda_{c/4}$ = Wing Sweep = 25°	
n = DES. L.F. = 7	
The 0.9 is to account for the fly-by-wire system	
The remainder is empirical	
$$W_{FC} = 0.9 \times 0.17 \left[(42 + 41/\cos 25^{\circ}) \times 22479 \times 7 \right]^{0.5} =$$	567
(f) Fuel Tanks - All CAS mission fuel (4552 lb) is in integral wing tanks with 1150 lb of this in a self-sealing cell. The 1041 lb is the fuel required to return to base in the CAS 160 N.M. radius mission. Additional integral fuselage fuel capacity is provided so no external fuel tanks are required for ferry.	
Est. weight wing integral tanks	107
Est. weight fuse. integral tanks	96
Est. weight self sealing cells in wing (see protection weight below)	
Capacity wing tanks 4922 lb	
Capacity fuse. tanks 3200 lb	
(g) Unusable Fuel - (1% CAS mission fuel)	46

		Weight
(h)	Fuel System - (excluding tanks, but includes fuel dump and provisions for aerial refueling) - Est.	170
(i)	Systems - (Est.)	
	Instruments	85
	Electrical	350
	Anti-ice	130
	Air Cond.	160
	Furnishings incl. ejection seat	330
	APU	120
	Armament Prov.	200
	Equipment (incl. oxygen and survival)	175

(j) SETOLS - Installed Weight

$= t\ d\ k\ A_c + 90$ where

t = trunk sheet thickness = .1875"

d = trunk sheet density = 142 lb/cu ft

k = factor for installation, ducts, attachments, brakes, etc. = 2.0

A_c = cushion ℄ area = 150 sq ft

The 90 lb is the est. weight of the roll stabilizing doors and mechanism.

		Weight
	W_{SETOLS} = (.1875/12) x 142 x 2 x 150 + 90 =	756
(k)	Protection	
	Fuel cells in wing (est.)	67
	Armor - pilot (allowance)	300
	Other (allowance)	100
(l)	CAS Loading Specified (8574 lb)	
	Installed avionics	770
	Crew	180
	4 - TERS	384
	12 - Mk 82 (droppable)	6840
	4 - Pylons	400
(m)	CAS Mission Fuel	4552
(n)	Contingency	55
	Gross Weight (lb)	24300

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(5) BALANCE - CAS MISSION LOADING

Item	Weight-Lb	Station*	Waterline*
Wing	2334	313.2	96.5
Horiz. Tail	307	496.25	90
Vert. Tail	206	487.5	141.6
Fuselage (incl Spd Brakes)	2192	274	67
Canopy	260	120	102.5
Engine	1618	400	73
Tail Pipe Ext.	43	444.5	73
Engine Sect.	146	382	77
Inlet Ducts	160	256.5	77
Eng. Cont., Start, Lub.,Oil	94	372	83.5
Flight Cont.	567	400	94
Fuel Tanks	203	295	77.7
Unusable Fuel	46	280	81.1
Fuel System	170	356	94.5
Systems			
Inst.	85	98	85
Elect. & Hyd.	350	236	78.5
Anti-Ice	130	364	94.5
Air Cond.	160	180	54.5
Furn. (Incl. Seat)	330	122.4	71.3
APU	120	210	47.5
Arm. Prov.	200	298	75.5
Equip.	175	147	67.6
CAS Mission Specified			
Avionics Inst.	(770)	(182)	(92.8)
1	385	159	93
2	55	151	78
3	330	214	95
4 TERS	384	300	76
4 Pylons	400	311	87.5
SETOLS	756	317	42
Protection			
Fuel Cell	67	292	97
Pilot	300	126	72
Other	100	402	73
Unassigned	55	300	67
Weight Empty	(12728)	(300.4)	(80.05)

* See three view, page 3 of the report.

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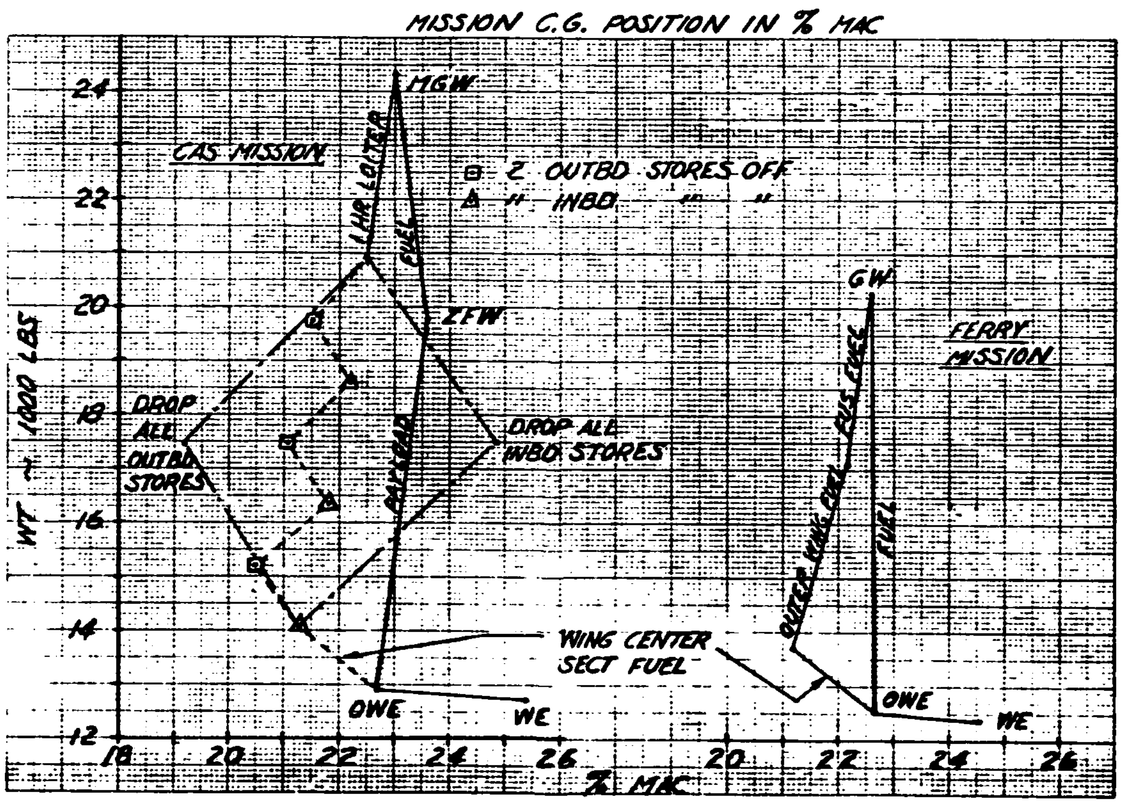
Item	Weight-Lb	Station*	Waterline*
Weight Empty	(12,728)	(300.4)	(80.05)
Pilot (Specified Wt.)	180	124	75
Operating Weight	(12,908)	(297.9)	(80)
12 - Mk 82 (droppable)	6,840	300	67
Zero Fuel Weight	(19,748)	(298.7)	(75.6)
Fuel (CAS Mission)	4,552	296.2	97
Gross Weight	(24,300)	(298.2)	(79.4)

$\bar{c}/4$ at Station 300 with $\bar{c}$ = 89.8 inches

The C.G. location and C.G. control are considered satisfactory. (See graph below).



* See bottom Page C-7

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GROUP WEIGHT STATEMENT

AIRCRAFT

(INCLUDING ROTORCRAFT)

ESTIMATED - ~~CALCULATED~~ - ~~ACTUAL~~

(CROSS OUT THOSE NOT APPLICABLE)

CONTRACT NO.	N62269-79-C-0438		
AIRCRAFT, GOVERNMENT NO.			
AIRCRAFT, CONTRACTOR NO.			
MANUFACTURED BY	Conceptual Point Design Study		
		MAIN	AUX
ENGINE Proposed		P&W	
ENGINE MODEL		STF - 529	
ENGINE NO.			
ENGINE TYPE		Turbofan	
PROPELLER MANUFACTURED BY			
PROPELLER MODEL			
PROPELLER NUMBER			
PAGES REMOVED	None	PAGE NO.	

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WEIGHT EMPTY

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1	WING GROUP					2334	
2	BASIC STRUCTURE-CENTER SECTION						
3	-INTERMEDIATE PANEL						
4	-OUTER PANEL						
5	-GLOVE						
6	SECONDARY STRUCTURE-INCL.WING FOLD WEIGHT			LBS.			
7	AILERONS - INCL. BALANCE WEIGHT	LBS.					
8	FLAPS - TRAILING EDGE						
9	- LEADING EDGE						
10	SLATS						
11	SPOILERS						
12							
13							
14	ROTOR GROUP						
15	BLADE ASSEMBLY						
16	HUB & HINGE - INCL. BLADE FOLD WEIGHT		LBS.				
17							
18							
19	TAIL GROUP					513	
20	STRUCT. - STABILIZER (INCL. LBS.SEC. STRUCT.)						
21	- FIN-INCL.DORSAL (INCL. LBS.SEC.STRUCT.)						
22	VENTRAL						
23	ELEVATOR - INCL.BALANCE WEIGHT	LBS.					
24	RUDDERS - INCL.BALANCE WEIGHT	LBS.					
25	TAIL ROTOR - BLADES						
26	- HUB & HINGE						
27							
28	BODY GROUP						
29	BASIC STRUCTURE - FUSELAGE OR HULL					2088	
30	- BOOMS						
31	SECONDARY STRUCTURE - FUSELAGE OR HULL						
32	- BOOMS						
33	- SPEEDBRAKERS					104	
34	- DOORS, RAMPS, PANELS & MISC.						
35	- Canopy					260	
36							
37	ALIGHTING GEAR GROUP - TYPE ** SETOLS					756	
38	LOCATION Bottom Fuse.		RUNNING	*STRUCT.	CONTROLS		
39	MAIN						
40	NOSE/TAIL						
41	ARRESTING GEAR						
42	CATAPULTING GEAR						
43							
44							
45	ENGINE SECTION OR NACELLE GROUP Engine Installation					146	
46	BODY - INTERNAL						
47	- EXTERNAL						
48	WING - INBOARD						
49	- OUTBOARD						
50							
51	AIR INDUCTION GROUP Inlet Ducts					160	
52	- DUCTS						
53	- RAMPS, PLUGS, SPIKES						
54	- DOORS, PANELS & MISC.						
55							
56							
57	TOTAL STRUCTURE					6361	

* CHANGE TO FLOATS AND STRUTS FOR WATER TYPE GEAR.
**LANDING GEAR "TYPE": INSERT "TRICYCLE", "TAIL WHEEL", "BICYCLE", "QUADRICYCLE", OR SIMILAR DESCRIPTIVE NOMENCLATURE.

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GROUP WEIGHT STATEMENT
WEIGHT EMPTY

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No.	Item	X AUXILIARY XX	MAIN X
58	PROPULSION GROUP		
59	ENGINE		1618
60	Tail Pipe Extension		43
61			
62	ACCESSORY GEAR BOXES & DRIVE		
63	EXHAUST SYSTEM		
64	ENGINE COOLING		
65	WATER INJECTION		
66	ENGINE CONTROL		24
67	STARTING SYSTEM		32
68	PROPELLER INSTALLATION		
69	SMOKE ABATEMENT		
70	LUBRICATING SYSTEM		24
71	FUEL SYSTEM		170
72	TANKS - PROTECTED Self Sealing Cells in Wing C.S.		67
73	- UNPROTECTED Integral Wing & Fuse.		203
74	PLUMBING, ETC.		
75			
76	DRIVE SYSTEM		
77	GEAR BOXES, LUB SY & ROTOR BRK		
78	TRANSMISSION DRIVE		
79	ROTOR SHAFTS		
80			
81	FLIGHT CONTROLS GROUP		567
82	COCKPIT CTLS. (AUTOPILOT LBS.)		
83	SYSTEMS CONTROLS		
84			
85			
86	AUXILIARY POWER PLANT GROUP		120
87	INSTRUMENTS GROUP		85
88	HYDRAULIC & PNEUMATIC GROUP		
89			
90	ELECTRICAL GROUP		350
91			
92	AVIONICS GROUP Specified		770
93	EQUIPMENT		
94	INSTALLATION		
95			
96	ARMAMENT GROUP (INCL. PASSIVE PROT. 400 LBS.)		600
97	FURNISHINGS & EQUIPMENT GROUP		
98	ACCOMMODATION FOR PERSONNEL		
99	MISCELLANEOUS EQUIPMENT		
100	FURNISHINGS Incl. Ejection Seat		330
101	EMERGENCY EQUIPMENT		
102	Equipment Incl. Oxygen & Survival		175
103	AIR CONDITIONING GROUP		160
104	ANTI-ICING GROUP		130
105	Contingency		55
106	PHOTOGRAPHIC GROUP		
107	LOAD & HANDLING GROUP		
108	AIRCRAFT HANDLING		
109	LOADING HANDLING		
110	BALLAST		
111	MANUFACTURING VARIATION		
112	TOTAL CONTRACTOR CONTROLLED		
113	TOTAL GFAE		
114	TOTAL WEIGHT EMPTY - PG 2-3		11884

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GROUP WEIGHT STATEMENT
USEFUL LOAD AND GROSS WEIGHT

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115	LOAD CONDITION	Close Air Support (CAS)					
116							
117	CREW (NO. one) Specified					180	
118	PASSENGERS (NO.)						
119	FUEL LOCATION TYPE JP-5	GALS.	669.4****			4552***	
120	UNUSABLE					46	
121	INTERNAL						
122							
123							
124							
125	EXTERNAL						
126							
127							
128	OIL (Incl. Unusable)					14	
129	TRAPPED						
130	ENGINE						
131							
132	FUEL TANKS (LOCATION)						
133	WATER INJECTION FLUID (	GALS.)					
134							
135	BAGGAGE						
136	CARGO						
137							
138	GUN INSTALLATIONS						
139	GUNS LOCAT.FIX.OR FLEX.QUANTITY CALIBER						
140							
141							
142	AMMO.						
143							
144							
145	SUPP'TS *						
146	WEAPONS INSTALL. ** Specified						
147							
148	4 - TER's					384	
149							
150	12 - Mk 82 (Droppable)					6840	
151	4 - Pylons					400	
152							
153							
154							
155							
156							
157							
158							
159							
160							
161							
162	SURVIVAL KITS						
163	LIFE RAFTS						
164	OXYGEN						
165	MISC.						
166							
167							
168							
169	TOTAL USEFUL LOAD					12416	
170	WEIGHT EMPTY						
171	GROSS WEIGHT					24300	

* IF REMOVABLE AND SPECIFIED AS USEFUL LOAD.

**LIST STORES, MISSILES, SONOBUOYS, ETC. FOLLOWED BY RACKS, LAUNCHERS, CHUTES, ETC. THAT ARE NOT PART OF WEIGHT EMPTY. LIST IDENTIFICATION, LOCATION, AND QUANTITY FOR ALL ITEMS SHOWN INCLUDING INSTALLATION.

*** 1150 lb of this is in a self-sealing cell installed in the wing center section integral tank. (The required fuel to return to base from the CAS mission 160 N.M. radius point is 1041 lb).

**** All fuel in wing for CAS mission

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GROUP WEIGHT STATEMENT
DIMENSIONAL AND STRUCTURAL DATA *****

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1	WING, ROTOR + TAIL GROUPS	WING	H TAIL	V TAIL	CANARD	ROTOR (BLADS/KIT)	
2							
3	RADIUS OR SPAN(FT)	41.00	15.33	8.44			
4	*SPAN AT .25 CHORD						
5	**ROOT CHORD(IN) - THEO.	126.1	70.0	90.0			
6	- MAX THICKNESS %	12	12	12			
7	**PLANFORM BREAK-CORD (IN)						
8	- MAX THICKNESS						
9	**TIP CHORD (IN) - THEO.	37.8	35.0	45.0			
10	- MAX THICKNESS %	10	10	10			
11	SWEEP ANGLE AT .25 CHORD Deg	25	35	40			
12	ASPECT RATIO	6	3.5	1.5			
13	TAPER RATIO	0.30	0.50	0.50			
14	MEAN AERODYNAMIC CHORD (in)	89.8	54.5	70.0			
15	AREAS *** sq ft	280	67.1	47.5			
16							
17	AREAS WING	SPD.BRK.	LE FLAPS	TE FLAPS	SLATS	SPOILERS	AIL
18	(SQ.FT.PER AIRCRAFT)						
19	FUS	SPD.BRK.	ELEV.	RUDDER	DORSAL		
20							
21							
22	ROTOR DISK AREAS - FWD		AFT		FOLDED	WING SPAN	
23	WING .25MAC TO H TAIL .25MAC (IN)		183.5	NOSE	TO WING	.25 MAC	
24	WING .25MAC TO V TAIL .25MAC (IN)			170.1		LEMAC	
25	WING BOX SPAN AT FUS.INTERSECTION			WING BOX	LENGTH AT	C.L.	
26							
27		CAPTURE	BLOW-IN	DUCT	MAX.DES.	CIRCUM-	
28	ENGINE INLETS	AREA	AREA	LENGTH	PRESSURE	FERENCE	
29	-MAIN						
30	AUXILIARY						
31		LENGTH	DEPTH	WIDTH	WET.AREA	VOLUME	VOL.PRESS
32	BODY + NACELLE GROUPS						
33	FUSELAGE OR HULL**** (in)	504					
34	BOOMS						
35	NACELLES (INBD.B.L.	)					
36	(OUTBD.B.L.	)					
37	ALIGHTING GEAR GROUP	LENGTH-OLEO EXT.		OLEO TRAVEL		LENGTH ARREST	
38		AXLE-CL. TRUNNION		EXT.TO COLLAPSED		HOOK TRUNNION	
39	- LOCATION					TO POINT	
40	- DIMENSION(INCHES)						
41							
42	PROPULSION GROUP	(S.L.S. UNINSTALLED THRUST IN LBS./ENGINE)					
43			MAXIMUM	INTERMEDIATE		MAX SLS	SHAFT RPM
44	ENGINES		RATING	RATING		SHAFT HP	AT MAX HP
45	MAIN (NO. one)		13202 lb				
46	AUXILIARY (NO.)						
47							
48				OUTPUT	INTER	NUMBER	
49	ROTOR DRIVE SYSTEM	DESIGN	INPUT	RPM AT	ROTOR	GEAR	TORQUE
50		H.P.	R.P.M.	ROTOR	R.P.M.	BOXES	FACTOR
51	1/2 HOUR RATINGS - MAIN						
52	- TAIL						
53	- INTERMEDIATE						
54	CONT.R/TINGS - MAIN						
55	- TAIL						
56	- INTERMEDIATE						
57							

THE NOTES FOR THIS PAGE MAY BE FOUND ON PAGE 8 OF PART I UNDERNEATH "AIRFRAME UNIT WEIGHT".

***** See drawings, pages 3 and 4 of the report for missing dimensions.

SANDAIRE

MIL-STD-1374 - TAB	GROUP WEIGHT STATEMENT	PAGE
NAME	DIMENSIONAL AND STRUCTURAL DATA	MODEL CAS SETOLS
DATE	(CONTINUED)	SAE-79-011 REPORT App. C

1	FUEL SYSTEM	X PROTECTED		XX UNPROTECTED		XX INTEGRAL	X
2	- INTERNAL * LOCATION	NO. TANKS	GALLONS	NO. TANKS	GALLONS	NO. TANKS	GALLONS
3	WING	1	169.1			2	723.8
4	FUSELAGE					1	470.6 **
5							
6	- EXTERNAL *	No External Tanks Required for Ferry					
7		Mission					
8	OIL						
9							
10							
11		QUANTITY X	GENERATOR	X	BATTERY RATING		EMERG
12		MAIN X	OUTPUT	X	(TYPE	)	GENERATR
13	ELECTRICAL GENERATING	GENERATRS X	D.C.	A.C. X	AMP-HOURS		(KVA)
14	SYSTEMS						
15							
16							
17		BODY					
18		PLUS INT	EXTERNAL	FUEL IN		DESIGN	ULTIMATE
19		CONTENTS	WEIGHT	WINGS		GROSS	LOAD
20	STRUCTURAL DATA - CONDITION	-LBS.	ON BODY	-LBS.		WEIGHT	FACTOR
21	FLIGHT - MANEUVER BFDGW					22479	7.0x1.5
22	- GUST						
23	LANDING MLDGW					21569	
24	MAXIMUM GROSS WEIGHT WITH	ZERO	WING FUEL				
25	CATAPULTING						
26	Gross Weight With CAS Loading					24300	
27	CRASH LIMIT LOAD FACTOR -	AXIAL		LATERAL		VERTICAL	
28	ULTIMATE LANDING SINK SPEED(FT/SEC)						
29	WING OR ROTOR LIFT ASSUMED FOR LDNG DSGN COND.						
30	STALL SPEED LDNG. CONFIGURATION-POWER OFF (KNOTS)						
31	APPROACH SPEED POWER ON (V-P KNOTS)						
32	ENGAGING SPEED (KNOTS)						
33	PRESSURIZED CABIN - ULTIMATE	DESIGN					
34	PRESSURE DIFFERENTIAL FLIGHT	(PSI)					
35	CARGO FLOOR AREA (DESIGN LOAD			LBS/SQ.FT.)			
36	HYDRAULIC SYSTEM OIL CAPACITY	(GALLONS)					
37	TAIL ROTOR CANT ANGLE (DEGREES)						
38							
39							
40	ROTOR TIP SPEED AT DESIGN LIMIT		R.P.M.	POWER	FT/SEC		
41	- MAIN						
42	- TAIL						
43							
44	DESIGN THRUST OR LIFT ON	WING		M ROTOR		T ROTOR	
45	ULTIMATE L.F. FOR THE ABOVE LOADS						
46							
47	MATERIAL BREAKDOWN IN PERCENT		STEEL	ALUM	TI	COMPOSITE	OTHER
48	OF STRUCT. WEIGHT (PAGE 2, LINE 57)						
49							
50	DESIGN SPEEDS AT S.L. (KNOTS)	LEVEL			DIVE		
51							
52	DESIGN SPEED AT BEST CRUISE		SPEED		ALTITUDE		
53	MAX. SPEED AND ALTITUDE		SPEED		ALTITUDE		
54							
55							
56	MODEL FIRST FLIGHT DATE						
57	AIRFRAME UNIT WEIGHT					8527	(from Pg C-9)

*TOTAL USABLE CAPACITY.

** This provides the additional fuel capacity for the 2500 NM specified ferry mission (Total fuel required for ferry 7687 lb)

SANDAIRE

MIL-STD-1374 PART I - TAB
NAME
DATE

GROUP WEIGHT STATEMENT
DESCRIPTION OF DIMENSIONAL
AND STRUCTURAL DATA

SAE-79-011

PAGE
MODEL CAS SETOL
REPORT App. C

REFER TO PARAGRAPH 5.1.1.4 OF

DETAILED REQUIREMENTS FOR INSTRUCTIONS FOR USE

SANDAIRE

MIL-STD-1374 PART I AIRFRAME UNIT WEIGHT PAGE
NAME MODEL CAS SETOLS
DATE SAE-79-011 REPORT App. C

THE AIRFRAME UNIT WEIGHT TO BE ENTERED ON LINE 56 OF PAGE 6 OF THE GROUP WEIGHT STATEMENT SHOULD BE DERIVED BELOW IN DETAIL SHOWING THOSE ITEMS DEDUCTED FROM WEIGHT EMPTY. THE ITEMS BELOW FOLLOW THE DEFINITION OF AIRFRAME UNIT WEIGHT CARRIED IN THE DOCUMENT "CONTRACTOR COST DATA REPORTING SYSTEM" DATED 5 NOVEMBER 1973. AIRFRAME UNIT WEIGHT IS THE SAME AS PREVIOUSLY CALLED AMPR AND DCPR AND IS NOT TO BE CONFUSED WITH WORK BREAKDOWN STRUCTURE (WBS) AIRFRAME COST DEFINITION.

	WEIGHT EMPTY		11884
	DEDUCT THE FOLLOWING ITEMS DESCRIBED IN PART II		
1	WHEELS BRAKES, TIRES & TUBES	Trunk	333
2	ENGINES - MAIN AND AUXILIARY		1618
3	RUBBER OR NYLON FUEL CELLS		67
4	STARTERS - MAIN AND AUXILIARY		32
5	PROPELLERS		
6	AUXILIARY POWER PLANT UNIT		120
7	INSTRUMENTS		85
8	BATTERIES & ELECTRICAL POWER SUPPLY & CONVERSION		100
9	AVIONICS		770
10	TURRETS & POWER OPERATED MOUNTS		
11	AIR CONDITIONING, ANTI-ICING AND PRESSURIZATION UNITS & FLUIDS		232
12	CAMERAS & OPTICAL VIEWFINDERS		
	AIRFRAME UNIT WEIGHT		8527

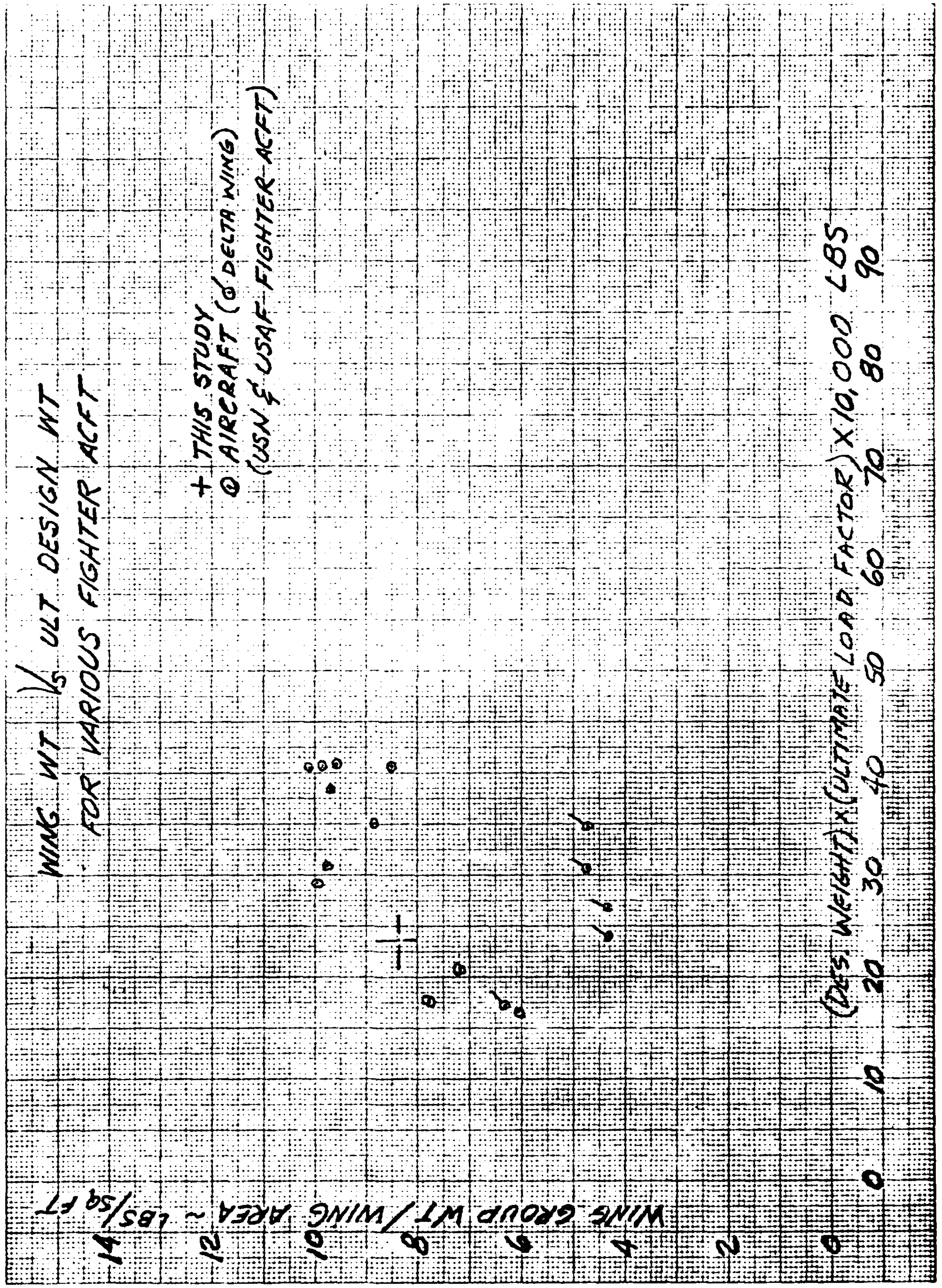
NOTES FOR PAGE 5:

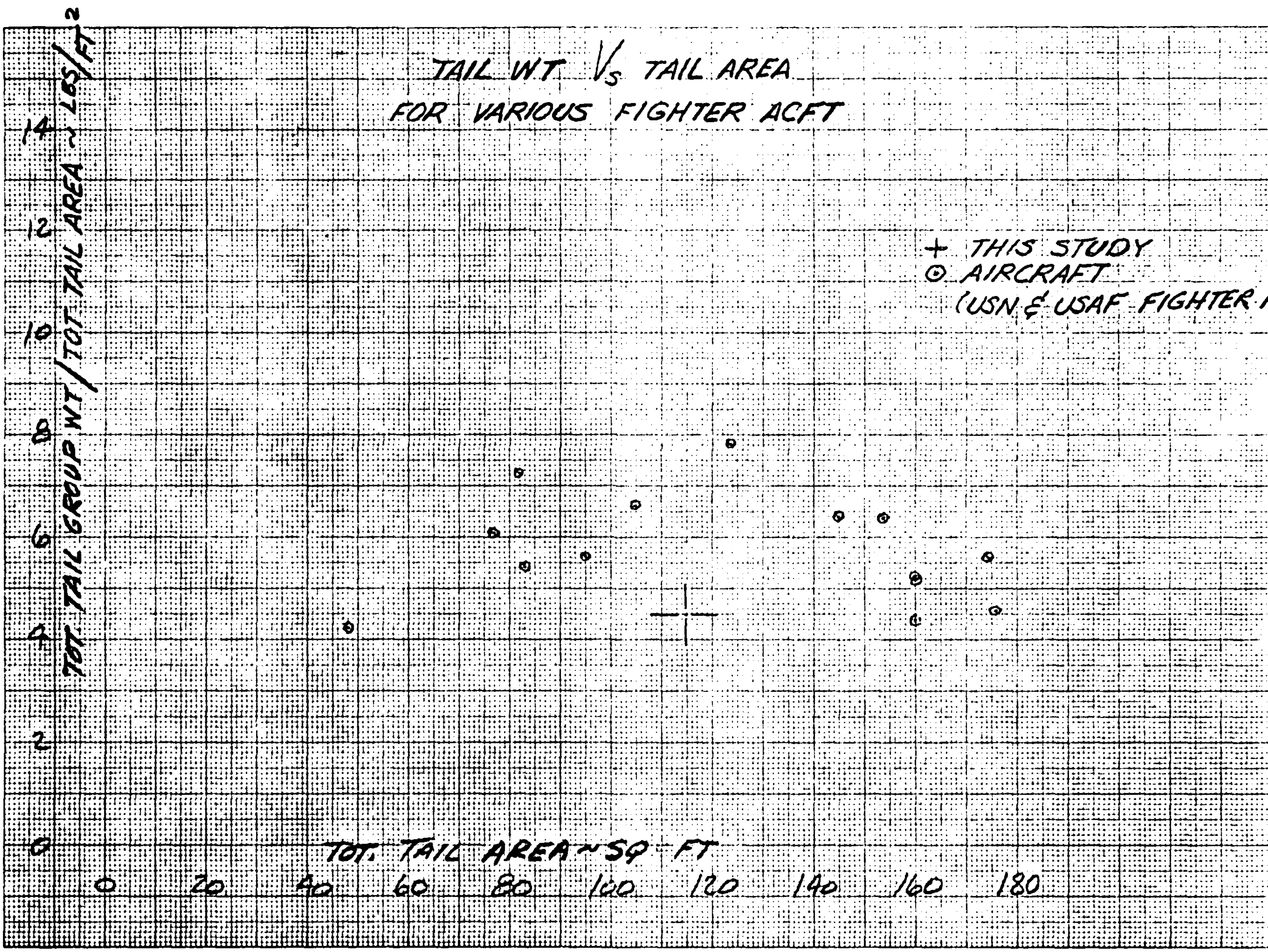
* INSERT INCHES FROM CENTER LINE OF THE ROTOR TO THE ELASTIC AXIS OF THE BLADE ATTACHMENT FOR THE ROTORS.

** PARALLEL TO THE CENTER LINE OF THE VEHICLE FOR WING AND TAIL.

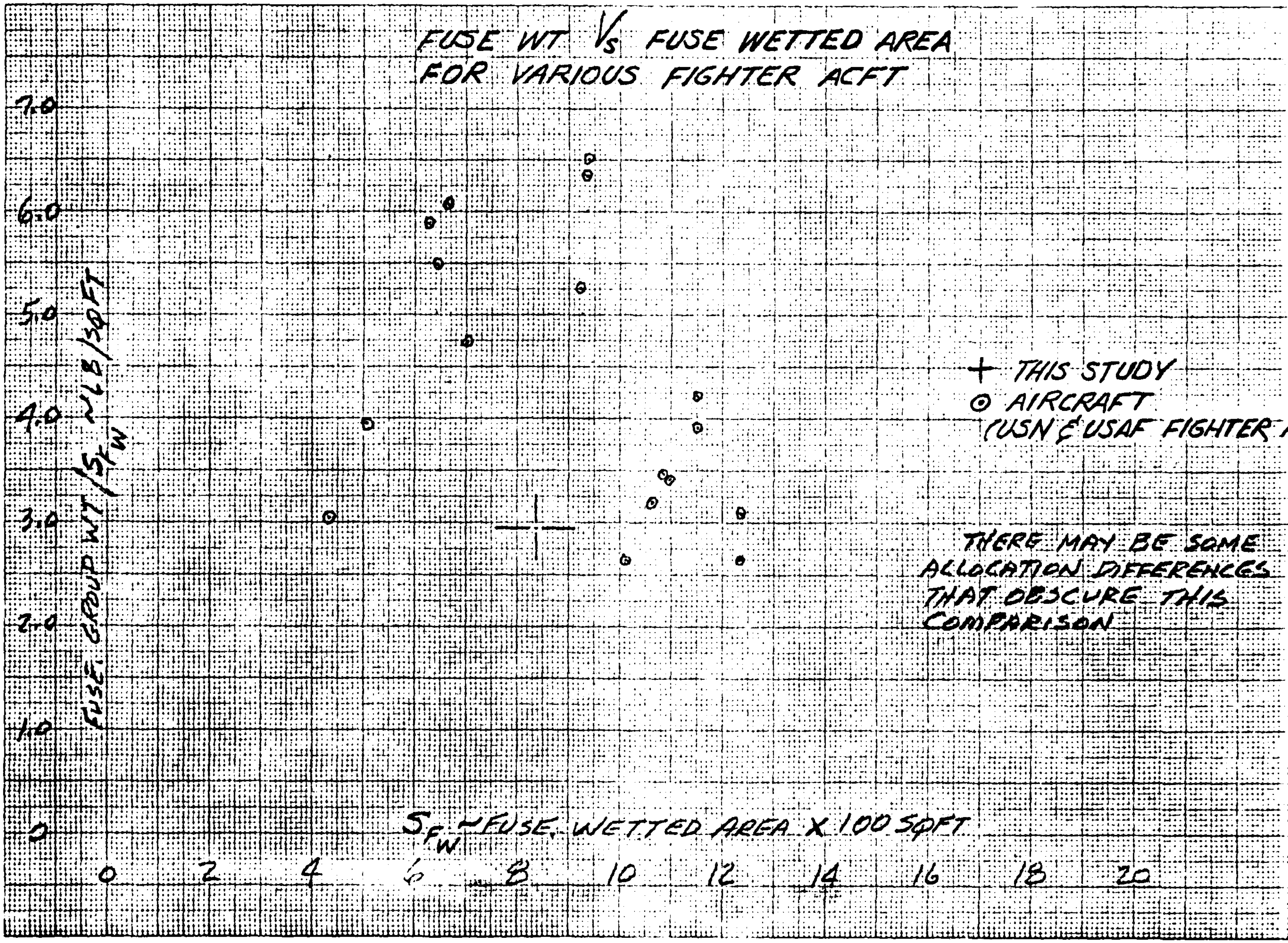
*** THEORETICAL FOR ROTORS AND CONTINUOUS WING, EXPOSED FOR NON CONTINUOUS WING AND ALL OTHERS.

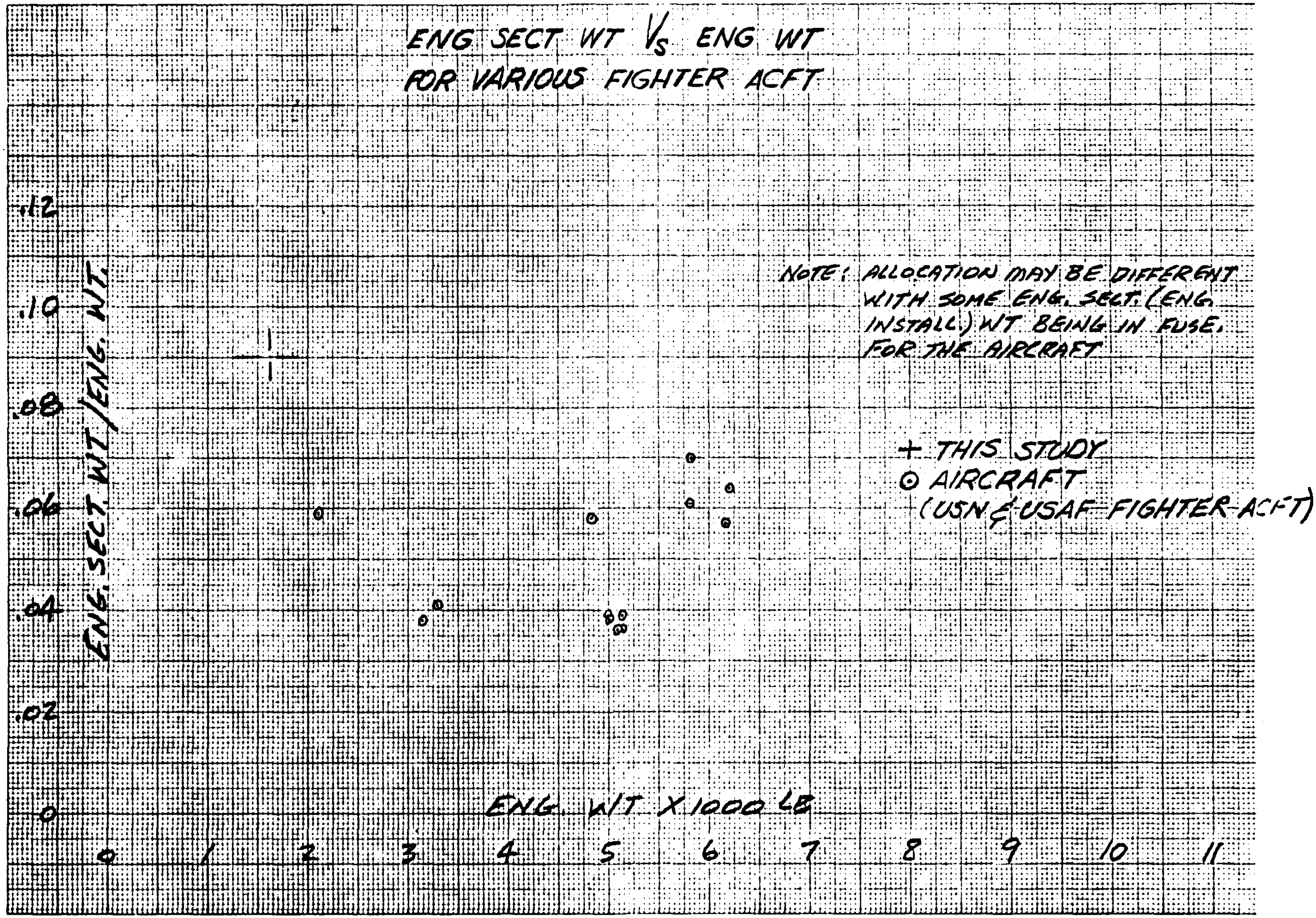
****NOSE TO AFT TIP OF FUSELAGE EXCLUDING EQUIPMENT PROTUBERENCES.

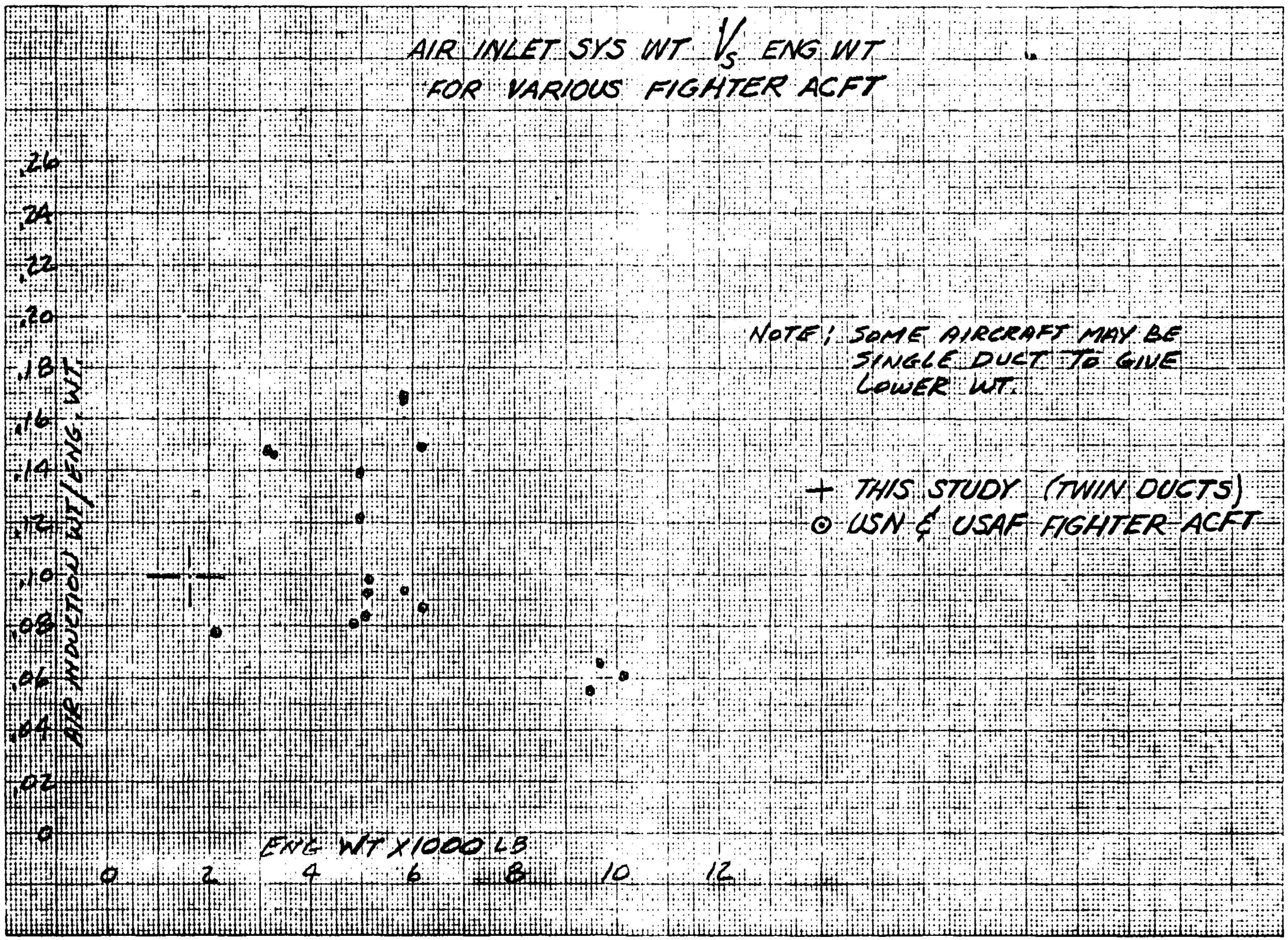

WING WT Vs ULT DESIGN WT
FOR VARIOUS FIGHTER ACFT
+ THIS STUDY
⊙ AIRCRAFT (⌀ DELTA WING)
(USN & USAF FIGHTER ACFT)
WING GROUP WT / WING AREA ~ LBS/SQ FT
0
2
4
6
8
10
12
14
(DES. WEIGHT) x (ULTIMATE LOAD FACTOR) X 10,000 LBS
0
10
20
30
40
50
60
70
80
90

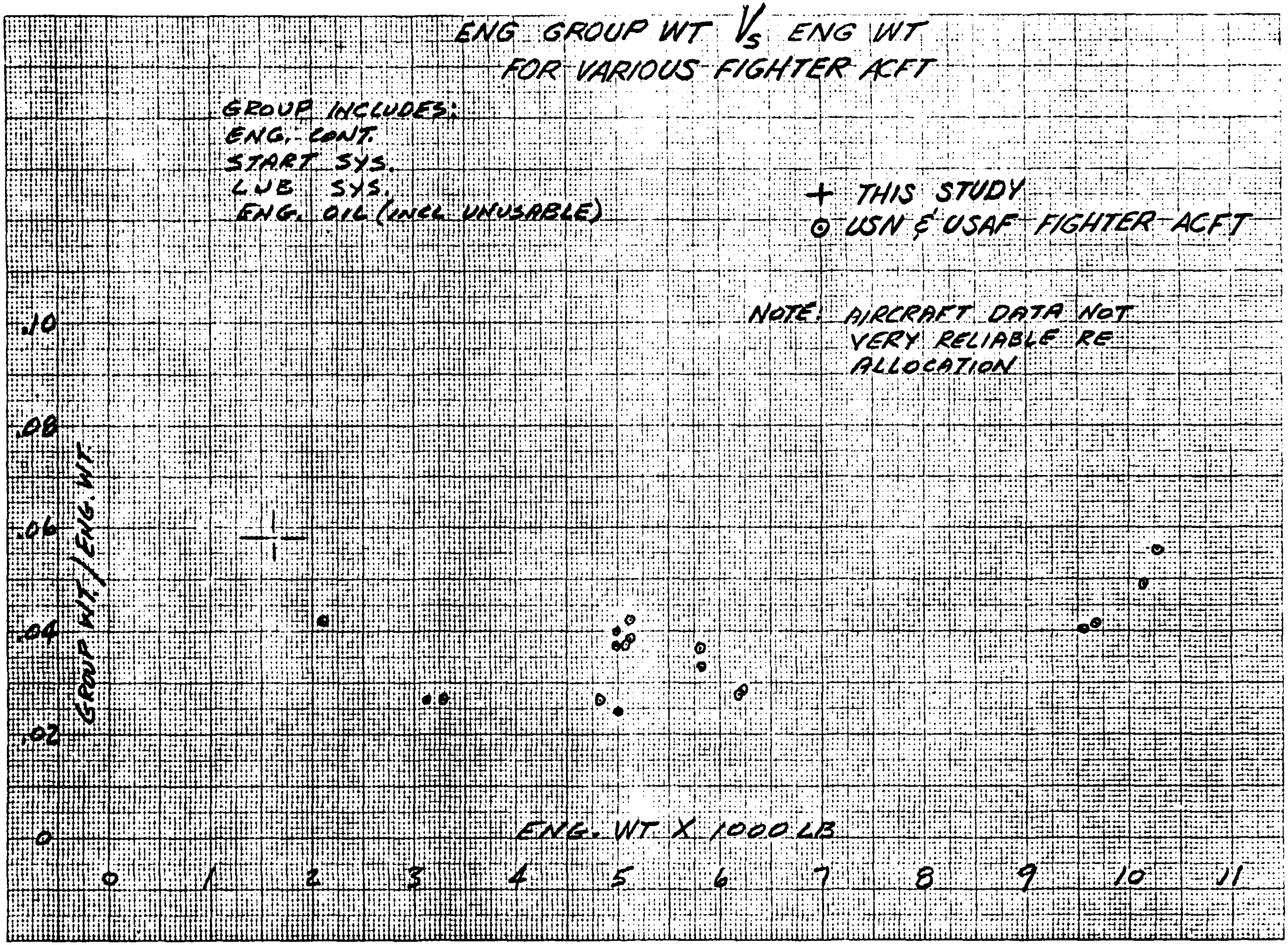


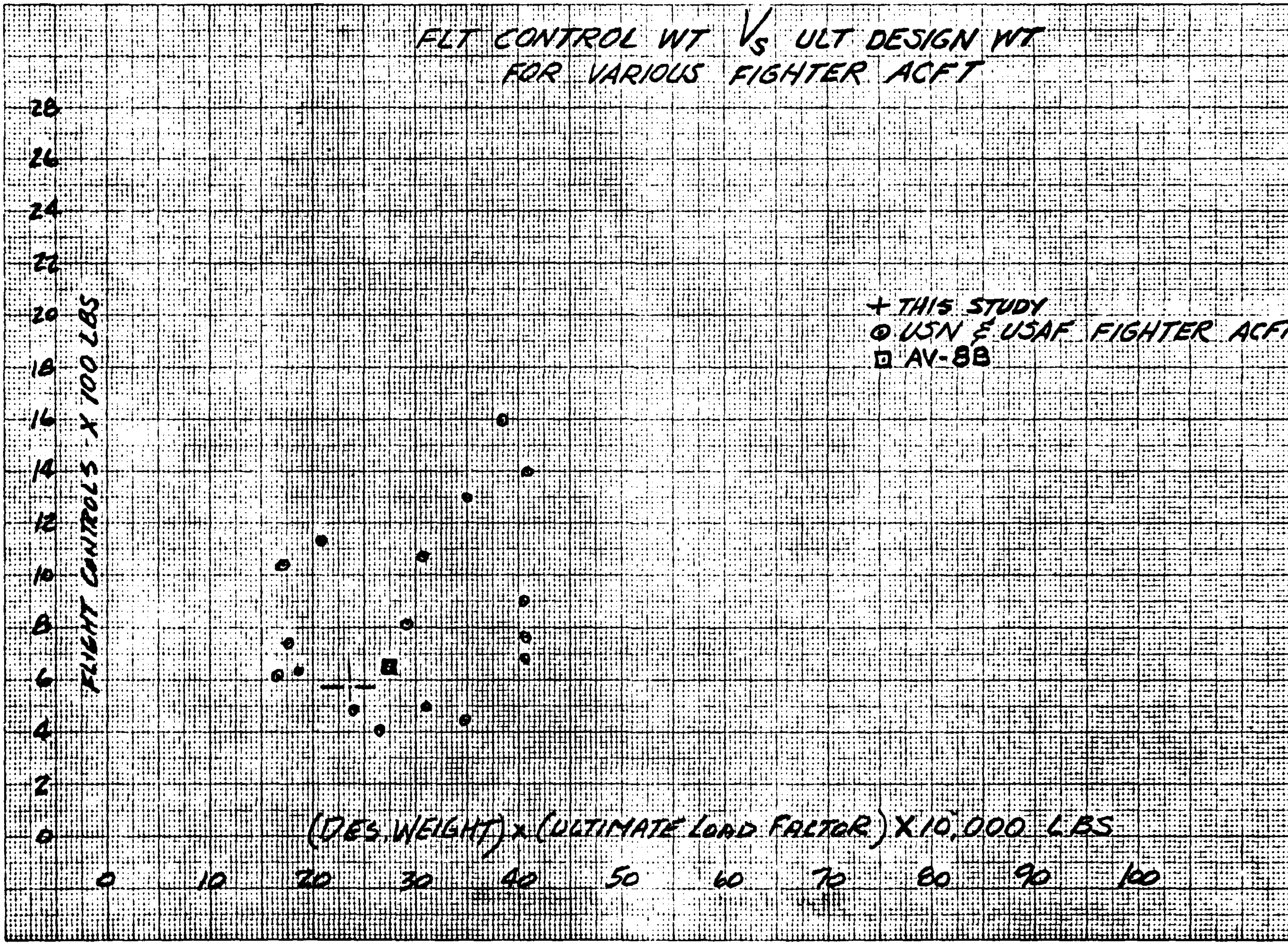
TAIL WT. Vs TAIL AREA
FOR VARIOUS FIGHTER ACFT
+ THIS STUDY
⊙ AIRCRAFT
(USN & USAF FIGHTER ACFT)
TOT. TAIL GROUP WT / TOT. TAIL AREA ~ LBS/FT²
TOT. TAIL AREA ~ SQ FT
0
2
4
6
8
10
12
14
0
20
40
60
80
100
120
140
160
180

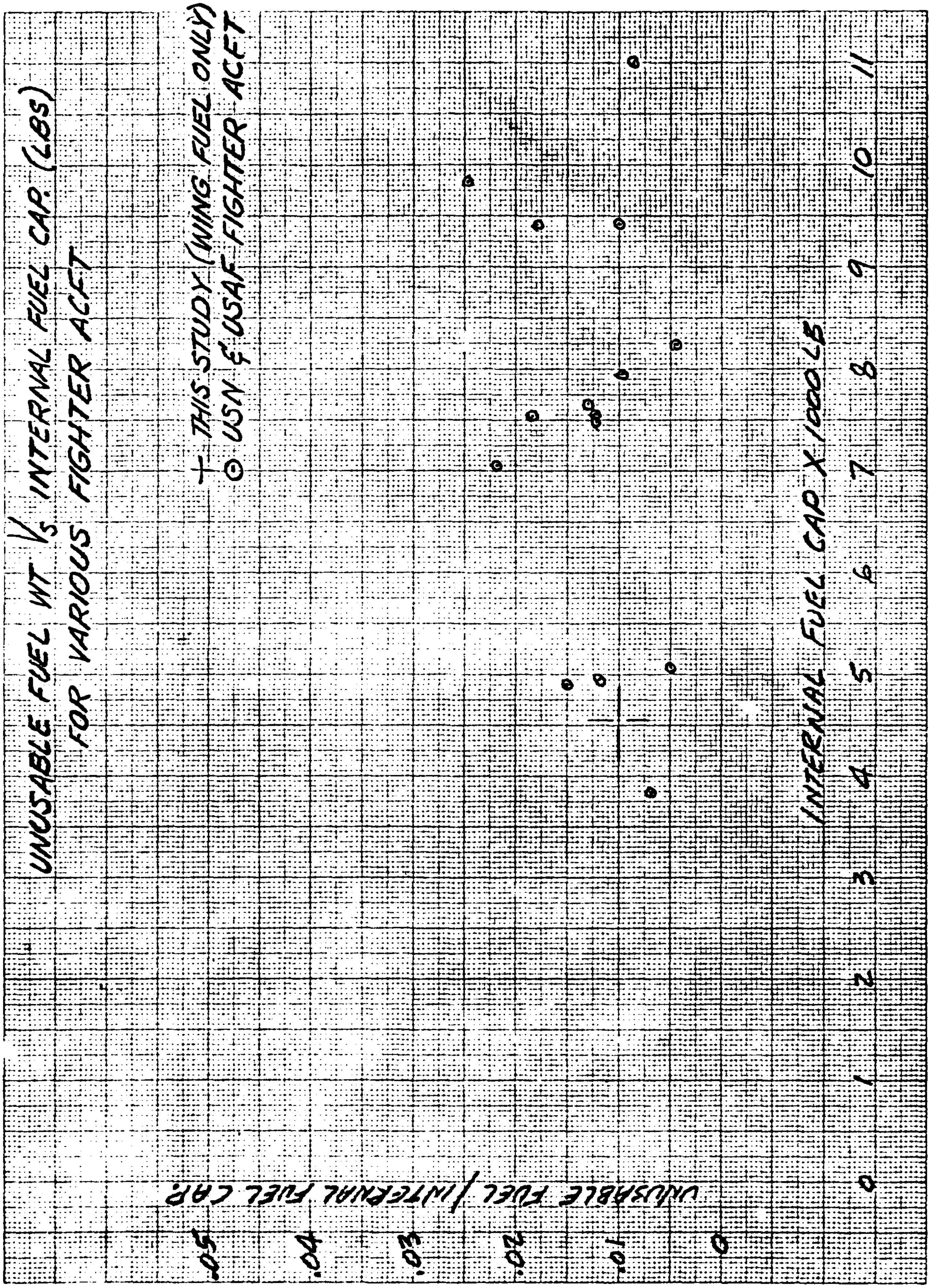

FUSE WT V_S FUSE WETTED AREA
FOR VARIOUS FIGHTER ACFT
7.0
6.0
5.0
4.0
3.0
2.0
1.0
0
FUSE. GROUP WT/S_{F_W} ~LB/SQFT
+ THIS STUDY
⊙ AIRCRAFT
(USN & USAF FIGHTER ACFT)
THERE MAY BE SOME
ALLOCATION DIFFERENCES
THAT OBSCURE THIS
COMPARISON
S_{F_W} ~FUSE. WETTED AREA X 100 SQFT
0
2
4
6
8
10
12
14
16
18
20

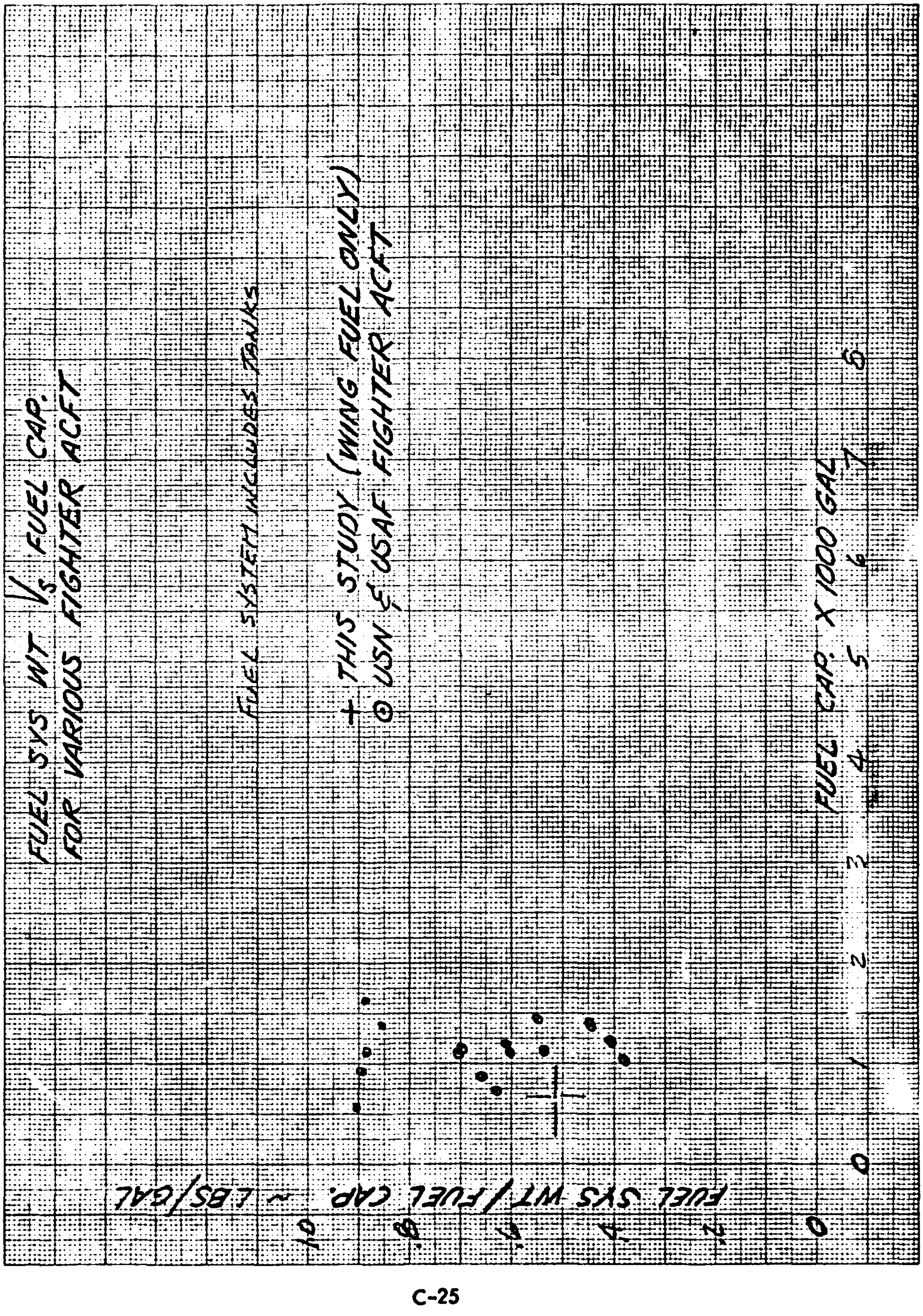

ENG SECT WT V_S ENG WT
FOR VARIOUS FIGHTER ACFT
NOTE: ALLOCATION MAY BE DIFFERENT WITH SOME ENG. SECT. (ENG. INSTALL.) WT BEING IN FUSE. FOR THE AIRCRAFT
+ THIS STUDY
⊙ AIRCRAFT (USN & USAF FIGHTER ACFT)
ENG. SECT. WT./ENG. WT.
ENG. WT X 1000 LB
.12
.10
.08
.06
.04
.02
0
0
1
2
3
4
5
6
7
8
9
10
11


AIR INLET SYS WT V_S ENG WT
FOR VARIOUS FIGHTER ACFT
NOTE; SOME AIRCRAFT MAY BE
SINGLE DUCT TO GIVE
LOWER WT.
+ THIS STUDY (TWIN DUCTS)
⊙ USN & USAF FIGHTER ACFT
AIR INDUCTION WT/ENG. WT.
.26
.24
.22
.20
.18
.16
.14
.12
.10
.08
.06
.04
.02
0
ENG WT X1000 LB
0
2
4
6
8
10
12


ENG GROUP WT Vs ENG WT
FOR VARIOUS FIGHTER ACFT
GROUP INCLUDES:
ENG. CONT.
START SYS.
LUB SYS.
ENG. OIL (INCL UNUSABLE)
+ THIS STUDY
⊙ USN & USAF FIGHTER ACFT
NOTE: AIRCRAFT DATA NOT VERY RELIABLE RE ALLOCATION
GROUP WT./ENG. WT.
.10
.08
.06
.04
.02
0
ENG. WT X 1000 LB
0
1
2
3
4
5
6
7
8
9
10
11

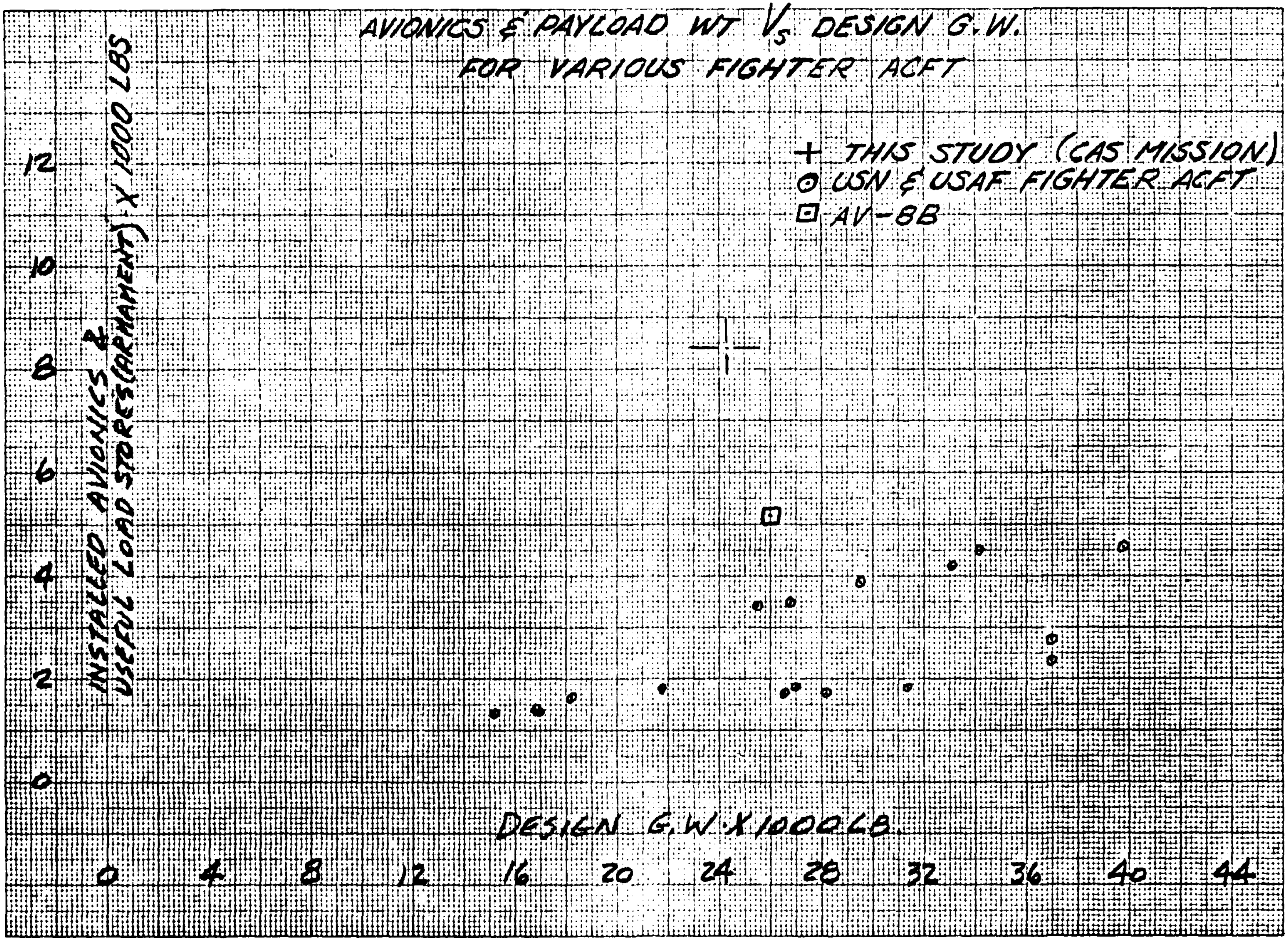

FLT CONTROL WT Vs ULT DESIGN WT
FOR VARIOUS FIGHTER ACFT
+ THIS STUDY
⊙ USN & USAF FIGHTER ACFT
◻ AV-8B
FLIGHT CONTROLS - X 100 LBS
(DES. WEIGHT) x (ULTIMATE LOAD FACTOR) X 10,000 LBS
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
0
10
20
30
40
50
60
70
80
90
100


UNUSABLE FUEL WT Vs INTERNAL FUEL CAP. (LBS)
FOR VARIOUS FIGHTER ACFT
+ THIS STUDY (WING FUEL ONLY)
⊙ USN & USAF FIGHTER ACFT
INTERNAL FUEL CAP X 1000 LB
0
1
2
3
4
5
6
7
8
9
10
11
UNUSABLE FUEL / INTERNAL FUEL CAP
0
.01
.02
.03
.04
.05


FUEL SYS WT Vs FUEL CAP.
FOR VARIOUS FIGHTER ACFT
FUEL SYSTEM INCLUDES TANKS
+ THIS STUDY (WING FUEL ONLY)
⊙ USN & USAF FIGHTER ACFT
FUEL CAP. X 1000 GAL
0
1
2
3
4
5
6
7
8
FUEL SYS WT / FUEL CAP. ~ LBS/GAL
0
.2
.4
.6
.8
1.0

SYS GROUP WT Vs DESIGN G.W.
FOR VARIOUS FIGHTER ACFT
GROUP INCLUDES
INSTRUMENTS
ELECTRICAL
HYDRAULIC & PNEUMATIC
ANTIICING
AIR COND.
FURNISHINGS INCL EJ.SEAT
EQUIPMENT
ARMAMENT PROVISION
APU
+ THIS STUDY
⊙ USN & USAF FIGHTER ACFT
(SINGLE PLACE)
□ AV-8B
GROUP WEIGHT - X 1000 LBS
0
1
2
3
4
5
DESIGN G.W. X1000 LB
0
4
8
12
16
20
24
28
32
36
40
44

END


AVIONICS & PAYLOAD WT V_S DESIGN G.W.
FOR VARIOUS FIGHTER ACFT
+ THIS STUDY (CAS MISSION)
⊙ USN & USAF FIGHTER ACFT
⊡ AV-8B
INSTALLED AVIONICS &
USEFUL LOAD STORES (ARMAMENT) X 1000 LBS
DESIGN G.W. X 1000 LB.
0
2
4
6
8
10
12
0
4
8
12
16
20
24
28
32
36
40
44

APPENDIX D

ENGINE - DATA

The Pratt & Whitney Advanced Study STF-529 Turbofan engine characteristics and installed engine performance data are presented on the following pages. The scale 1.0 engine as used has the following characteristics:

Rated T_{Max} (SLS, Std)	13202 lb
Thrust to Weight Ratio	8.2
Airflow	225.7 lb/sec
Bypass Ratio	1.54
Inlet Diameter	35 inches
Length as installed with a 26.5" tailpipe extension	145 inches
Engine and Fan Bleed Config.	See Page D-12
Required Fan Bleed	39 lb/sec
Installation Factor	0.95
Weight with Subsonic Nozzle and Fan Bleed	1618 lb

The weight of 1618 pounds includes a thrust reverser; however, the aircraft design of this report does not include a full thrust reverser. For landing, vanes or other means are required at the nozzle to dissipate the approximately 2500 pound thrust that results from the minimum throttle setting that is required to provide 39 lb/sec fan bleed for SETOLS trunk pressurization. Also, vanes at the nozzle may be required for slow speed directional control on the land, water and other surfaces. Therefore, any weight reduction due to removal of the reverser is assumed added for these additional nozzle modifications.

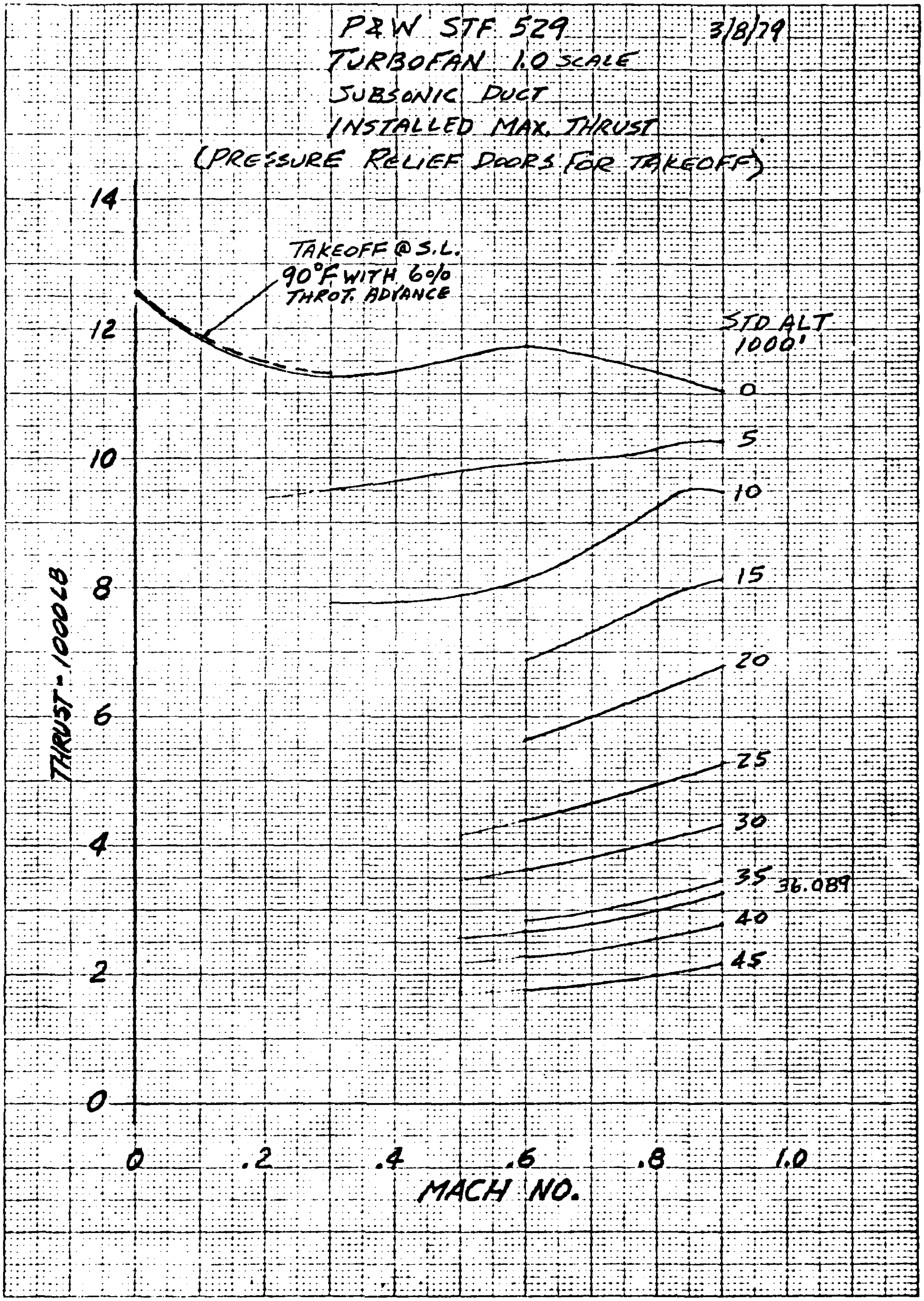
Engine performance is taken from a P&W computer printout for the engine with an afterburner and, as no afterburner was used for this design, the data was corrected to account for the performance deterioration due to its installation. The ratio of the engine thrust (at sea level state conditions) with and without the afterburner is used in conjunction with an engine installation factor (.95) to correct for installed thrust as shown in the following equation. The fuel flow is used as read from the printout.

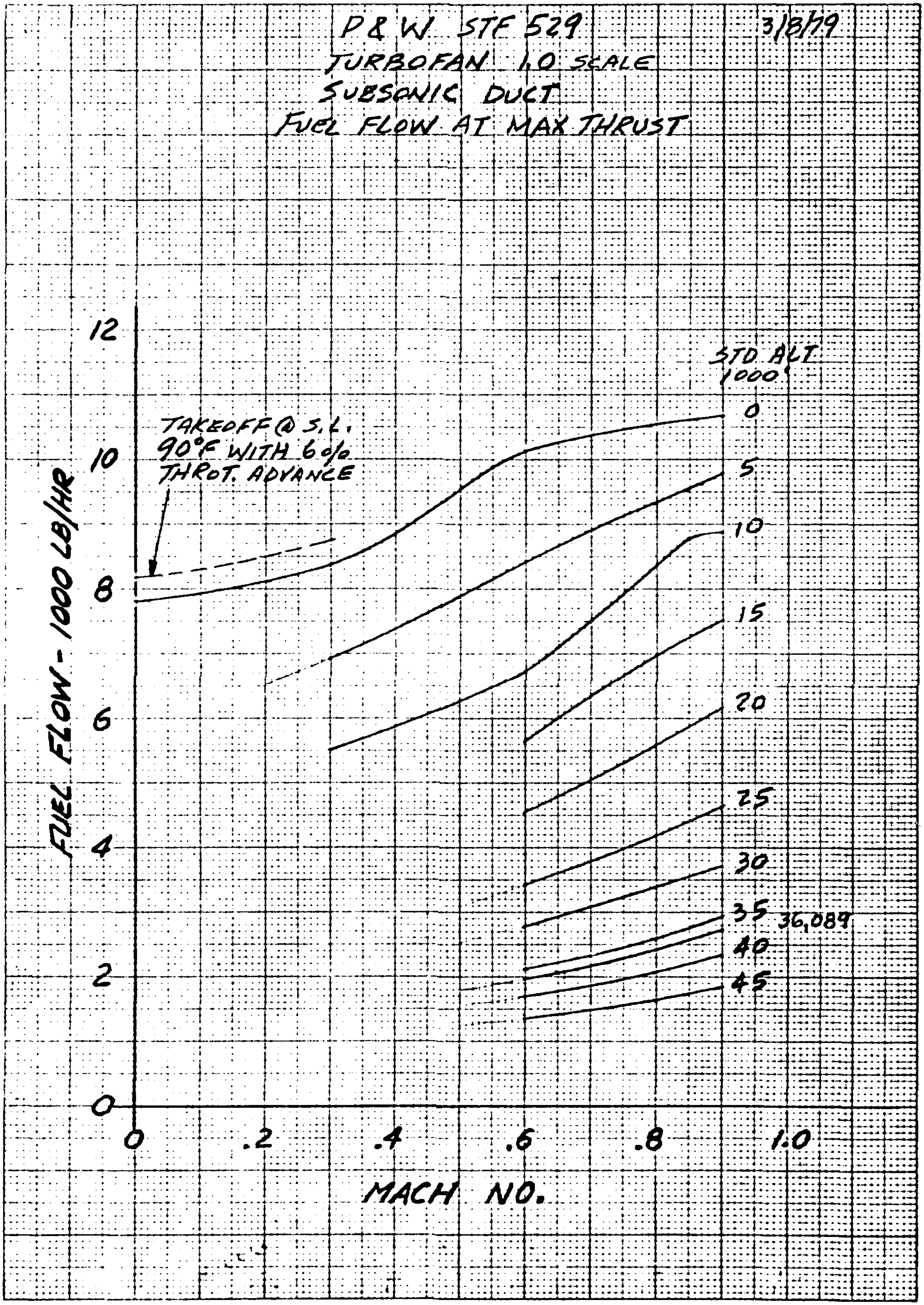
$$T_{Inst} = T_{Printout} \times \frac{13202}{12990} \times .95 = .9655 \times T_{Printout}$$

SANDAIRE

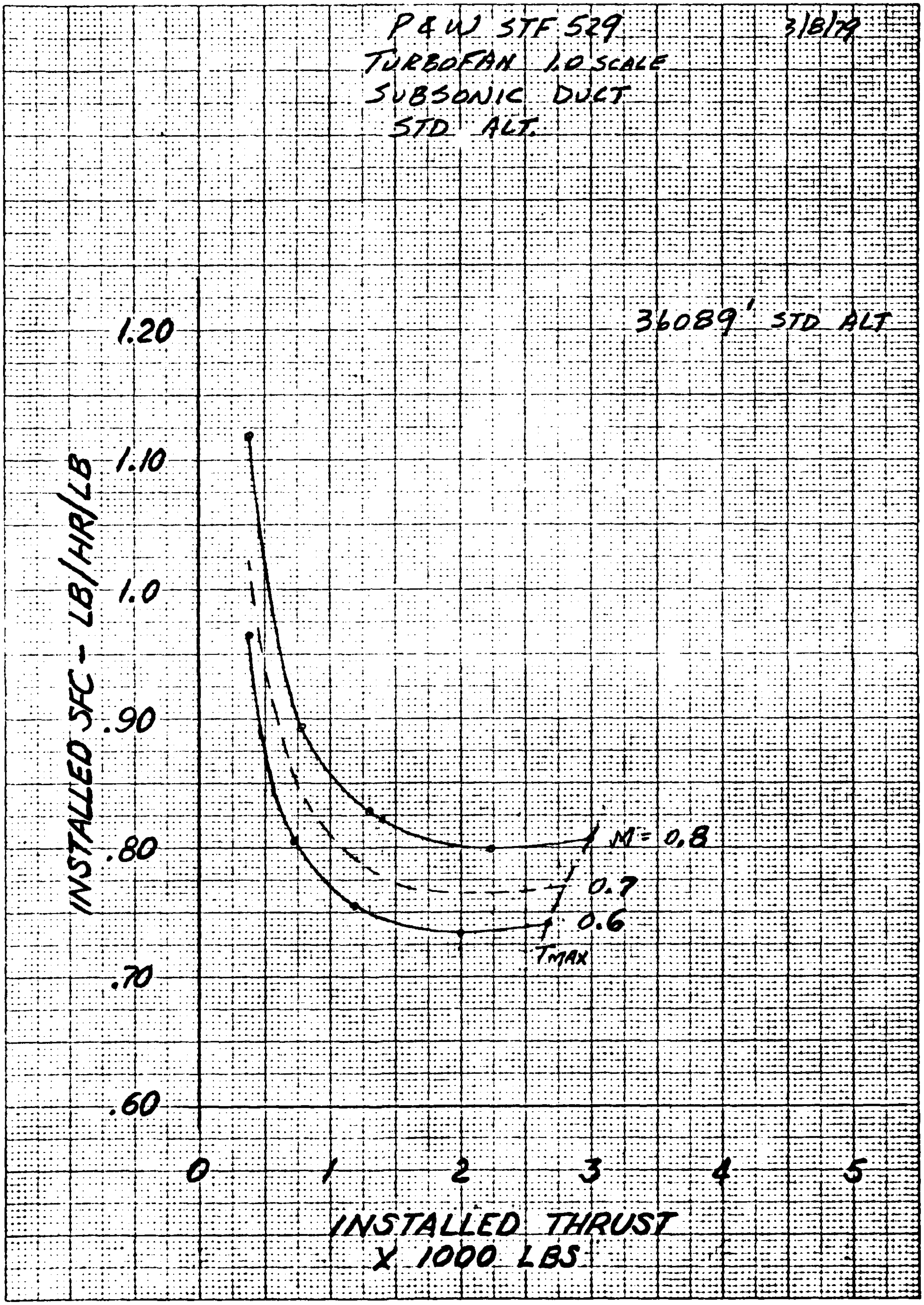
The .95 installation factor may be optimistic even for the 1995 time period. But if the factor were decreased 5% to .9, the effect would be minimal. The most significant effect would be to the takeoff distance (increase approximately 210 feet) and to the rate of climb which now exceeds the design requirement. The maximum speed would only be decreased approximately .003 Mach No. due primarily to the steepness of the drag rise curve.

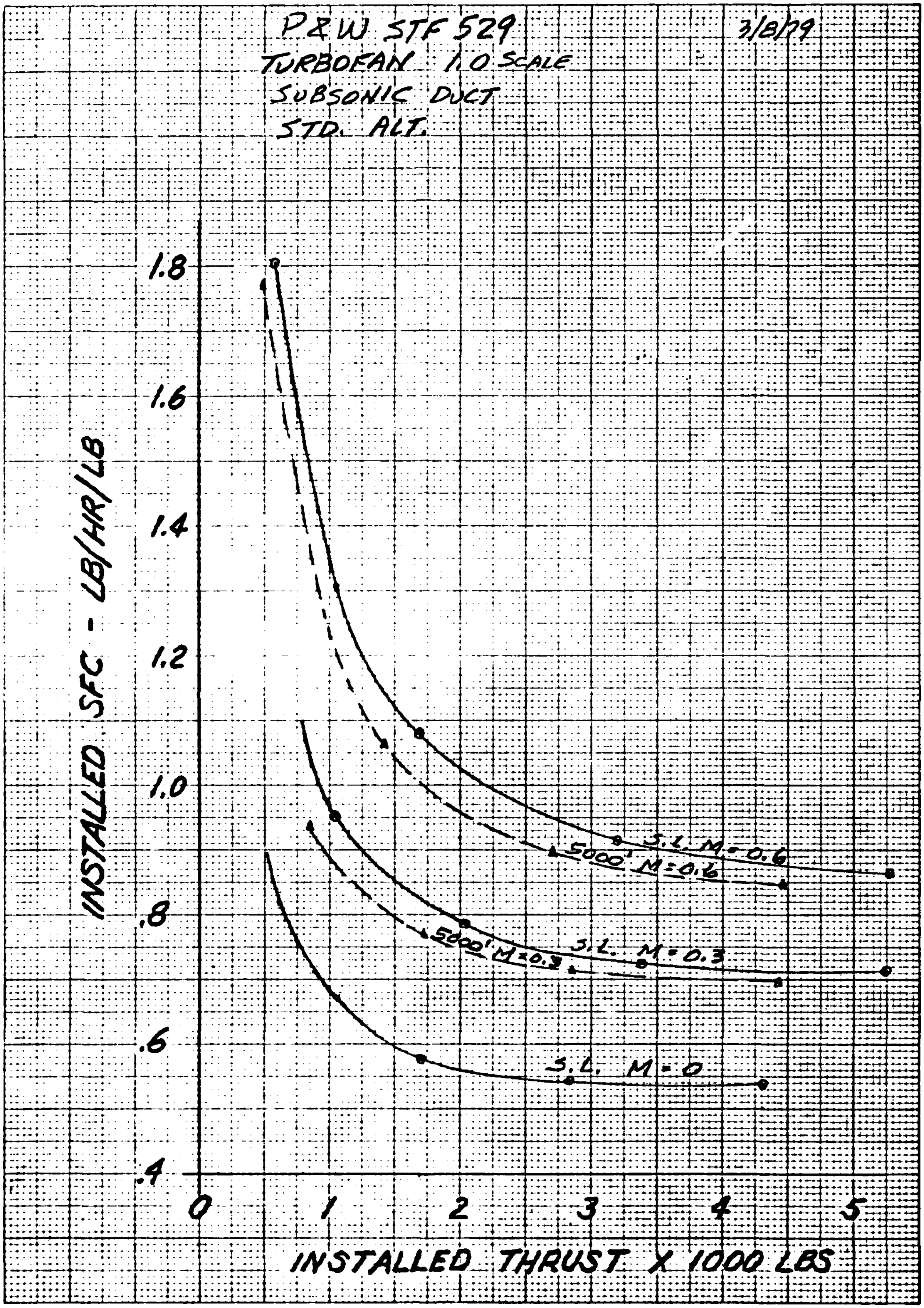
The calculated installed engine performance is shown on the following pages.

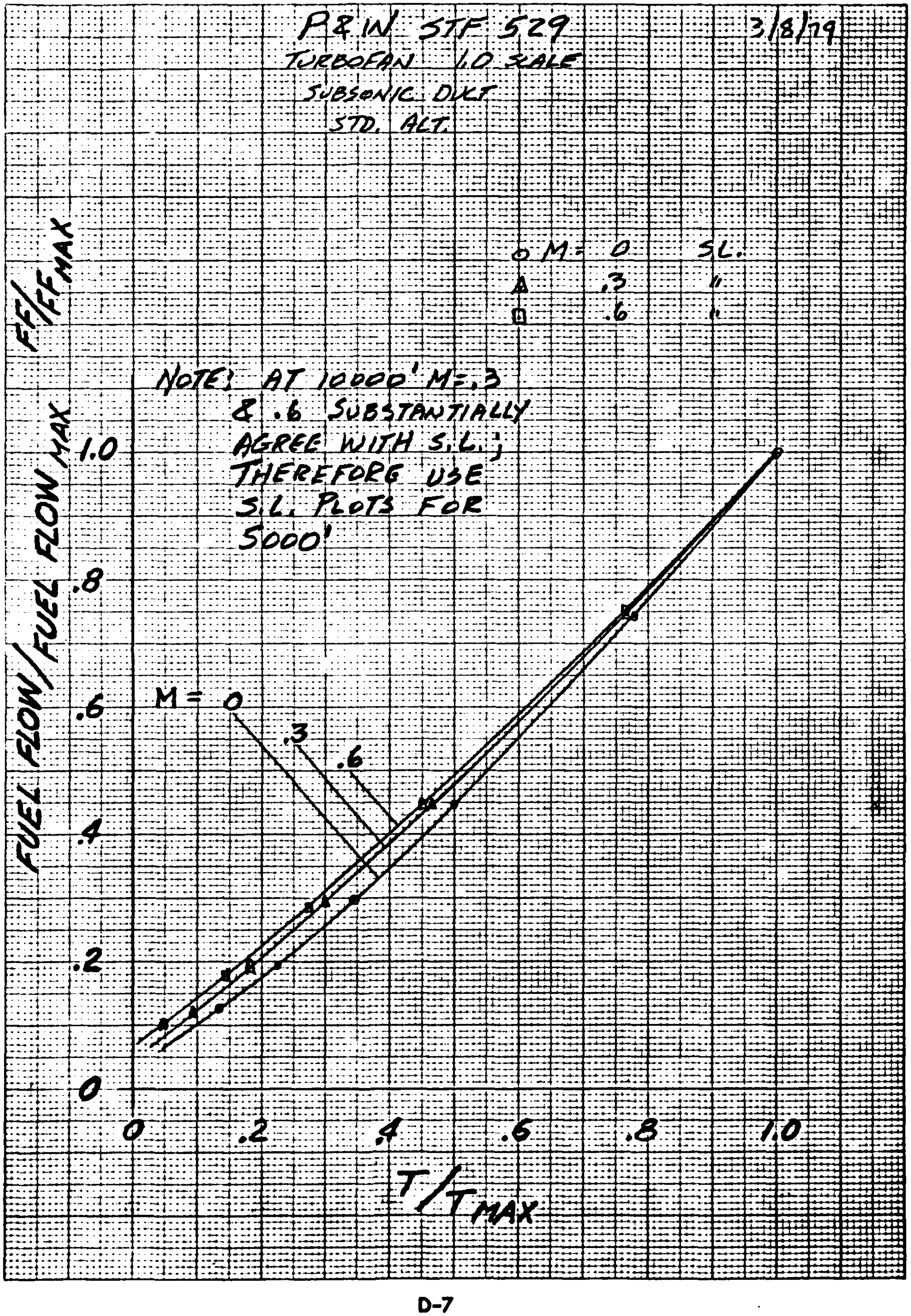

P&W STF 529
3/8/79
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
INSTALLED MAX. THRUST
(PRESSURE RELIEF DOORS FOR TAKEOFF)
TAKEOFF @ S.L.
90°F WITH 6%
THROT. ADVANCE
STD ALT
1000'
0
5
10
15
20
25
30
35
36.089
40
45
THRUST - 1000 LB
14
12
10
8
6
4
2
0
0
.2
.4
.6
.8
1.0
MACH NO.

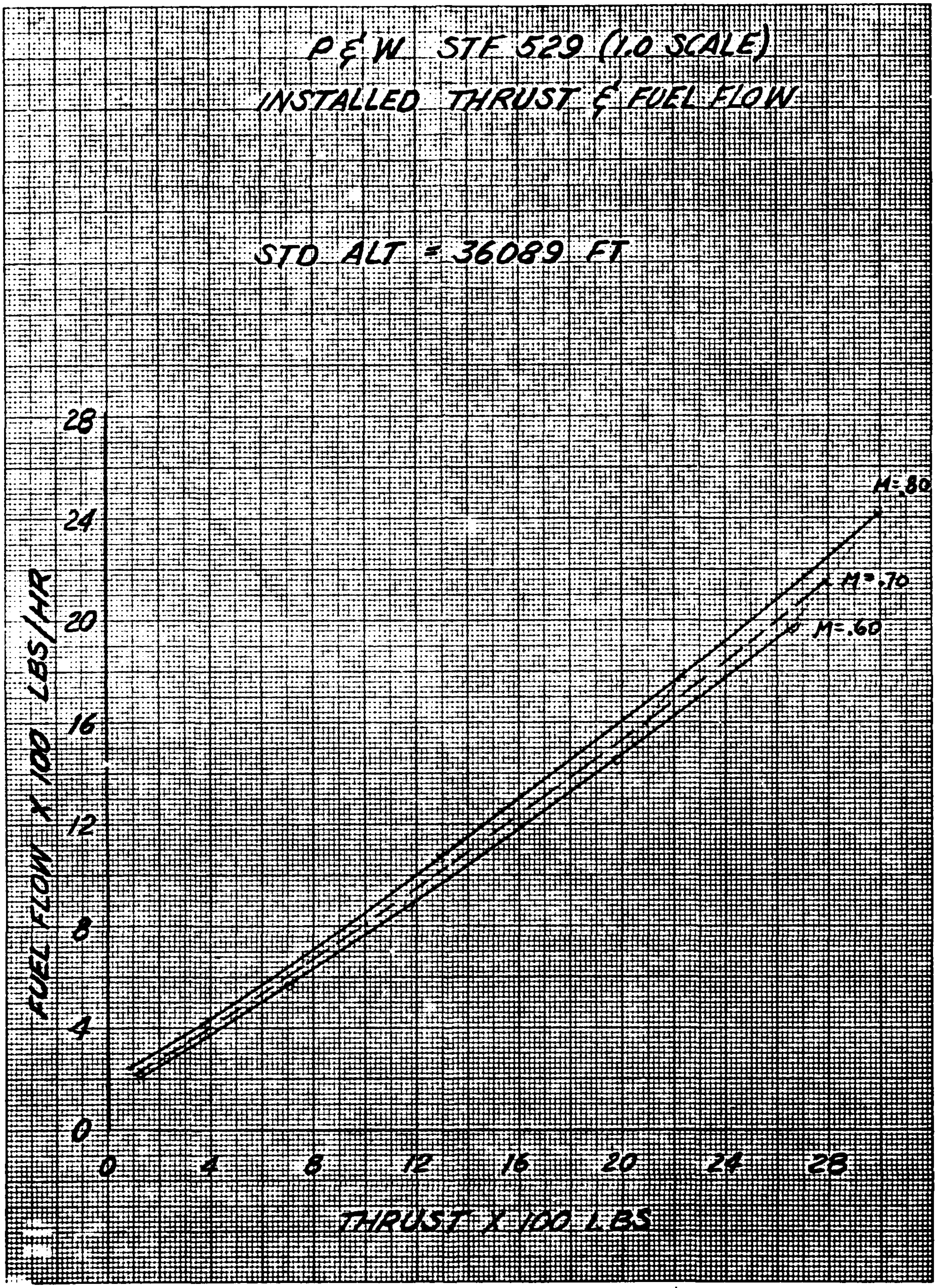


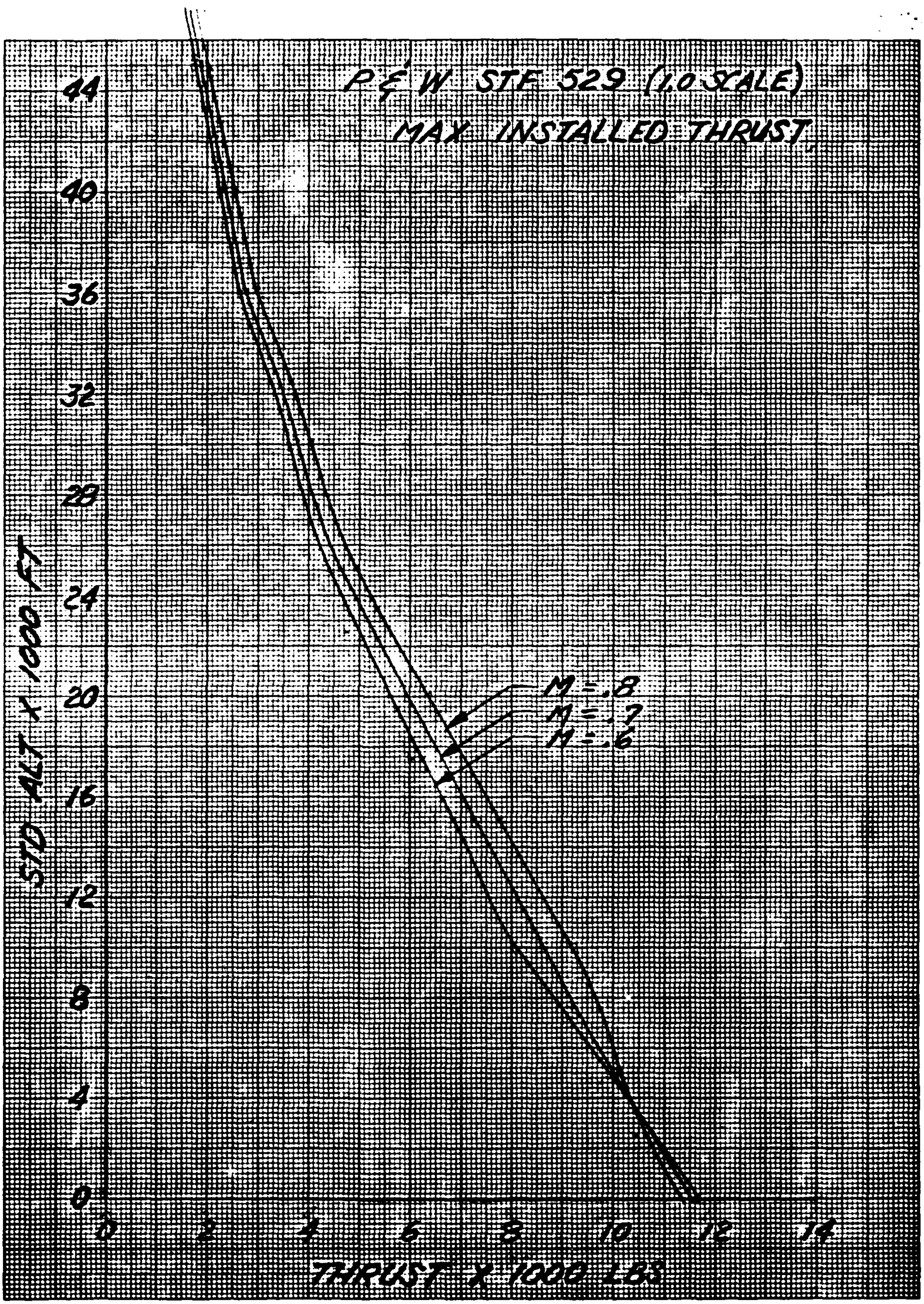
P & W STF 529
3/8/79
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
FUEL FLOW AT MAX THRUST
TAKEOFF @ S.L.
90°F WITH 6% THROT. ADVANCE
STD ALT 1000
0
5
10
15
20
25
30
35
36,089
40
45
FUEL FLOW - 1000 LB/HR
12
10
8
6
4
2
0
0
.2
.4
.6
.8
1.0
MACH NO.

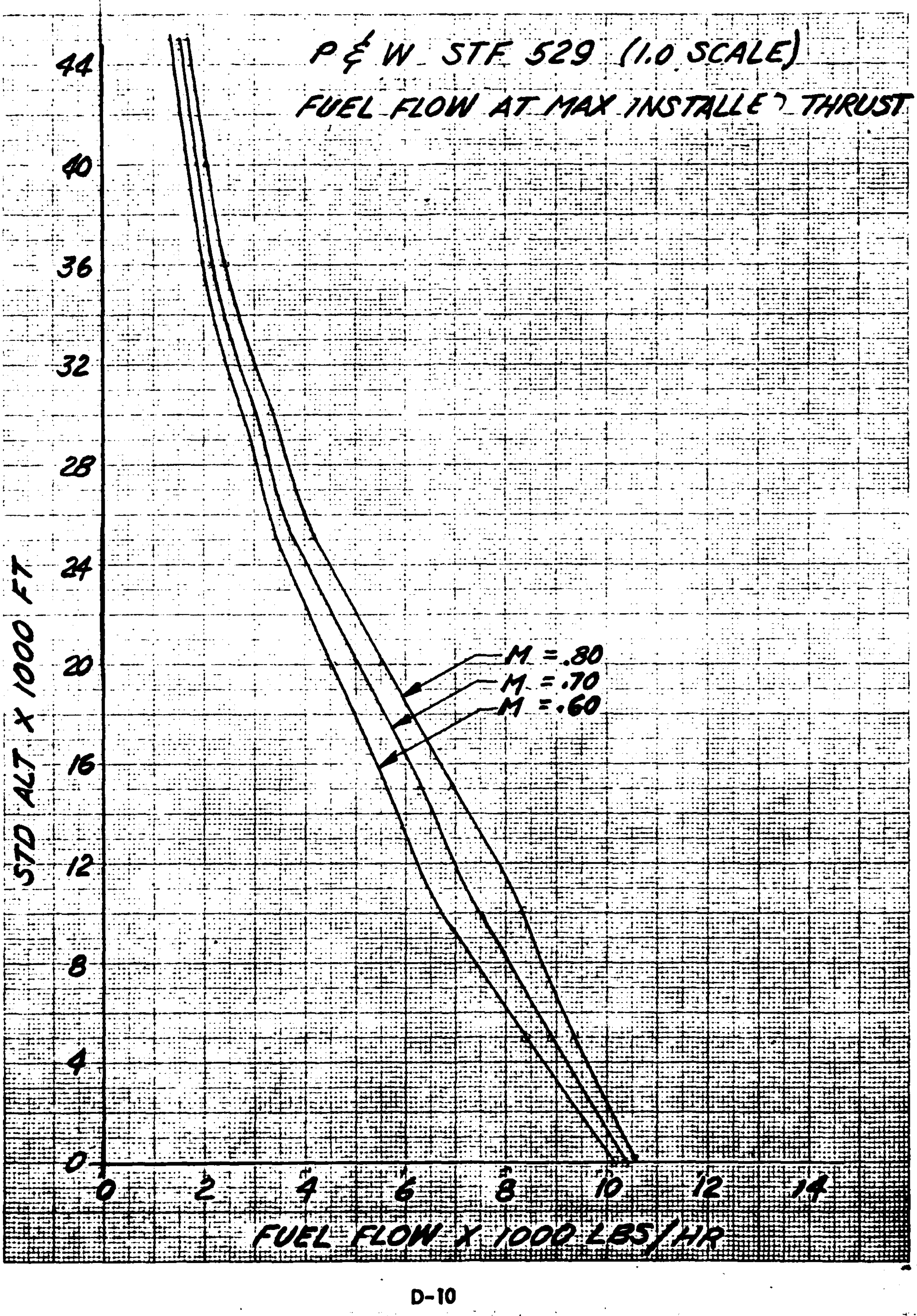

P&W STF 529
3/8/79
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
STD ALT.
36089' STD ALT
INSTALLED SFC - LB/HR/LB
1.20
1.10
1.0
.90
.80
.70
.60
M = 0.8
0.7
0.6
TMAX
0
1
2
3
4
5
INSTALLED THRUST
X 1000 LBS


P&W STF 529
3/8/79
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
STD. ALT.
INSTALLED SFC - LB/HR/LB
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
0
1
2
3
4
5
S.L. M=0.6
5000' M=0.6
S.L. M=0.3
5000' M=0.3
S.L. M=0
INSTALLED THRUST X 1000 LBS

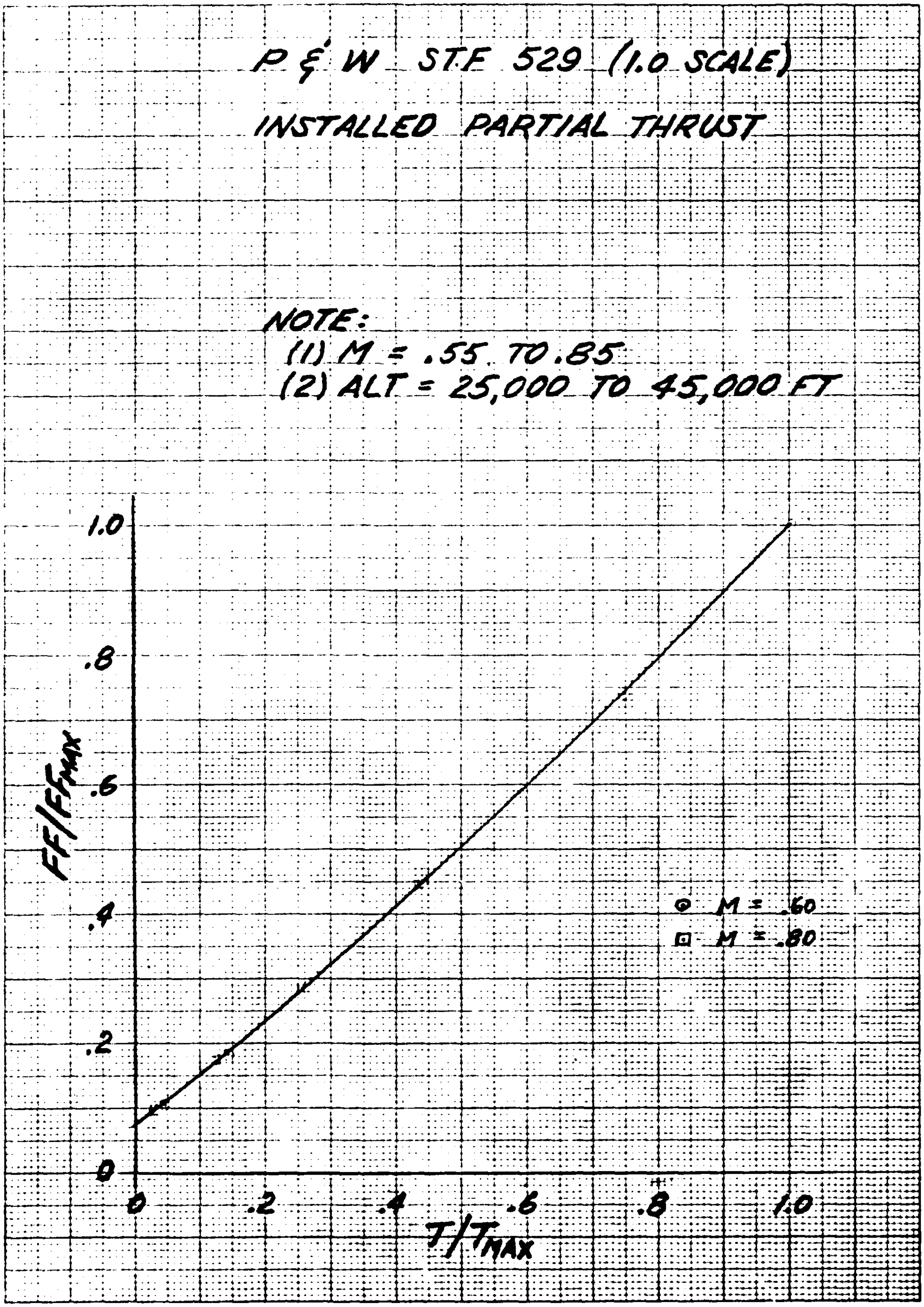

P&W STF 529
3/8/79
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
STD. ALT.
○ M = 0 S.L.
△ .3 "
□ .6 "
NOTE: AT 10000' M=.3
& .6 SUBSTANTIALLY
AGREE WITH S.L.;
THEREFORE USE
S.L. PLOTS FOR
5000'
M = 0
.3
.6
FUEL FLOW/FUEL FLOW MAX FF/FFMAX
0
.2
.4
.6
.8
1.0
T/TMAX

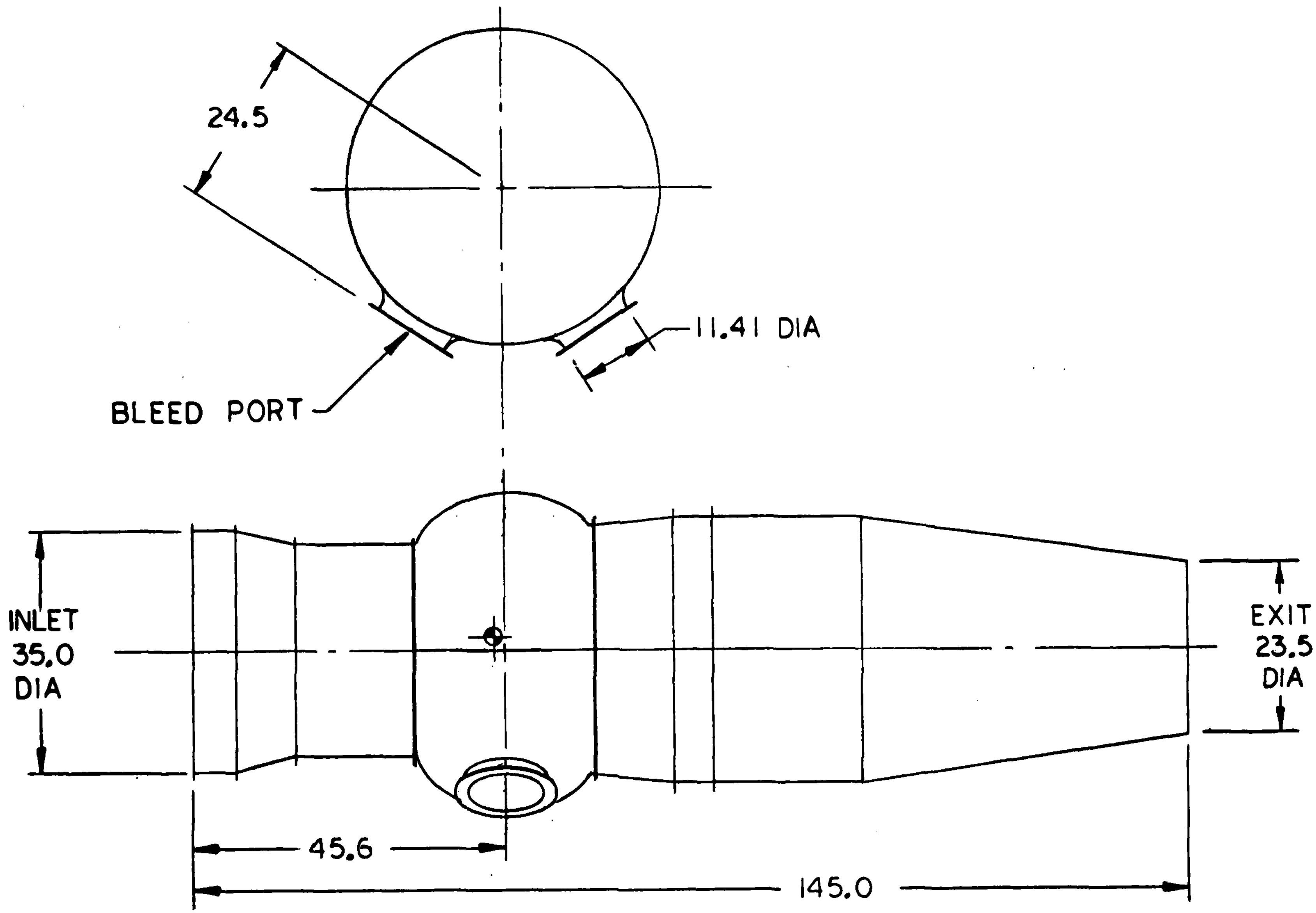

P & W STF 529 (1.0 SCALE)
INSTALLED THRUST & FUEL FLOW
STD ALT = 36089 FT
M=.80
M=.70
M=.60
FUEL FLOW X 100 LBS/HR
0
4
8
12
16
20
24
28
THRUST X 100 LBS


P&W STF 529 (1.0 SCALE)
MAX INSTALLED THRUST
STD ALT X 1000 FT
44
40
36
32
28
24
20
16
12
8
4
0
0
2
4
6
8
10
12
14
THRUST X 1000 LBS
M=.8
M=.7
M=.6



P & W STF 529 (1.0 SCALE)
FUEL FLOW AT MAX INSTALLED THRUST
STD ALT X 1000 FT
0
4
8
12
16
20
24
28
32
36
40
44
M = .80
M = .70
M = .60
0
2
4
6
8
10
12
14
FUEL FLOW X 1000 LBS/HR


P & W STF 529 (1.0 SCALE)
INSTALLED PARTIAL THRUST
NOTE:
(1) M = .55 TO .85
(2) ALT = 25,000 TO 45,000 FT
FF/FFMAX
1.0
.8
.6
.4
.2
0
T/TMAX
.2
.4
.6
.8
1.0
M = .60
M = .80



ADV STUDY
P & W STF-529
TURBOFAN ENGINE

APPENDIX E

PRELIMINARY WORK

The preliminary work was started well before candidate engine information was available. Parametric engine data, that were fairly representative, were used to do tradeoff studies for two engine vs one engine and APU vs engine bleed for trunk pressurization. As a result, a one engine configuration was selected with engine fan bleed used to pressurize the trunk. This work is summarized in the following Section (1).

General Electric candidate engine data became available late in February, and the selected aircraft configuration was reworked around a scale 0.914 G.E. F101/F15-A1 Turbofan engine which was required to meet 3000 foot takeoff ground run at S.L., 89.8°F. The fan bleed used at this time was about 45% greater than used for the later, final design which was developed after considerably more analysis and study of the operation of the SETOLS. Obviously, the higher bleed is adverse to engine size requirement; however, the Pratt & Whitney STF 529 turbofan engine data became available early in March and, after inspection and comparison of the data, it was decided to use the P&W engine in lieu of the G.E. engine. Therefore, the G.E. engine configuration was not reworked for update to the lower bleed requirement. The work with the G.E. engine is summarized in the following Section (2).

Incorporation of the P&W STF 529 turbofan required only one iteration to arrive at the final design with 24,300 pound gross weight and 280 square foot wing area. The final work is covered in detail in the report.

(1) Preliminary Work with the Parametric Engine Data

(a) The parametric engine data are given on the following pages and were used pending receipt of data requested of G.E. and P&W.

Reference G.E. Report R72AEG206, June 1972, Pre Study Data, GE16/F4 Study A1 Turbofan, for data level and variation.

Scale	1.0
Rated T_{Max} (S.L. Std Temp)	18360 lb (No AB)
D_{Inlet}	44 inches
L_E (without nozzle)	$1.84 \times D_{Inlet}$
L_{Noz}	$0.8 \times D_{Inlet}$
W_{Eng} (including nozzle)	2623 lb

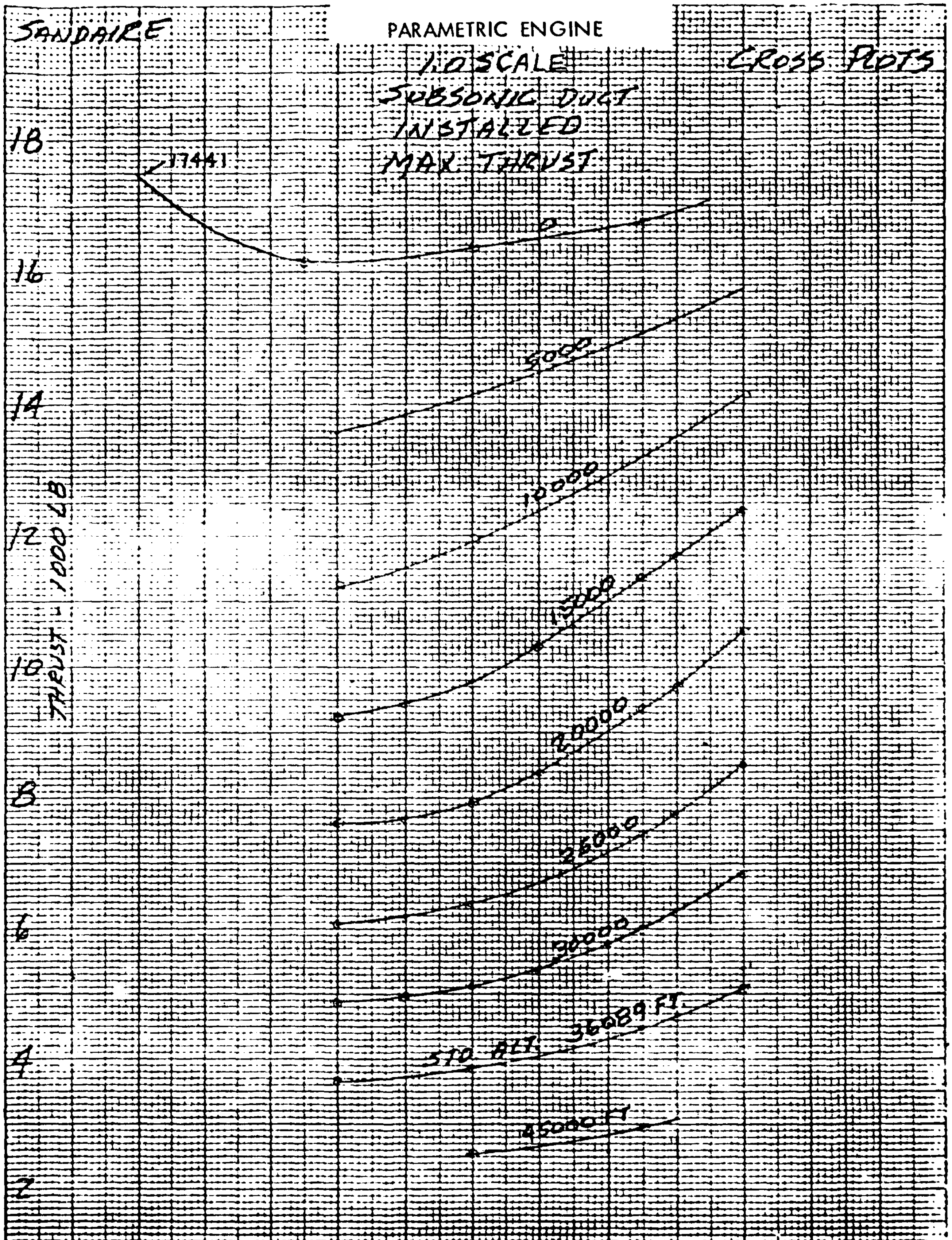
Scaling

$$D_{Inlet} = \text{Linear from 44 inches at } \left[\frac{T_{Scale}}{T_{Basic}}\right] = 1.0 \text{ to } 34.5 \text{ inches}$$

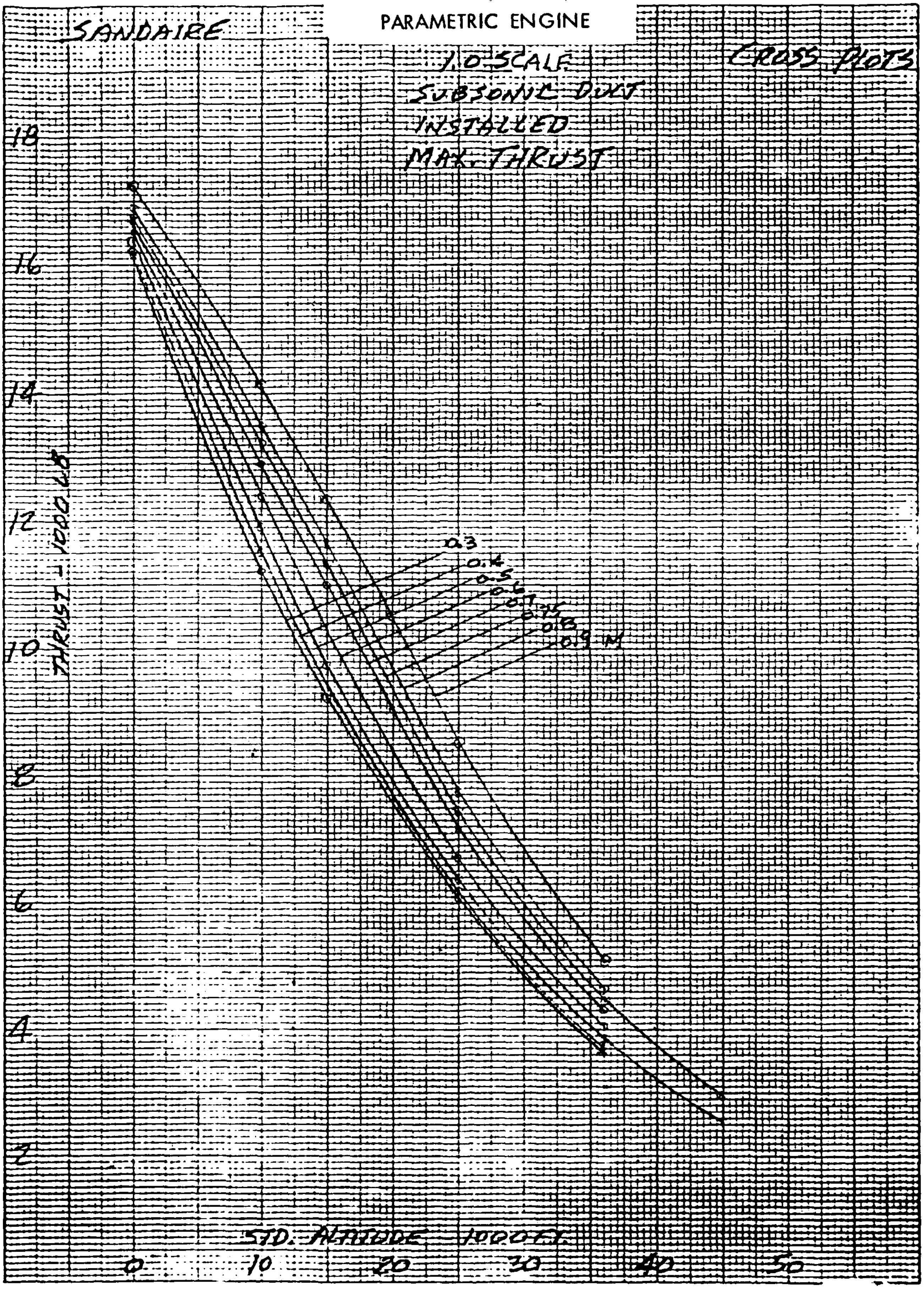
$$\text{at } \left[\frac{T_{Scale}}{T_{Basic}}\right] = 0.5$$

W_{Eng} based on $T/W = 7$

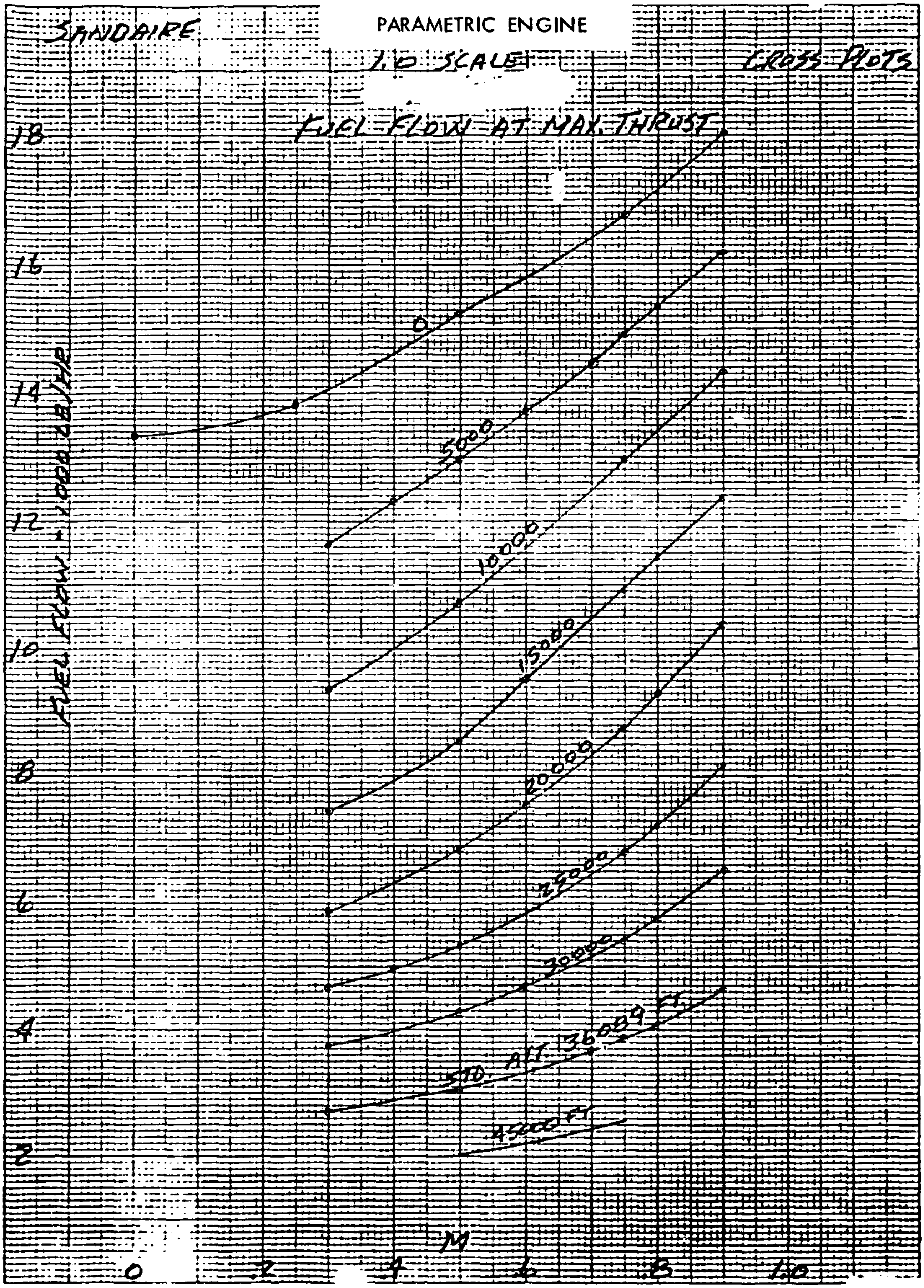
An installation loss factor of 0.95 was applied to the thrust and an advanced technology factor of 0.90 was applied to the fuel flow of the above-referenced G.E. report. Pressure relief doors are assumed for takeoff.

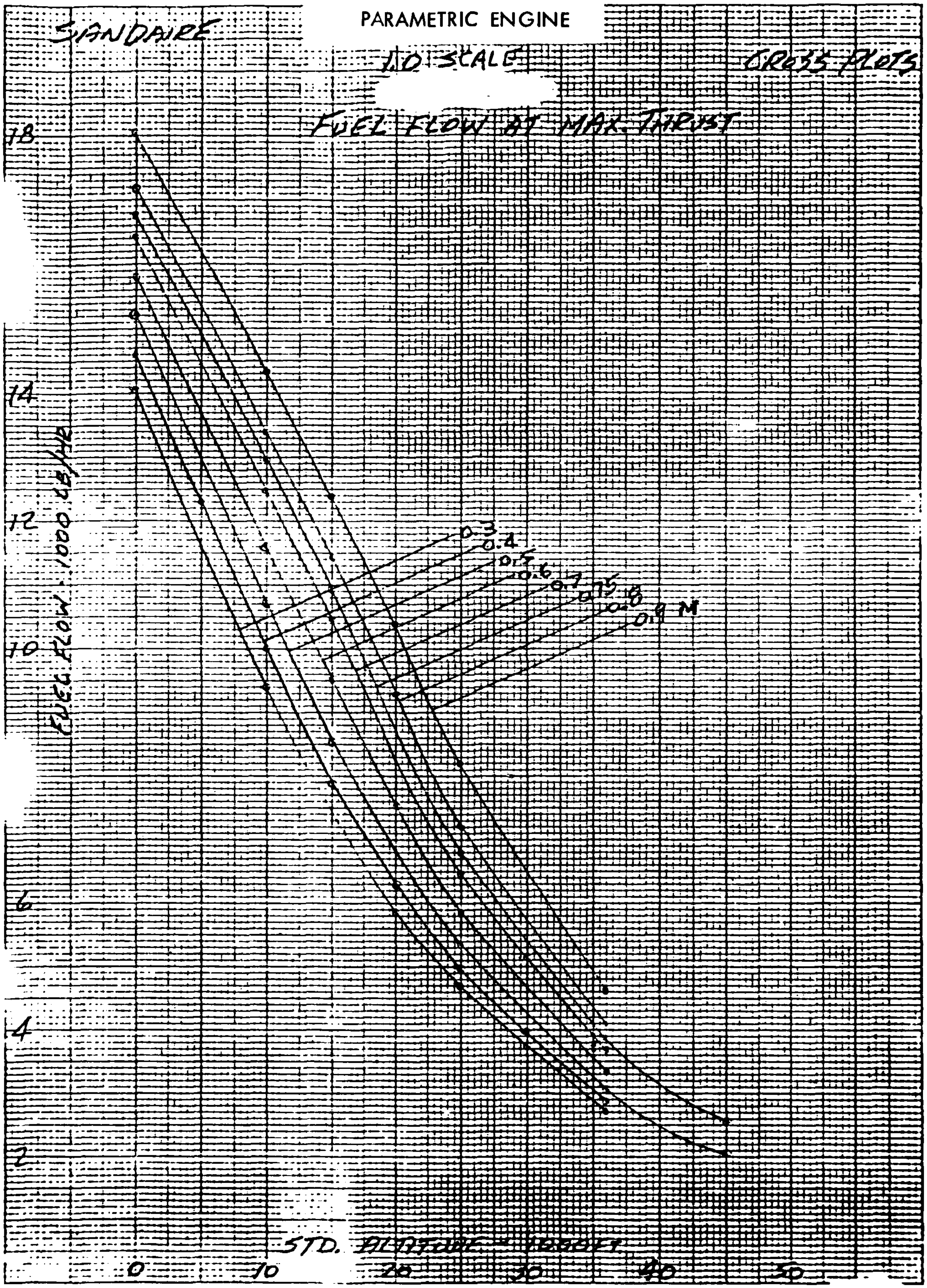


SANDAIRE
PARAMETRIC ENGINE
1.0 SCALE
SUBSONIC DUCT
INSTALLED
MAX THRUST
CROSS PLOTS
17441
5000
10000
15000
20000
25000
30000
STD ALT 36089 FT
45000 FT
THRUST - 1000 LB
18
16
14
12
10
8
6
4
2
M
0
.2
.4
.6
.8
1.0

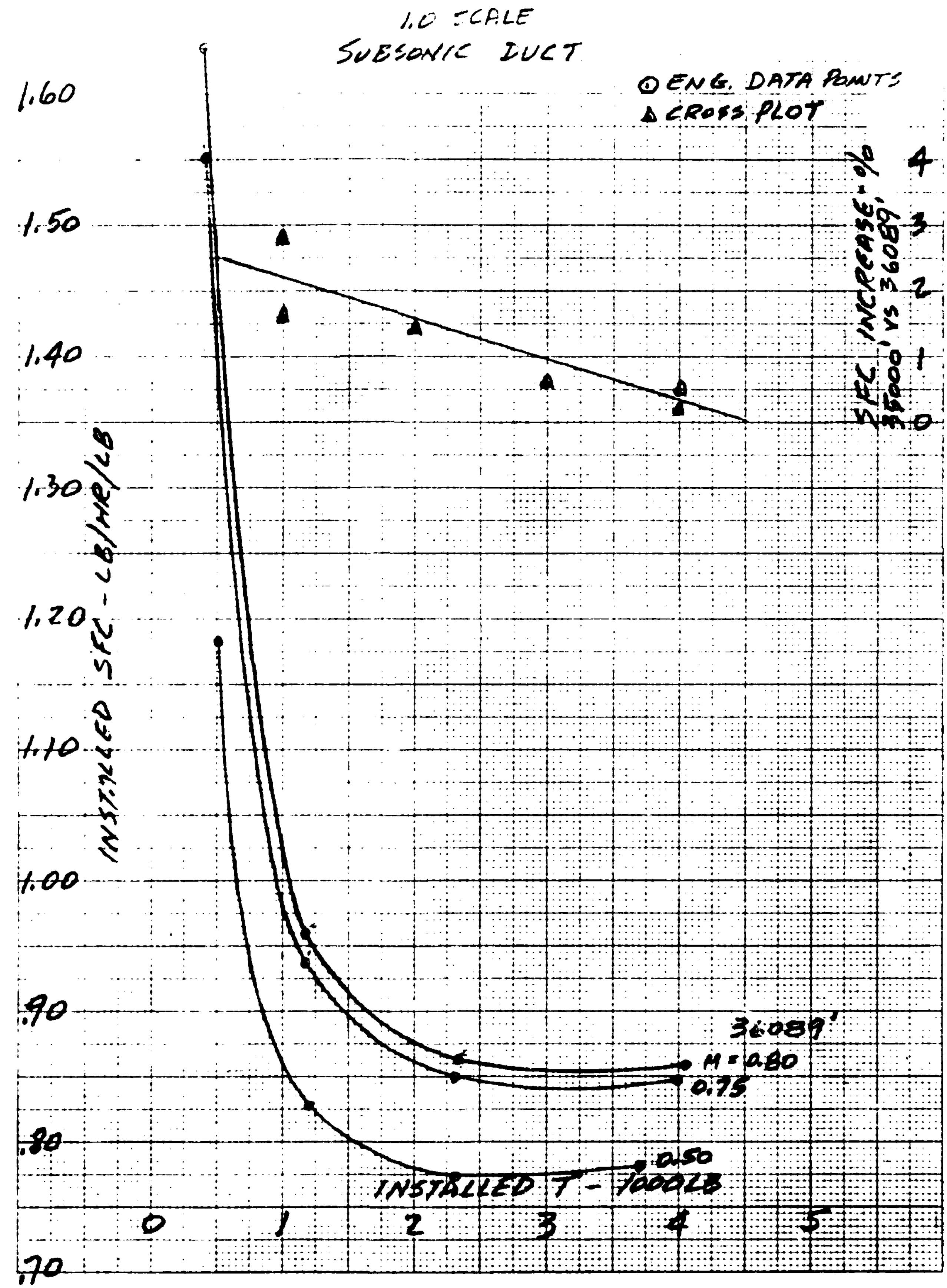

SANDAIRE
CROSS PLOTS
1.0 SCALE
SUBSONIC DUCT
INSTALLED
MAX. THRUST
THRUST - 1000 LB
0.3
0.4
0.5
0.6
0.7
0.75
0.8
0.9 M
STD. ALTITUDE - 1000 FT
18
16
14
12
10
8
6
4
2
0
10
20
30
40
50

PARAMETRIC ENGINE





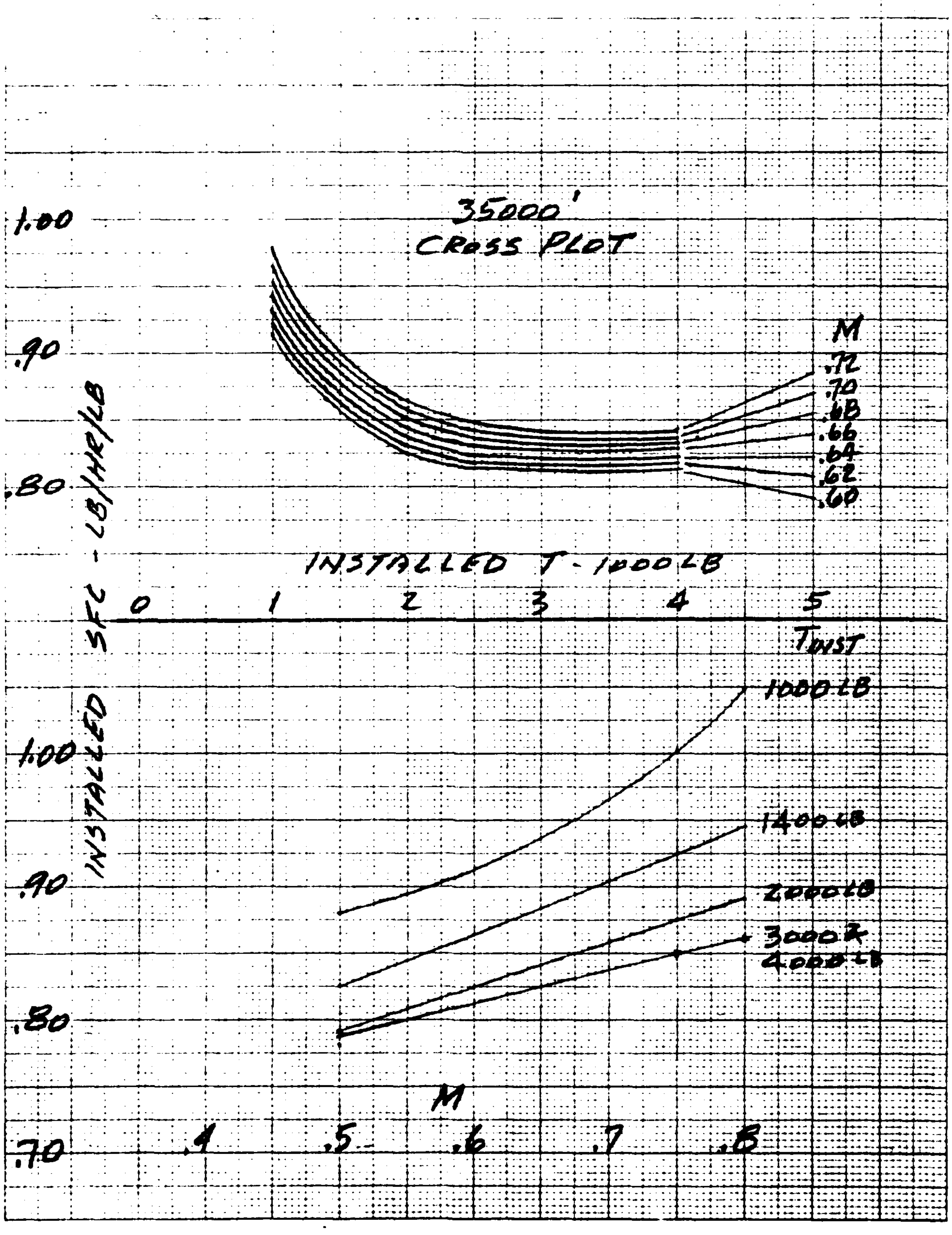
PARAMETRIC ENGINE
SANDAIRE
1.0 SCALE
CROSS PLOTS
FUEL FLOW AT MAX THRUST
FUEL FLOW 1000 LB/HR
STD. ALTITUDE - 1000 FT
0.3
0.4
0.5
0.6
0.7
0.75
0.8
0.9 M
18
14
12
10
6
4
2
0
10
20
30
40
50


1.0 SCALE
SUBSONIC DUCT
ENG. DATA POINTS
CROSS PLOT
1.60
1.50
1.40
1.30
1.20
1.10
1.00
.90
.80
.70
INSTALLED SFC - LB/HR/LB
SFC INCREASE - %
35000' VS 36089'
4
3
2
1
0
36089'
M = 0.80
0.75
0.50
INSTALLED T - 1000 LB
0
1
2
3
4
5

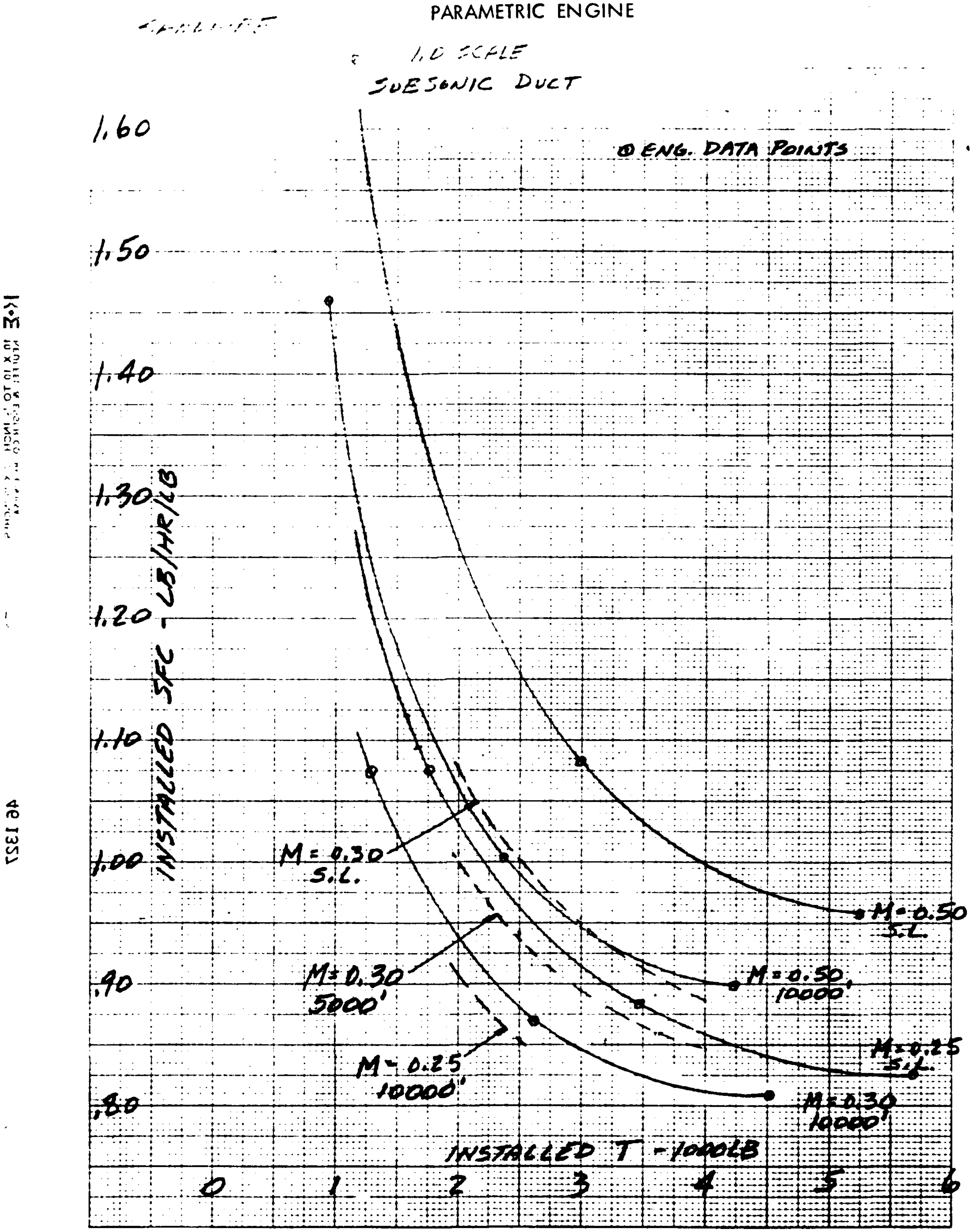
PARAMETRIC ENGINE

1.0 SCALE

SUBSONIC DUCT



PARAMETRIC ENGINE



(b) Calculation methods and bases are similar to the final report, and any differences are small enough so that conclusions reached are reliable.

(c) One Engine Configuration, fan bleed to pressurize trunk, parametric engine, gull wing. (Ref: Tech. Clar. Memo to NADC 12/19/78)

Two Iterations were made in arriving at the following

Design G.W.	29,500 lb
Rated T_{Max} (Std, SL) Scale 1.0	18,360 lb
Wing area (excludes chord extensions at gull)	350 sq ft
Cushion PSI	1.25
Trunk PSI	2.5
Trunk ℄ Perimeter	47.1 ft
Daylight Gap (Ave)	0.75 ins
Trunk ℄ Area	164 sq ft
Bleed required	87.7 lb/sec
Thrust loss due to bleed	4500 lb
Takeoff Ground Run S.L., 89.8°F, μ = 0.10, 4.5 min at T_{Max} prior T.O.	2830 ft
V_{To}	222 ft/sec
Takeoff over 50 Ft. obstacle	4000 ft
Wing AR	6
λ	0.3
b	45.8 ft
C_{Root}	11.8 ft
$\Lambda_{c/4}$	25°
$\bar{c}$	8.4 ft
t/c	.14-.12
Airfoil	Advanced
$\Delta M_{Drag\ Rise}$	.08 for advanced airfoil vs NASA low drag series airfoil
a.c. Wing	.26 $\bar{c}$
C_{L_α} Wing	.076/deg
Design a.c. Aircraft	0.35 $\bar{c}$
Design Aft C.G.	0.30 $\bar{c}$
C_R/L_{Fuse} = 11.8/40.6 =	.29

(c) (continued)

$(\text{Nose} - LEC_R)/L_{Fuse} = 15/40.6 =$	.37
a.c. Horiz Tail Off	$0.21\,\bar{c}$
q/q_{Tail}	0.95
Downwash Factor	0.55

<u>Horizontal Tail</u>

AR	2.6
λ	0.46
b	15.7 ft
C_R	8.3 ft
$\Lambda_{c/4}$	35°
t/c	.12-.10
C_{L_α}	.050/deg
ℓ_H $(\bar{c}/4_W - \bar{c}/4_H)$	14.5 ft
S_H	95 sq ft

<u>Vertical Tail</u>

C_{N_β} Design	.0010
AR	1.40
λ	0.42
b	9.7 ft
C_R	9.7 ft
$\Lambda_{c/4}$	35°
t/c	.12-.10
$(\text{Nose} - \bar{c}/4)/L_{Fuse.} = 22.3/40.6 =$	.55
C_{N_β} Tail Off	-.0023
ℓ_V $(\bar{c}/4_W - \bar{c}/4_V)$	15.5 ft
C_{L_α}	.054/deg
S_V	67 sq ft

<u>Fuselage</u>

Effective Depth	6.3 ft
Effective Width	4.8 ft
Effective Length	40.6 ft

(c) (continued)

Drag Aircraft - Low Speed

$C_{D_{Min}}$ = .0213

C_L	$C_D - C_L^2/6\pi$
.01	.0213
.1	.0212
.2	.0212
.3	.0214
.4	.0217
.5	.0223
.6	.0235
.7	.0260
.8	.0307

Drag Aircraft - With Mach No.
Advanced Airfoil $\Delta M = + .08$

C_L =	0	.2	.4	.6	.8
M	ΔC_{D_M}	M for same ΔC_{D_M}			
.74	0	.716	.692	.667	.643
.78	.0005	.756	.732	.707	.683
.82	.0015	.796	.772	.747	.723
.84	.0025	.816	.792	.767	.743
.86	.0039	.836	.812	.787	.763
.88	.0069	.856	.832	.807	.783
.90	.0136	.876	.852	.827	.803
.91	.0188	.886	.862	.837	.813
.92	.0260	.896	.872	.847	.823

Drag Stores

	12-MK-82 & 4-TERs	4 TERs Alone
ΔC_D at M = 0.60	.0115	.0019
0.70	.0115	.0019
0.80	.0132	.0022
0.88	.0174	.0029

SANDAIRE

(c) (continued)

CAS Mission

	Distance, N.M.	Fuel, Pounds
5 minutes at T_{Max} at S.L.		1110
Climb to 35000 feet, M = 0.65	36	682
Cruise, M = 0.7	124	691
Descent to 5000 feet		
Loiter 60 minutes, M = 0.3		2229
Drop MK-82, Retain TERs		
Climb to 35000 feet, M = 0.65	16	289
Cruise, M = 0.64	144	507
Descend to S.L.		
Loiter 10 minutes, M = 0.25		256
Reserve 5% initial fuel		303
Total	320	6067

RAD = 160 NM

Weight

	Pounds
Wing	2943
Wing Extension for SETOLS	921
Horizontal Tail	404
Vertical Tail	300
Fuselage, including duct structure and speed brakes	1587
Canopy	409
Engine	2623
Bleed	127
Tail Pipe Extension	171
Engine Section	248
Inlet ducts	275
Engine cont, start, lub, oil	160
Flight controls	636
Wing Integral Tanks	133
Unusable Fuel	93
Fuel System, including Fuel Dump & Aerial Refueling Provisions	268
Systems including ejection seat	2041
Specified Load	8574
SETOLS	1008
Protection	454
CAS Mission Fuel	6067
Gross Weight	29442

(d) Two-Engine Configuration
With APU to pressurize trunk, parametric engine, gull wing
(Ref: Tech. Clar. Memo to NADC 12/19/78)

First Iteration was with GW = 24000 pounds, Two 8000-pound thrust engines, and wing area = 285 sq ft, which was too low a gross weight.

Second Iteration was as follows:

Design G.W.	27000 lb
Rated T_{Max} (Std, SL)	
Scale 0.490	9000 lb
Wing Area (excludes chord estensions for SETOLS at gull)	320 sq ft
Cushion PSI	1.25
Trunk PSI	2.5
Trunk ℄ Perimeter	45 ft
Daylight Gap (Ave)	0.75 ins
Trunk ℄ area	150 sq ft
Trunk air required (from APU)	83.8 lb/sec

Takeoff is based on the failure of one engine at the point in the takeoff run where the distance to "fail and go" equals the distance to "fail and stop". Fuel for 4.5 minutes at T_{Max} is consumed prior to takeoff.

For S.L. 89.8°F, V_{TO} = 222 ft/sec

For two-engine acceleration (μ = 0.10)

V/V_T	0	.2	.4	.6	.8	1.0
ΔS_G(ft)	0	65	198	340	494	660

For one engine acceleration, rudder will trim the asymmetric thrust down to just over 0.6 V_{TO}. Below this speed it is assumed that asymmetric braking will be used. Braking μ = 0.22 is required at 0.2 V_{TO}, 0.16 at 0.4 V_{TO} and .03 at 0.6 V_{TO}. For takeoff on water, equivalent asymmetric braking is assumed. Taking into account the deflected rudder drag and the asymmetric brake-retarding force

V/V_{TO}	0	.2	.4	.6	.8	1.0
ΔS_G (ft)	0	398	997	1229	1499	2089

(d) (continued)

For deceleration after engine failure, reverse thrust is assumed = $0.5\ T_{Max}/Eng$, and braking $\mu = 0.30$. For this case, rudder will trim the asymmetric reverse thrust down to just over $0.4\ V_{TO}$. Asymmetric braking required is $\mu = 0.10$ at $0.2\ V_{TO}$ and .03 at $0.4\ V_{TO}$. Taking into account the rudder drag and the reduced μ_{Ave} for asymmetric braking

V/V_{TO}	0	.2	.4	.6	.8	1.0
μ Ave	.25	.285	.30	.30	.30	
ΔS_G (ft)	78	227	380	574	839	0

Allowing two seconds reaction time at the engine failure point (V_1)

V_1/V_{TO}	.4	.6	.8
S_G 2-engine	263	603	1097
2 seconds at V_1	178	266	355
For "Fail and Go" S_G 1-engine (Go)	4817	3588	2089

In rotation for lift off, the trunk center of pressure will shift aft, thus decreasing the vertical tail arm by about 50%. The rudder deflection before rotation is calculated to be 11.5^{o}; it will have to be increased to 22.3^{o} causing 270 lb increased drag. This will increase

ΔS_G (.8 to 1.0 V/V_{TO}) 109 ft

Using 50% of this adds 55 feet to each of the above one engine acceleration distances to give

S_G 1-engine (Go corrected)	4872	3643	2144
Total S_G "Fail and Go"	5313	4512	3596

(d) (continued)

For "Fail & Stop"

V_I/V_{TO}	.4	.6	.8
S_G 2-engine	263	603	1097
2 sec at V_I	178	266	355
For "Fail & Stop" S_G 1-eng (Stop)	305	685	1259
Total S_G "Fail & Stop"	746	1554	2711

Plotting $S_{G_{\text{Fail \& Go}}}$ and $S_{G_{\text{Fail \& Stop}}}$

vs V_I/V_{TO} shows an intersection at

$V_I = .88\ V_{TO}$ and a distance of 3220 ft

(S.L. 89.8°F)

If the reaction time at V_I is reduced to 1.0 sec, $V_I = .88\ V_{TO}$ and the distance is 3020 ft (S.L. 89.8°F)

Therefore, this iteration is close to the desired takeoff run, and the calculation is continued.

Using the same wing and tail shapes, as used for the one-engine configuration above, and the same fuselage except for the required increase in width for two engines

$S = 320$ sq ft
$S_H = 87$ sq ft
$S_V = 64$ sq ft

<u>Drag Aircraft - Low Speed</u>

$C_{D_{Min}} = .0226$

Use drag coefficient for the one engine configuration above plus $\Delta C_D = .0013$; correct stores ΔC_D for wing area change.

(d) (continued)

CAS Mission

	Distance, N.M.	Fuel, Pounds
5 minutes at T_{Max} at S.L.		1089
Climb to 35000 feet, M = 0.65	33	618
Cruise, M = .68	127	679
Descend to 5000 feet		
Loiter 60 minutes, M = 0.3		2128
Drop MK 82, Retain TERs		
Climb to 35000 feet, M = 0.65	14	249
Cruise M = 0.62	146	483
Descend to S.L.		
Loiter 10 minutes, M = 0.25		242
Reserve 5% initial fuel		289
Total	320	5777

Rad = 160 N.M.

Weight

	Pounds
Wing	2581
Wing Extension for SETOLS	790
Horizontal Tail	354
Vertical Tail	280
Fuselage, including duct structure and speed brakes	1710
Canopy	400
Engines	2572
Tail Pipe Extensions	367
Engine Section	231
Inlet Ducts	272
Engine, Cont, Start, Lub, Oil	149
Flight Controls	600
Wing Integral Tanks	120
Unusable Fuel	80
Fuel System including Fuel Dump & Aerial Refueling Provisions	255
Systems including Ejection Seat (No APU in Systems Weight)	1981
APU (Special for SETOLS & also used for self-starting)	545
Specified Load	8574
SETOLS	961
Protection	451
CAS Mission Fuel	5777
Gross Weight	29050

E-17

(d) (continued)

The assumed G.W. for this iteration is 27000 pounds; therefore, the engine size must increase to retain takeoff distance and G.W. must increase above 29050 pounds which will result in the two-engine configuration being considerably heavier than the one-engine configuration; therefore it was decided to do no more work on the two-engine configuration.

(e) One Engine Configuration
With APU to pressurize trunk, parametric engine, gull wing.
(Ref: Tech. Clar. Memo to NADC 12/19/78)

Sufficient work had been accomplished at this point so that a fairly accurate choice of gross weight could be made as follows:

Design G.W.	27500 lb
Rated T_{Max} (Std, SL)	
Scale 0.708	13000 lb
Wing Area (Excludes chord extensions for SETOLS at gull)	326 sq ft
Cushion PSI	1.25
Trunk PSI	2.5
Trunk ℄ Perimeter	45.5 ft
Daylight Gap (Ave.)	0.75 ins
Trunk ℄ area	153 sq ft
Trunk air required (from APU)	84.7 lb/sec
Takeoff Ground Run	
S.L. 89.8°F, $\mu = 0.10$, 4.5 min at T_{Max} prior T.O.	2836 ft
V_{To}	222 ft/sec

Using the same wing and tail shapes, as used for the one-engine-configuration-with-bleed above, and the same fuselage except the duct structure is smaller for the 0.708 scale engine

$$S = 326 \text{ sq ft}$$
$$S_H = 89 \text{ sq ft}$$
$$S_V = 65 \text{ sq ft}$$

Drag Aircraft - Low Speed

$$C_{D_{Min}} = .0217$$

Use drag coefficient for the one-engine-configuration-with-bleed above plus $\Delta C_D = .0004$; correct stores ΔC_D for wing area change.

(e) (continued)

CAS Mission

	Distance N.M.	Fuel Pounds
5 minutes at T_{Max} at S.L.		786
Climb to 35000 feet, M = 0.65	73	948
Cruise, M = 0.7	87	463
Descend to 5000 feet		
Loiter 60 minutes, M = 0.3		1980
Drop MK 82, Retain TERs		
Climb to 35000 feet, M = 0.65	24	313
Cruise, M = 0.62	136	430
Descend to S.L.		
Loiter 10 minutes, M = 0.25		216
Reserve 5% initial fuel		270
Total	320	5406

RAD = 160 NM

Weight

	Pounds
Wing	2655
Wing Extension for SETOLS	818
Horizontal Tail	366
Vertical Tail	287
Fuselage, including duct structure and speed brakes	1546
Canopy	403
Engine	1857
Tail Pipe Extension	181
Engine Section	167
Inlet Ducts	245
Engine Cont, Start, Lub, Oil	108
Flight controls	610
Wing Integral Tanks	122
Unusable Fuel	82
Fuel System, including Fuel Dump and Aerial Refueling Provisions	239
Systems, including Ejection Seat (No APU in Systems Weight)	1981
APU (Special for SETOLS & also used for self-starting)	552
Specified Load	8574
SETOLS	973
Protection	449
CAS Mission Fuel	5406
Gross Weight	27621

E-19

(e) (continued)

This configuration does show an advantage over the above one-engine-with-bleed configuration of about 6% less gross weight; however, the big unknown is the weight, size, cost and availability of an APU of the size needed, one that will deliver 84.7 lb/sec to pressurize the trunk. This is essentially a small aircraft engine. It must have the same reliability as the primary engine but can have shorter life due to minimum usage.

The weight used for the APU installation is based on producing the required airflow with 65% of the weight the primary engine uses to do the job. If the primary engine delivered a fan bleed of 84.7 lb/sec, it is equivalent to loss in thrust of

$$(84.7/87.7) \times 4500 = 4346 \text{ lb}$$

For a T/W = 18360/2623 = 7.0 for the primary engine (Page E-2), the engine weight assignable to the bleed is 4346/7 = 621 lb (compared to 404 lb used in the above APU estimate for the uninstalled APU). The weight of 552 lb shown includes 148 lb for installation, controls, air intake, exhaust, firewall, etc. If the APU in an actual application required another 300 lb, which would pyramid to about 700 lb in gross weight if the design was completely recycled, the advantage decreases to 4% in gross weight. Therefore, it was decided to use engine bleed and work on reducing the amount of bleed required.

Further study of available SETOLS technology and tests showed that the daylight gap (trunk to ground clearance in the takeoff and landing run) was more nearly 0.25 inch than the 0.75 inch used. This would immediately reduce the required airflow by 67%. Concurrently, it was decided that additional trunk nozzles were needed to help provide air lubrication outboard of the ground tangent line, and they would normally exhaust to the atmosphere. Using 20% of the total nozzle area for this purpose increases the flow by 35%. This distribution was used on the final trunk configuration. The net summation results in a reduction of thrust loss due to bleed from 24.5 to 18.3% and is considered a simpler and more reliable design than coping with the unknown and costly development of an APU of the size required.

(2) Preliminary Work with the General Electric F101/F15A1 Turbofan Engine

(a) At this point in the design progress, the first candidate engine information become available, and data are shown by the following seven pages.

(a) (continued)

G.E. Engine Data
Ref. G.E. Report R77AEG631, dtd 12/1/77
for GE F101/F15 Al Turbofan, Scale 1.0

The G.E. Report listed 3155 lb for the basic engine (less nozzle) and 76.85 inches length (less nozzle). Length and weight shown on the next page, for the engine with nozzle, are estimated.

Rated T_{Max}	=	18217 lb
Airflow	=	351.7 lb/sec
Bypass Ratio	=	1.86
Fan Pressure Ratio	=	2.37
Airflow/Rated T_{Max}	=	.0193
T/W	=	5.77 (without nozzle, without bleed)

Engine performance, with installation factor of 0.95 applied to thrust, is shown on the following pages.

All data are for a scale = 1.0 engine. Scaling factors, for $\pm$ 30% rated T_{Max}, are

Airflow varies as T_{Max}

Engine Dimensions vary as $(T_{Max})^{0.5}$

Engine Weight (including nozzle) varies as T_{Max} (T/W is constant). This will not apply outside the $\pm$ 30% scaling.

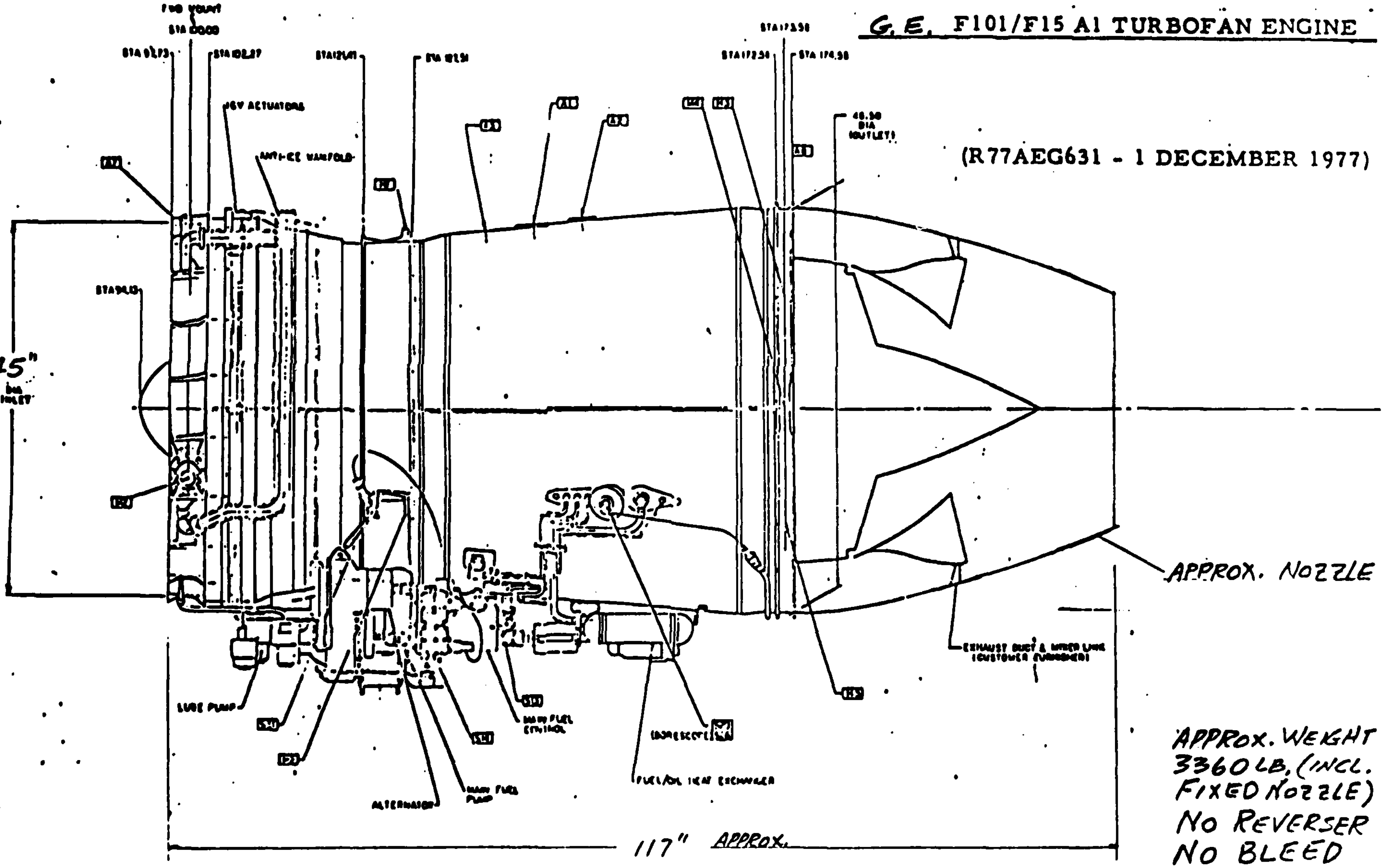
(S.F.C. vs T/T_{Max}) does not change with scale.

Rated T_{Max} loss due to fan bleed = (Bleed Airflow - lb/sec)/.0193

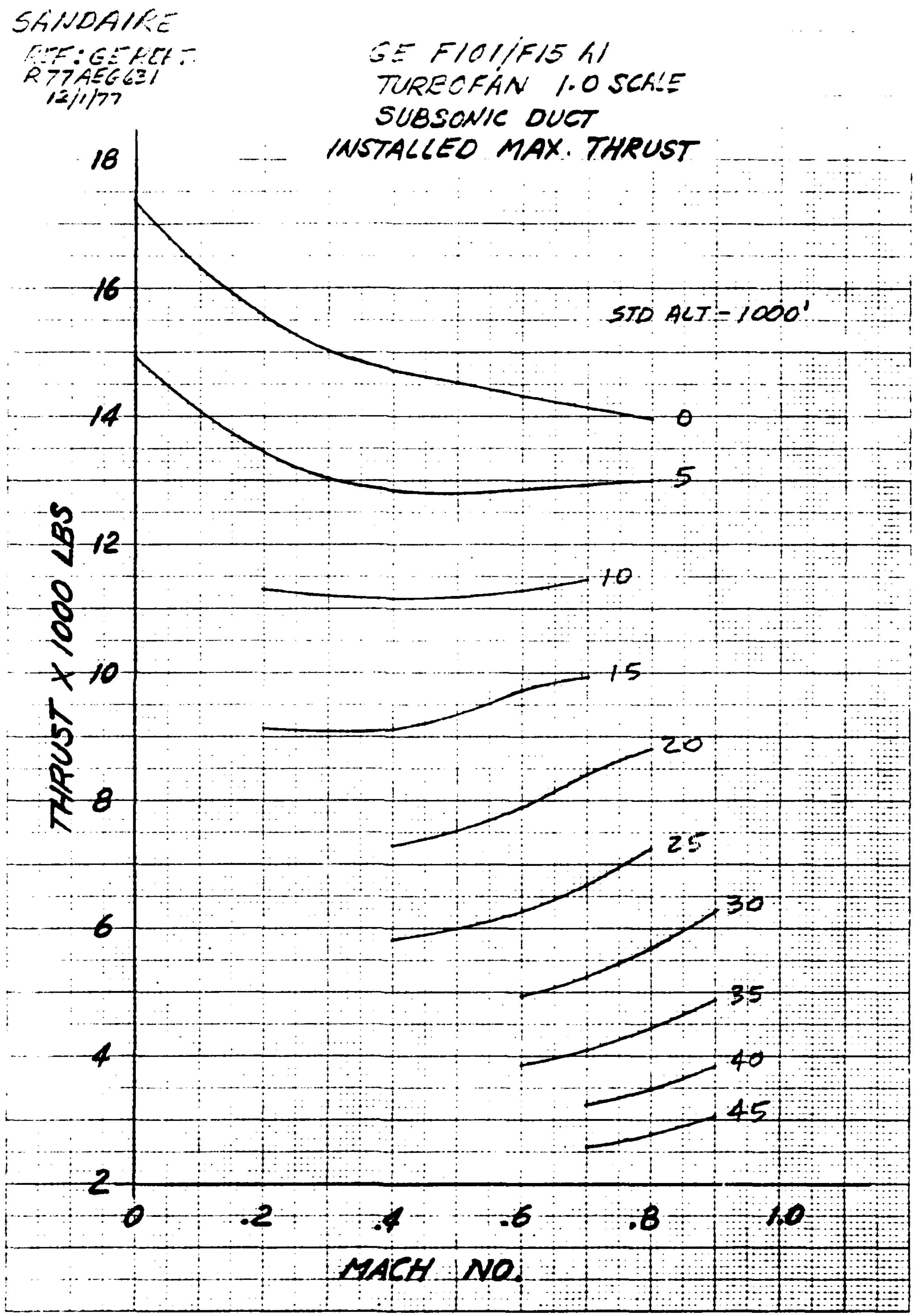
PRELIMINARY INSTALLATION

G.E. F101/F15 A1 TURBOFAN ENGINE

(R77AEG631 - 1 DECEMBER 1977)



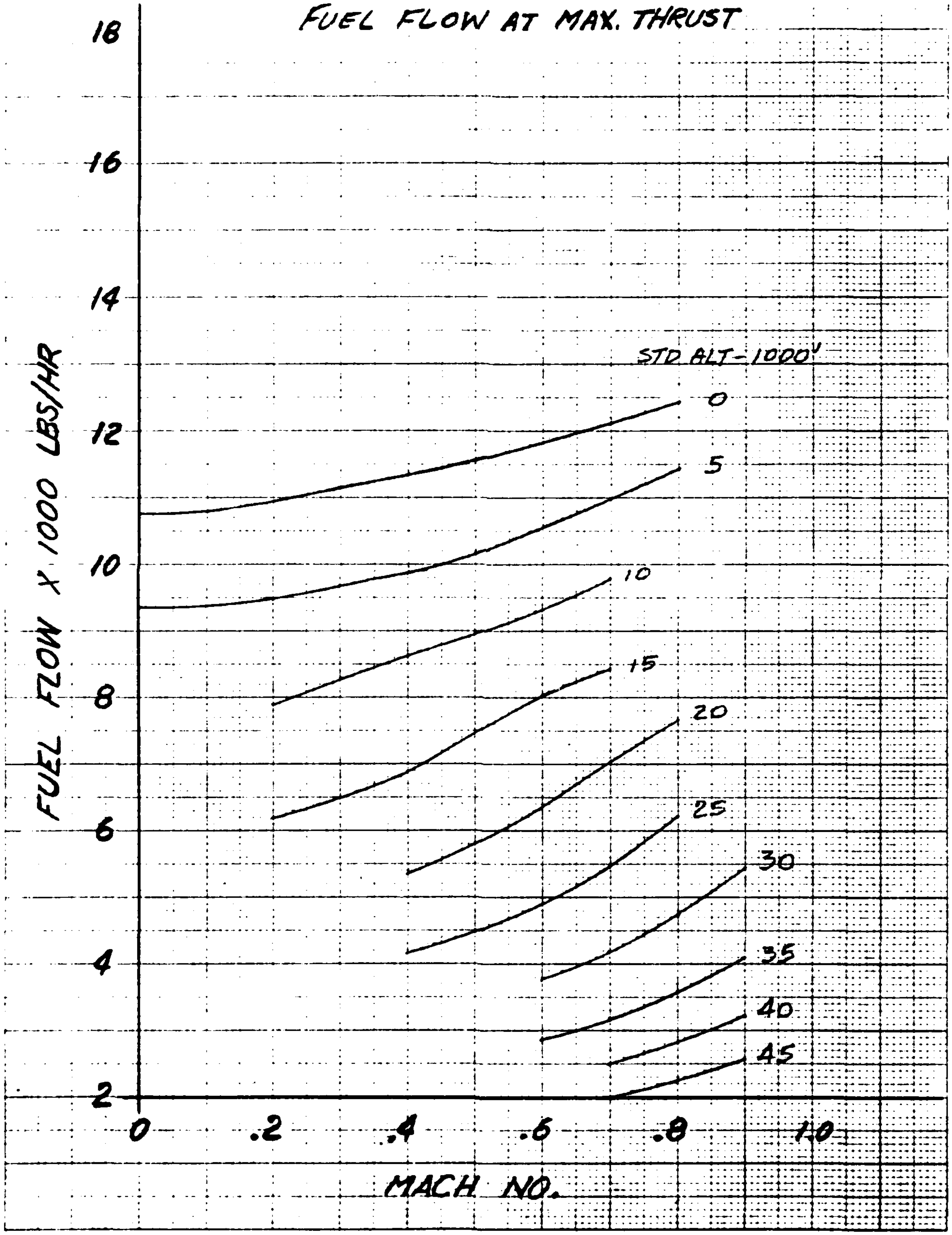
APPROX. WEIGHT
3360 LB. (INCL.
FIXED NOZZLE)
NO REVERSER
NO BLEED



SANDAIRE
REF: GE REPT
R77AEG631
12/1/77
GE F101/F15 A1
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
INSTALLED MAX. THRUST
STD ALT - 1000'
THRUST X 1000 LBS
MACH NO.
0
5
10
15
20
25
30
35
40
45

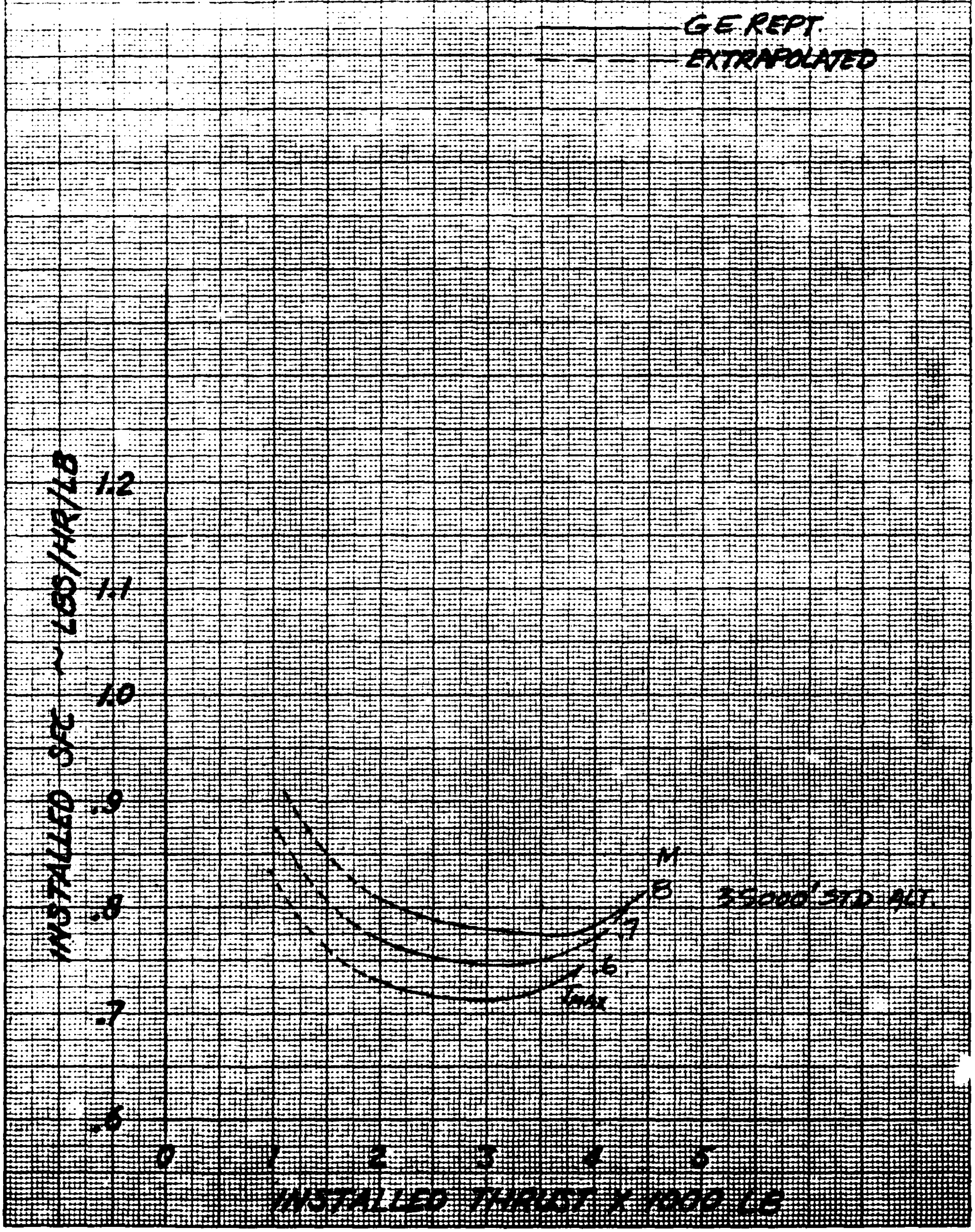
12/1/77

GE F101/F15 A1
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
FUEL FLOW AT MAX. THRUST



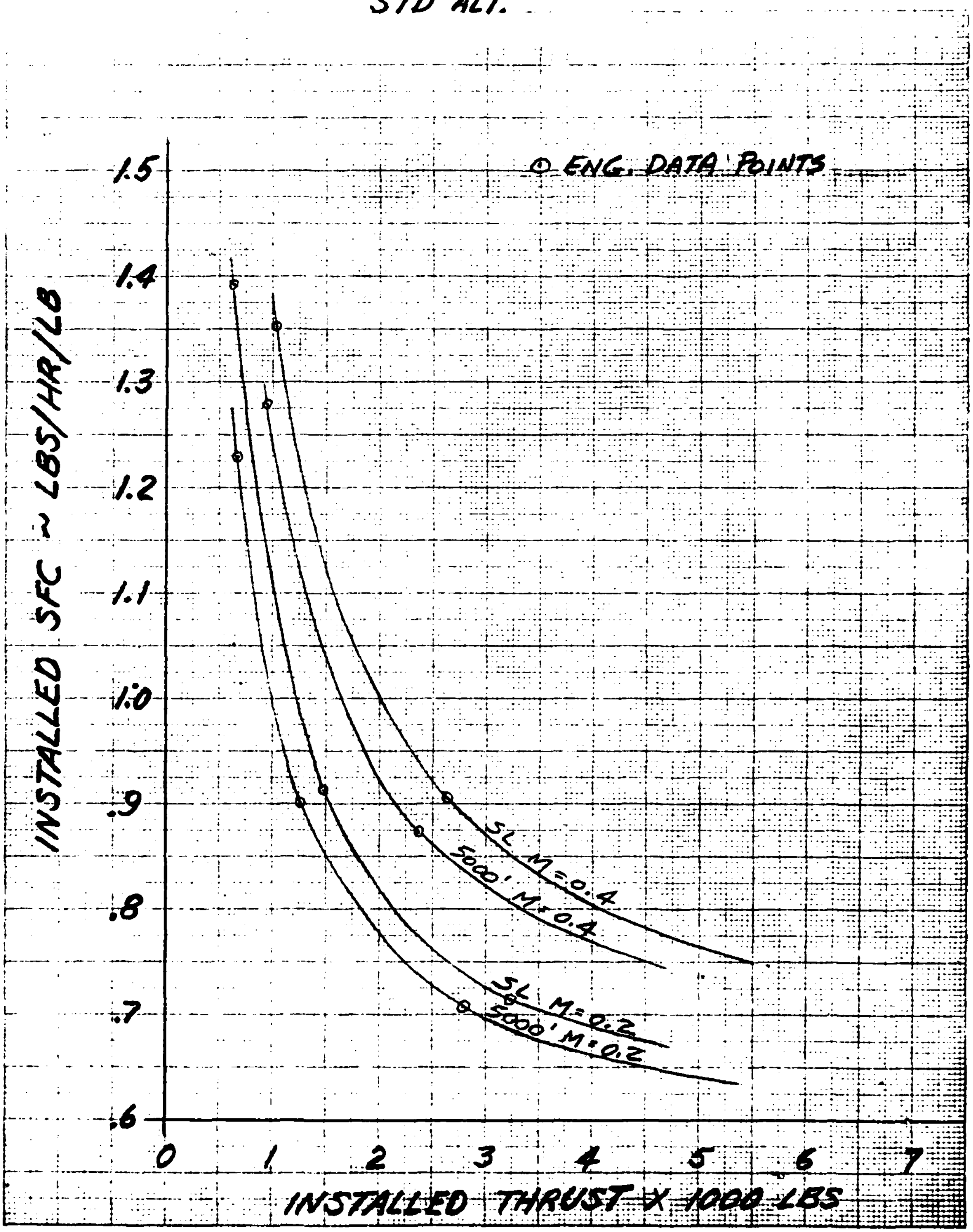
SANDAIRE
REF: G.E. REPT.
R77 AEG 631
12/1/77

GE F101/F15 A1
TURBOFAN 1.0 SCALE
SUBSONIC DUCT



SANLAIRE
REF: G.E. REPT
R77 AEG 631
12/1/77

GE F101/F15 A1
TURBOFAN 1.0 SCALE
SUBSONIC DUCT
STD ALT.



REF GE

R77AEG631

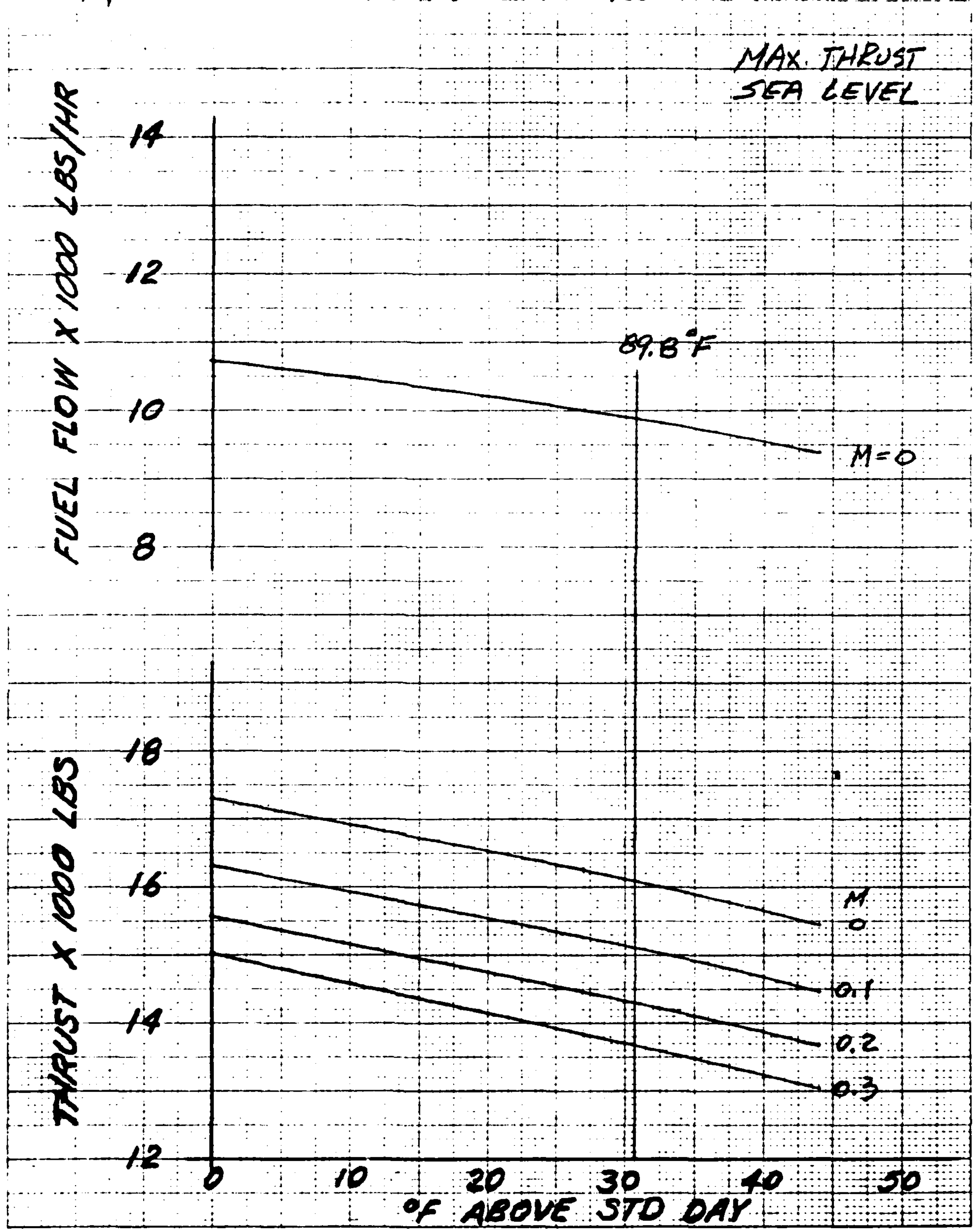
12/1/77

GE F101/F15 A1

TURBOFAN 1.0 SCALE

SUBSONIC DUCT

INSTALLED DATA



(b) One Engine Configuration, Fan Bleed to Pressurize Trunk, GE F101 Engine

Changes

At this point, some updating changes were made. Fan bleed calculations were revised, but a decision to go to the final configuration described on Page E-20 was held in abeyance. The spare trunk orifice flow to the atmosphere was selected as 40% vs the final 20%. Flap characteristics had been partially developed and slight changes in the takeoff parameters were made. A high wing configuration had been selected (gull eliminated) and the trunk was made rectangular in planform (2.5/1) with rounded ends. The trunk was considered fully retracted with fuselage doors to close the opening. The total effect of the changes was not great, and no big change is apparent except that associated directly with the G.E. engine vs the parametric engine used previously.

For the first iteration

Design G.W.	-	29300 lb
Rated T_{Max} (Std, S.L.) Scale 0.914	-	16650 lb
Wing Area	-	348 sq ft
Cushion PSI	-	1.25
Trunk PSI	-	2.5
Trunk ℄ Perimeter	-	51.8 ft
Daylight Gap (Ave)	-	0.25 ins
Trunk ℄ Area	-	162.8 sq ft
Bleed required	-	62.4 lb/sec
Thrust loss due to bleed	-	3233 lb
Takeoff Ground Run S.L. 89.8°F, μ =.065 4.5 min at T_{Max} prior T.O.	-	3000 ft
V_{To}	-	230 ft/sec

Using the same wing and tail shapes, as used in Section (1) above, and a fuselage that will house a retracted trunk (this was later determined to be extremely difficult if not impossible and was abandoned in favor of stowing the trunk externally on an appropriately shaped fuselage bottom), calculation showed.

$$S = 348 \text{ sq ft}$$
$$S_H = 94 \text{ sq ft}$$
$$S_V = 74 \text{ sq ft}$$

(b) (Continued)

Drag Aircraft - Low Speed

$$C_{D_{Min}} = .0201$$

Use drag coefficient for the one-engine-configuration-with-bleed in Section (1) minus $\Delta C_D = .0012$; correct stores ΔC_D for wing area change.

CAS Mission

	Distance, N.M.	Fuel, Lb
5 minutes at T_{Max} at S.L.		818
Climb to 35000, M = 0.65	51	710
Cruise, M = 0.7	109	528
Descend to 5000 ft		
Loiter 60 minutes, M = 0.3		1847
Drop MK82, retain TERs		
Climb to 35000, M = 0.65	21	284
Cruise, M = 0.66	139	427
Descend to S.L.		
Loiter 10 minutes, M = 0.23		209
Reserve 5% initial fuel		254
Total	320	5077

RAD = 160 N.M.

Weight

	Pounds
Wing	2955
Horizontal Tail	398
Vertical Tail	348
Fuselage including duct structure and speed brakes	1523
Trunk doors and mechanism	936
Canopy	410
Engine	3071
Bleed	121
Tail Pipe Extension	109
Engine Section	287
Inlet Ducts	258
Engine Cont, Start, Lub, Oil	185

SANDAIRE

(b) (Continued)

Weight	Pounds
Flight Controls	623
Wing Integral Tanks	124
Unusable Fuel	85
Fuel System including Fuel dump & Aerial Refueling Provisions	224
Systems including Ejection Seat	2041
Specified Load	8574
SETOLS	1083
Protection	447
CAS Mission Fuel	5077
Gross Weight	28879

This is 1.4% less than the selected G.W. of 29300 pounds for design. This reduction in weight will result in a G.W. of about 28400 pounds if the design is recycled.

At this point in the design progress, the P&W STF 529 engine data became available. The favorable characteristics of this engine were recognized, and the final design was developed around this engine. The final G.W. was 24300 pounds as shown in the report.

APPENDIX F

PERFORMANCE DATA

It was deemed not feasible or expedient to include all the calculations for the following performance data; therefore, it is presented in graphic form. All data is for standard atmospheric conditions except as noted.

(1) CAS and ferry mission calculations.

(2) Best cruise Mach No. and altitude for clean configuration.

(3) Maximum specific range vs cruise Mach No. for clean configuration.

(4) Best endurance Mach No. and altitude for CAS loading less 40% fuel.

(5) Maximum endurance for CAS loading less 40% fuel.

(6) Service ceiling, CAS mission.

(7) Max rate of climb, S.L., 5000 ft. and 15,000 ft., standard air, and S.L. for 89.8°F.

(8) Max. rate of climb at S.L., CAS mission, SETOLS down and up at takeoff speed, flaps 40°.

Max. rate of climb at S.L., CAS mission less 60% fuel (MLDGW), 21569 lb., SETOLS down and up, at landing approach speed, flaps 50°.

(9) Max. sustained maneuver load factor at S.L. and 5000 ft., CAS mission less 40% fuel, 22479 lb.

(10) Max. sustained maneuver load factor at S.L. and 5000 ft., clean configuration.

(1) The following computer data was used in establishing the CAS and Ferry Mission profiles:

CAS MISSION
4 Pylons
4 TER's
12 Mk 82

Operation	Δ Fuel (Lbs.)	Weight (Lbs.)	Alt Ft.	Mach No.	Δ Dist N.Mi.	Δ Time Min.
Initial	-	24300	SL	-	-	-
WU & TO, 5 min	651	23649	SL	-	0	5.0
Climb-out	810	22839	36089	.60	79	13.2
Cruise-out	415	22424	36089	.69 Ave	81	12.3
Desc.	0	22424	5000	-	0	0
Loiter	1635	20789	5000	.31	0	60
Drop Mk-82	0	13949	5000	.31	0	0
Climb-back	260	13689	36089	.70	26	3.6
Cruise-back	375	13314	36089	.595 Ave	134	23.6
Desc.	0	13314	SL	-	0	0
Loiter	178	13136	SL	.235	0	10
Reserve (5% Initial)	228	12908	-	-	-	-

Radius 160 N.Mi.

Total Fuel: 4552 Lbs.

Total Time: 2.1283 Hrs.

Mission Profile is on Page 14 of the Report.

FERRY MISSION
4 Pylons
All Internal Fuel

Operation	Δ Fuel (Lbs)	Weight (Lbs)	Alt Ft.	Mach No.	Δ Dist N.Mi.	Δ Time Min.
Initial	-	20243	SL	-	-	-
WU & TO,5 min	651	19592	SL	-	0	5.0
Climb-out	583	19009	45800	.70	81	11.7
Cruise-out	2000	17009	48000	.775	720	97.2
Cruise-out	2000	15009	50200	.775	814	109.9
Cruise-out	1899	13110	54200	.775	885	119.5
Descend	0	13110	SL	-	0	0
Loiter	170	12940	SL	.23	0	10
Reserve (5% Initial)	384	12556	-	-	-	-

Range: 2500 N.Mi.

Total Fuel: 7687 Lbs.

Total Time: 5.888 Hrs.

Mission Profile is on Page 16 of the Report.

(2)&(3) Best Cruise and Maximum Range

(a) Cruise Mach No. at altitude for the clean configuration (with 4 pylons), is plotted as specific range in N. Mi/lb fuel vs Mach No., Pages F-6 to F-9. The corresponding instantaneous rate of climb is plotted to show altitude limits on cruise, pages F-10 to F-13. These data were used to determine best cruise altitude and Mach No. which are combined to give 0.99 maximum specific range and best cruise altitude vs weight on Pages F-14 and F-15. Also shown are time, fuel and distance for climb in the ferry mission, Pages F-16 to F-19. These plots were used in the calculation of ferry range.

(b) Included here are similar plots for the CAS loading (4 pylons + 4 TER's + 12 Mk 82) and the CAS mission calculations. Note that for "cruise back", data are plotted for determination of best "cruise back" altitude considering both "climb back" from 5000 ft. and "cruise back". The 36089 ft. was the best, pages F-20 to F-24.

(4) & (5) Endurance

(a) Calculations for loiter at 5000 ft and sea level are plotted on Pages F-25 and F-26. Loiter is at $(L/D)_{Max}$ which was justified by plotting fuel flow vs Mach No. where the minimum fuel flow is essentially at $(L/D)_{Max}$, Page F-27.

(b) Endurance at higher altitude was calculated; however, insufficient engine data are available at the low thrust associated with minimum fuel flow. Extrapolation was necessary, and some error undoubtedly results. An example is on Page F-28. This and other data, not shown, resulted in Page F-29. The above probability of error may cause the irregularities.

(6) Service Ceiling

The data of Section (7) are combined with other data, not presented, to calculate the service ceiling which is based on climb at maximum rate of climb and on varying weight due to the fuel consumed in climb. The weight at start of climb is 24300 - 5 min at T_{Max} = 23649 lb.

For the CAS mission loading, fuel to climb to 36089 ft. from Section (1) is 810 lb. This is at climb M = .60; adjusting to best climb speed, fuel reduces to 774 lb. Page F-30 shows total fuel consumed in climb vs altitude in percent of fuel to climb to 36089 ft. The altitude for 300 ft/min and 500 ft/min instantaneous rate of climb vs instantaneous weight is also plotted on Page F-30. Using the two plots together, the service ceiling (300 ft/min R/C) is 36800 ft for the CAS loading with weight of 23649 lb. at start of climb.

(7) Rate of Climb for S.L., 5000 ft. and 15000 ft., standard air, is plotted vs Mach No., for several weights at CAS loading (4 pylons, 4 TER's, 12 Mk 82), CAS loading after bomb drop (4 pylons, 4 TER's), and clean with 4 pylons. P&W engine performance at S.L., 89.8°F at the Mach No. for maximum rate of climb is not available; therefore, no estimate was made for this temperature. (Pages F-31 to F-40).

The maximum rate of climb at any weight may be obtained from these data in combination with Page F-30 which gives the fuel consumed in climb. The maximum rate of climb for the CAS mission takeoff at 24300 lb less 5 min. at T_{Max} = 23649 lb is

Alt ft	Max Rate of Climb ft/min
0	9880
5000	8250
15000	5000

Also, maximum speed in level flight is plotted vs altitude for the three store loadings, Page F-41.

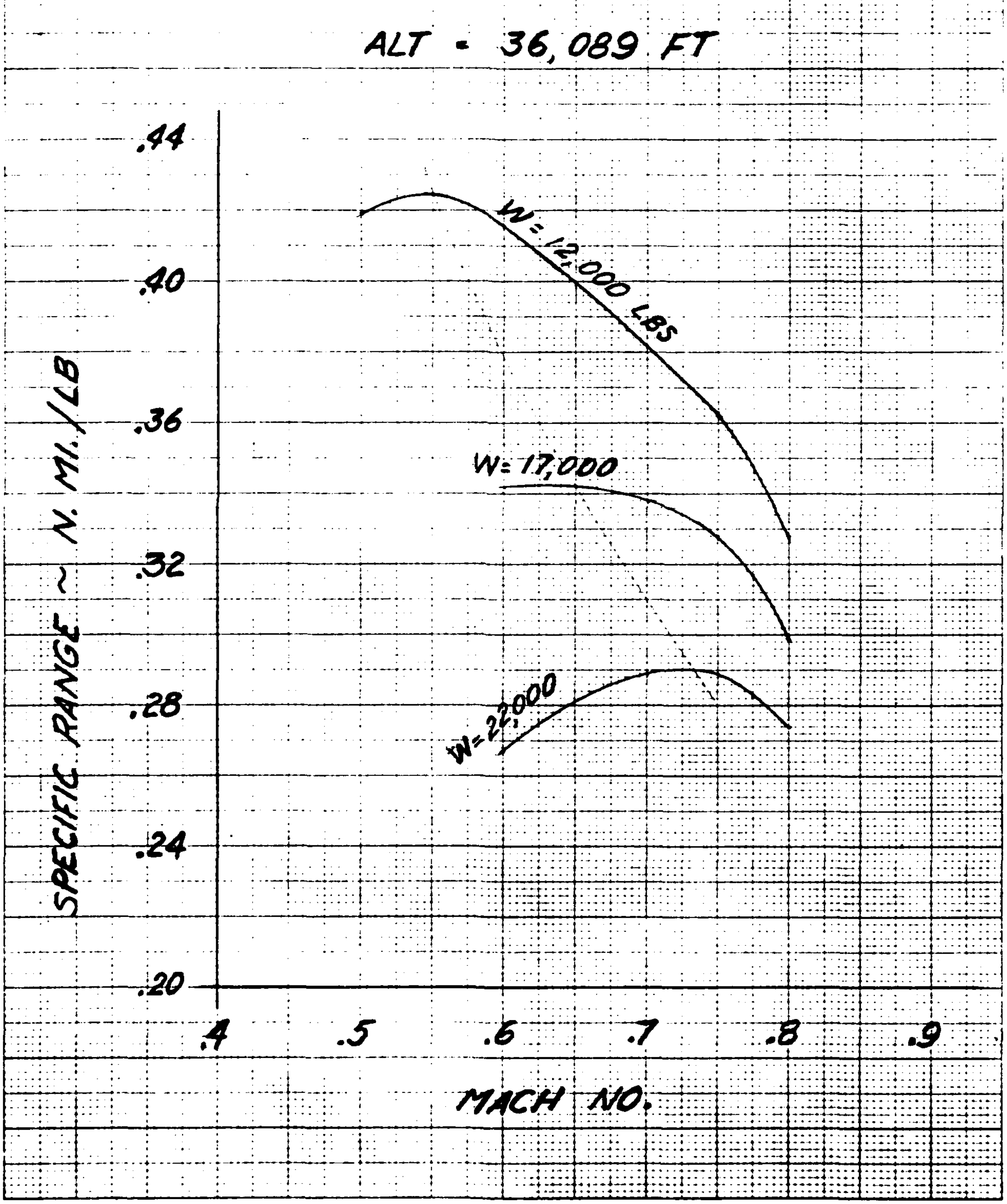
(8) Max rate of climb in takeoff and landing - see Appendix B, pages B-26 and B-29.

(9)&(10) Max sustained maneuver load factor; the results of the calculations are shown by the plot, page F-41.

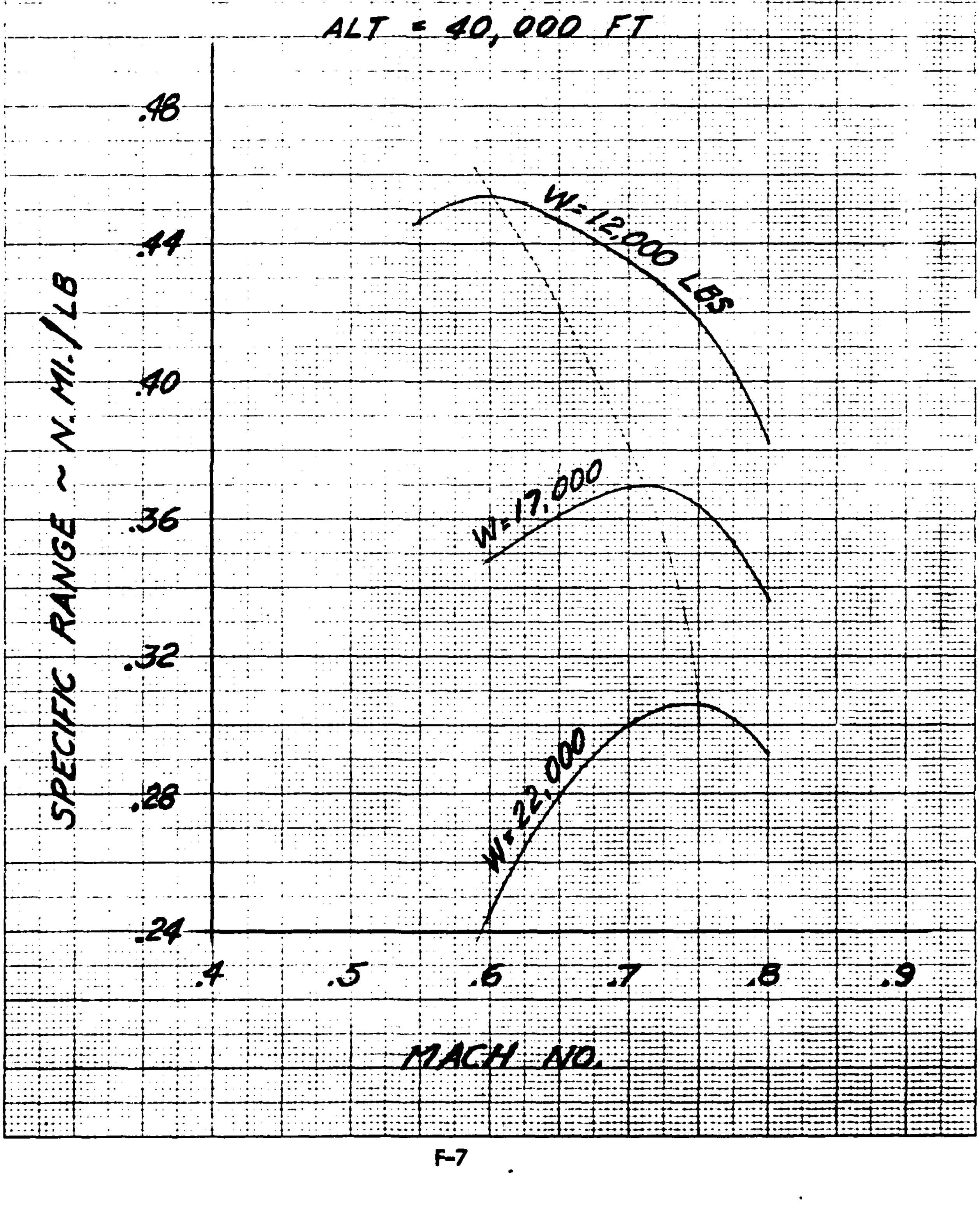
CAS - SETOLS

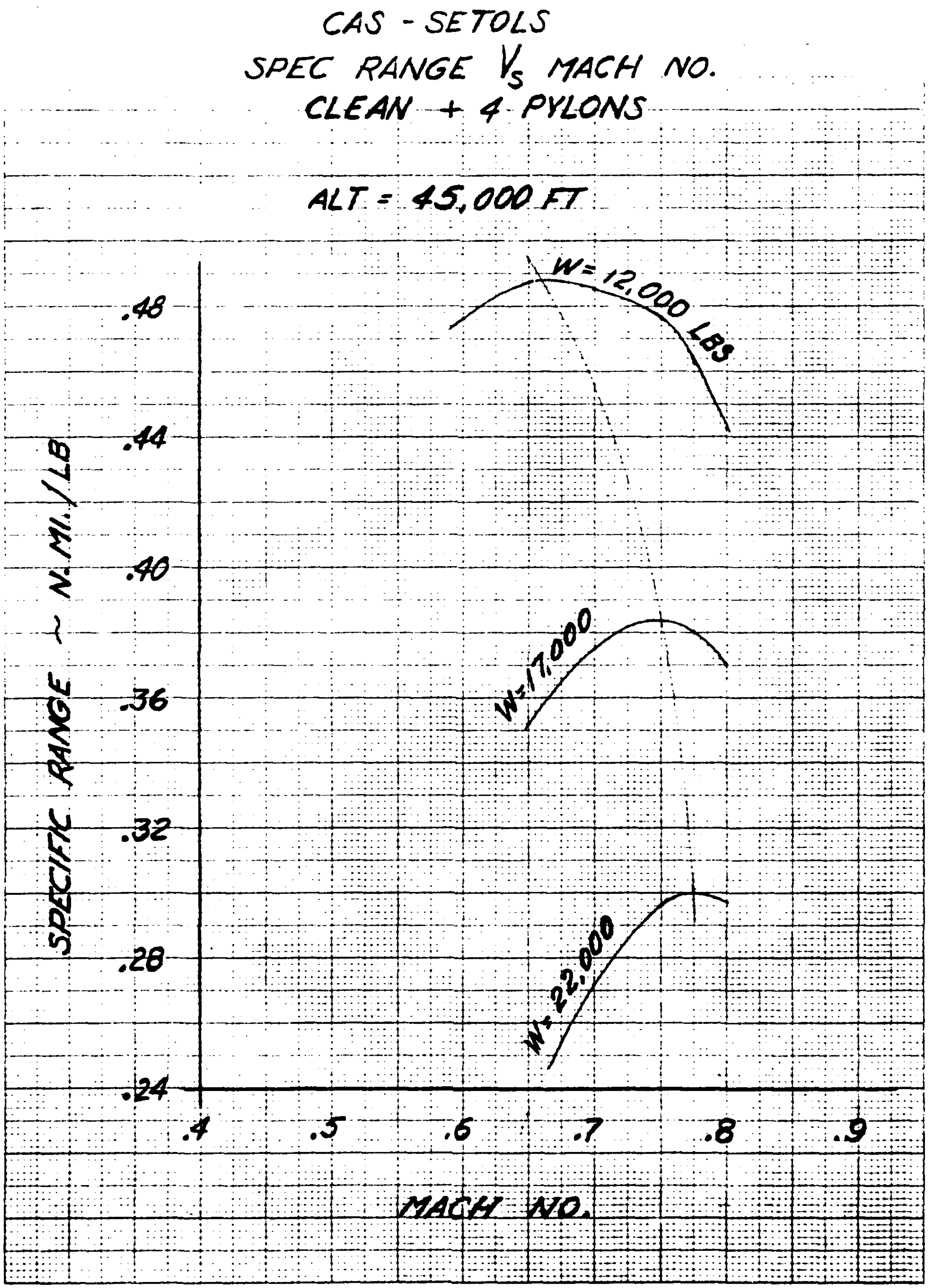
SPEC RANGE Vs MACH NO.

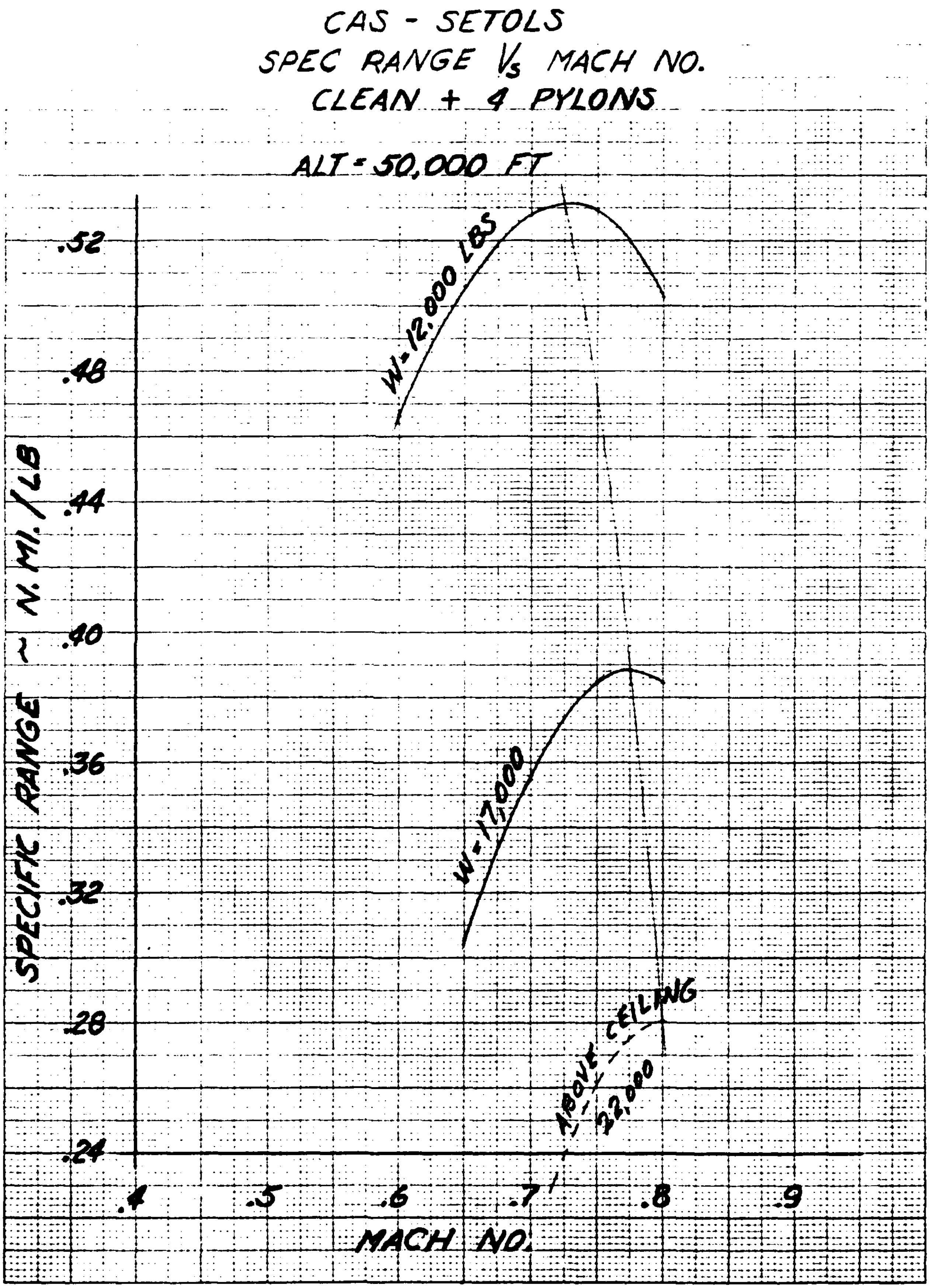
CLEAN + 4 PYLONS

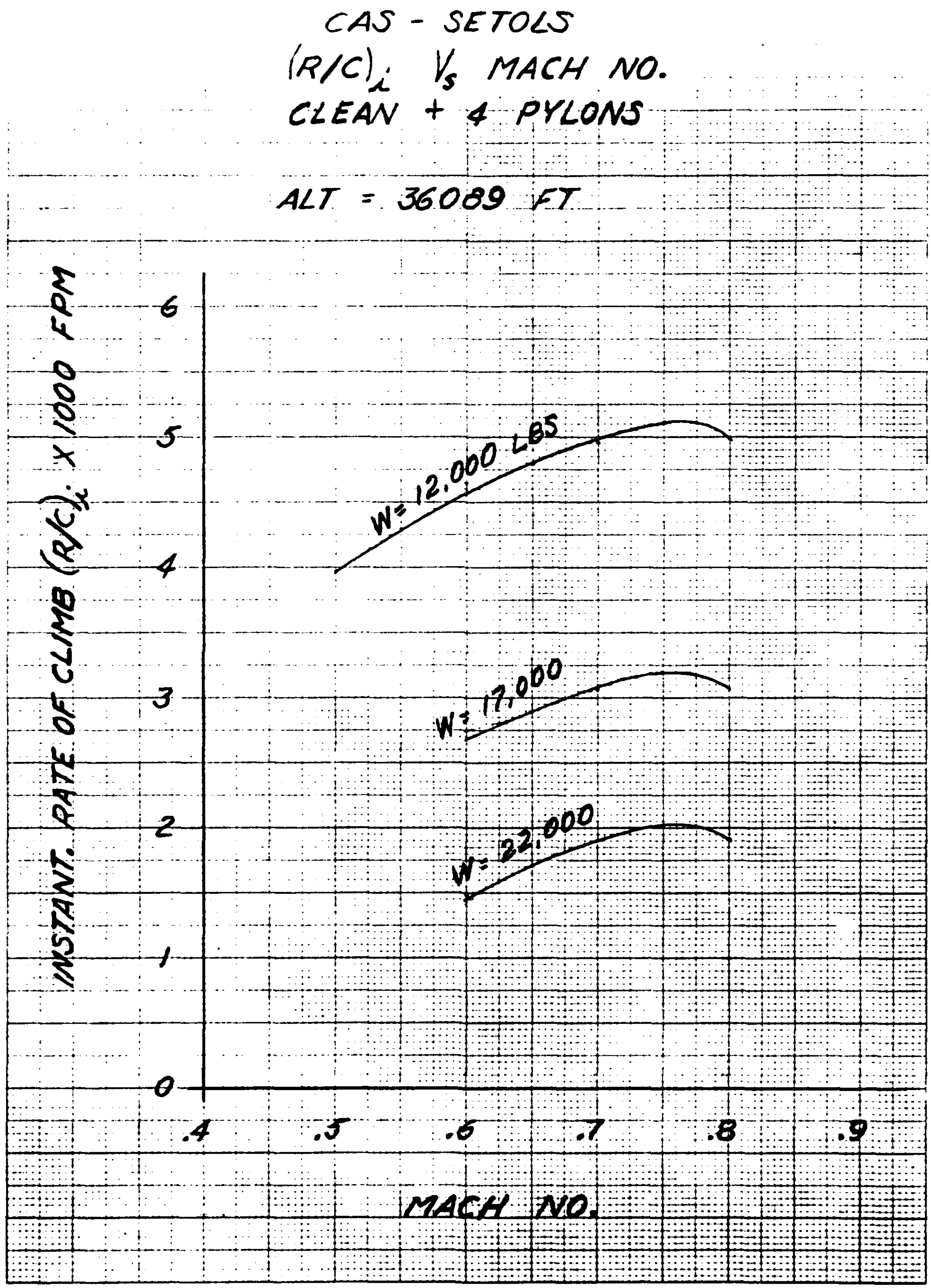


CAS - SETOLS
SPEC RANGE Vs MACH NO.
CLEAN + 4 PYLONS




CAS - SETOLS
SPEC RANGE Vs MACH NO.
CLEAN + 4 PYLONS
ALT = 45,000 FT
W = 12,000 LBS
W = 17,000
W = 22,000
SPECIFIC RANGE ~ N.MI./LB
.48
.44
.40
.36
.32
.28
.24
.4
.5
.6
.7
.8
.9
MACH NO.

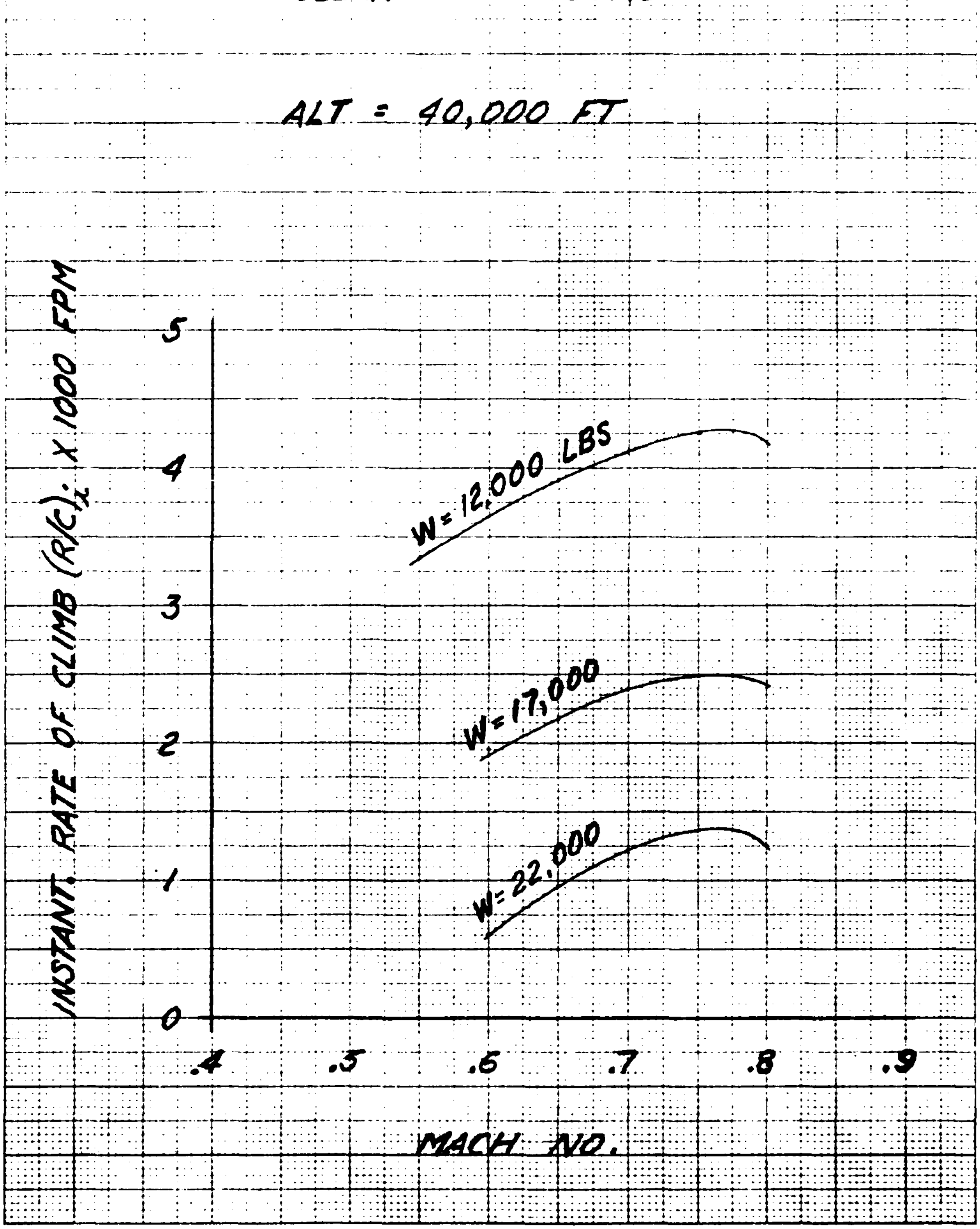

CAS - SETOLS
SPEC RANGE Vs MACH NO.
CLEAN + 4 PYLONS
ALT = 50,000 FT
W = 12,000 LBS
W = 17,000
ABOVE CEILING
22,000
SPECIFIC RANGE ~ N. MI. / LB
.52
.48
.44
.40
.36
.32
.28
.24
.4
.5
.6
.7
.8
.9
MACH NO.

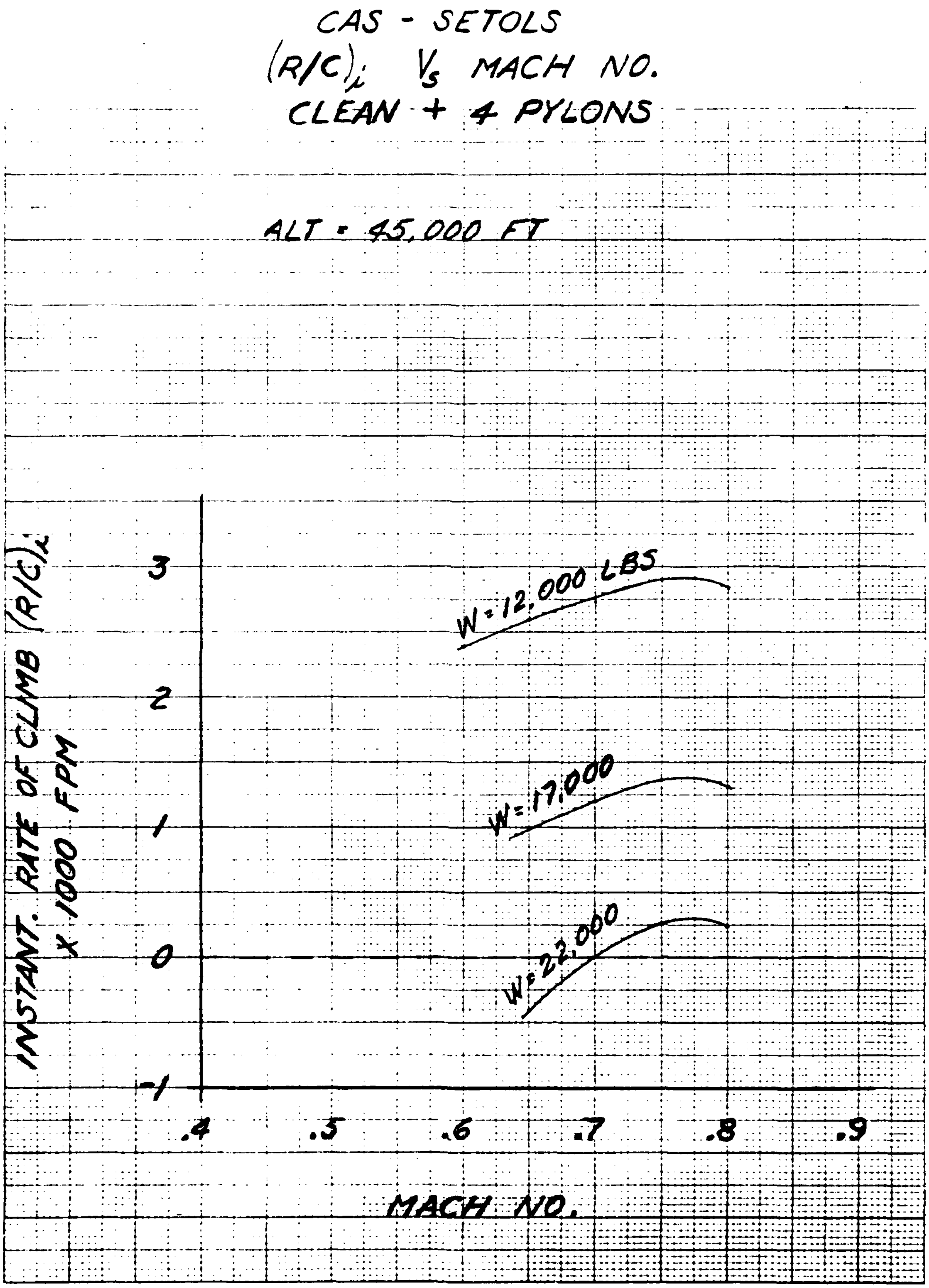

CAS - SETOLS
(R/C)i Vs MACH NO.
CLEAN + 4 PYLONS
ALT = 36089 FT
INSTANT. RATE OF CLIMB (R/C)i X 1000 FPM
W= 12,000 LBS
W= 17,000
W= 22,000
0
1
2
3
4
5
6
.4
.5
.6
.7
.8
.9
MACH NO.

CAS - SETOLS

$(R/C)_i$ Vs MACH NO.

CLEAN + 4 PYLONS



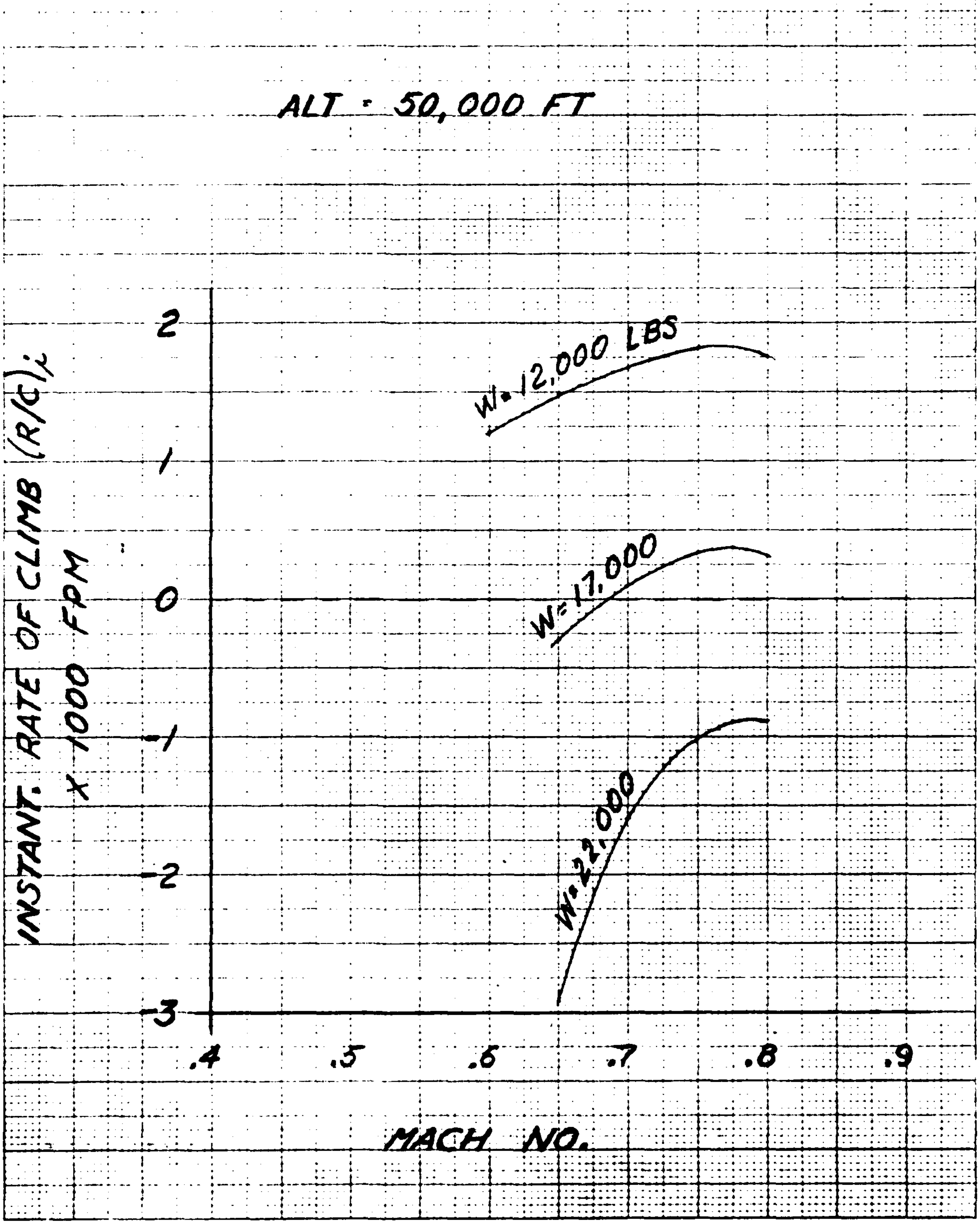


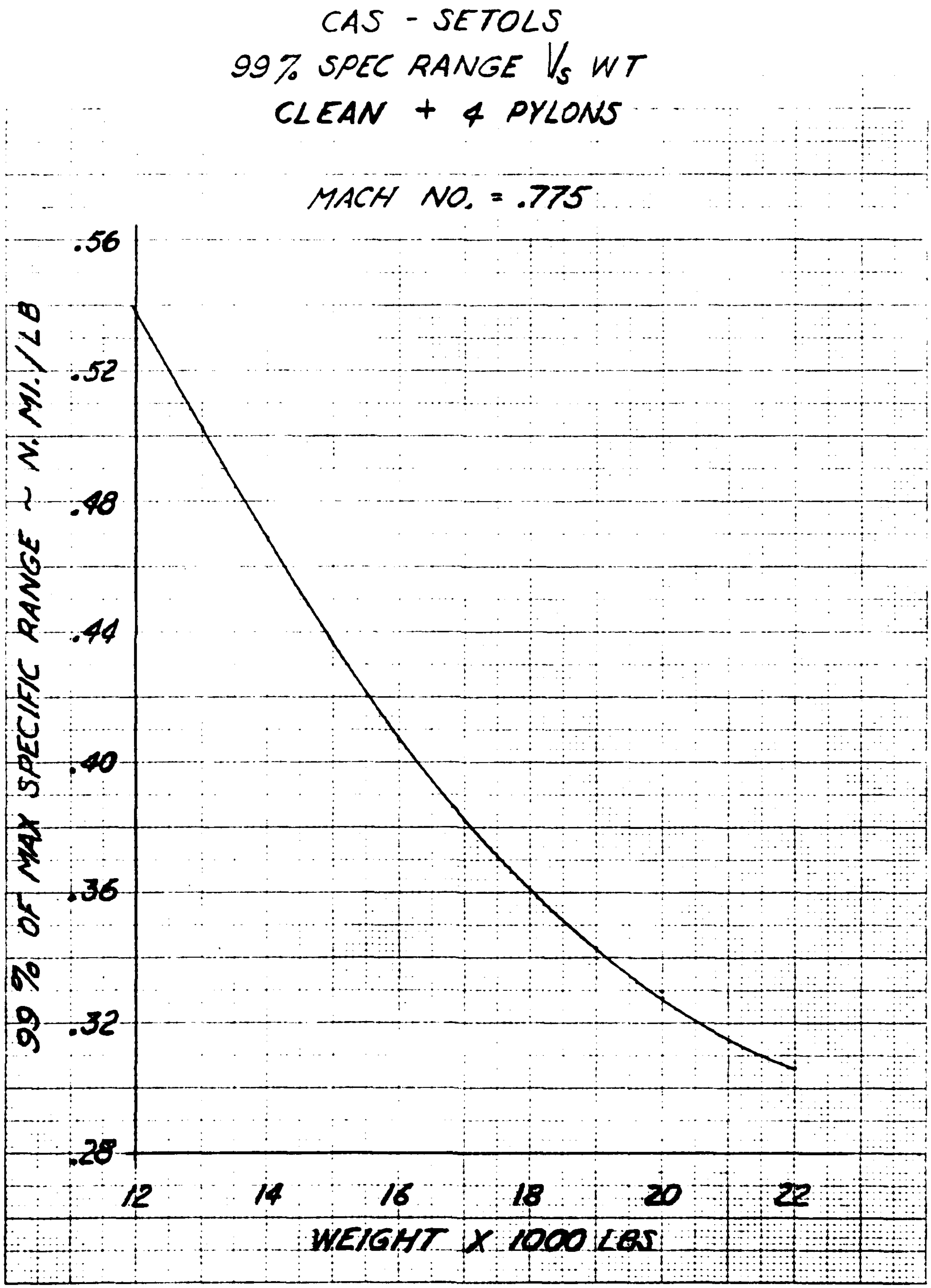
CAS - SETOLS
(R/C)i Vs MACH NO.
CLEAN + 4 PYLONS
ALT = 45,000 FT
INSTANT. RATE OF CLIMB (R/C)i
X 1000 FPM
3
2
1
0
-1
W=12,000 LBS
W=17,000
W=22,000
.4
.5
.6
.7
.8
.9
MACH NO.

CAS - SETOLS

$(R/C)_i$ Vs MACH NO.

CLEAN + 4 PYLONS

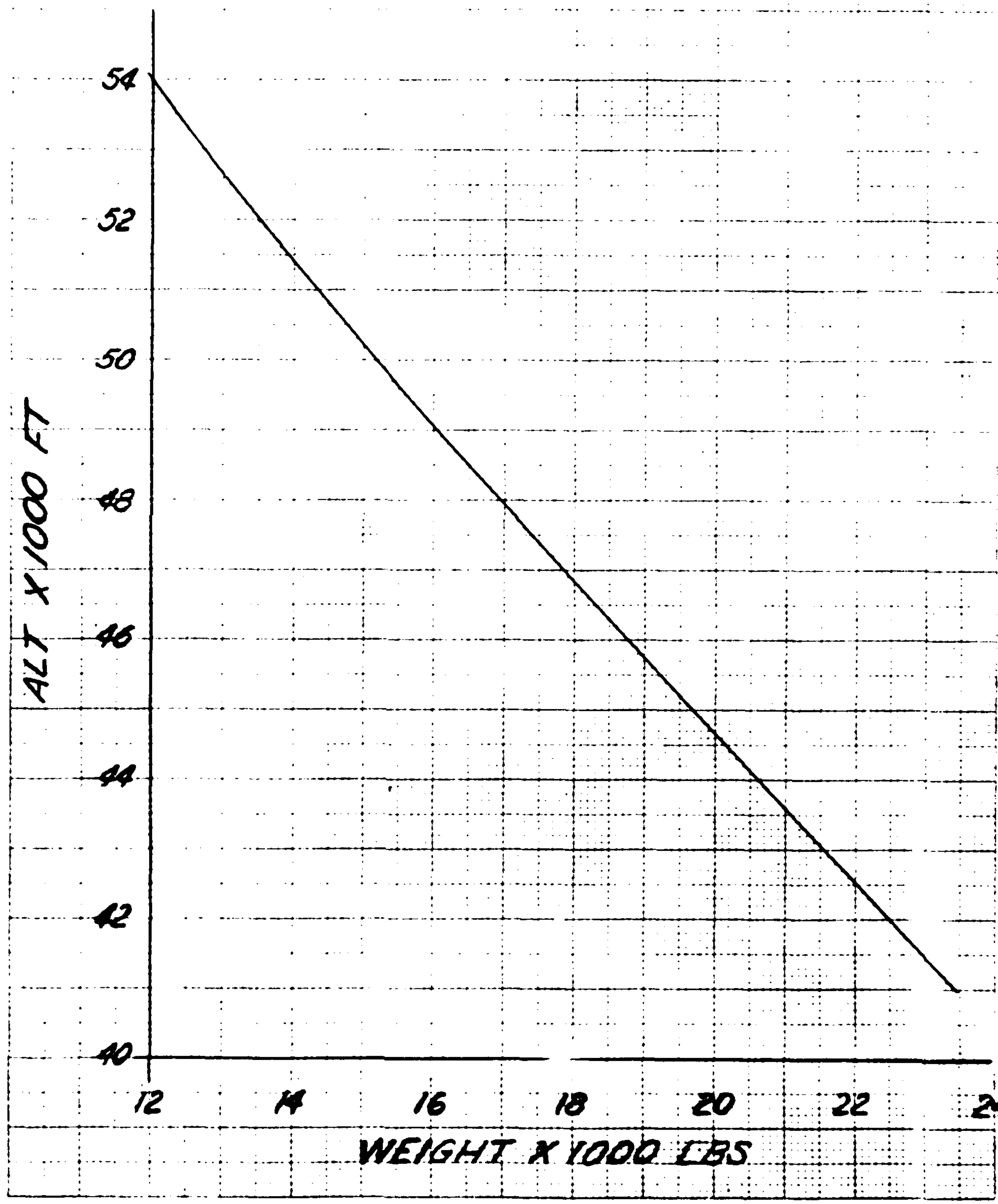



CAS - SETOLS
99% SPEC RANGE Vs WT
CLEAN + 4 PYLONS
MACH NO. = .775
99% OF MAX SPECIFIC RANGE ~ N. MI./LB
.56
.52
.48
.44
.40
.36
.32
.28
12
14
16
18
20
22
WEIGHT X 1000 LBS

BEST CRUISE ALTITUDE

CLEAN + 4 PYLONS (FERRY)

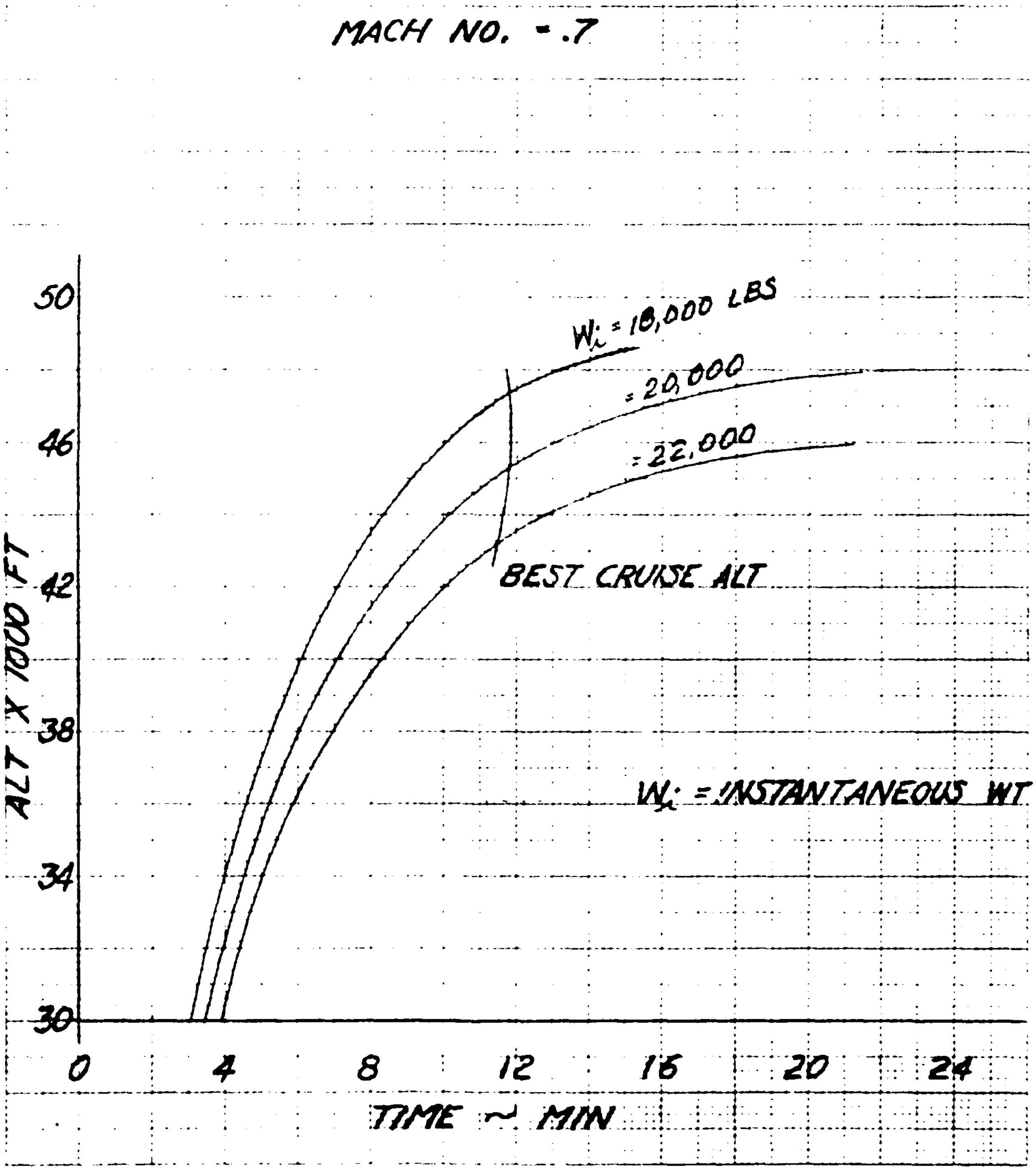
$M_{CR} = .775$



CAS - SETOLS

CLIMB DATA

CLEAN + 4 PYLONS



AD-A088 351   SANDAIRE SAN DIEGO CA   F/G 1/2
CONCEPTUAL POINT DESIGN STUDY OF A NEW CTOL SETOLS CAS AIRCRAFT--ETC(U)
JUN 79 P D SORENSEN, R L BAYLESS, E F NOEL   N62269-79-C-0438

UNCLASSIFIED   NADC-78155-20   NL

3 of 3

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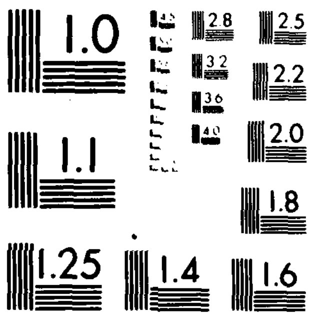
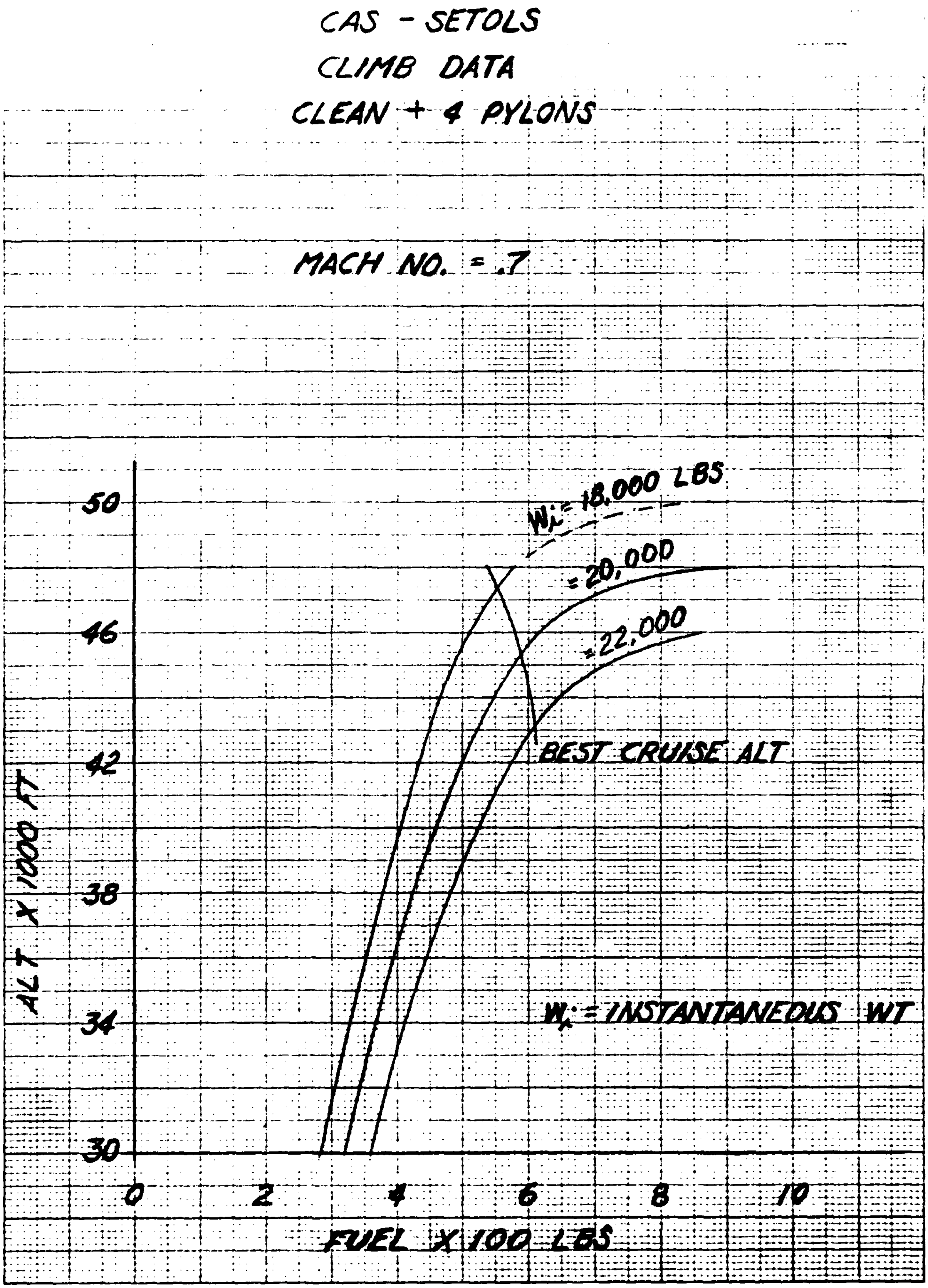
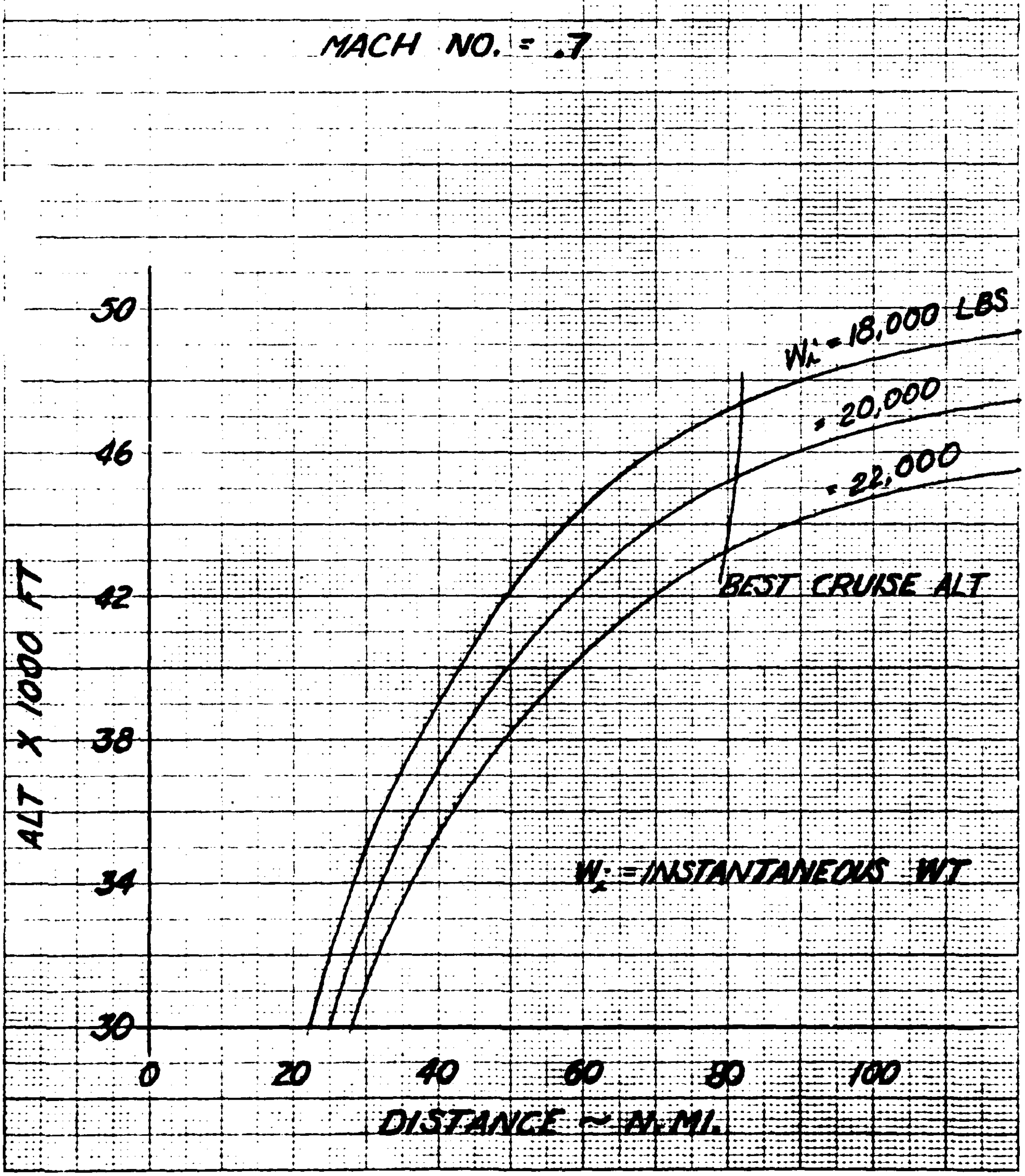


MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS 1963 A



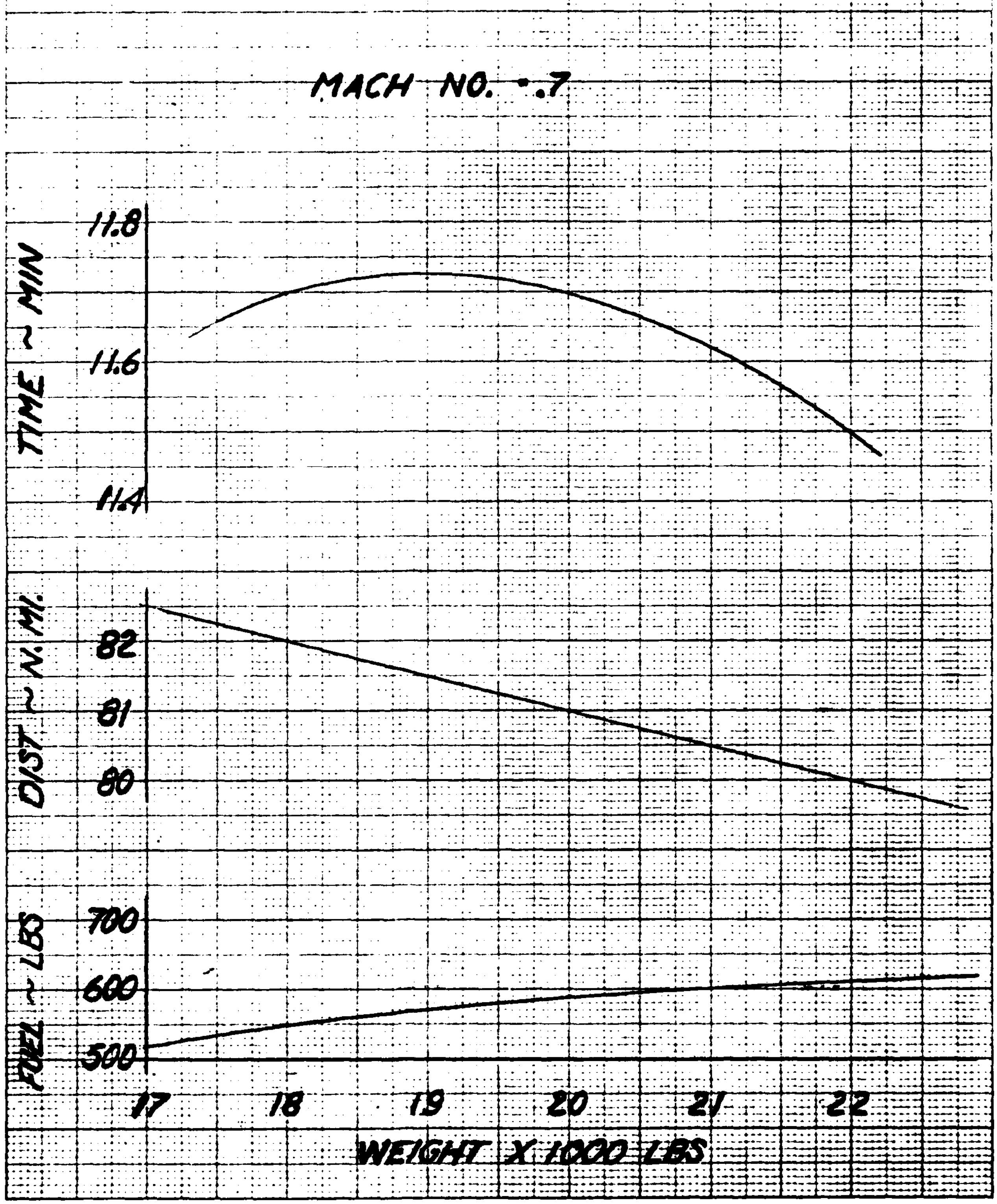
CAS - SETOLS
CLIMB DATA
CLEAN + 4 PYLONS
MACH NO. = .7
Wi = 18,000 LBS
= 20,000
= 22,000
BEST CRUISE ALT
Wi = INSTANTANEOUS WT
ALT X 1000 FT
50
46
42
38
34
30
0
2
4
6
8
10
FUEL X 100 LBS

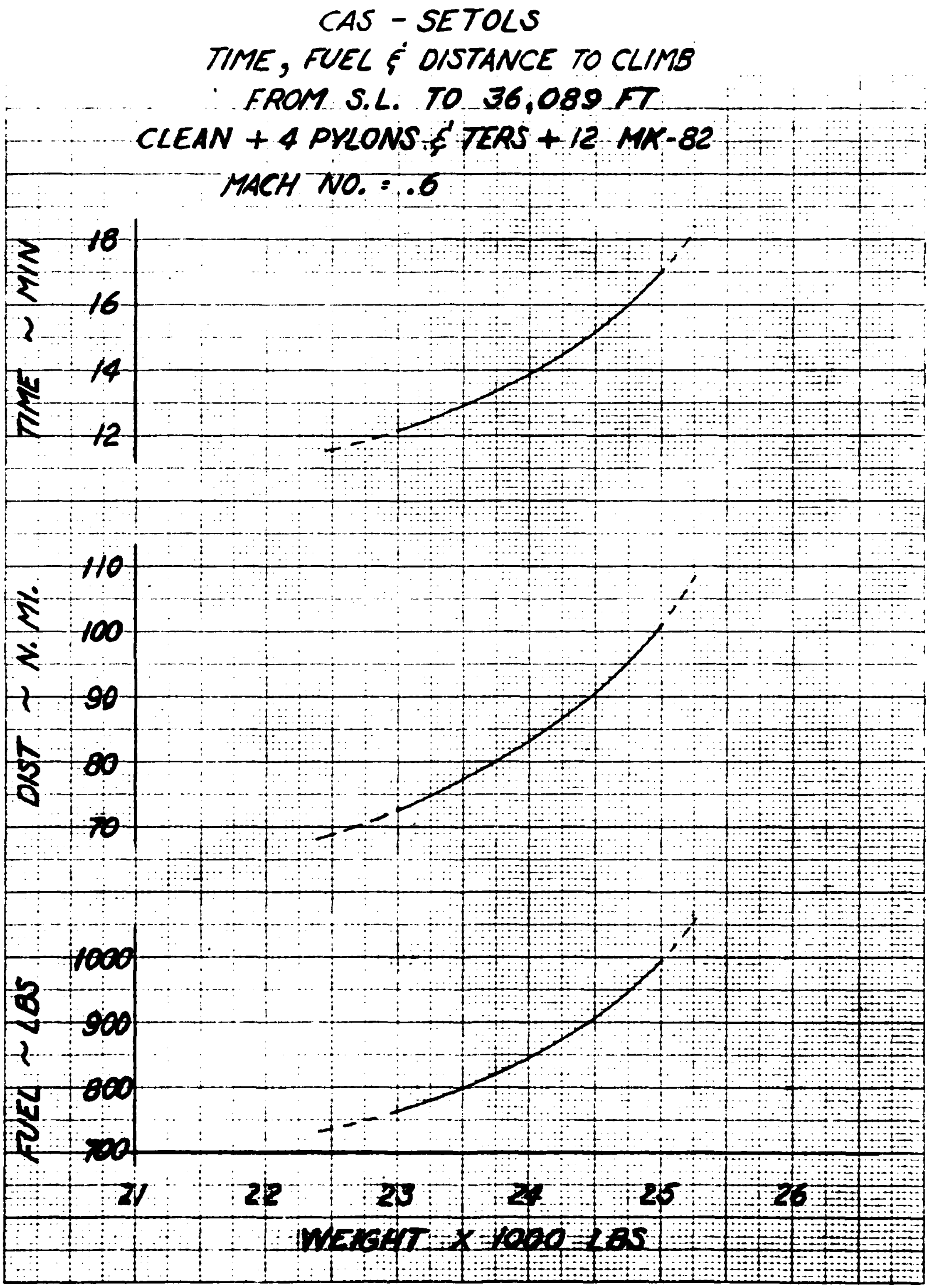
CAS - SETOLS
CLIMB DATA
CLEAN + 4 PYLONS



CAS - SETOLS

TIME, FUEL & DISTANCE TO CLIMB FROM S.L. TO BEST CRUISE ALT. CLEAN + 4 PYLONS



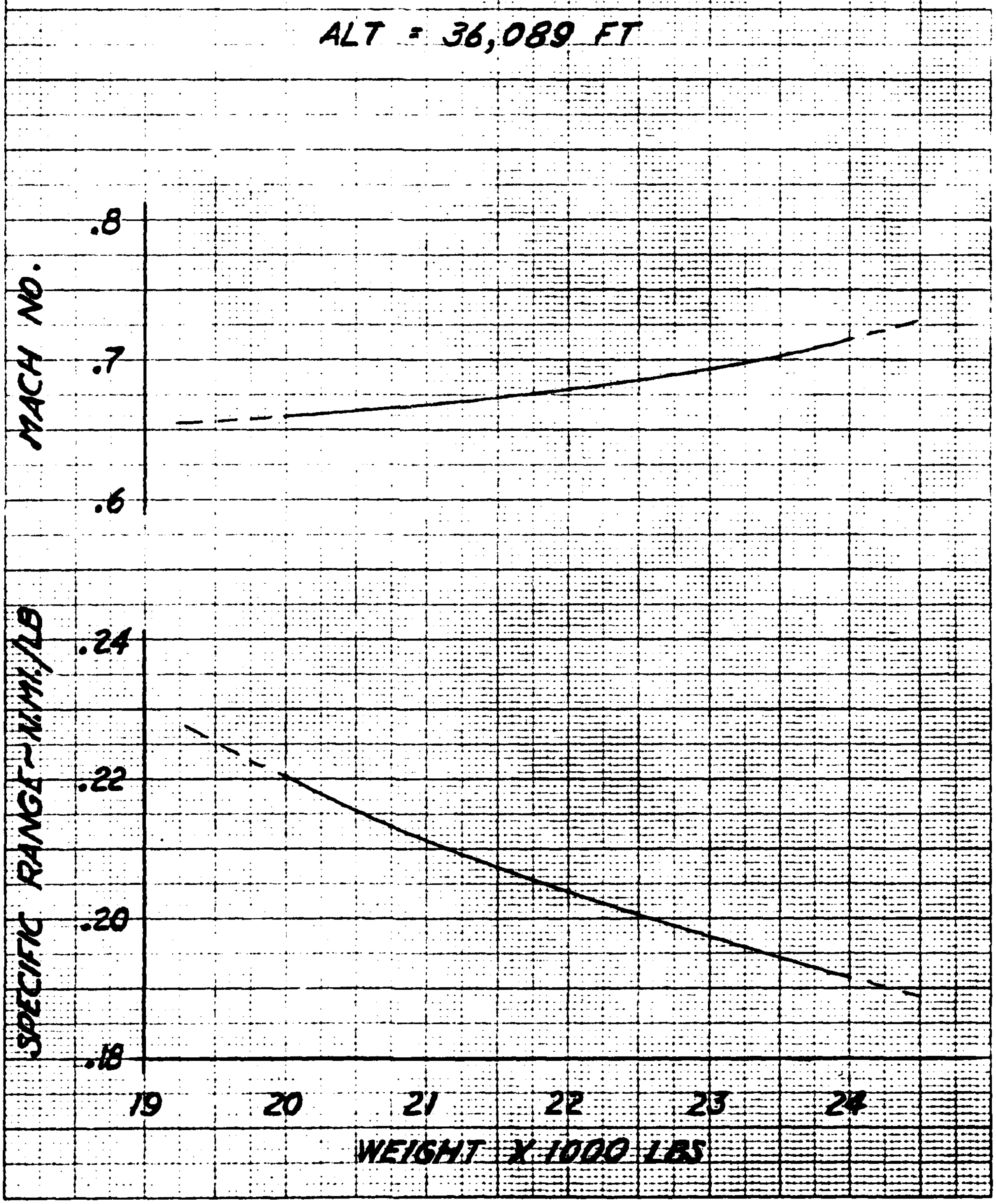


CAS - SETOLS
TIME, FUEL & DISTANCE TO CLIMB
FROM S.L. TO 36,089 FT
CLEAN + 4 PYLONS & TERS + 12 MK-82
MACH NO. = .6
TIME ~ MIN
18
16
14
12
DIST ~ N. MI.
110
100
90
80
70
FUEL ~ LBS
1000
900
800
700
21
22
23
24
25
26
WEIGHT X 1000 LBS

CAS - SETOLS

CRUISE OUT-CAS MISSION

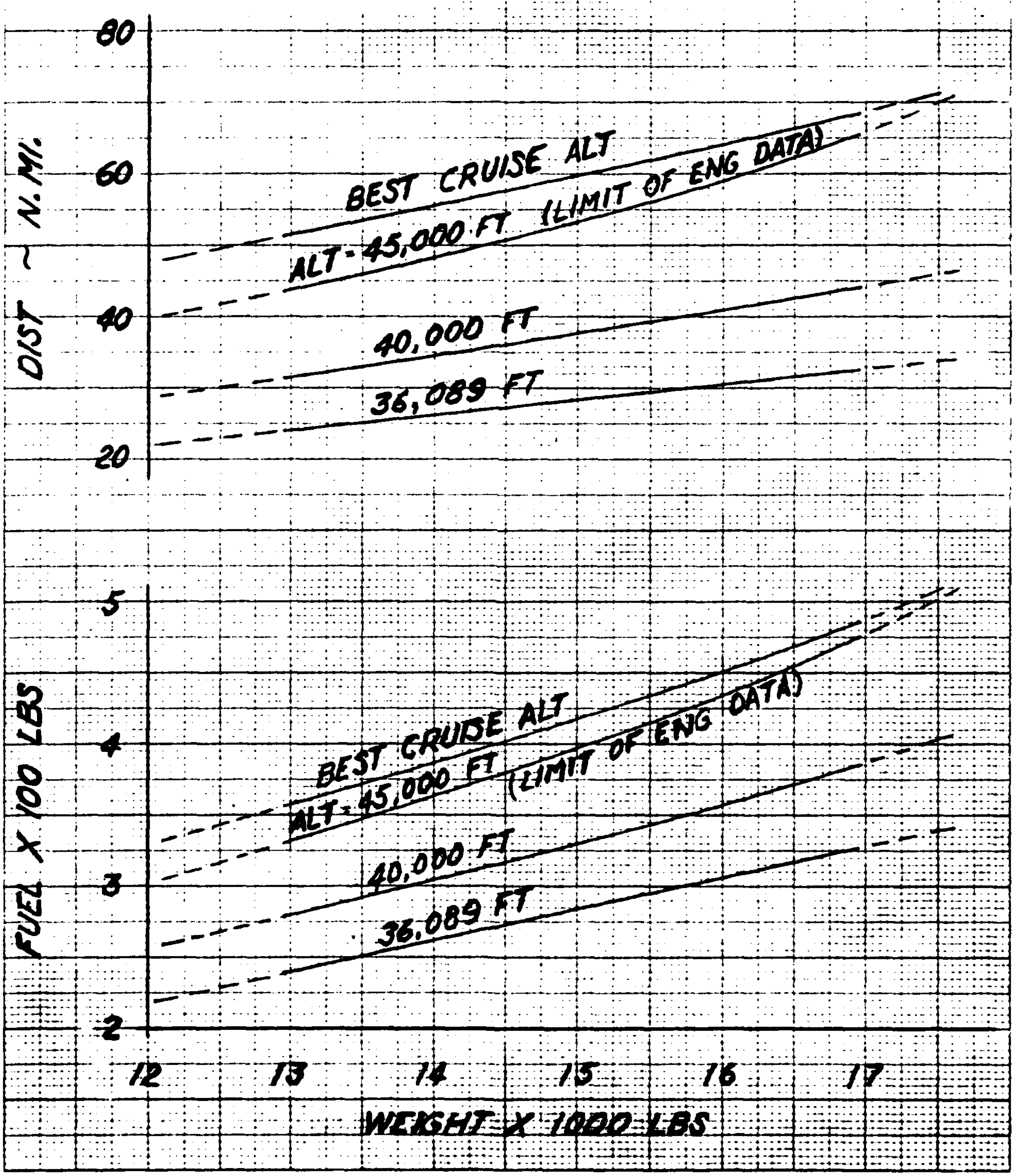
CLEAN + 4 PYLONS & TERS + 12 MK-82



CAS - SETOLS

CLIMB BACK FROM 5000 FT - MACH NO. = .7

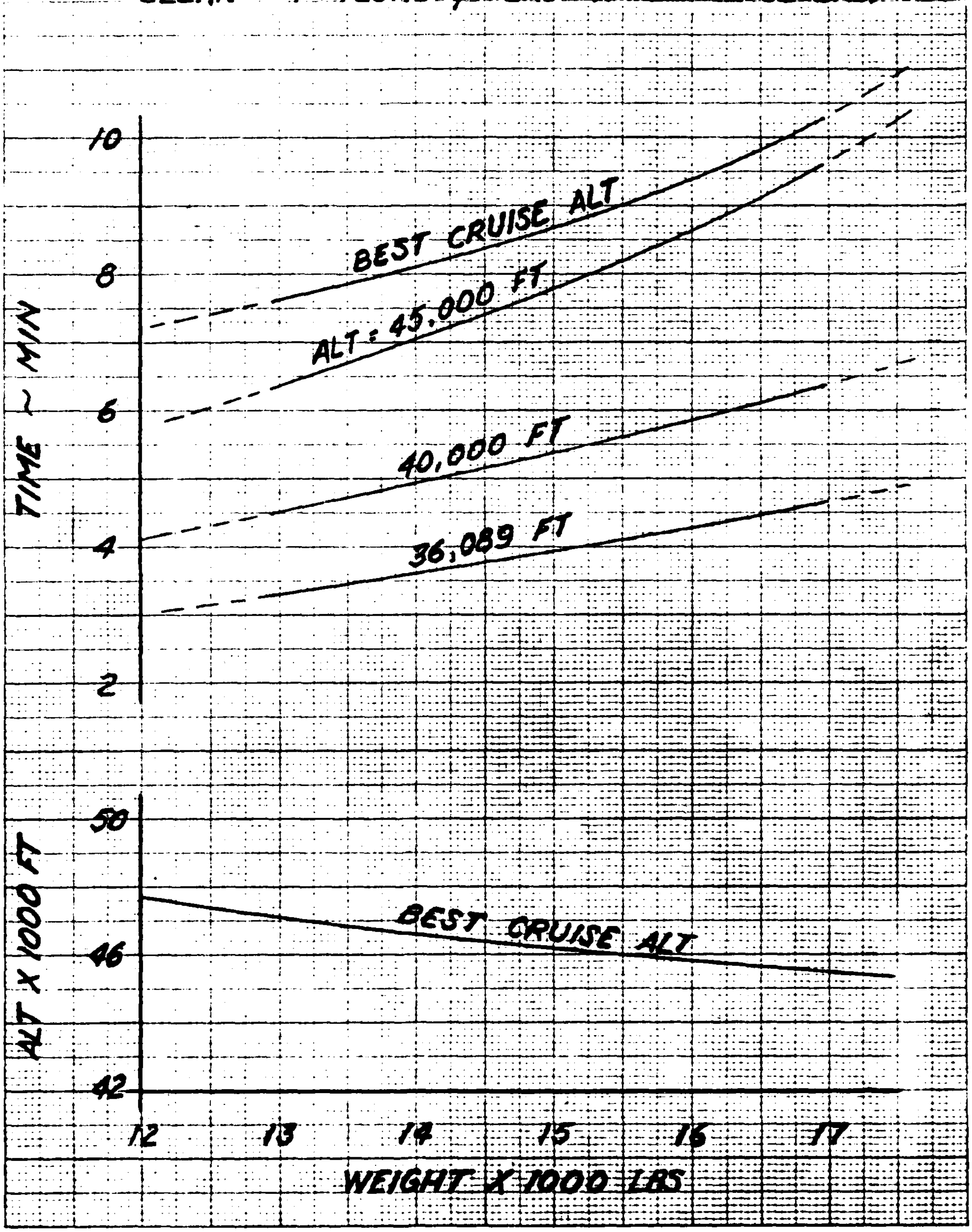
CLEAN + 4 PYLONS & TERS - AFTER MK-82 DROP



CAS - SETOLS

CLIMB BACK FROM 5000 FT - MACH NO.=.7

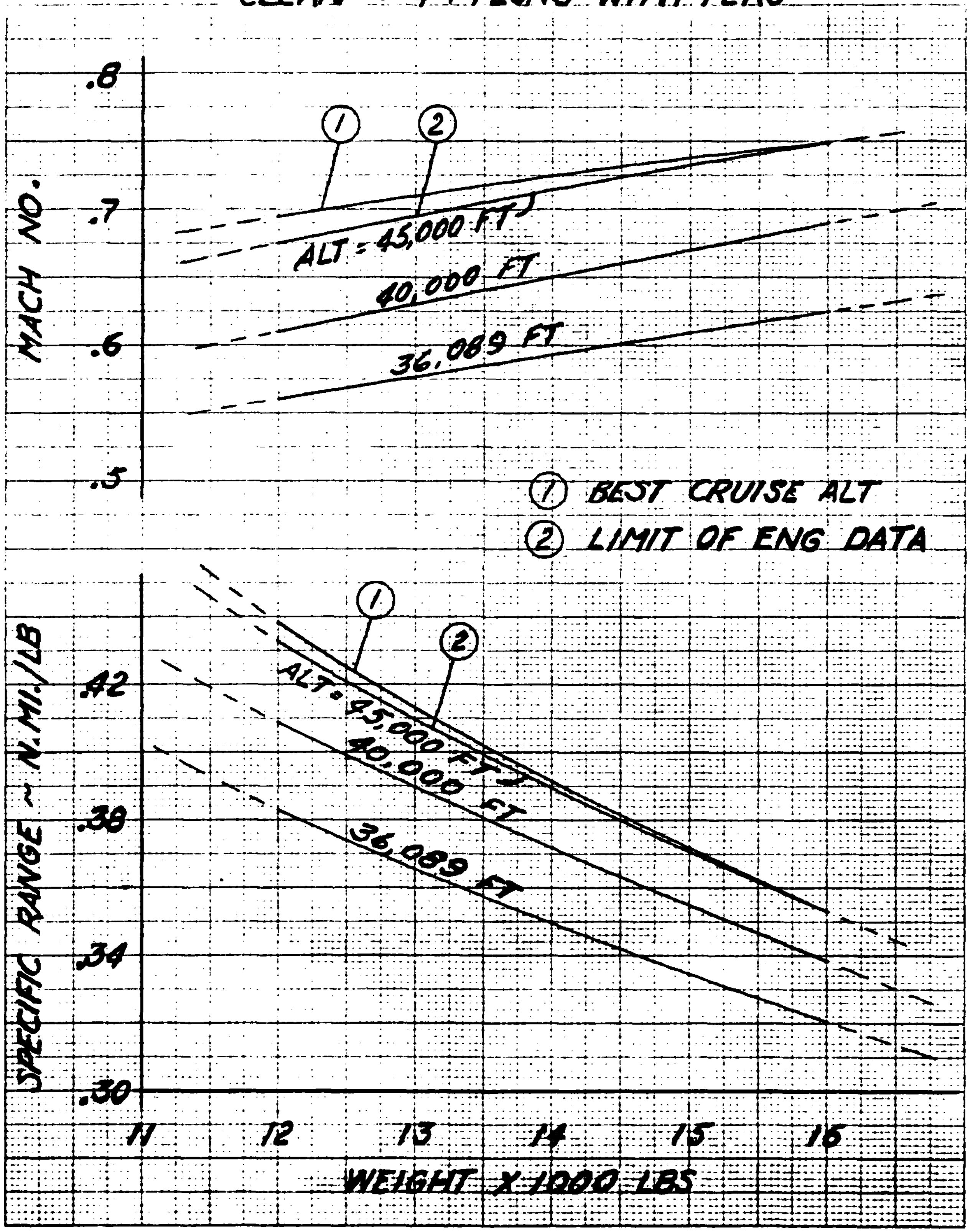
CLEAN + 4 PYLONS & TERS -AFTER MK-82 DROP



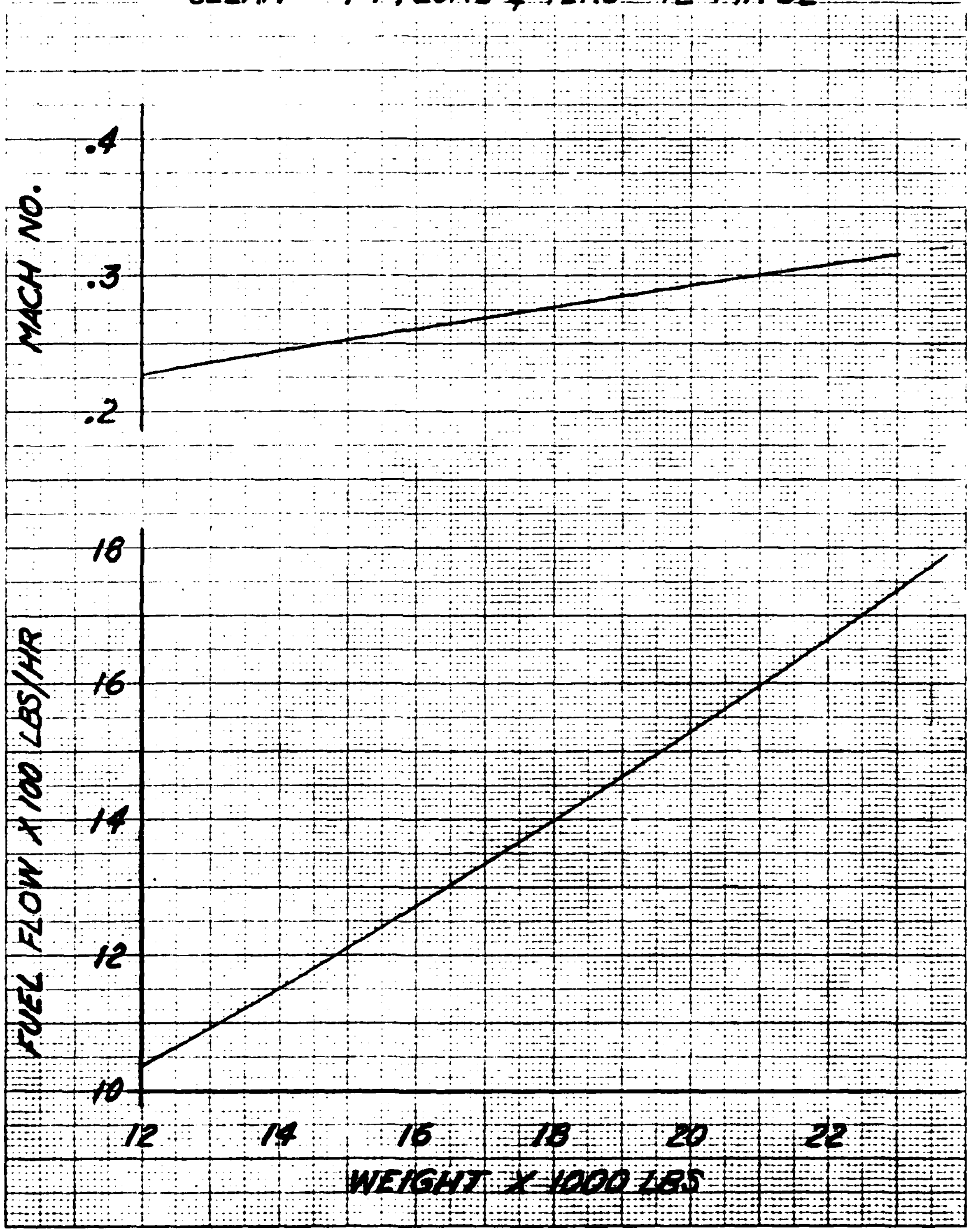
CAS - SETOLS

CRUISE BACK - AFTER MK-82 DROP

CLEAN + 4 PYLONS WITH TERS

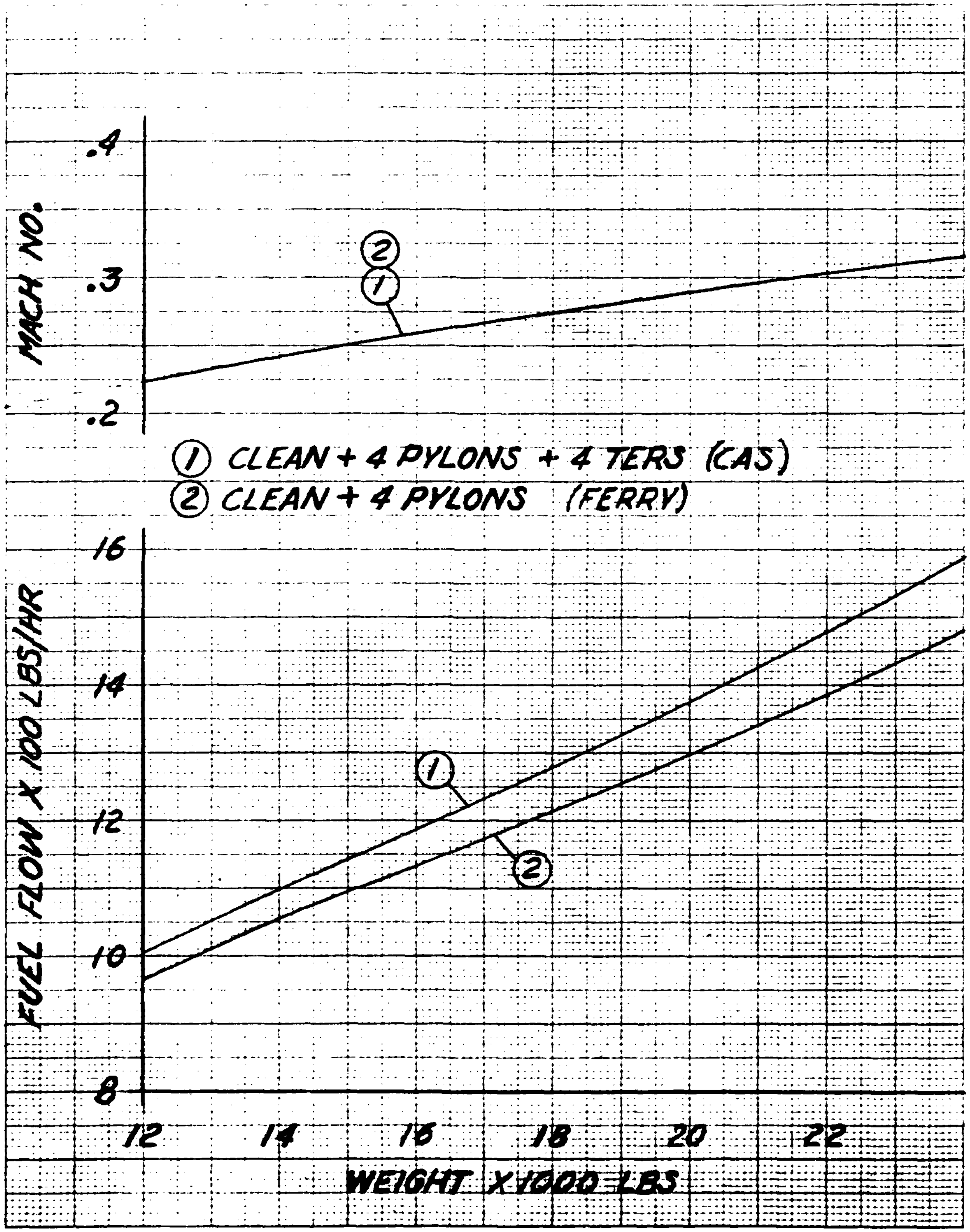


CAS - SETOLS
LOITER AT 5000 FT & $(L/D)_{MAX}$
CLEAN + 4 PYLONS & TERS + 12 MK-82

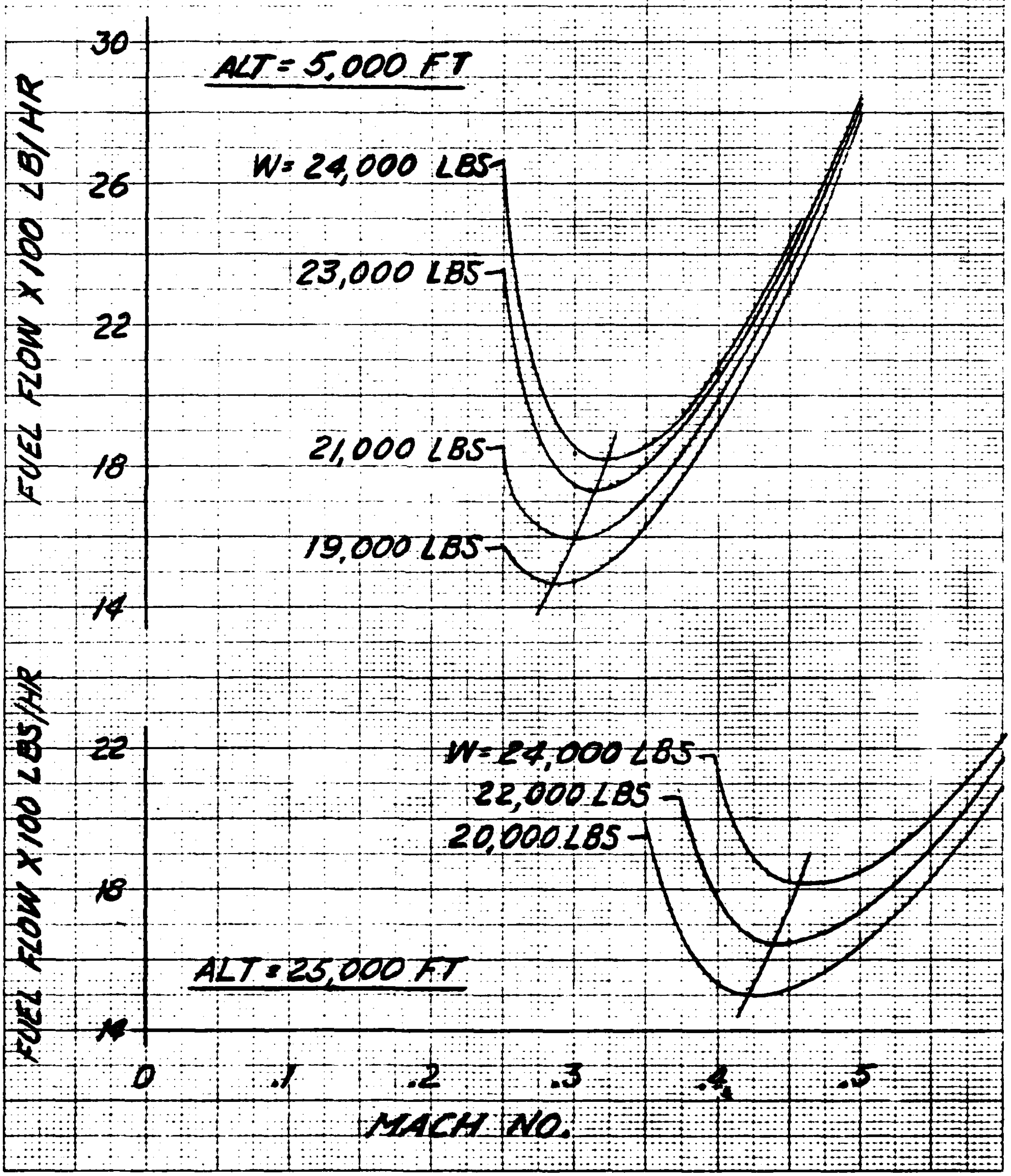


CAS - SETOLS

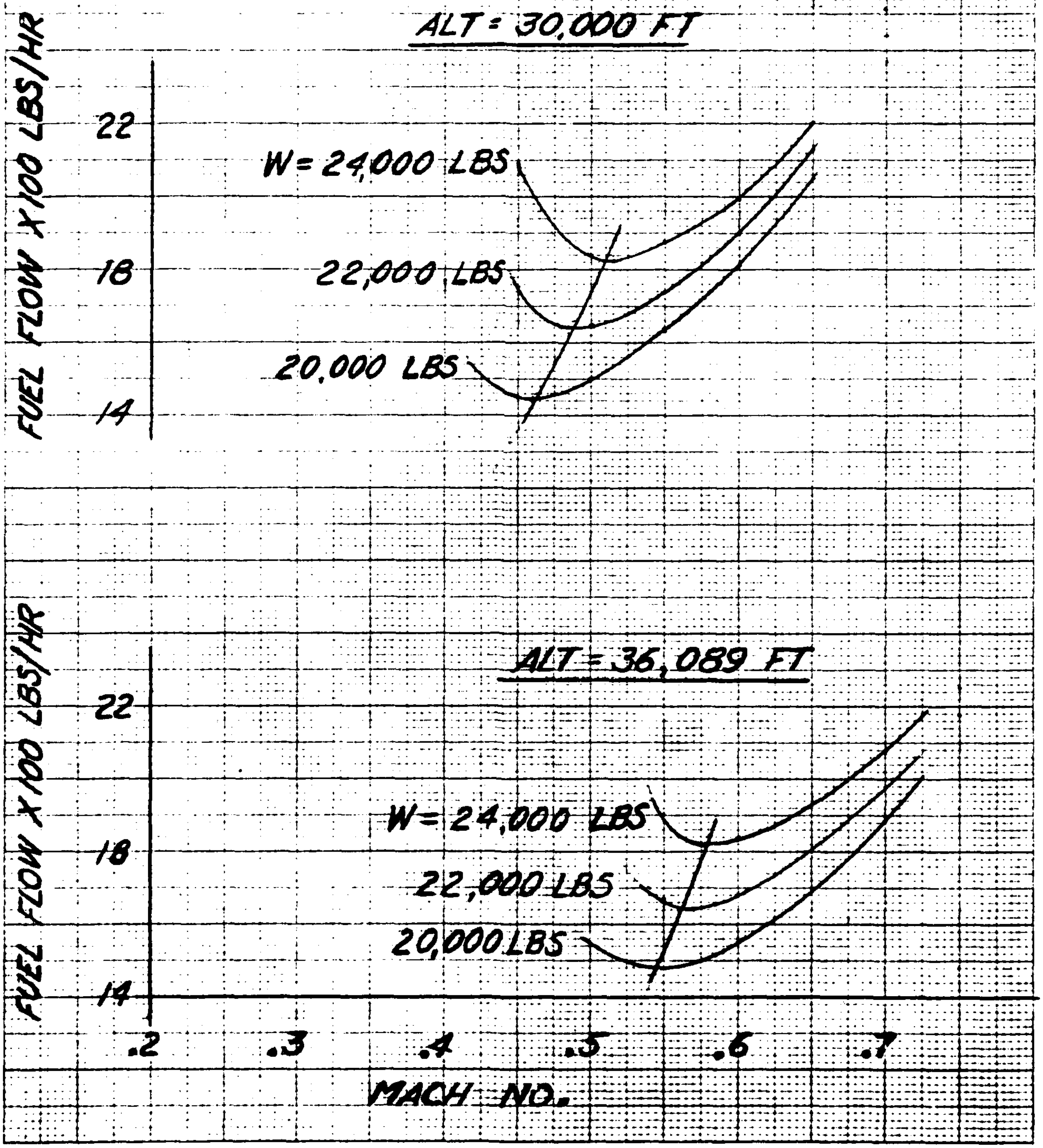
LOITER AT S.L. & $(L/D)_{MAX}$



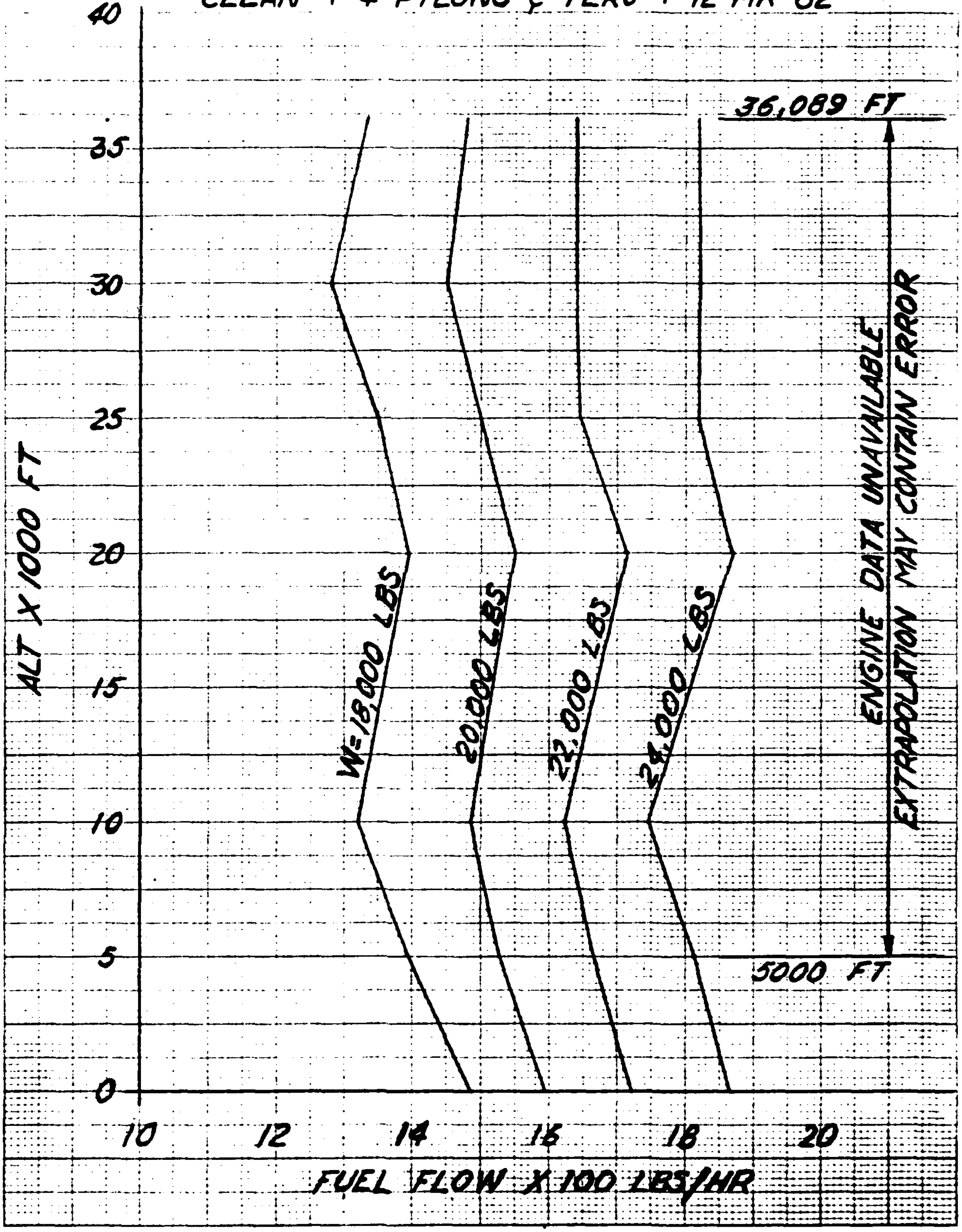
CAS - SETOLS
LOITER OR MAX ENDURE. FUEL FLOW
CLEAN + 4 PYLONS & TERS + 12 MK-82



CAS - SETOLS
LOITER OR MAX ENDURE. FUEL FLOW
CLEAN + 4 PYLONS & TERS + 12 MK-82



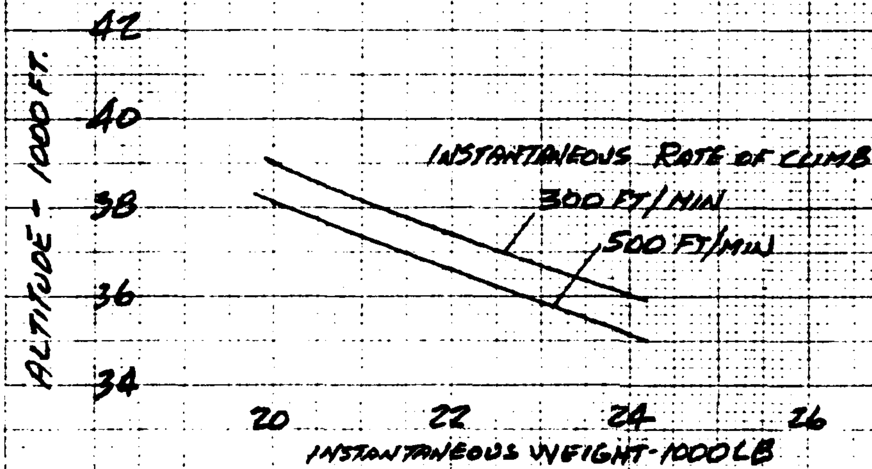
CAS - SETOLS
MAX ENDURANCE (MIN FUEL FLOW)
CLEAN + 4 PYLONS & TERS + 12 MK-82

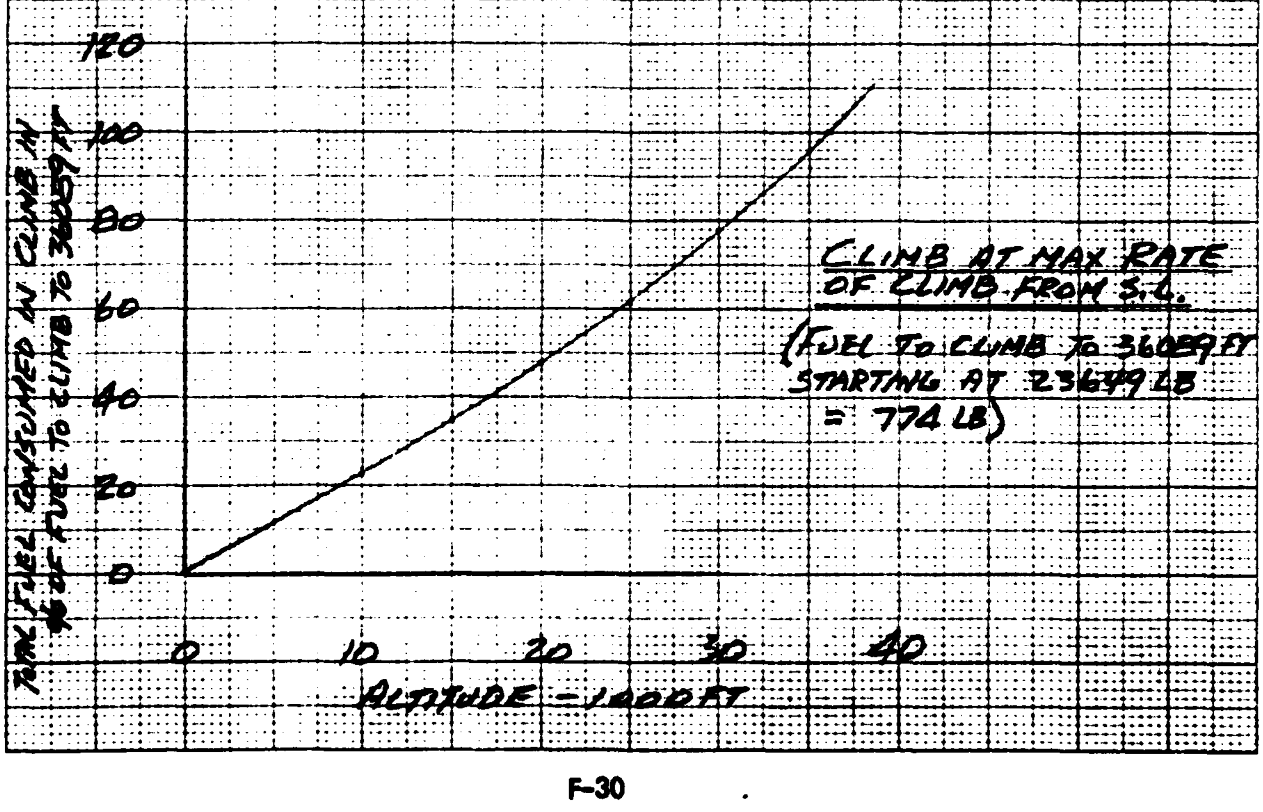


CAS · SETOLS

CLEAN + 4 PYLONS + 4 TERS + 12 MK 82

SERVICE CEILING DETERMINATION
BASIC DATA



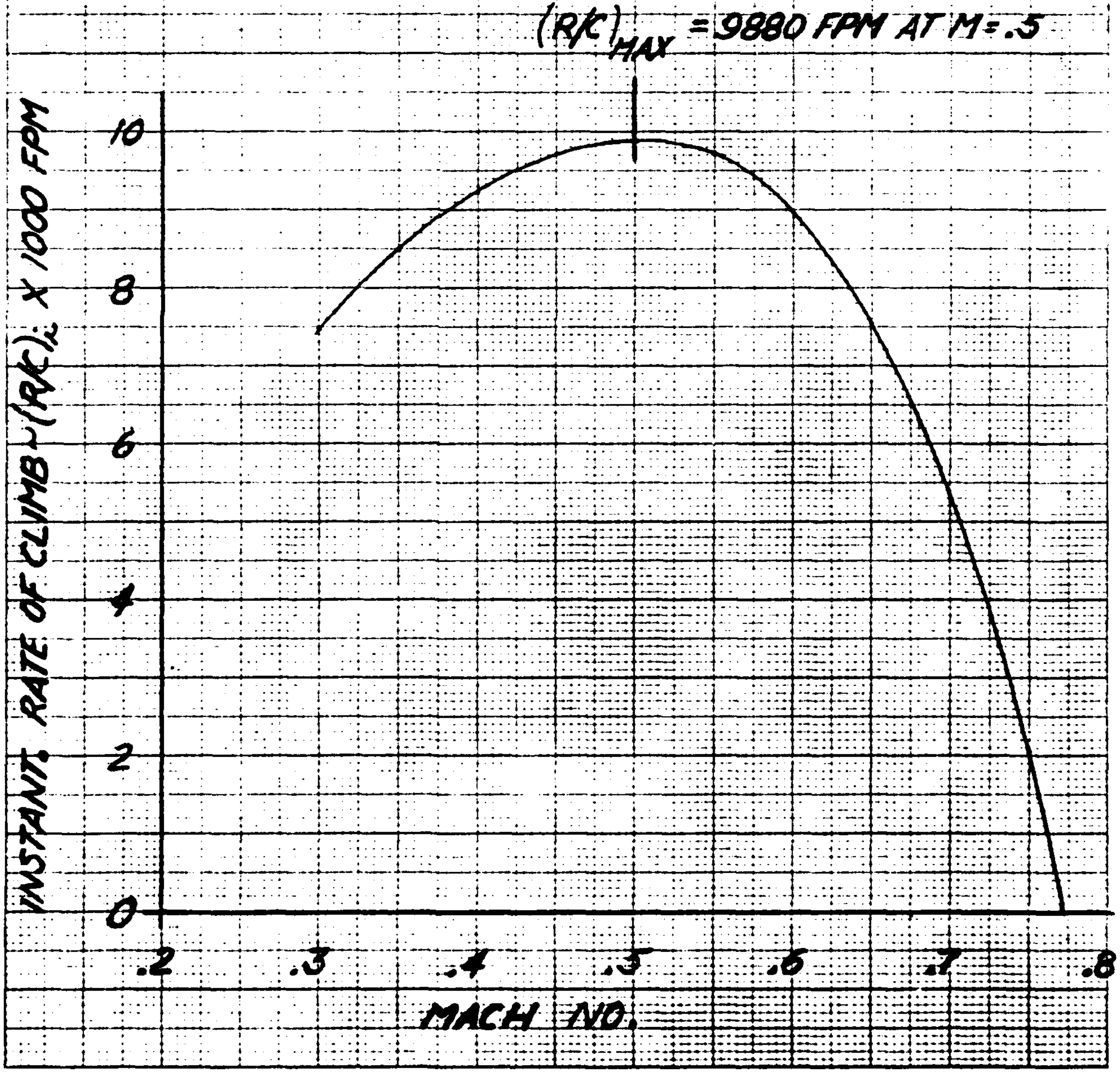


CAS - SETOLS

MAX RATE OF CLIMB AT S.L.

CLEAN + 4 PYLONS & TERS + 12 MK-82

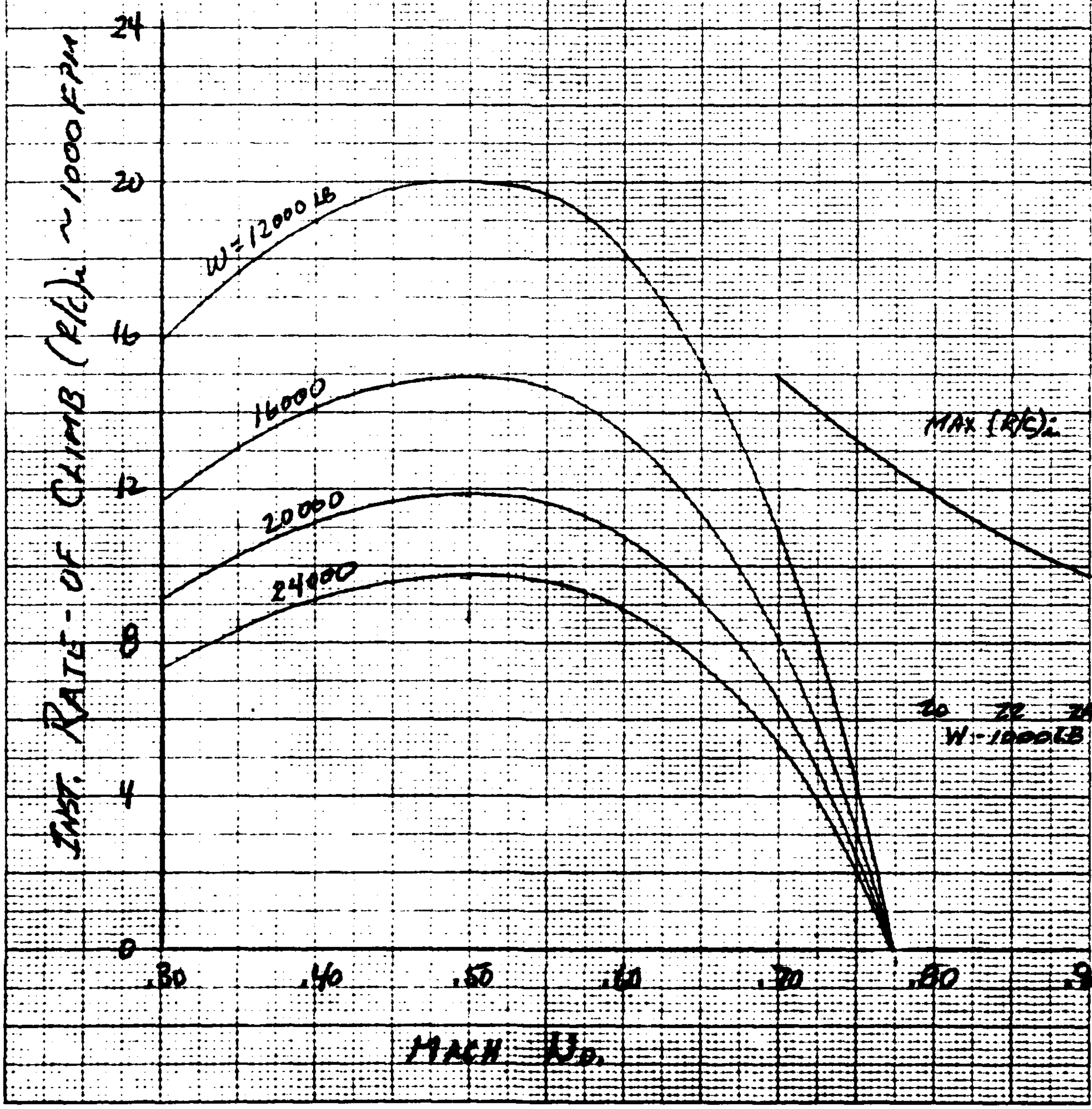
WT = 23,649 LBS = 24,300 LESS 5 MIN FUEL AT T_{MAX}



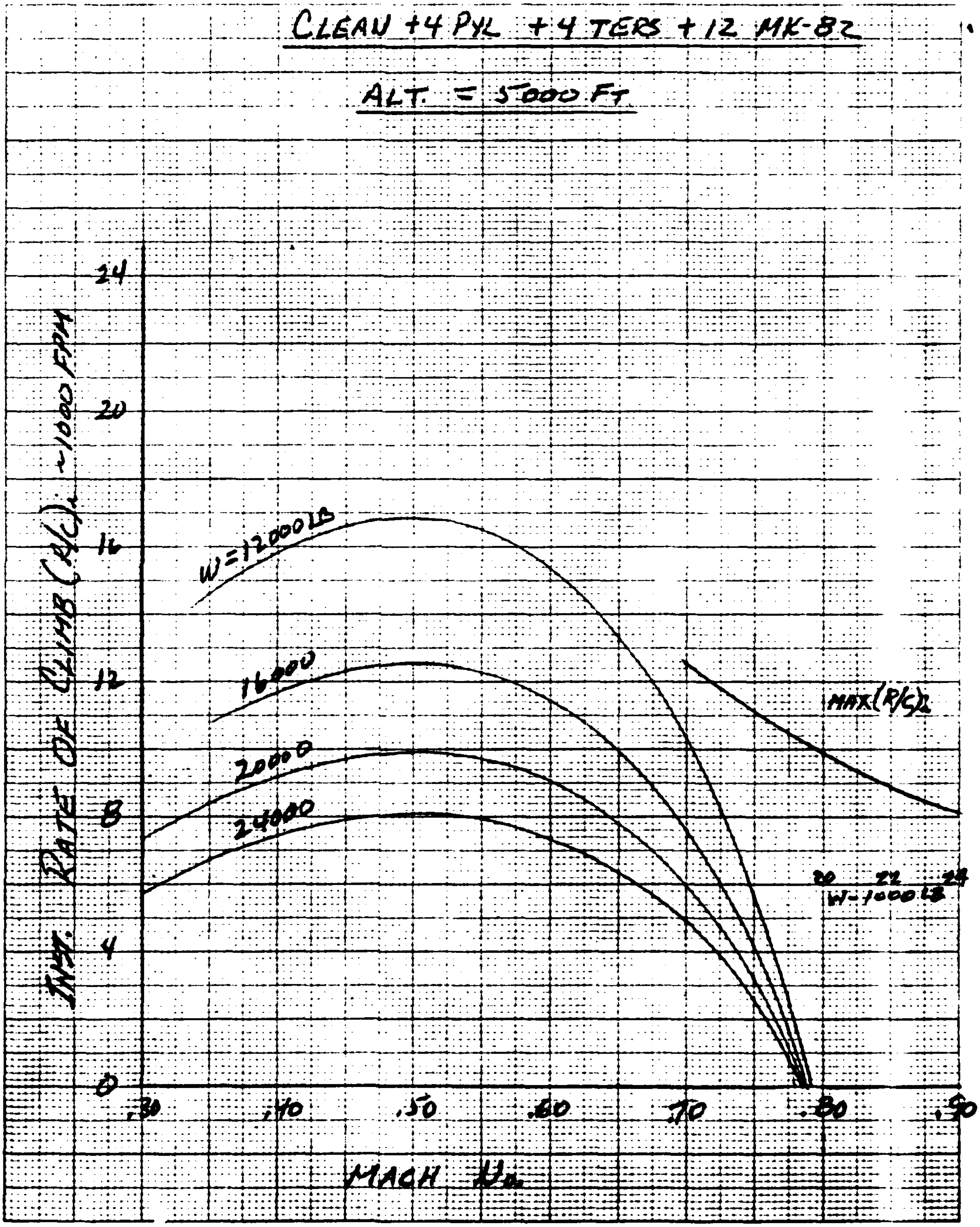
CAS - SETOLS

CLEAN + 4PYL + 4 TERS + 12 MK-82

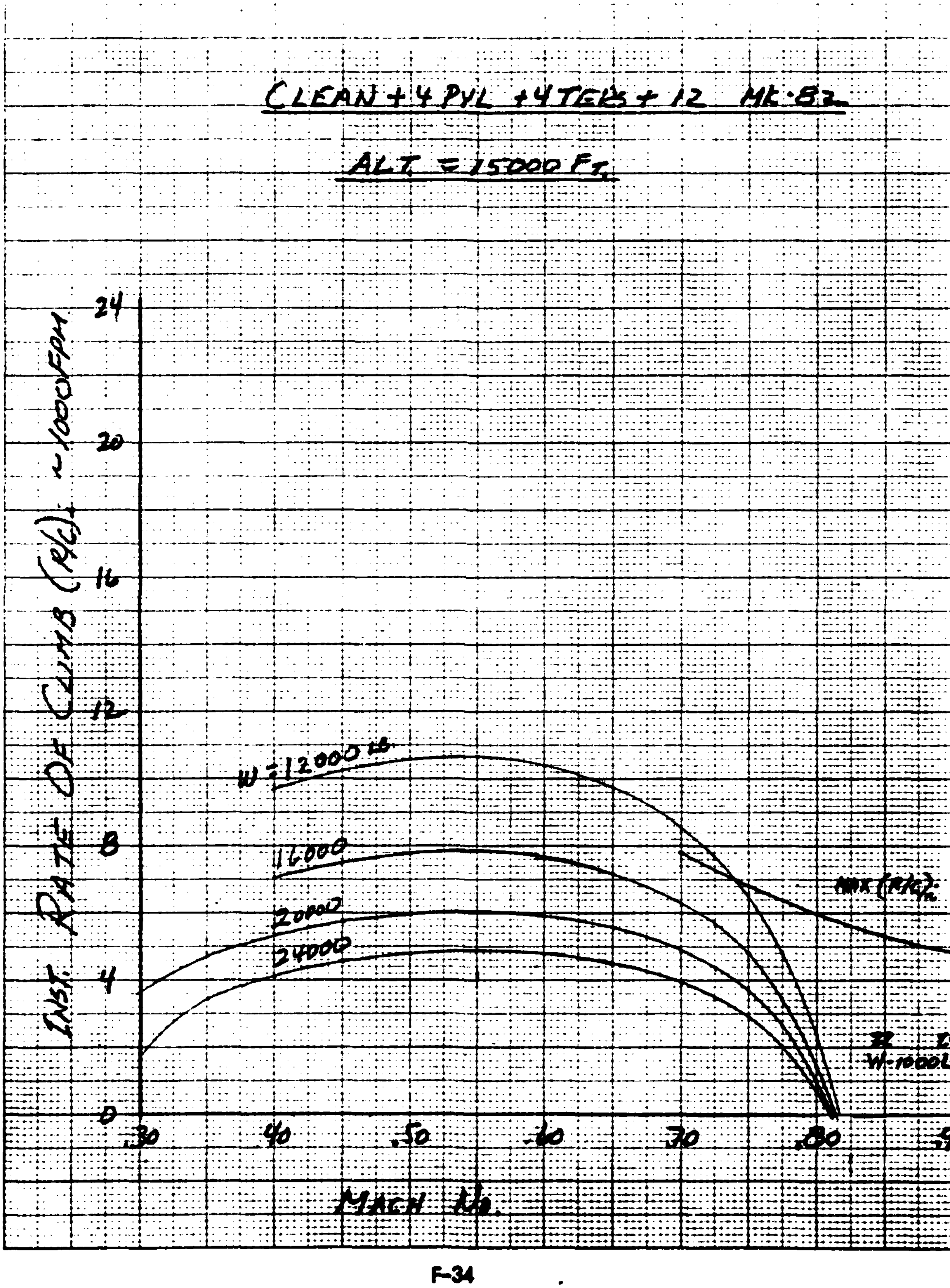
SEA LEVEL



CAS - SETOLS


CLEAN +4 PYL +4 TERS +12 MK-82
ALT. = 5000 FT
INST. RATE OF CLIMB (R/C) ~1000 FPM
24
20
16
12
8
4
0
W=12000LB
16000
20000
24000
MAX (R/C)
20 22 24
W-1000LB
.30
.40
.50
.60
.70
.80
.90
MACH NO.

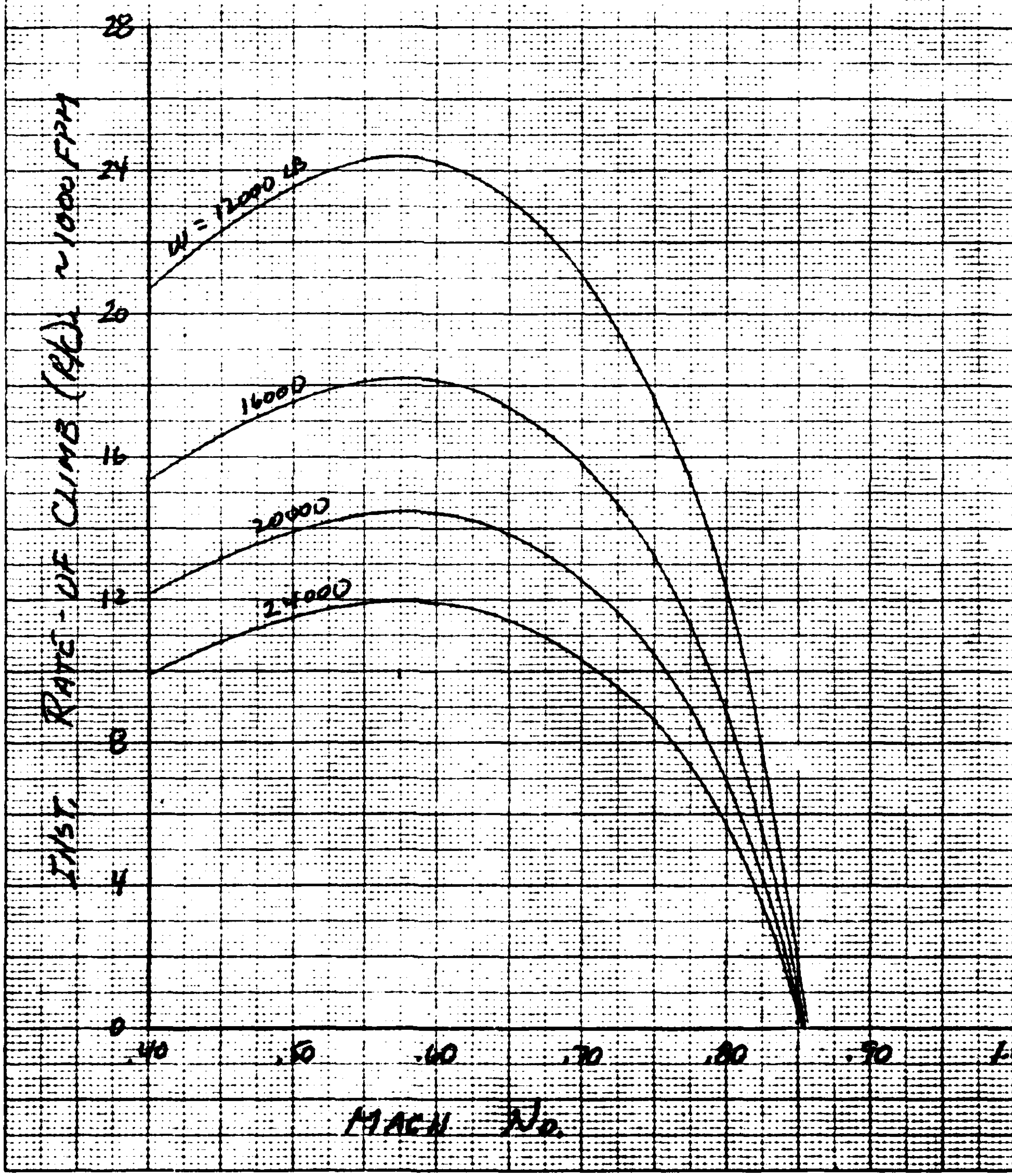
CAS - SETOLS


CLEAN + 4 PYL + 4 TERS + 12 MK-82
ALT. = 15000 FT.
INST. RATE OF CLIMB (R/C)i ~ 1000 FPM
24
20
16
12
8
4
0
W = 12000 LB
16000
20000
24000
MAX (R/C)i
W-1000 LB
.30
.40
.50
.60
.70
.80
.90
MACH NO.

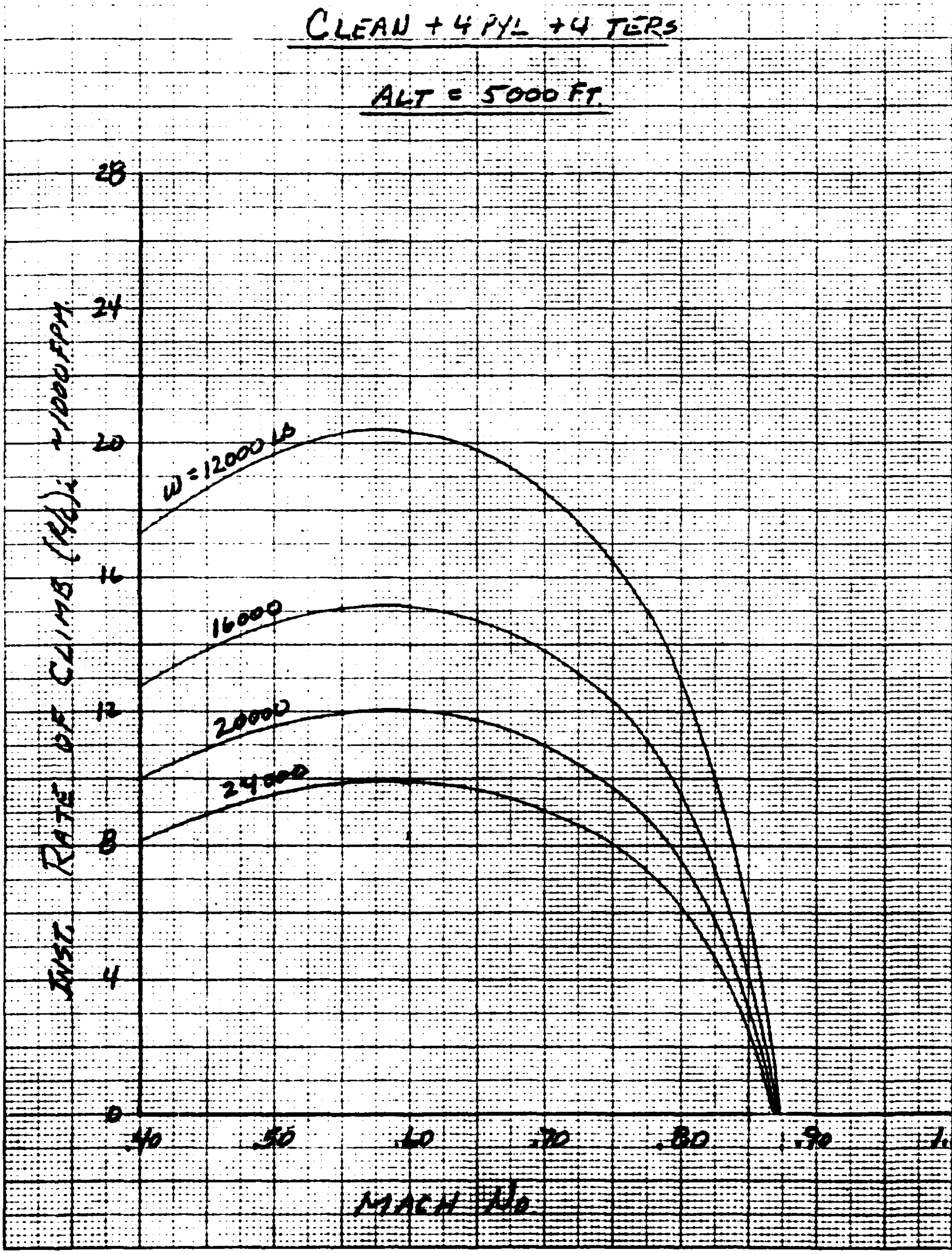
CAS - SETOLS

CLEAN + 4 PYL + 4 TERS

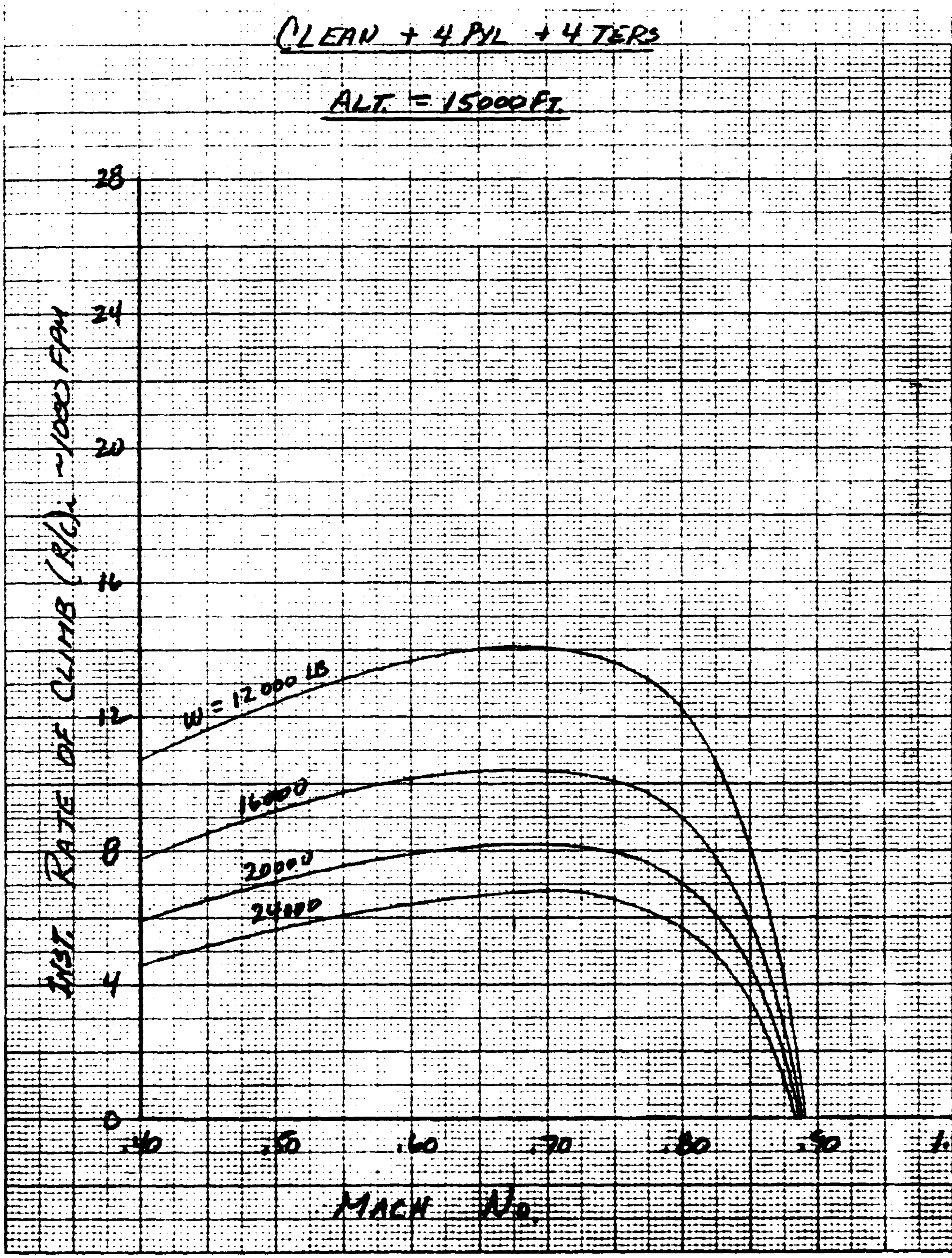
SEA LEVEL



CAS-SETOLS


CLEAN + 4 PYL + 4 TERS
ALT = 5000 FT
INST. RATE OF CLIMB ($\dot{H}_c$), ~1000 FPM
28
24
20
16
12
8
4
0
W = 12000 LB
16000
20000
24000
.40
.50
.60
.70
.80
.90
1.0
MACH No.

CAS - SETOLS

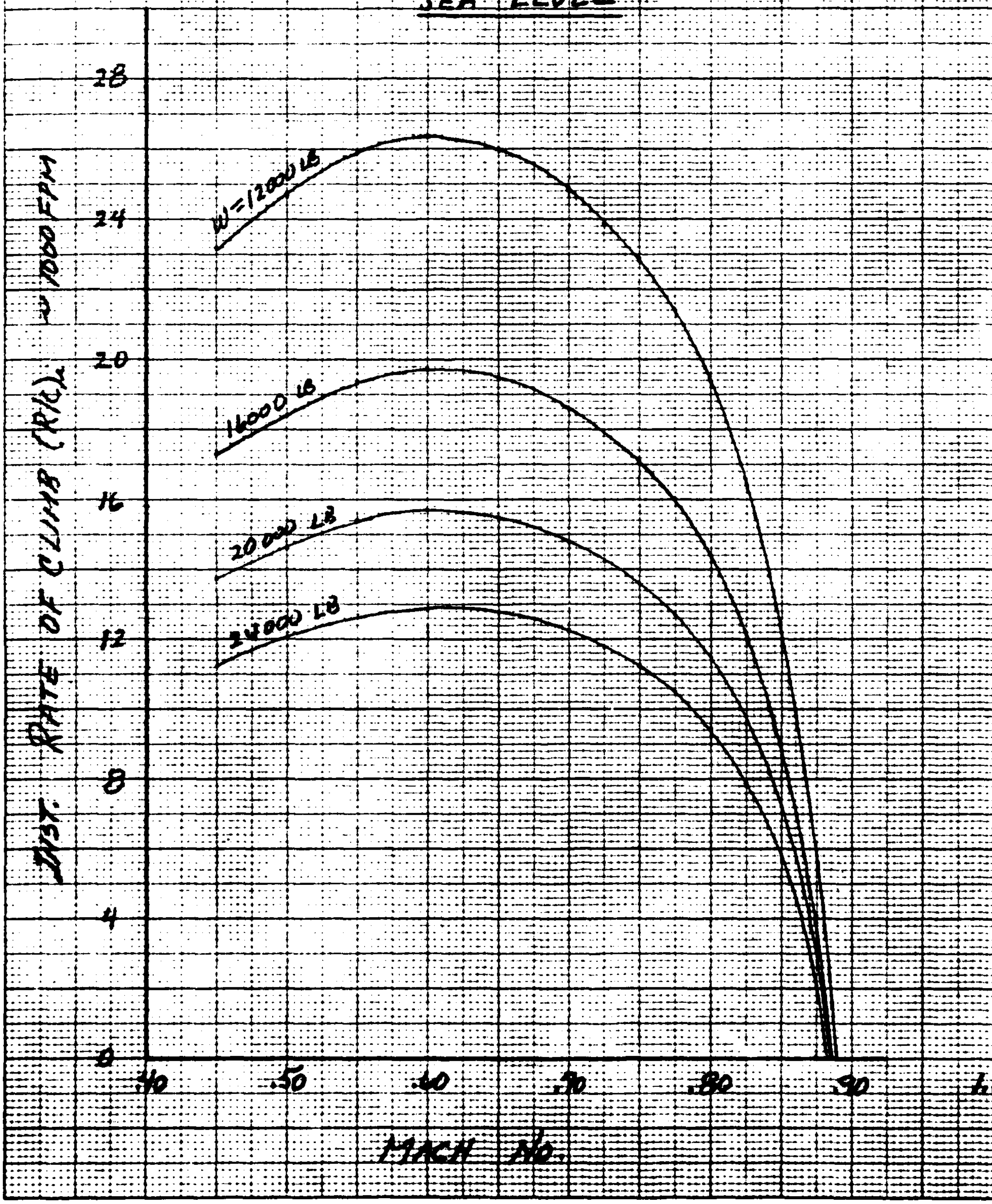


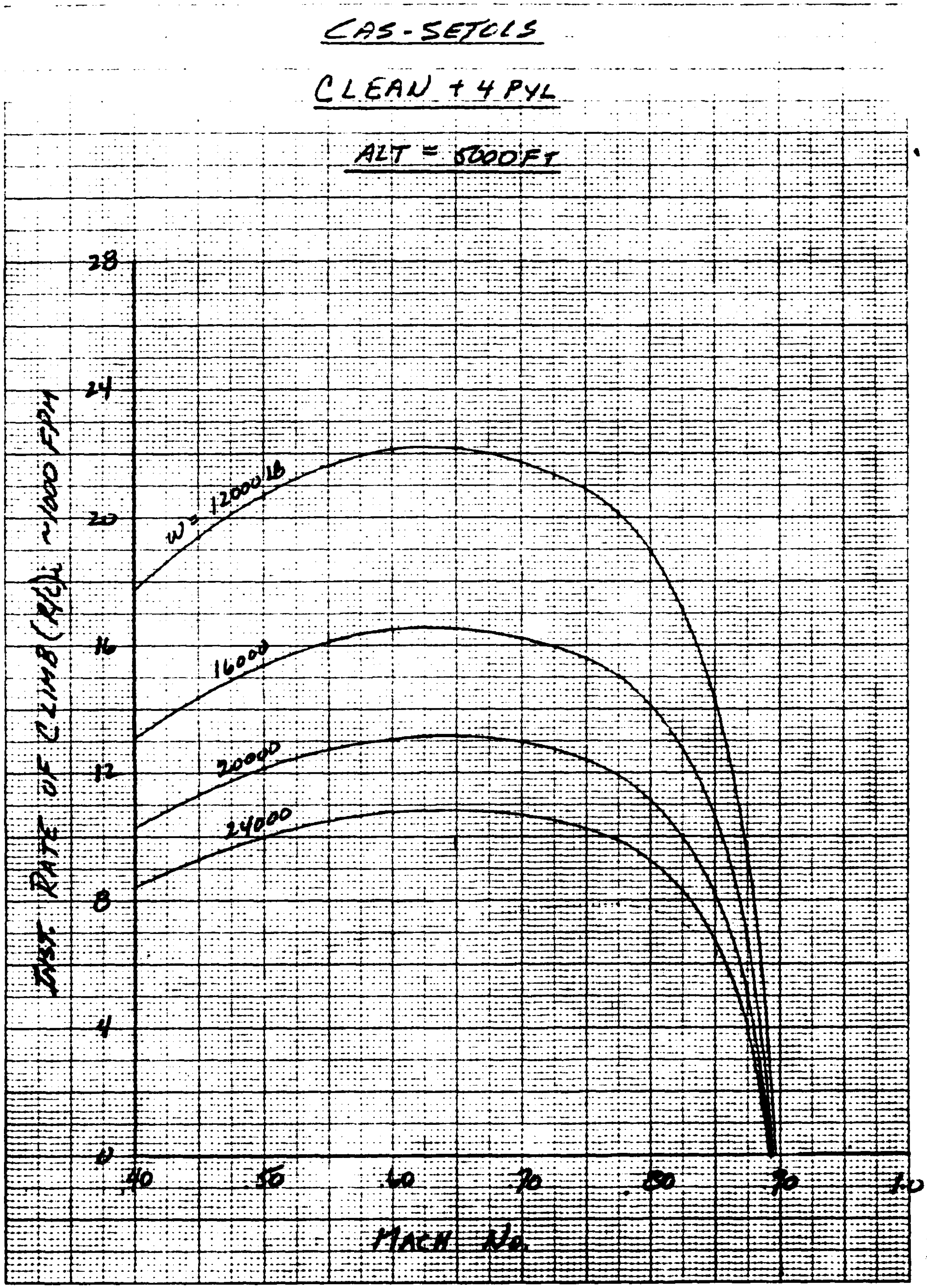
CLEAN + 4 PYL + 4 TERS
ALT. = 15000 FT.
INST. RATE OF CLIMB (R/C)i ~1000 FPM
MACH NO.
W = 12000 LB
16000
20000
24000
0
4
8
12
16
20
24
28
.40
.50
.60
.70
.80
.90
1.0

CAS-SETOLS

CLEAN + 4 PYL.

SEA LEVEL

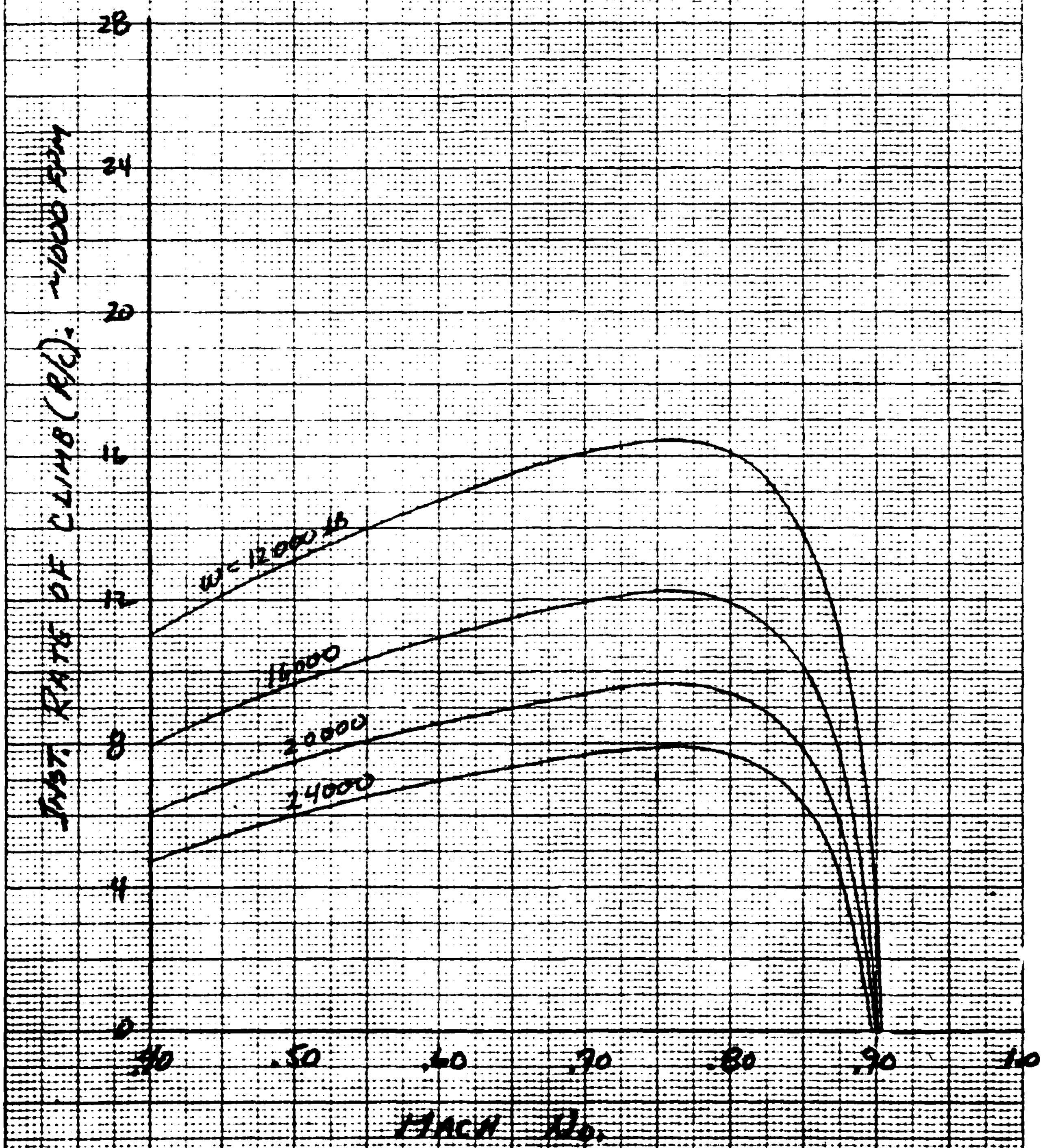



CAS-SETOLS
CLEAN + 4 PYL
ALT = 5000 FT
INST. RATE OF CLIMB (R/C) ~1000 FPM
W = 12000 LB
16000
20000
24000
28
24
20
16
12
8
4
0
.40
.50
.60
.70
.80
.90
1.0
MACH No.

CAS - SETOLS

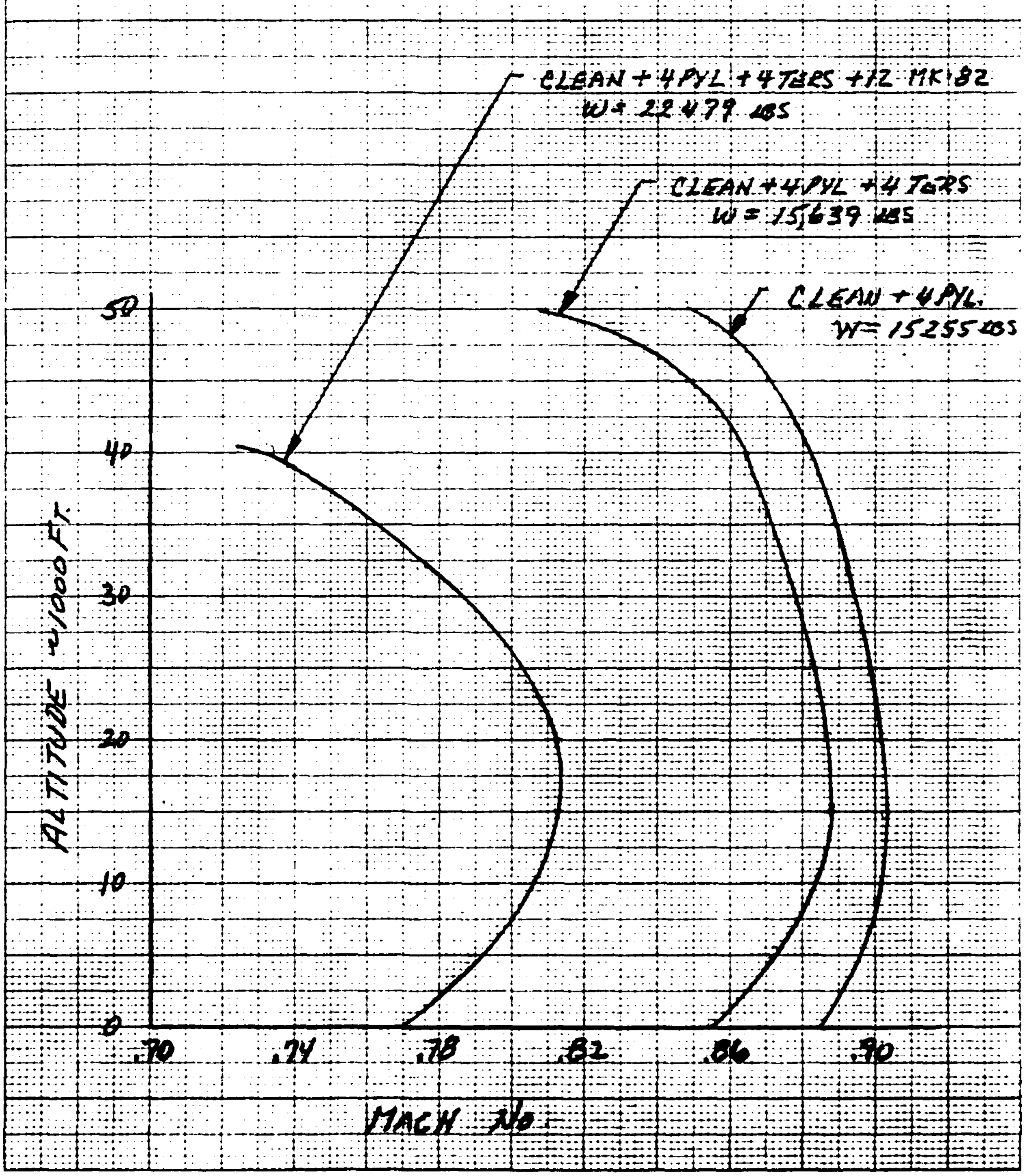
CLEAN + 4 PYL

ALT = 15000 FT



MAXIMUM LEVEL FLT. SPEED

(40% MISSION FUEL USED)



CAS - SETOLS

MAX. SUSTAINED MANEUVER LOAD FACTOR

$W = W_{MAX} - .4$ [CAS MISSION FUEL = 4552 LBS]