UNCLASSIFIED

AD NUMBER
AD365474
CLASSIFICATION CHANGES
TO: unclassified
FROM: confidential
LIMITATION CHANGES
TO: Approved for public release, distribution unlimited
FROM: Distribution authorized to DoD only; Specific Authority; 10 NOV 1965. Other requests shall be referred to Aeronautical Systems Division, Attn: ASZXS, Wright-Patterson ADB, OH 45433.
AUTHORITY
88 CG/SCCMF ltr, 24 Aug 2006; 88 CG/SCCMF ltr, 24 Aug 2006

THIS PAGE IS UNCLASSIFIED

UNCLASSIFIED

AD NUMBER
AD365474
CLASSIFICATION CHANGES
TO: confidential
FROM: secret
LIMITATION CHANGES
TO: Distribution authorized to DoD only; Specific Authority; 10 NOV 1965. Other requests shall be referred to Aeronautical Systems Division, Attn: ASZXS, Wright-Patterson AFB, OH 45433.
FROM: Controlling DoD Organization: Aeronautical Systems Division, Attn: ASZXS, Wright-Patterson AFB, OH 45433.
AUTHORITY
30 Jun 1976 per Grp-3; Aeronautical Systems Div. ltr dtd 10 Nov 1965

THIS PAGE IS UNCLASSIFIED

SECURITY MARKING

The classified or limited status of this report applies to each page, unless otherwise marked.
Separate page printouts MUST be marked accordingly.

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U.S.C., SECTIONS 793 AND 794. THE TRANSMISSION OR THE REVELATION OF ITS CONTENTS IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

NOTICE: When government or other drawings, specifications or other data are used for any purpose other than in connection with a definitely related government procurement operation, the U. S. Government thereby incurs no responsibility, nor any obligation whatsoever; and the fact that the Government may have formulated, furnished, or in any way supplied the said drawings, specifications, or other data is not to be regarded by implication or otherwise as in any manner licensing the holder or any other person or corporation, or conveying any rights or permission to manufacture, use or sell any patented invention that may in any way be related thereto.

CATALOGED BY DDC
AS AD No. 365474

SECRET

FZM-4176-I
30 June 1964

ADVANCED MANNED PRECISION STRIKE SYSTEM

ADDITIONAL TASK

AFSC PLANNING STUDY NO. 799086

TASK I BASELINE AMPSS CONFIGURATION

TASK II PARAMETRIC STUDIES

(TITLE UNCLASSIFIED)

DDC
RECEIVED
SEP 23 1965
TISIA D

DDC CONTROL
NO. 54392

COPY NO. 14

Best Available Copy

GENERAL DYNAMICS | FORT WORTH

SECRET

GD Control No.
FW, 64/100/588-I

SECRET

FZM-4176-I
30 June 1964

ADVANCED MANNED PRECISION STRIKE SYSTEM

ADDITIONAL TASK

AFSC PLANNING STUDY NO. 799086

Task I Baseline AMPSS Configuration

Task II Parametric Studies

AF33(615)-1174

DDC CONTROL
NO. 54392

GENERAL DYNAMICS/FORT WORTH

Control Number
FW/64/100/588-I

SECRET

TABLE OF CONTENTS

1.0 INTRODUCTION

This report summarizes the studies conducted by General Dynamics/ Fort Worth from early May through June on the AMPSS project. It is in direct response to the additional task and statement of work issued by the Air Force in Exhibit A (16 April 1964) under contract AF33(615)1174, S/A #2 (64-2394).

For convenience the report is issued in two volumes. Volume I consists of the airplane studies conducted in response to Tasks I and II of the statement of work. The avionics, or military subsystem studies, Task III, are shown in Volume II.

The statement of work suggested that the new configuration should be an updated design from those previously studied under the contract modified only to the extent necessary to incorporate the new specified flight envelope.

Configuration 2110, presented in General Dynamics/Fort Worth report FZM-4124, dated 27 April 1964, was used as the baseline configuration. This configuration (2110) was a refinement of the designs shown at the termination of the funded study in February 1964 and documented in report FZM-4038, General Dynamics/Fort Worth.

Since this report covers only that work accomplished under the "additional tasks" it does not in itself provide all of the background studies that lead to the final configuration selection. The reader is urged to refer to reports FZM-4038 and FZM-4124 for more detailed studies, particularly in the area of the airplane structures and subsystems design.

2.0 SUMMARY

2.1 BASELINE CONFIGURATION

A gross weight of 395,000 lbs. is necessary to meet the revised AMPSS requirements with an aluminum airframe. The configuration is essentially the same as that reported in General Dynamics/Fort Worth report FZM-4124. This configuration features a variable sweep wing, a design wing loading 175 lbs. per sq.ft., two weapons bays, four Pratt & Whitney STF200C35.1 engines, with static take-off ratings of 22,900 lbs. each, located in duct nacelles off the lower fuselage to provide favorable interference in supersonic flight. A flight view of the airplane is shown in Figure 2.1-1.

The basic performance for this configuration is summarized in the table below:

	Dash N.Mi.	Total Range Required	Total Range Achieved
Subsonic	2000 @ Mach .85	6300	6320
Supersonic	2000 @ Mach 2.2	3300	4410

As can be seen from the table the subsonic performance is much more difficult to achieve, and in fact, determines the gross weight of the vehicle.

2.2 PARAMETRIC STUDIES

A number of trades were investigated as specified in the statement of work. Some of the more significant results are:

1. Using the optimum amount of titanium in the airframe reduces the airplane gross weight to approximately 360,000 lbs. for the basic mission.
2. A reduction of the take-off distance from 6000 to 5000 yields a point design airplane gross weight in excess of 500,000 lbs.

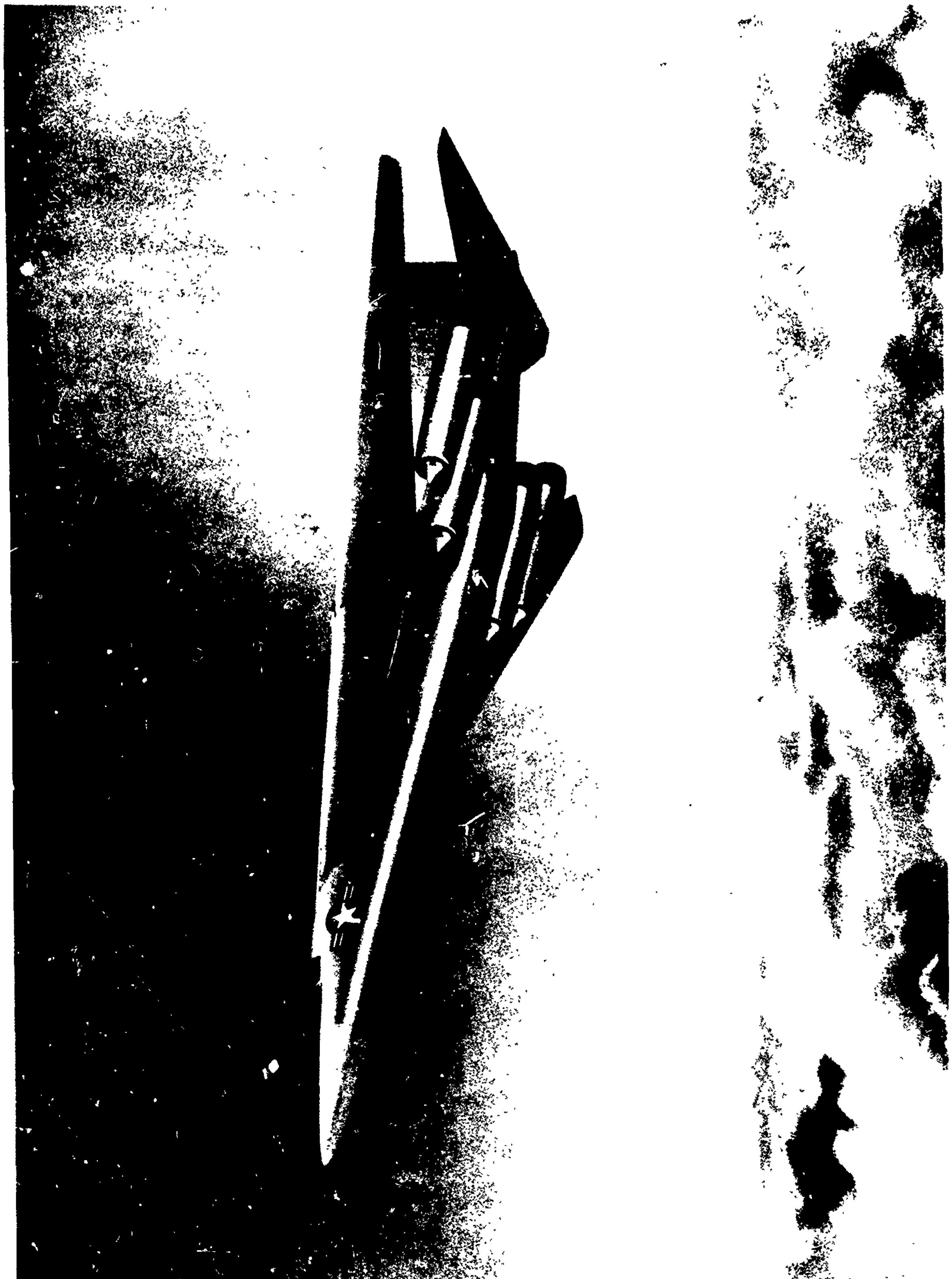


Fig. 2.1-1 AMPSS FLIGHT PHOTO

3. Increasing the take-off distance from 6000 to 7000 feet would reduce the point design airplane gross weight to approximately 340,000 lbs.

4. Combining a 7000-foot take-off dis ce and optimum use of titanium the airplane gross weight is reduced approximately 80,000 lbs. to around 310,000 lbs.

5. The incorporation of a 120 ft./sec. gust in the structural criteria would require an increase of 50,000 lbs. in the point design airplane gross weight.

6. Extending the Mach 1.2 dash capability to several hundred hours has negligible effect on the aircraft gross weight.

7. Reducing the total time below 500 feet from 2500 hours to 500 hours produces a negligible effect.

2.3 AVIONICS (MILITARY SUBSYSTEMS)

During the basic AMPSS study contract, October 1963, through January 1964, General Dynamics, in association with over 30 avionics manufacturers, conducted studies to define an integrated complement of Military Subsystems suitable in performance and achievable in the time period of the early 1970's. Functional areas wherein development efforts were either essential or desirable were defined, and the associated risks and performance implications were reviewed. The basic studies were responsive primarily to the mission of low-altitude penetration emphasized in the RFP, with a capability for high altitude flexibility also included. During the spring months of 1964, additional analyses were made; concentrated effort was placed upon higher altitude target identification, weapons delivery, and survival. Emphasis in the studies conducted during this contract and reported herein has been placed on the further evaluation of the risks, costs, schedules, and confidence levels associated with the recommended developments and on an analysis of the maintainability and reliability of the selected equipments.

Two developments are considered essential to achieve satisfactory mission performance: (1) the real-time correlator and cross-hair laying techniques required for successful operation of the side looking radar and (2) the target alerting technique, buffer storage unit, and associated refinements required for implementation of the integrated displays. These studies will cost approximately $2,125,000 through the completion of feasibility demonstrations.

In the case of less critical developments, where other elements could be used to achieve reduced but acceptable performance, $955,000 is estimated for the work on the IR Line Scanner and the Low Light Level Viewfinder. Refinement of the Decoy Rockets could be achieved for $1 to $8 million dollars, depending upon the extent of flight testing accomplished.

In the case of the noncritical developments, which have been recommended by General Dynamics because of the possibility of greatly improved future performance, an additional $7,000,000 would be needed to establish feasibility. Not enumerated nor discussed are the normal development advancements that are anticipated to be available in the microminiaturization of elements and improvements in the performance of radars, platforms, computers, sensors, and components. It is believed that these improvements will accrue without specific emphasis from the AMPSS program.

3.0 BASELINE CONFIGURATION

3.1 DESIGN SELECTION

3.1.1 Conceptual Design

Configuration 2120 was evolved from SLAMP Configuration 2110 (reported in FZM-4124) after incorporation of new requirements outlined in Air Force document 64ASZXS-32, Exhibit A, and updated STF200C-35.1 engine data package, and a landing gear reflecting the criteria described in SETFL Report 164.

New or modified requirements included an increased sea level dash capability, revised runway criteria, an increased analysis on radar cross-section and IR emission reduction, overpressure and thermal load capabilities, and a specified gust sensitivity.

No external changes to the configurations are required to extend the flight envelope at sea level for either sustained operation at Mach 0.9 or for five minutes of flight at Mach 1.2. Weight penalties are incurred for increased structural and fuel system capabilities.

The new landing gear criteria was incorporated into the design of the airplane by resizing the tires and tire spacing while retaining the same retracting mechanism concept. The envelope for the main landing gear at a gross weight of 354,000 lbs., resulted in approximately the same volume and weight; therefore, no modification was required to the configuration at this weight due to landing gear criteria. Refer to Section 5.0 of this report for detailed design data on the landing gear and flotation studies.

An investigation was made to determine what possible improvements to the configuration could be incorporated to reduce the radar cross-section and IR emissions. The results indicated that substantial improvements could be made by altering the overall conceptual arrangement; however, resulting penalties in performance would occur. These improvements were not incorporated in the design because the time period for this study was limited. Internal changes were made, such as substitution of non-conductive materials, which would not affect the overall conceptual design and arrangement. Refer to Section 6.0 of this report for more detailed data.

3.1.2 Gross Weight Study

The variation of range with gross weight (Figure 3.1-5) was determined by evaluating three designs having the same wing loading, wing geometry, tail volumes, and conceptual arrangements. Configuration 2111A, identical to 2110 except as modified to meet the new requirements, served as a baseline with a gross weight of 354,000 lbs. In addition, Configurations 2112A and 2113A, 450,000 and 275,000 lbs., respectively, were designed and evaluated. This broad range of gross weights was selected in an effort to produce a more accurate shape for the performance growth curve.

The following conceptual arrangement characteristics were retained.

1. Variable sweep wing concept and high wing location on the fuselage, with a W/S = 175 lbs./ft.^{2}, AR = 8.91, λ = .25, Λ_{LE} = 16°/72.5°, NACA 64-012 at pivot and NACA 64-009 tip airfoil sections, pivot location at 16.4% of semispan and 26% of local chord with wing in the Λ_{LE} = 16° position.

2. Fuselages designed in the same manner with the same 4-man crew compartment, electronic systems, nose radome and navigational radar. Landing gear designed to new criteria, mounted and retracted into the fuselage between split bomb bays.
3. Horizontal and vertical tails located in a conventional manner using a constant tail volume and tail length. (C_{VT} = .0755, C_{HT} = .8.) Tail geometry also remained constant.
4. P&W STF200C-35.1 engines located in siamese nacelles mounted on the fuselage below the wing and scaled to produce the same thrust/weight ratio for the different design gross weights.

Plots of aerodynamic wetted area (A_{wet}) and weight ratio (Wo/We) versus gross weight are plotted in Figures 3.1-1 and 3.1-2 to show the consistency of the design data.

3.1.3 Wing Loading Study

Due to significant changes in engine performance characteristics, it was felt advisable to reevaluate the effects of wing loading on subsonic range performance. Two additional airplanes, Configurations 2114 and 2115 with wing loadings of 187 lbs./sq.ft. and 163 lbs./sq. ft., respectively, were designed for this study employing the building block technique in lieu of complete layouts.

The fuselage and its contents from Configuration 2111A was used for both airplanes. Each wing with its corresponding tail was mated to this fuselage. Fuel tanks were filled to their maximum capacity to ensure comparison on a maximum density basis and engines were sized for the required 6000-foot take-off.

The aerodynamic wetted areas and weight ratios for these two airplanes were then plotted on the curves derived from the gross weight study (Figures 3.1-3 and 3.1-4) to provide characteristics for a family with a gross weight of 354,000 lbs. and wing loadings of 163, 175, and 187 lbs. per square foot. Thrust to weight effects were determined by varying engine scale on these airplanes, and the range contours and take-off and dash restrictions of Figure 3.1-6 were established. From an inspection of this plot, it can be seen that a wing loading of 175 lbs./ft.^{2} yields maximum range for an airplane designed to take off in 6000 feet.

The selection of the wing loading must also be predicated on ride quality considerations. The gust sensitivity goal established for the AMPSS airplanes was 0.02 $\frac{\text{RMS g's}}{\text{RMS ft./sec.}}$ at the crew station at mid-dash or target gross weight. For an airplane with a take-off gross weight of 395,000 lbs. and a wing area of 2257 sq.ft. (W/S = 175), the mid-dash gross weights are 253,000 and 285,000 lbs. for the design unrefueled and refueled missions, respectively. The gust sensitivities corresponding to these weights are 0.0216 and 0.0194, bracketing the requirement and, therefore, considered acceptable.

3.1.4 Final Configuration Selection

A design gross weight of 395,000 lbs. (Configuration 2120) was selected from the performance growth curve of subsonic range versus gross weight. This design gross weight is required to meet the range requirements of 6300 n.mi. total with 2000 n.mi. at Mach .85 at sea level.

Figures 3.1-1 and 3.1-2 were used as a criteria for designing the final configuration to assure accurate performance predictions

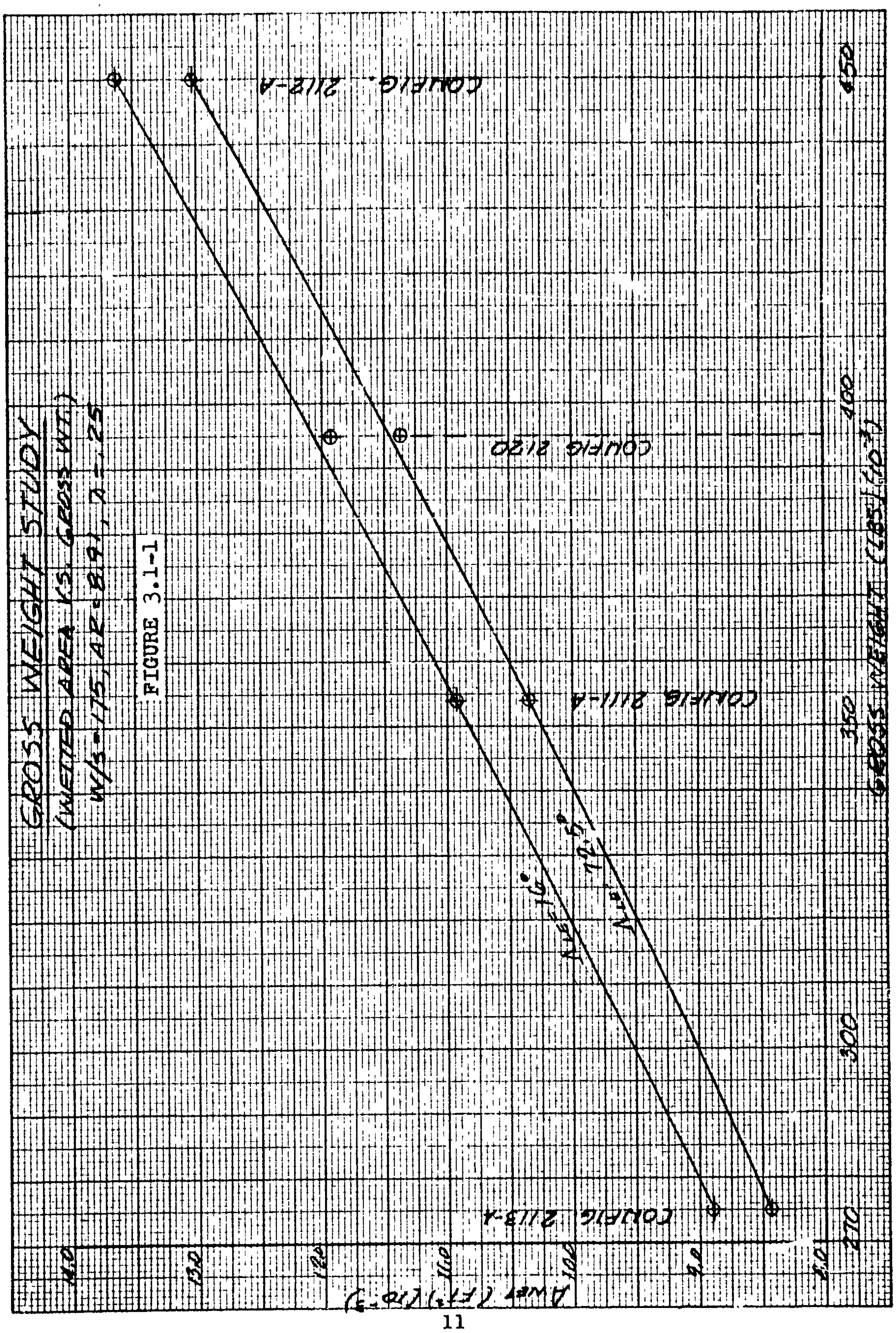

GROSS WEIGHT STUDY
(WETTED AREA VS. GROSS WT.)
W/S=175, AR=8.91, λ=.25
CONFIG. 2112-A
CONFIG 2120
CONFIG 2111-A
CONFIG 2113-A
ΛLE=16°
ΛLE=12.5°
14.0
13.0
12.0
11.0
10.0
9.0
8.0
Awet (FT²)(10⁻³)
270
300
350
400
450
GROSS WEIGHT (LBS)(10⁻³)

FIGURE 3.1-1

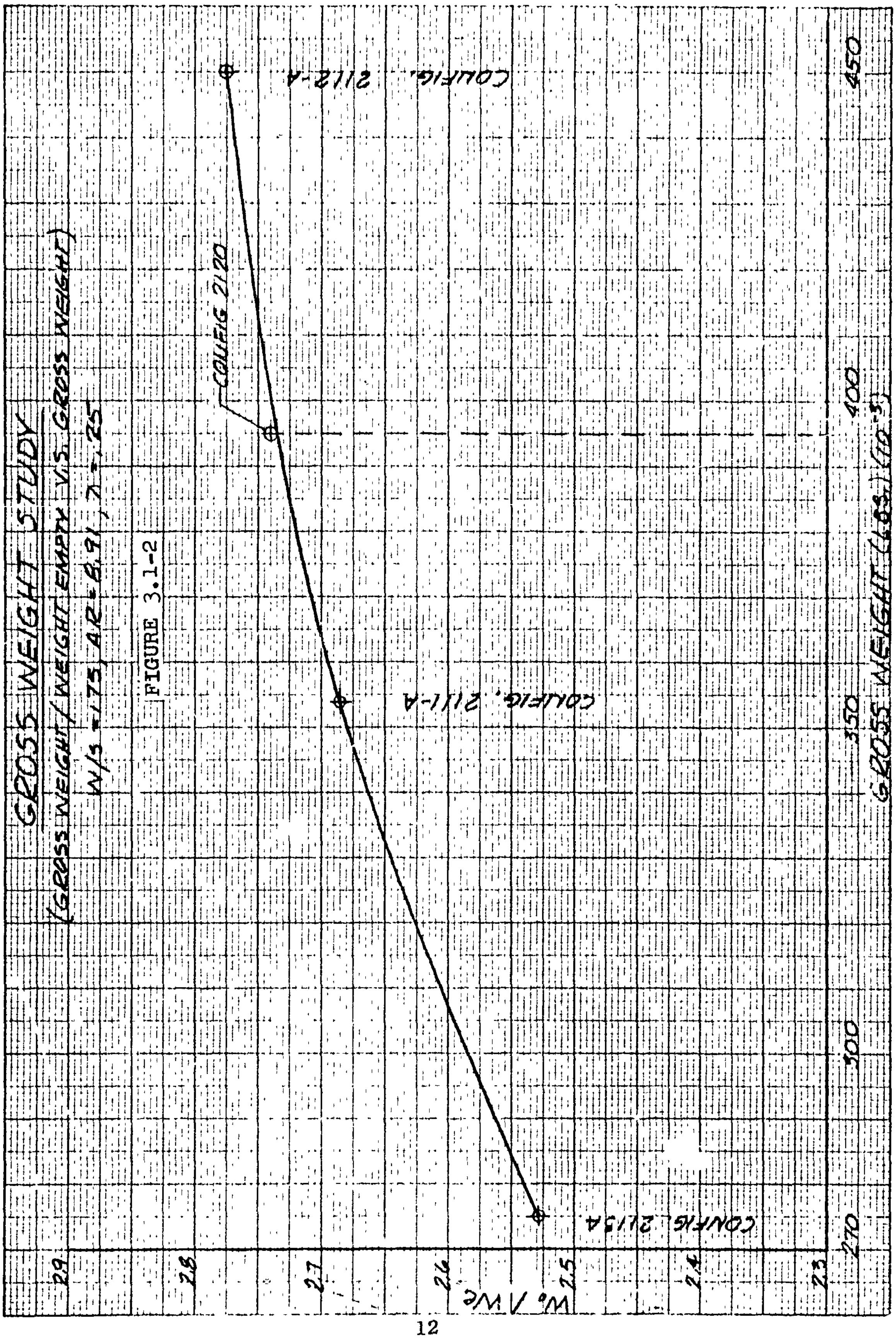

GROSS WEIGHT STUDY
(GROSS WEIGHT / WEIGHT EMPTY V.S. GROSS WEIGHT)
W/S = 175, AR = 8.91, λ = .25
CONFIG. 2112-A
CONFIG 2120
CONFIG. 2111-A
CONFIG. 2113A
Wo / We
2.9
2.8
2.7
2.6
2.5
2.4
2.3
270
300
350
400
450
GROSS WEIGHT (LBS.) (10^{-3})

FIGURE 3.1-2

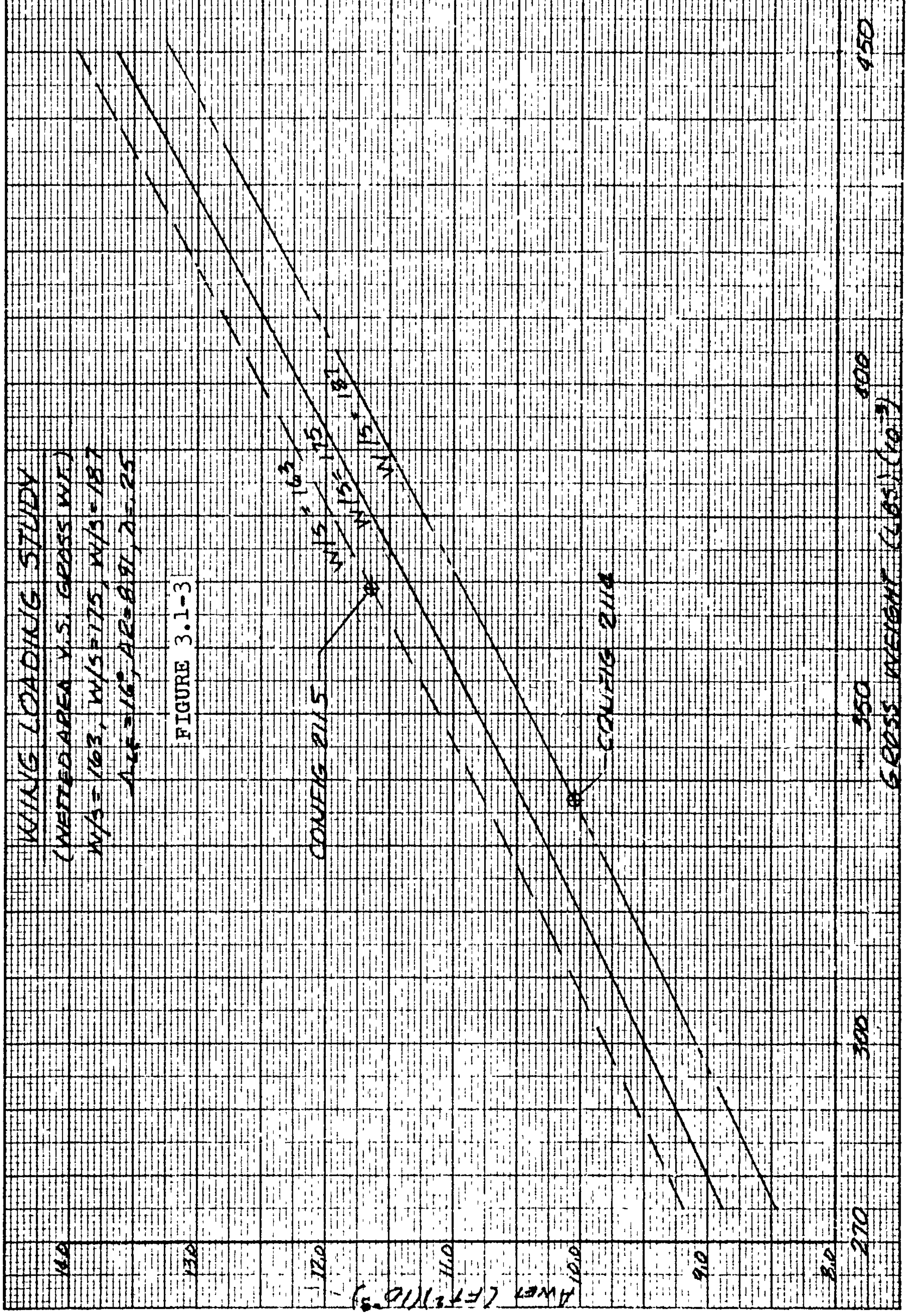

WING LOADING STUDY
(WETTED AREA V.S. GROSS WT.)
W/S=163, W/S=175, W/S=187
Λ_LE=16°, AR=8.91, λ=.25
W/S=163
W/S=175
W/S=187
CONFIG 211.5
CONFIG 211.4
270
300
350
400
450
GROSS WEIGHT (LBS) (10^-3)
8.0
9.0
10.0
11.0
12.0
13.0
14.0
A_WET (FT^2)(10^-3)

FIGURE 3.1-3

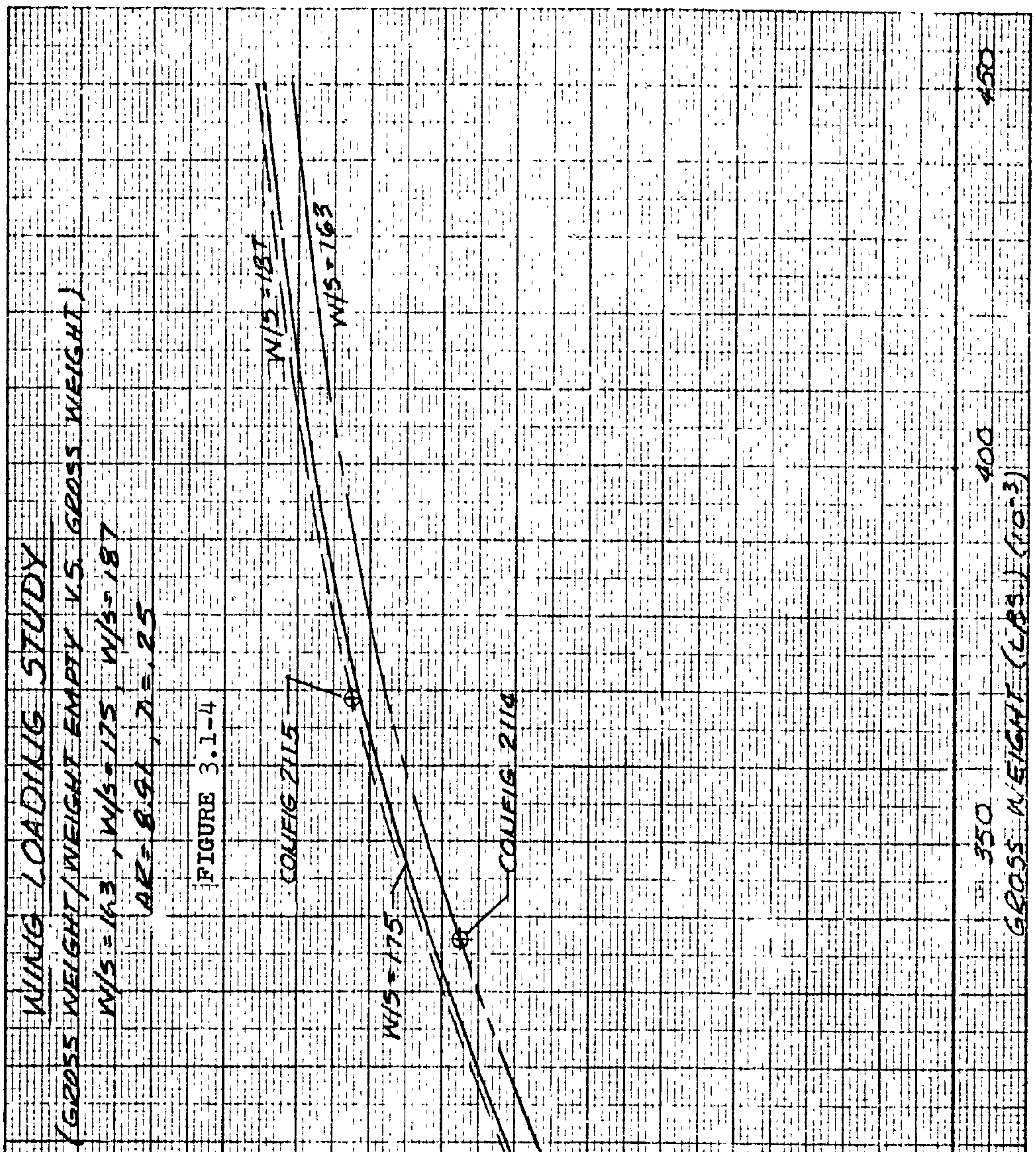

WING LOADING STUDY
(GROSS WEIGHT/WEIGHT EMPTY VS. GROSS WEIGHT)
W/S=163 , W/S=175 W/S=187
AR=8.91 , λ=.25
CONFIG 2115
CONFIG 2114
W/S=175
W/S=187
W/S=163
2.9
2.8
2.7
2.6
2.5
2.4
2.3
Wg/We
270
300
350
400
450
GROSS WEIGHT (LBS.) (10⁻³)

FIGURE 3.1-4

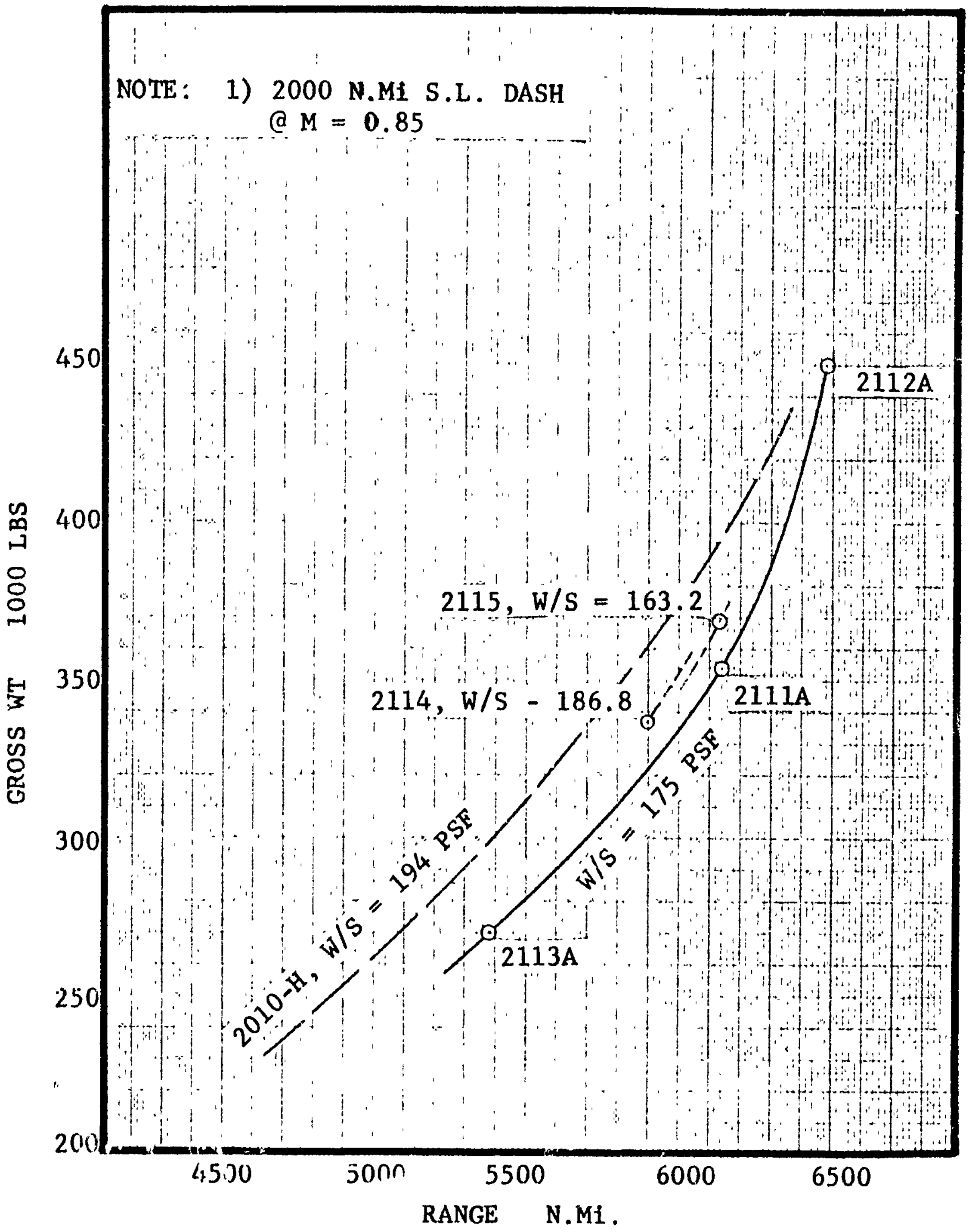


FIG. 3.1-5

AMPSS - GROSS WEIGHT EFFECTS ON RANGE

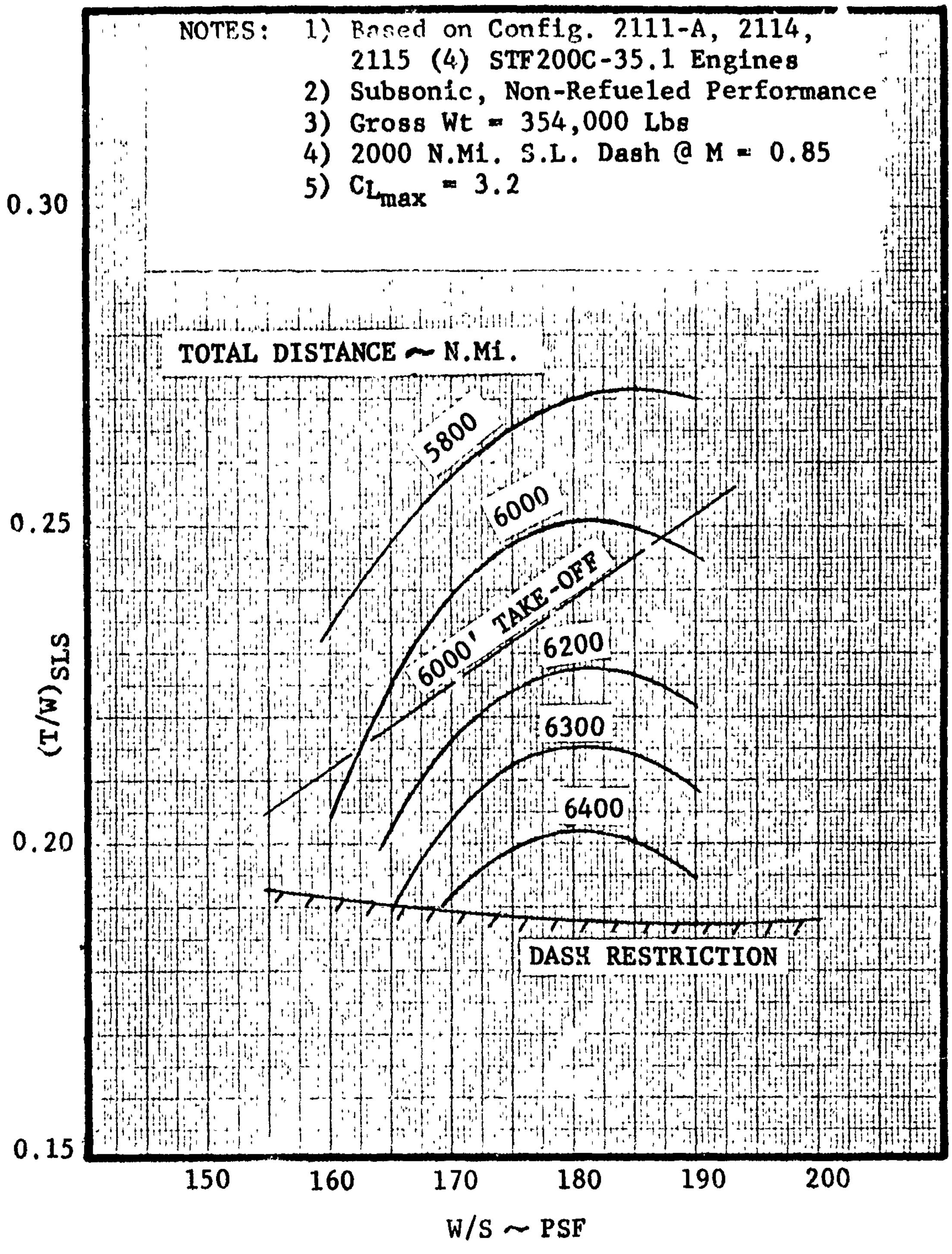


FIG. 3.1-6

AMPSS - THRUST TO WEIGHT RATIO VS WING LOADING for the design gross weight selection. The curves from Figure 3.1-1 give a A_{wet} (16°) = 12,050 sq.ft., and A_{wet} (72.5°) = 11,450 sq.ft., for a design gross weight of 395,000 lbs. The curve from Figure 3.1-2 gives a Wo/We = 2.735 for a design gross weight of 395,000 lbs. Configuration 2120, as determined from the layout, is plotted on Figures 3.1-1 and 3.1-2 with a A_{wet} (16°) = 11,939 sq.ft., A_{wet} (72°) = 11,392 sq.ft., and Wo/We = 2.74.

Note that the wetted areas and weight ratio for Configuration 2120 are slightly better than those predicted on the growth curve. This was possible because of a slightly improved fuselage shape that resulted in a smaller vertical tail volume requirement. The smaller vertical tail and improved fuselage shape reflected lower wetted areas and structural weights.

It was discovered that the growth curves were very sensitive to size and shape of the fuselage. The minimum fuselage cross-section was determined in most cases by the fixed contents which are: navigation radar antenna, crew compartment, weapons bay, and landing gear. The length was determined by these same constraints including the volume required for fuel tankage. The slenderness ratio of the fuselage could be increased to give better supersonic aerodynamic characteristics if the weapons bay and landing gear bay restrictions were lifted. However, it was found that designing the fuselage to a minimum length with the required cross-sectional area resulted in improved subsonic range performance due to lower surface area and structural weights.

3.2 DESIGN DESCRIPTION

3.2.1 Configuration Description

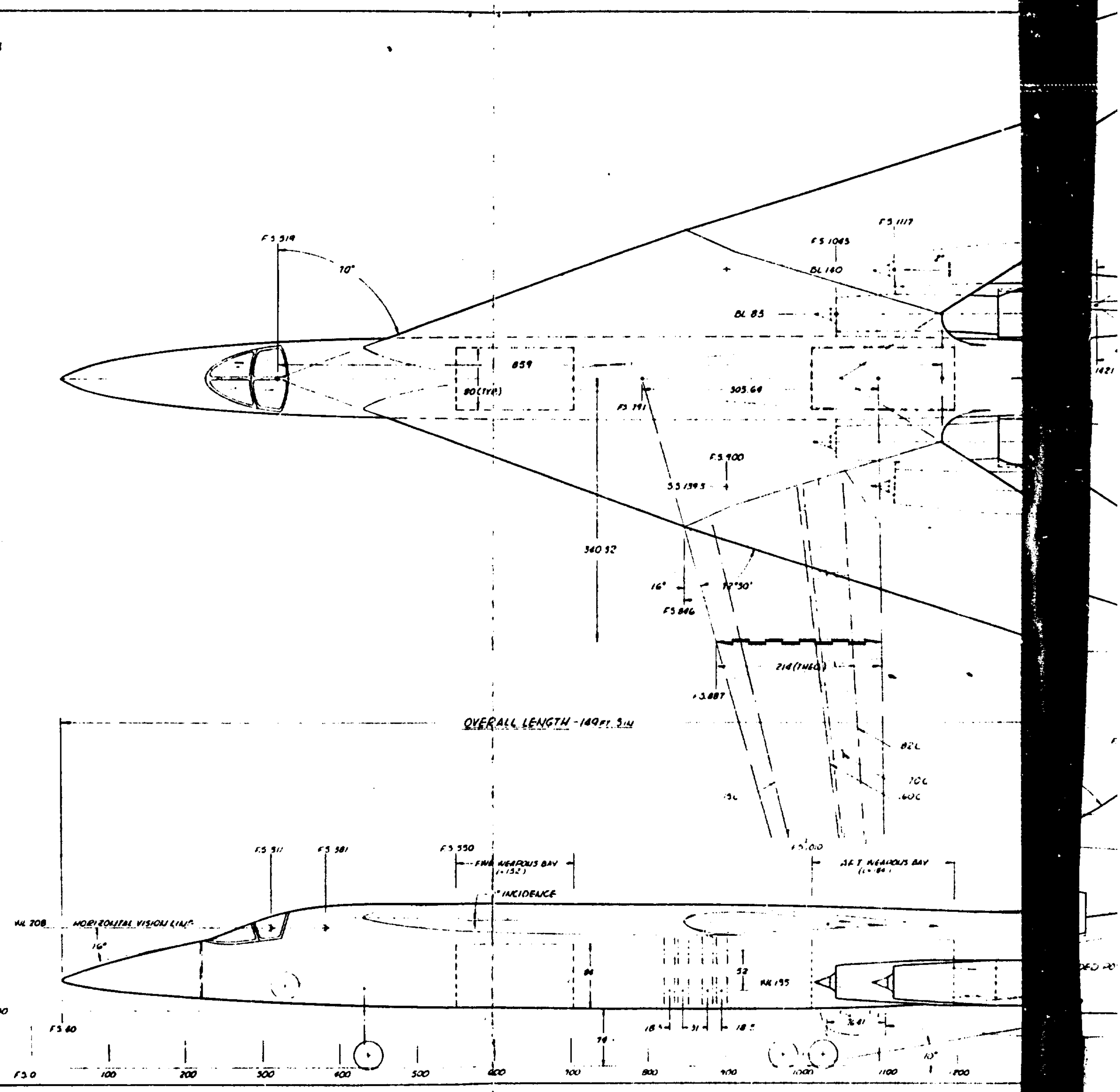
Configuration 2120 shown in Drawing FW6401083, Figure 3.2-1, meets the requirements of the AMPSS Statement of Work (Contract AF33(615)-1174). The design gross weight is 395,000 lbs. with a wing reference area of 2257 square feet, a wing loading of 175 lbs. per square foot and a total aerodynamic wetted area of 11,939 square feet. The airplane conceptual arrangement from Configuration 2110 as reported in FZM-4124 is retained. However, modifications have been made to include the new flight envelope and internal modifications as stated in the statement of work (see Paragraph 3.1.1).

The airplane utilizes the variable sweep wing concept that is necessary to meet the AMPSS mission requirements. The wing is located in a high position in relation to the fuselage and is positioned to a leading edge sweep of 16° for take-off and 72.5° for the dash portion of the mission. Intermediate positions are used during the cruise portion of the mission and for an emergency landing condition with fuel tanks empty and without payload.

The fuselage slenderness ratio (length/cross-section) has been designed primarily for the subsonic portion of the mission since studies indicated that it would determine the aircraft gross weight. Minimum aerodynamic wetted area is one of the prime criteria for low subsonic drags which would dictate a low slenderness ratio fuselage rather than a high slenderness ratio. The fuselage is 108 inches wide, 136 inches high and 1743 inches long.

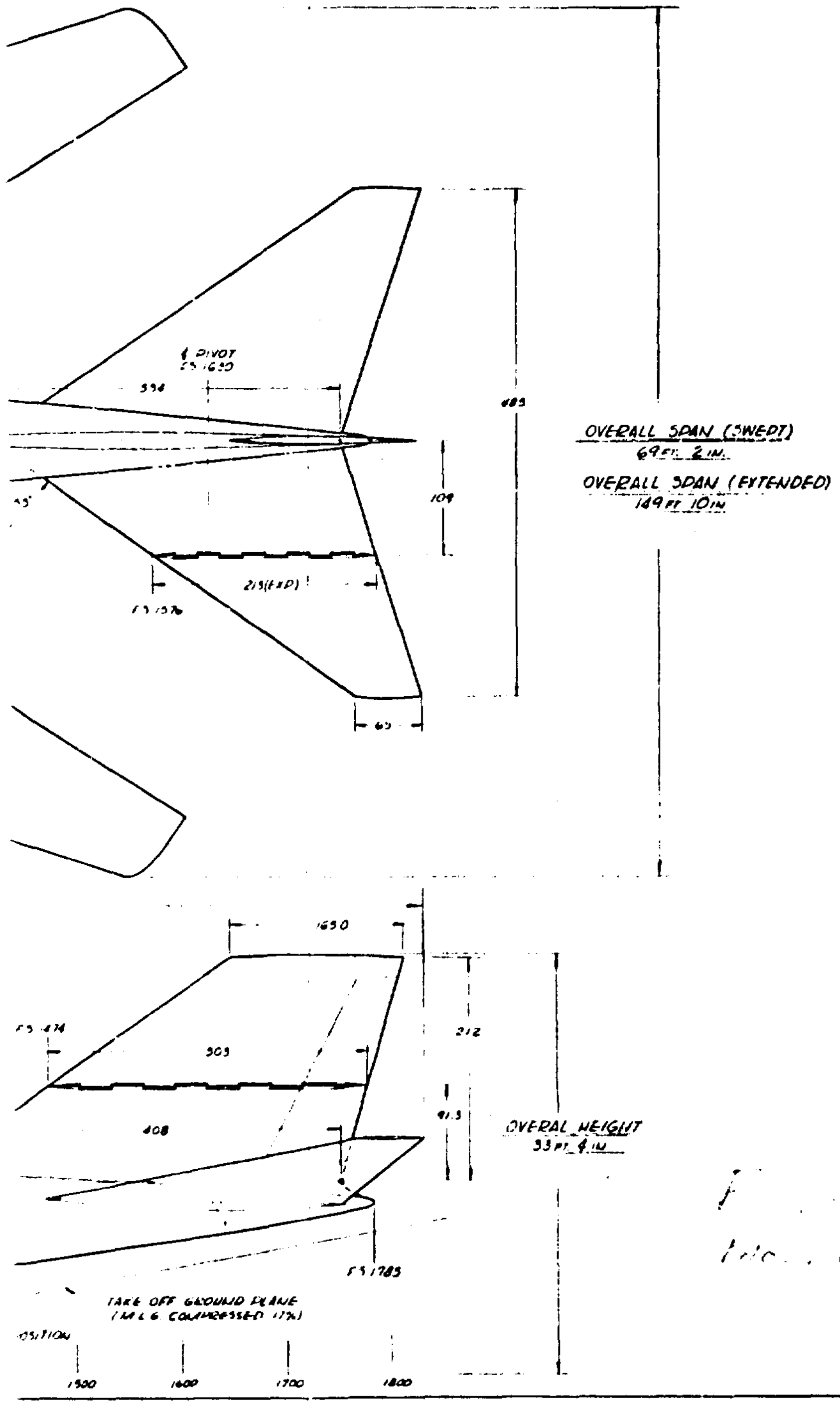
The horizontal and vertical tail is located in a conventional manner on the fuselage. The all moveable horizontal tail is differentially operated for roll and pitch control. It has a dihedral

SECRET



SECRET

SECRET



BASIC DATA

WING	
EXTENDED POSITION	
AREA (THEO-GLOVE NOT INCL)	2257 SQ FT
AREA L.E. GLOVE (THEO)	625 SQ FT
ASPECT RATIO	8.41
TAPER RATIO	[illegible]
LEADING EDGE SWEEP — (GLOVE 70°)	16°
AIRFOIL SECTION	(PIVOT) NACA 64-0[illegible]
	(TIP) NACA 64-00[illegible]
LEADING EDGE SLAT AREA (TOTAL)	218 SQ FT
TRAILING EDGE FLAP AREA (DOUBLE SLOTTED TYPE)	502 SQ FT
TRAILING EDGE SPOILER AREA	248 SQ FT
SWEPT POSITION	
AREA (THEO-GLOVE INCL)	2655 SQ FT
ASPECT RATIO	1.60
TAPER RATIO (AVG C$_t$ / AVG C$_r$)	.86
LEADING EDGE SWEEP — (GLOVE 70°)	78.5°
HORIZONTAL TAIL	
AREA (THEO) — 565 SQ FT (EXPOSED)	670 SQ FT
ASPECT RATIO	2.48
TAPER RATIO	.197
LEADING EDGE SWEEP	55°
AIRFOIL SECTION	5% BICONVEX
VERTICAL TAIL	
AREA	421 SQ FT
ASPECT RATIO	1.36
TAPER RATIO	.409
LEADING EDGE SWEEP	55°
AIRFOIL SECTION	4% BICONVEX
RUDDER AREA	78.5 SQ FT
POWER PLANT	
PRATT & WHITNEY STF 200C-35.1 (SCALED 47.6%)	4
ALIGHTING GEAR	
MAIN GEAR 37 x 14.5 TYPE VII TIRES	16
NOSE GEAR 37 x 14.5 TYPE VII TIRES	2
DESIGN GROSS WEIGHT	395,000 LBS

SECRET

FD 64 270 41

PRELIMINARY DESIGN DRAWING

STUDY-
AMPSS GENERAL ARRANGEMENT
CONFIG 2120 (U)

GD
GENERAL DYNAMICS | FORT WORTH

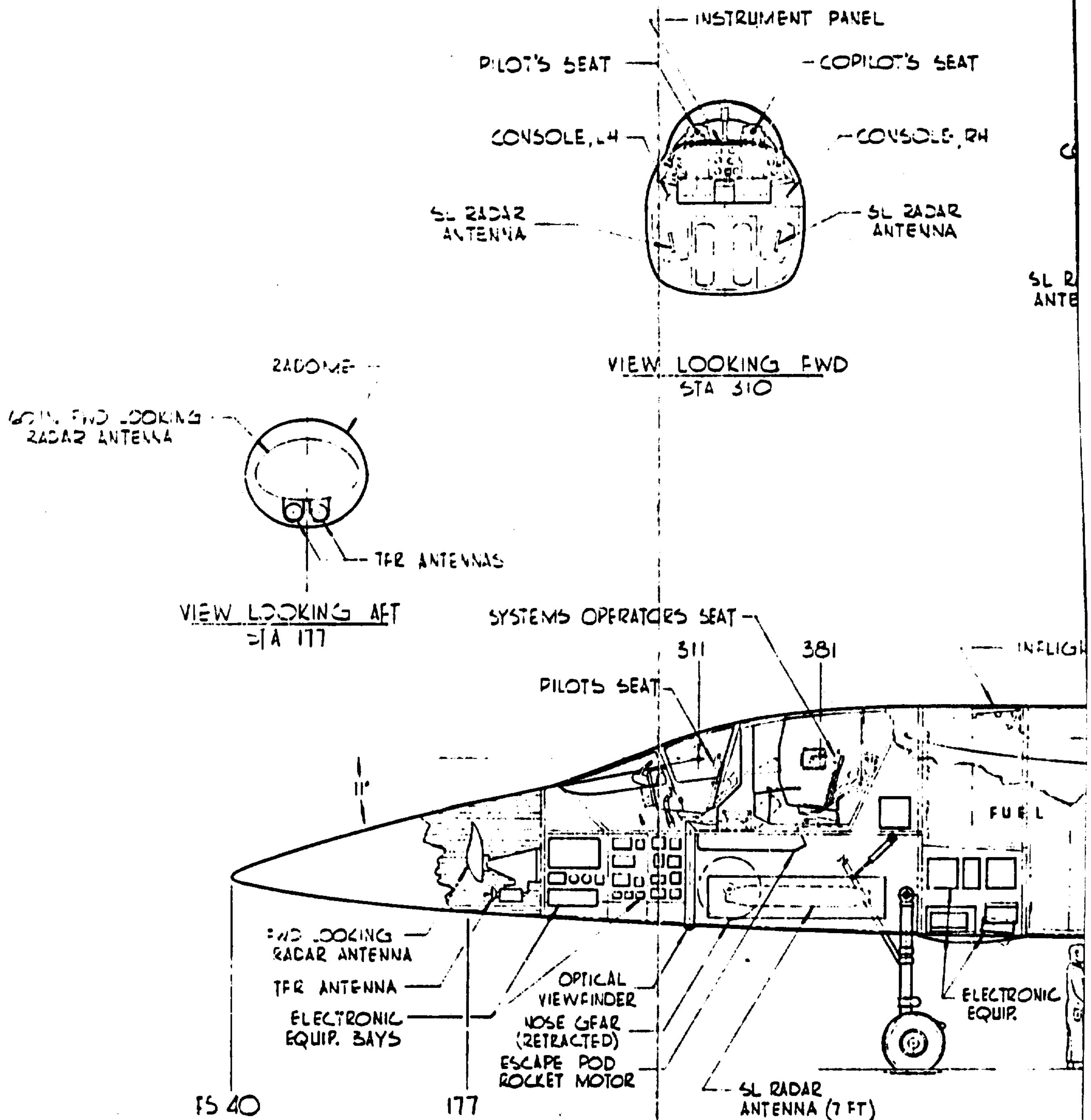
FW6401083

of 15° to help alleviate the effects from the engine blast. Structural weight penalties have been imposed upon this surface because of its near location to the engine nozzles. The surface could be relocated to eliminate this undesirable feature by mounting it on the vertical tail in a "T" tail arrangement. However, a stability and control problem might occur if the surface were moved to this position.

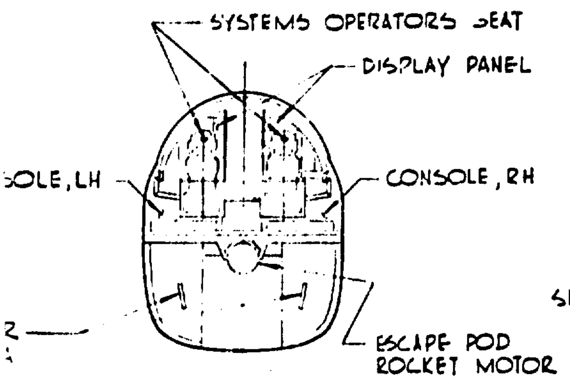
Four Pratt & Whitney STF200C-35.1 turbofan engines scaled to 42.6% are used for this configuration. The engines are located on the fuselage in a siamese staggered nacelle arrangement. A translating double cone external compression inlet design is used for the air induction system. A possible problem exists with foreign object damage to the air induction system because of the location of the main landing gear directly below the inlets. This problem will be alleviated in a similar manner as accomplished on the F-111. Deflector doors mounted on the fuselage will extend simultaneously with the landing gear to protect the inlets.

The design payload for this airplane is 20,000 lbs. and is carried in two separate weapon bays located forward and aft of the wing box carry-through structure and the main landing gear stowage bay. Two separate weapon bays have been provided to obtain a more desirable airplane balance, since carriage of the entire 20,000 lbs. either forward or aft of the airplane center of gravity would create a problem. The basic mission payload is 10,000 lbs. and is carried only in the forward bay. An auxiliary tank with a capacity of 18,000 lbs. of JP-4 is placed in the aft weapons bay when the 10,000 lb. mission payload is carried. This tank is limited to 10,000 lbs. of JP-4 for a design take-off gross weight of 395,000 lbs. However, an inflight refueling gross weight capability of 403,000 lbs. is available.

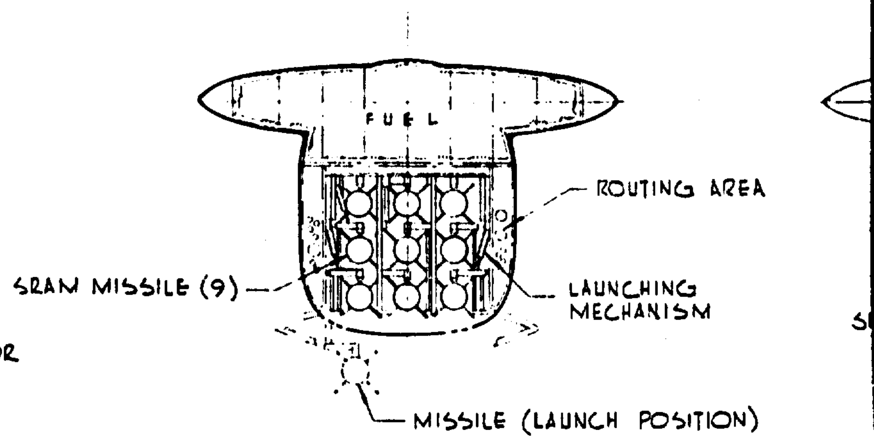
SECRET



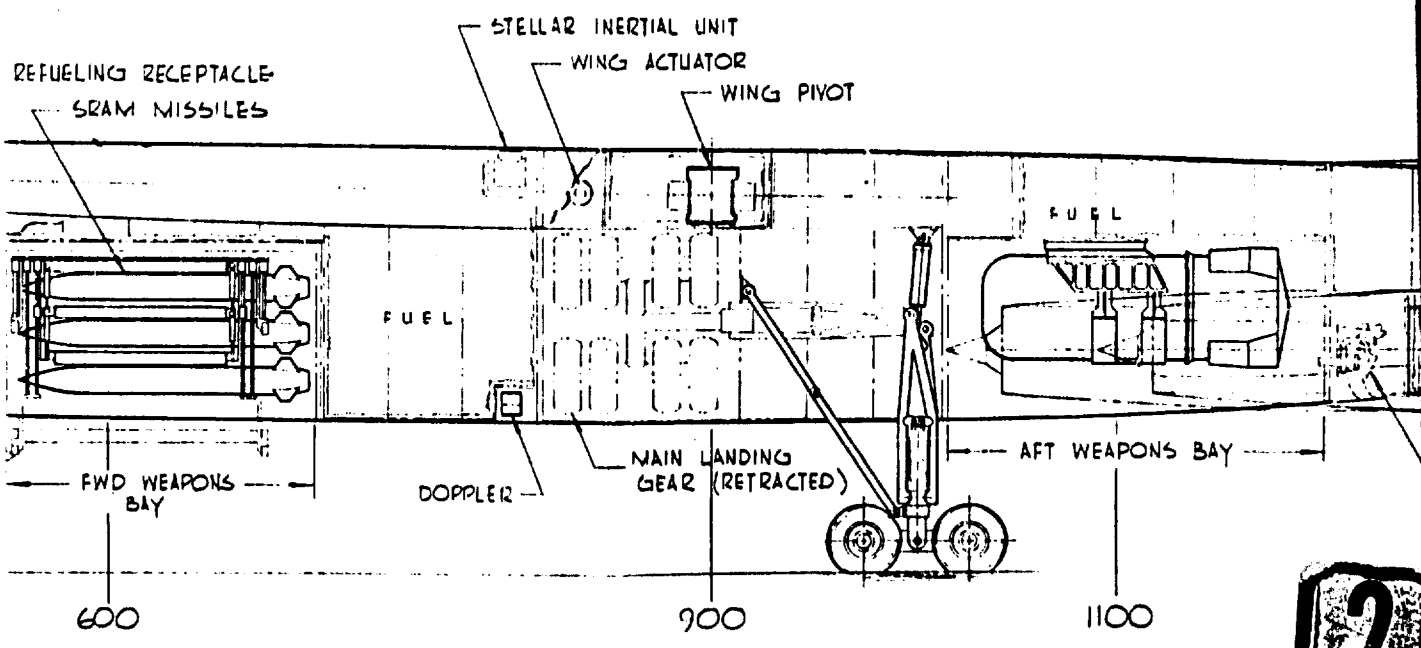
SECRET

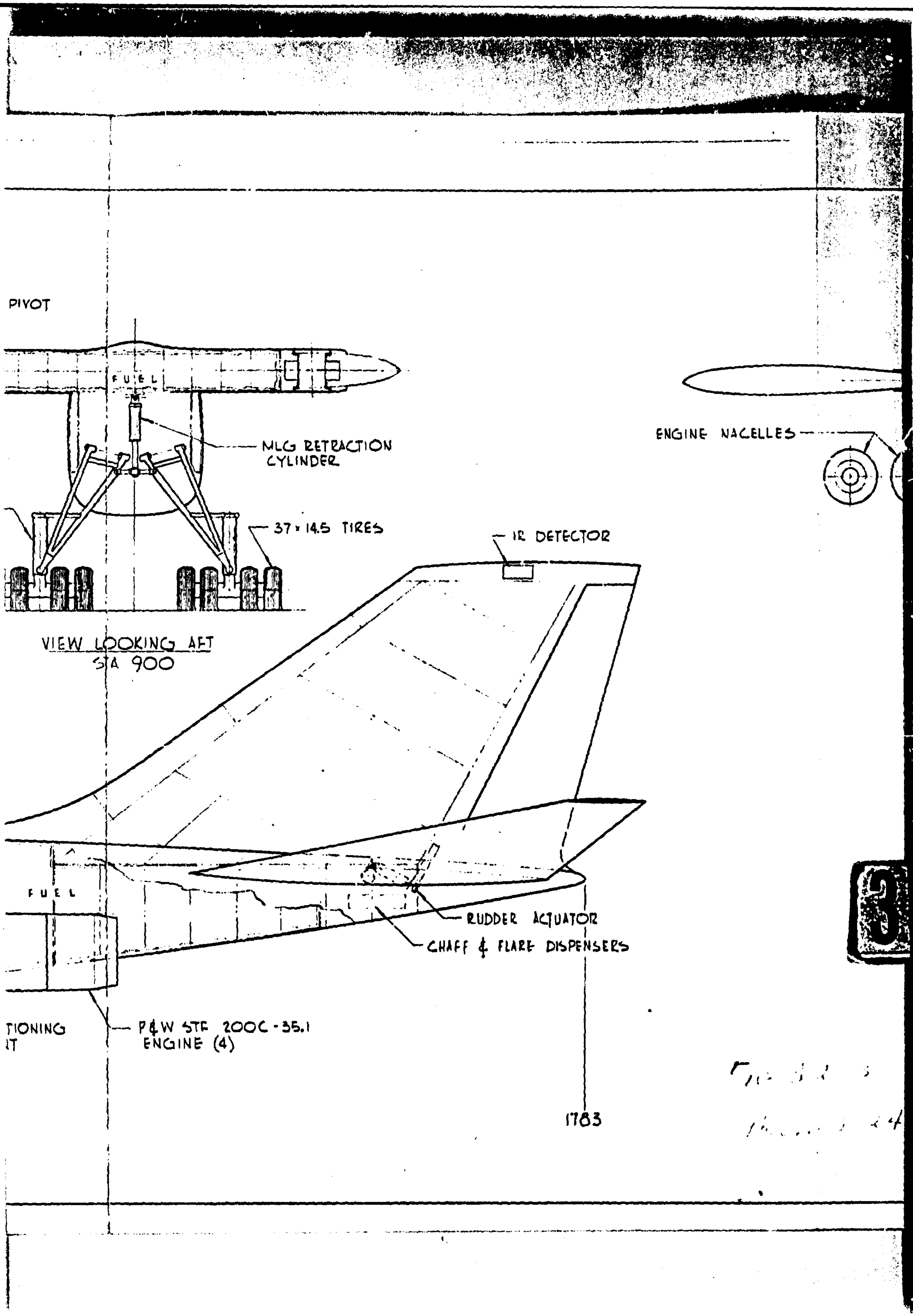


VIEW LOOKING FWD
STA 381



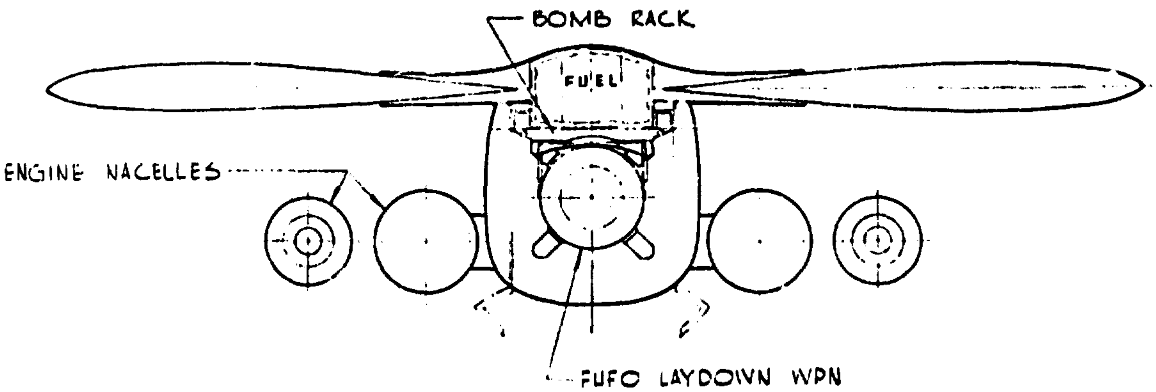
VIEW LOOKING FWD
STA 600



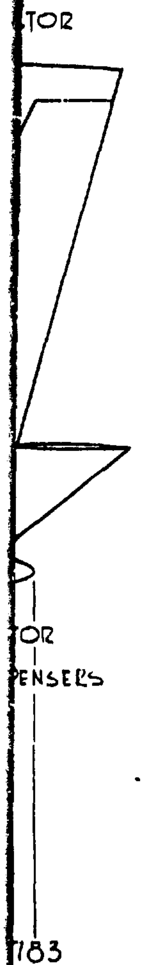


PIVOT
FUEL
MLG RETRACTION
CYLINDER
37 × 14.5 TIRES
VIEW LOOKING AFT
STA 900
ENGINE NACELLES
IR DETECTOR
FUEL
RUDDER ACTUATOR
CHAFF & FLARE DISPENSERS
TIONING
IT
P&W STF 200C-35.1
ENGINE (4)
1783

SECRET



VIEW LOOKING AFT
STA 1100



4

SECRET

DOWNGRADED AT 12 YEAR INTERVALS NOT AUTOMATICALLY DECLASSIFIED. DOD DIR 5200.10

THIS MATERIAL CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18 U.S.C., SECTIONS 793 AND 794, THE TRANSMISSION OR REVELATION OF WHICH IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

FW 64 276 93

PRELIMINARY DESIGN DRAWING

STUDY AMPSS INBOARD PROFILE CONFIGURATION 2120 (U)				
BY W.B. BALL	CHECKED [illegible]	APPROVED [illegible]	SCALE 1/40	DATE 6-16-64
GD GENERAL DYNAMICS | FORT WORTH			FW 6401084	

Code 81755

Department 6

3.2.2 Interior Arrangement

A high density design concept (see Drawing FW6401084, Figure 3.2-2) has been retained through the configuration studies for the AMPSS airplane and is employed for Configuration 2120. A considerable amount of the total fuel is contained in the fuselage since the fuel volume capacity of the wing is limited. The fuselage contains 73.5% (184,185 lbs.) of the total airplane fuel (250,595 lbs.).

A 60" x 36" radar antenna is located in the nose radome compartment. It has a $\pm 90^\circ$ scan in azimuth, $\pm 10^\circ$ yaw and pitch, and 0° to -20° tilt motion. The terrain following radar antenna and its equipment are also located in this compartment. The nose cross-section was dictated by this equipment and the length was predicated by the aerodynamic requirements resulting in a total radome volume of 220 cubic feet.

The four-man crew compartment requires a minimum of 320 cubic feet for the AMPSS mission, however, this configuration has 484 cubic feet available. This volume is a result of the cross-sectional requirements in the middle portion of the fuselage (dictated by the size of the weapons bay, main landing gear, and fuselage fuel tankage) and the nose radome size (dictated by the forward looking radar).

The electronic equipment basic volume is 132 cubic feet of which 30 cubic feet is contained in the crew compartment. The remaining 102 cubic feet of equipment are located beneath the crew compartment and on either side of the nose landing gear wheel well. The installed volume of 404 cubic feet includes the side looking radar antennas and the decoy rocket package located between the nose radome and the nose wheel bay.

The nose landing gear is located directly below the crew compartment and occupies a volume of 138 cubic feet. It has a conventional design with (2) 37 x 14.5 tires and retracts forward to the stowed position.

The forward fuselage fuel tank, containing 103,245 lbs. of JP-4, starts at the aft nose wheel bay bulkhead and terminates at the forward main landing gear bay bulkhead. This includes the fuel tankage contained in the wing glove, and above the weapons bay.

The forward weapons bay envelope is 80 inches wide, 83 inches high and 152 inches long with a total volume of 583 cubic feet. It is capable of carrying a basic load of 9 attack missiles (SRAM) or a number of alternate loadings. Refer to Section 3.1.2.4, FZM-4038-II-1 for a description of the missile arrangement in this bay.

The wing box carry-through structure fuel tank is located above the main landing gear bay and has a capacity of 13,700 lbs. of JP-4.

The main landing gear wheel well occupies a volume of 814 cubic feet. Eight 37 x 14.5 tires per landing gear bogie in a dual twin tandem arrangement are used for this design. Because the airplane has a variable sweep wing, the landing gear attachment is limited to the fuselage. The carriage structure and mechanism folds the gear bogie inboard and then retracts it forward to the stowed position. A secondary power equipment bay is located in this area above the main landing gear and below the wing box structure. The outer wing panel fuel tank contains 52,700 lbs. of JP-4 and is bounded by the front and rear spars extending from the wing pivot root rib to the tip chord rib.

The aft weapons bay is located immediately aft of the main landing gear bay. It has an evelope 80 inches wide, 83 inches high and 185 inches long and occupies a volume of 712 cubic feet. This bay is capable of carrying a number of various loadings such as (4) CLAMP, (1) MK-53, or (4) MK-43 weapons, or a tank with 18,000 lbs. of JP-4 capacity

The aft fuselage tank contains a total of 70,950 lbs. of JP-4 including the tank area above the weapons bay and the tank from the aft weapons bay bulkhead to the vertical stabilizer rear spar/fuselage bulkhead.

3.2.3 Detail Design

The configuration arrangement, design gross weight and wing loading for Configuration 2120 has changed from Configuration 2010-H as reported in FZM-4038-II-1. However, the general concept for the detail design of this airplane is similar to Configuration 2010-H. Those areas which are not affected by the above changes are as follows: (1) crew compartment and furnishings, (2) weapons bay design and arrangement, (3) electrical and electronic equipment installation, (4) hydraulic system, (5) fuel and oil system (larger fuel pumps added to fuel system for Mach 1.2 capability), (6) air-conditioning system, and (7) age and training. Refer to Section 3.1.2, FZM-4038, for a description of these systems.

The areas which are affected by the changes in the configuration are: (1) structural arrangement, (2) powerplant installation, and (3) alighting system. A brief description of the detail design for these areas is presented below.

3.2.3.1 Structural Arrangement

Aluminum is the basic material used for construction for this airplane wherever temperatures and strength allowables are acceptable. The basic concepts of fabrication are: (1) integrally machined bulkheads and frames, (2) machined longerons, and (3) honeycomb sandwich panels for skins, floors, webs, and panels.

This type of construction is retained for the structural design of the fuselage. The fuselage structural arrangement of Configuration 2120 is identical to Configuration 2010-H from the nose section to the aft weapons bay bulkhead. From this station and aft the structure is modified to accept the attachment of the engine nacelles and the horizontal stabilizer. Structural weight penalties have been imposed on the empennage because of the temperature and acoustical environment. The type of construction will be changed in this area to either heavier gage aluminum plate or brazed stainless steel sandwich panels. The acoustical environment will be the criteria for selection of the type of construction in this area. The aft fuselage fuel bulkhead will serve as a primary load carrying member. It will accept the actuator and hinge loads from the horizontal tail, and the vertical tail rear spar loads.

The structural design for the horizontal and vertical tail is identical to that described for Configuration 2010-H. However, thicknesses on the skin panels for the horizontal tail will be increased due to engine blast.

The structural design for the outer wing panel, wing pivot and forward glove section is similar to Configuration 2010-H. The type of construction for the aft glove fairing remains the same as that for Configuration 2010-H. However, the glove has been reshaped at the outboard edge by sweeping it inboard toward the fuselage.

3.2.3.2 Propulsion Design

The propulsion system for Configuration 2120 consists of four Pratt & Whitney STF200C-35.1 turbofan engines scaled to 42.6%. They are located below the wing and mounted on the fuselage in a siamese staggered nacelle arrangement. The nacelles have been located in this region and staggered to minimize the supersonic drag. The separate podded engine nacelle concept gives flexibility to the type and size of engines that can be installed without disturbing the overall configuration design. In addition, this type of engine installation provides good accessibility and simplifies engine removal from the rear of the nacelle, thereby, reducing maintenance time.

The engine accessories and the hydraulic pumps are mounted within the contour of the engine, and the alternators are mounted inside the fuselage with a remote drive connecting them with the engines. This arrangement concept for the accessories contributes to a nacelle shape with a higher slenderness ratio and less total cross-sectional area.

The engine nacelles are supported by two main frames which attach to the fuselage. These two frames are aligned with the inboard engine mounts. The inboard nacelle cowling structure is attached to the two main engine mounting rings. Longerons on the outboard side of this nacelle extend aft for attachment of the outboard engine rings. The nacelle cowling for the outboard nacelle is attached to these engine frames. The nacelle cowling aft of the rear engine mounts will be steel or waffle panel type construction. Forward of the rear engine mount aluminum waffle panel construction will be used. Engine mounting rings and longerons will be integrally machined from steel.

The air induction system for each engine consists of a separate circular inlet with external compression design controlled by a two-cone variable diameter spike that translates fore and aft. The spike is translated forward for transonic speed ranges, and translated aft to control the flow of air at a high ai_ ow pressure recovery. To increase the inlet performance a portion of the boundary layer is diverted by the inboard and center stub pylons. Part of the fuselage boundary layer air is passed internally and used for secondary cooling supply. In a similar manner the boundary layer air from the inboard nacelle is bypassed through the center stub pylon and exits at the nacelle base area.

3.2.3.3 Alighting System

The alighting system consists of the main and nose landing gears, tires, wheels, brakes, steering and anti-skid systems. Tire selection and arrangement are discussed in Paragraph 5.1, Runway and Flotation Studies.

Arrangemen and Capabilities

A conventional tricycle gear arrangement is used. The Inboard Profile Drawing (FW6401084, Figure 3.2-2) shows the locations of the nose and main gears. Take-off tail clearance is 10° and roll clearance is 11° at the 10° take-off angle. The turnover angle of 59° is within the requirement established by ARDCM 80-1. Long stroke shock struts are provided to assure relatively soft landings and to provide for maximum shock absorption during taxi or uneven surfaces. The nose and main gears are designed to withstand landing loads at sink speeds in excess of the 8 fps required.

Materials

Primary load carrying members are steel, heat treated to a maximum of 220 KSI. High strength aluminum alloy is used for the secondary structure.

Brake discs are steel. Beryllium can be used, should the weight advantage justify the increased cost. Wheels are machined from aluminum alloy forgings.

Main Landing Gear

For the airplane configuration developed, the most feasible location for landing gear retraction is inside the fuselage. A skewed axis pivot is employed, which causes the gear to enter its well through the bottom of the fuselage, thereby preserving the structural integrity of the fuselage sides. During retraction, the shock strut is moved from the vertical to a position perpendicular to the pivot axis, and the truck is rotated about its longitudinal axis approximately 90°. This arrangement assures:

1. Minimum fuselage cutout
2. Minimum fuselage cross-section
3. Zero engine inlet air interference

Forward retraction provides for failsafe gravity-drag extension in the event of hydraulic failure. Hydraulically operated downlocks are provided to prevent premature retraction. Uplatches provide positive positioning of the gear in its retracted position.

Nose Landing Gear

The nose landing gear is a conventional, forward retracting design. Retraction, extension and downlocking are accomplished by a single hydraulic cylinder. Uplocking is actuated by a separate cylinder. Steering is accomplished by two hydraulic cylinders driving a rack and pinion system mounted inside the shock strut housing.

Wheels and Brakes

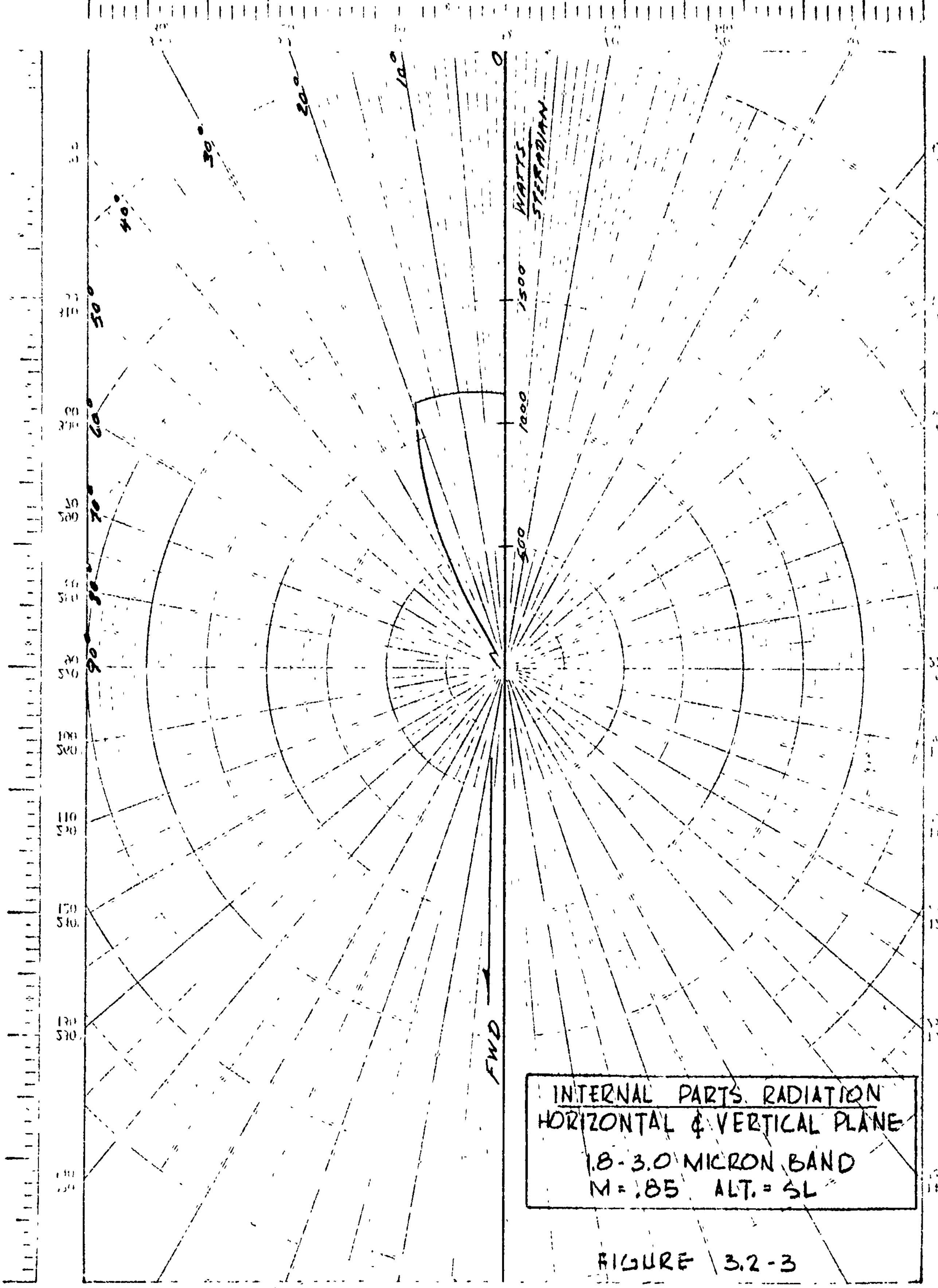
Brake energy capacities are in accordance with MIL-W-5013 and were calculated using Method II of that specification. The brakes are installed inside the axle housings between the inboard tires and are driven by shafts from the outboard wheel hubs. This arrangement precludes brake heat transfer through the wheels into the tire beads. Direct slipstream exposure assures optimum disc cooling. Discs are removed for maintenance through access panels in the housings without wheel removal or jacking. Individual brake and anti-skid devices are provided for each set of duals for a total of four brakes per main gear. The lower part of each housing protrude and is reinforced to support the airplane, thereby eliminating the necessity for non-frangible wheels.

3.2.4 Infrared Emission

As requested in the Air Force Statement of Work several methods of reducing the infrared emission on the AMPSS airplane are discussed in the following paragraphs. The most probable attack mode of an infrared seeking intercept weapon will be from the tail-on (or aft) aspect.

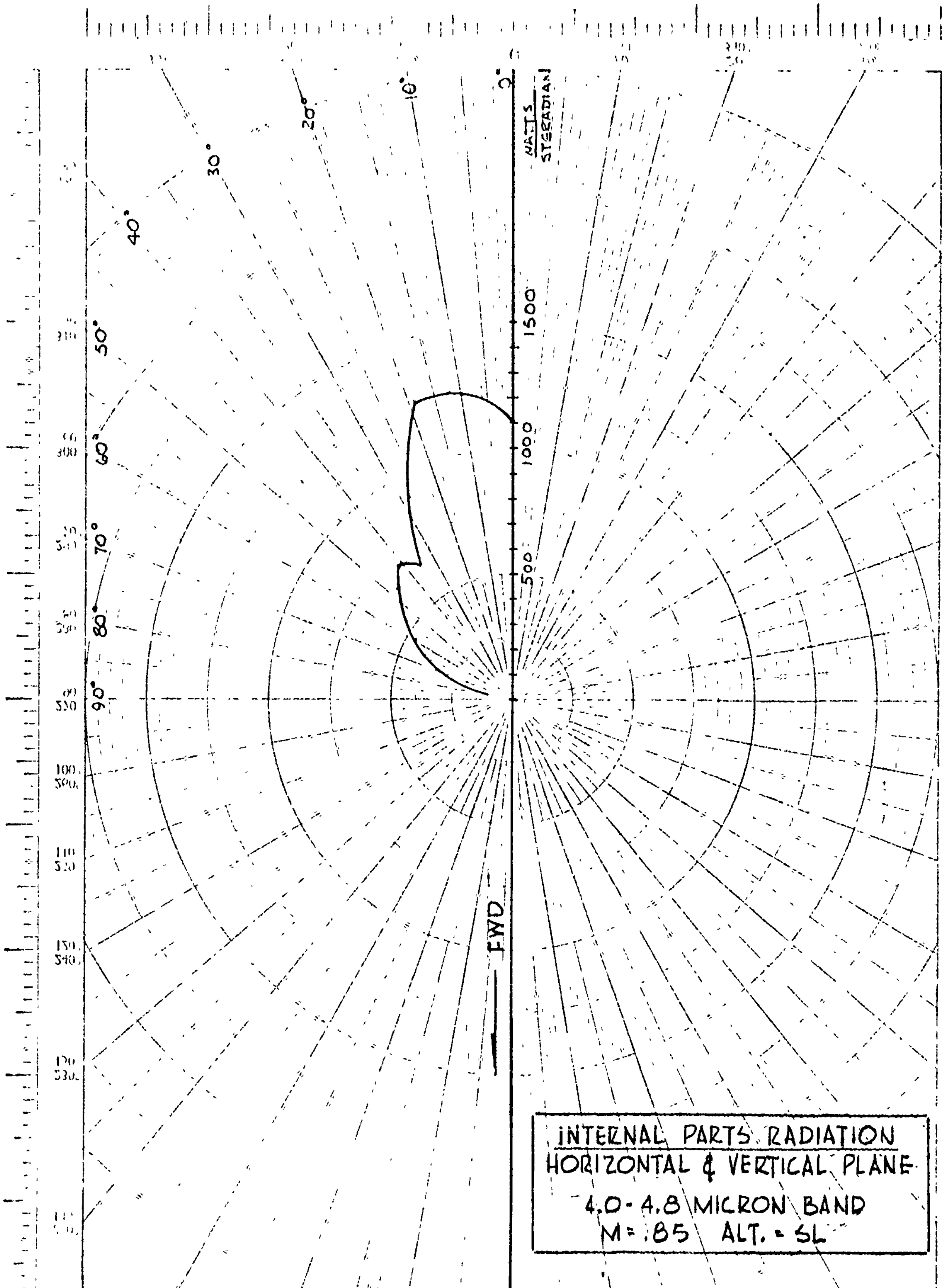
This evaluation of infrared signature was made for the design mission conditions of Mach .85 at sea level. At this flight condition, the engine power setting is below both the military and normal power settings. Also, in the tail-on quadrant, the major contributor to the infrared signature will be the radiation from the internal parts of the engine. An extensive and accurate analysis of the radiation from the internal parts of an engine requires experimental data of infrared emission taken at various angles from the tail-on axis. Since these data are not available, the present analysis was based on the assumption that the various internal parts radiated as black bodies at the exhaust gas temperatures of their respective regions (i.e., engine, duct, and mixed exhaust gas temperatures). Also, the radiation view factors were approximated from the basic dimensions of the engines. The radiation from the plume and air-frame were neglected in this analysis since, in the tail-on aspect, they will be insignificant compared to the radiation from the engine internal parts for this flight condition. Most infrared detectors are sensitive to radiation in the wavelength bands of 1.8 to 3.0 and 4.0 to 4.8 microns. Therefore, the engine internal parts infrared signatures are presented for these wavelength bands in Figures 3.2-3 and 3.2-4, respectively.

Suppression of internal parts radiation is somewhat difficult, however, several methods, with associated performance penalties, are available. The use of transpiration cooled plugged nozzles has been shown to considerably reduce internal parts radiation; however, they generally introduce performance penalties. Also, the use of long tailpipes extensions will substantially reduce the solid angle from which the maximum radiation intensity can be seen. The screening of the internal parts by injection of screening particles such as aluminum or carbon dust is another technique which is effective from all viewing angles.



INTERNAL PARTS RADIATION
HORIZONTAL & VERTICAL PLANE
1.8-3.0 MICRON BAND
M = .85 ALT. = SL

FIGURE 3.2-3



INTERNAL PARTS RADIATION
HORIZONTAL & VERTICAL PLANE
4.0 - 4.8 MICRON BAND
M = .85 ALT. = SL

FIGURE 3.2-4

3.3 PERFORMANCE

The performance capabilities of Configuration 2120 are presented in this section. Configuration 2120 is an improved "Heavy Type B" design similar to Configuration 2110 previously reported in GD/FW report FZM 4124. Configuration 2120 has a design gross weight of 395,000 lbs and an extended wing area of 2257 ft^2 resulting in a takeoff wing loading of 175 PSF. Four 0.426-scale P&W STF 200C-35.1 engines are used to produce a sea level static T/W = .2318.

Only the performance capabilities are presented in this section. Performance methods and equations used are the same as previously reported in Section 3.3.2 of FZM 4038-II-1. Aerodynamic data, propulsion data, and weight data used in computing the performance of Configuration 2120 are presented or discussed in Sections 3.4, 3.5, and 3.8, respectively. A tabulated summary of Configuration 2120 performance is presented in Table 3.3-I.

3.3.1 Takeoff Performance

As indicated in Section 3.1, the method of configuration selection resulted in a design which has the minimum engine size allowed by takeoff requirements (6000 feet over 50 ft. obstacle - sea level, standard day). Figure 3.3-1 presents the takeoff distance over a 50 ft. obstacle, ground roll, velocity at 50 ft, and velocity at unstick versus gross weight for runway altitudes of sea level and 3000 ft and for temperatures of standard (59°F) and 90°F. The data are for a wing sweep of 16 degrees with the C.G.

TABLE 33-1
AMPSS SUMMARY PERFORMANCE

CONFIGURATION	NON-REFUELED	REFUELED 1000 N MI PRS TANKER	REFUELED RADIUS TANKER	500 N MI POST-DASH STAGE NON-REFUELED	500 N MI POST-DASH STAGE REFUELED 1000 N MI PRS TANKER	[illegible] N MI POST-DASH STAGE NON-REFUELED	[illegible] N MI POST-DASH STAGE REFUELED 1000 N MI PRS TANKER	500 N MI POST-DASH STAGE NON-REFUELED	500 N MI POST-DASH STAGE REFUELED 1000 N MI PRS TANKER	5 MIN @ M 1.2 NON-REFUELED	5 MIN @ M 1.2 REFUELED 1000 N MI PRS TANKER
TAKE-OFF WEIGHT OR REFUEL WEIGHT, W_o (LB)	395,000	403,000	403,000	395,000	403,000	395,000	403,000	395,000	403,000	395,000	403,000
ENGINES	←			P & W	STF 200C-35	1 →					
NO. & SCALE	(4) 426 %										
AUGMENTATION, %	74										
ENGINE SIZING BASIS	TAKE OFF										
TAKE-OFF C_{LMAX} (FREE AIR)	3.2										
S_w (REF) (SQ.FT.)	2257										
AR (REF) (UNSWEPT)	8.91										
t/c (PIVOT-TIP) (UNSWEPT), %	.2 / 9										
T_{SLS}/W_o	23.8										
W_o/W_e (NO PL IN W_e)	2.94	3.00	3.00	2.94	3.00	2.94	3.00	2.94	3.00	2.94	3.00
W_o/S_w (PSF)	175	178.8	178.8	175	178.8	175	178.8	175	178.8	175	178.8
A_{WET}/S_w (UNSWEPT)	5.2?										
ZERO FUEL WEIGHT (W_e) (LB) (NO P.L.)	134,405										
PAYLOAD (LB)	0,000										
TAKE-OFF FUEL ALLOWANCE (LB)	6,300										
FUEL RESERVE (LB)	4,910	5,320	5,320	4,910	5,320	4,910	5,320	4,910	5,320	4,410	5,320
CRUISE MACH	.73										
DASH MACH	.85										
INITIAL-FINAL SUBSONIC CRUISE ALT. (1000 FT)	21.4 / 43.9	20.4 / 43.9	20.4 / 43.9	21.4 / 43.9	20.4 / 43.9	21.4 / 43.9	20.4 / 43.9	21.4 / 43.9	20.4 / 43.9	21.4 / 43.9	20.4 / 43.9
INITIAL-FINAL SUPERSONIC CRUISE ALT. (1000 FT)	47.3 / 60.0	45.0 / 57.5									
LIFT-OFF--TOUCHDOWN VELOCITY (KTS)	140 / 86.3										
TAKE-OFF DISTANCE OVER 50' (FT)	6000										
LANDING DISTANCE OVER 50' (FT)	3280										
REFUEL RADIUS (N.MI.)	–	2170	1920	–	2170	–	2170	–	2170	–	2170
RANGE (DASH/TOTAL) (N.MI.)											
SUBSONIC DASH: MAX. DASH	3690 / 3690	3000 / 7110	3390 / 7180	3060 / 4060	3830 / 6470	3410 / 4410	3230 / 6820	3270 / 4770	3510 / 7190	3450 / 3450	[illegible] / [illegible]
SUBSONIC DASH: 2000 N MI DASH	2000 / 6340	2000 / [illegible]	2000 / 9100	2000 / 5110	2000 / 8830	2000 / 5910	2000 / 9320	2000 / [illegible]	2000 / 8630	2000 / 5100	[illegible] / 8510
SUBSONIC DASH: ZERO DASH	0 / 8440	0 / 12030	0 / 8410	0 / 7210	0 / 11910	0 / 7240	0 / 11850	0 / 7260	0 / [illegible]	0 / [illegible]	0 / [illegible]
SUPERSONIC DASH AT M 2.2: MAX DASH	2680 / 2350	2170 / 6700	—	—	—	—	—	—	—		—
SUPERSONIC DASH AT M 2.2: 2000 N MI DASH	2000 / 4400	2000 / 7020	—	—	—	—	—	—	—	—	—
SUPERSONIC DASH AT M 2.2: ZERO DASH	0 / 8820	0 / 11290	—	—	—	—	—	—	—	—	—

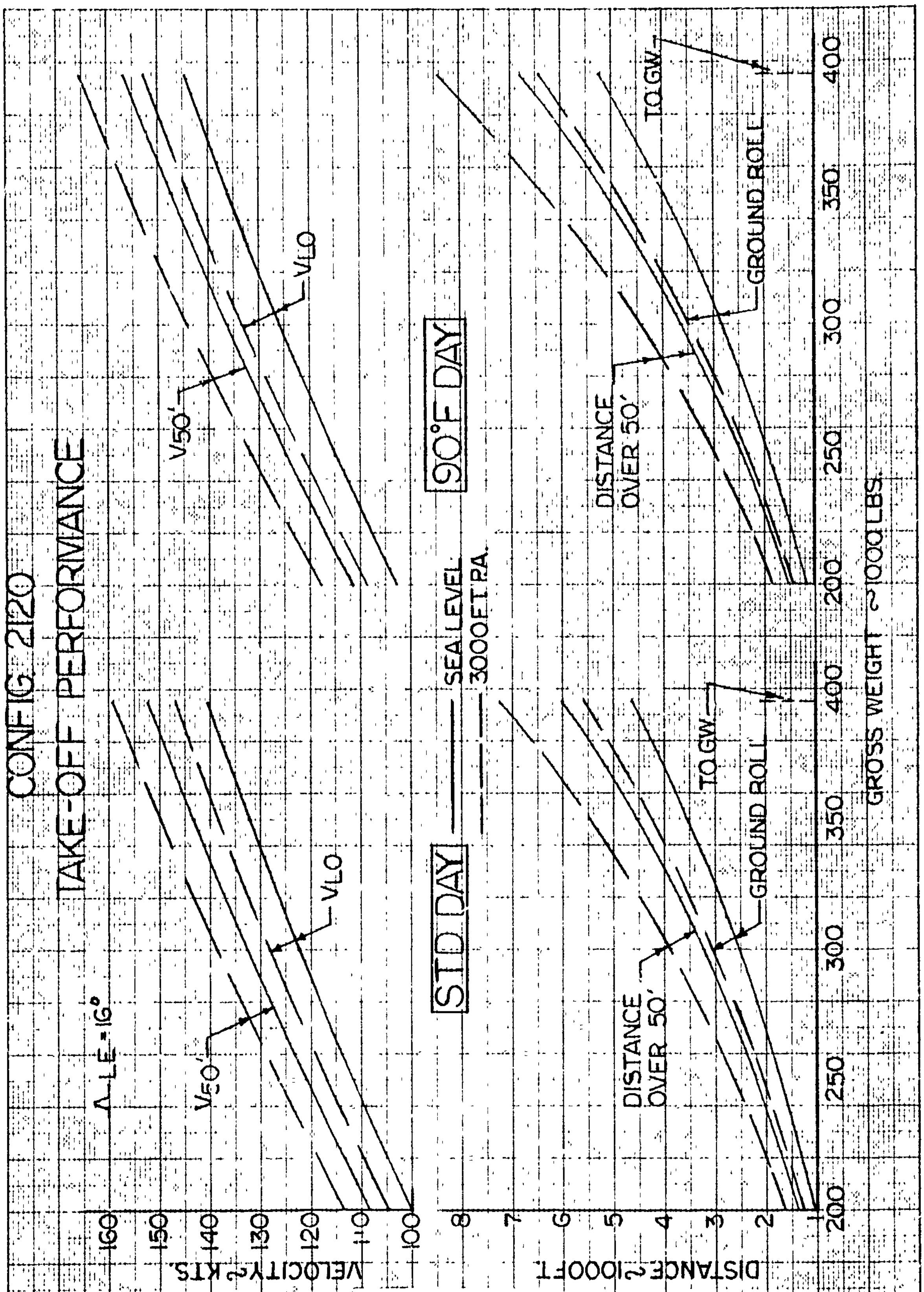

CONFIG 2120
TAKE-OFF PERFORMANCE
Λ LE = 16°
STD DAY
90°F DAY
SEA LEVEL
3000 FT PA.
V50'
VLO
DISTANCE OVER 50'
GROUND ROLL
TO GW
VELOCITY ~ KTS.
DISTANCE ~ 1000 FT.
GROSS WEIGHT ~ 1000 LBS.
100
110
120
130
140
150
160
1
2
3
4
5
6
7
8
200
250
300
350
400

FIGURE 3.3-1

at 33.4 percent MAC. The trimmed C_{LMAX} is 3.2 at 50 ft and 3.16 in the presence of the ground. The wing is at 1.0 degrees angle of attack during ground run. Total takeoff distance over a 50-ft obstacle (sea level, standard day) is 6000 ft.

The takeoff calculations follow MIL-C-5011A rules such that lift off velocity is 110 percent of power-off stall speed and velocity at 50 ft is 120 percent of power-off stall speed. The lifting effect of the vertical component of thrust is included in the takeoff distance calculations.

3.3.2 Critical Field Length

The critical field length for Configuration 2120 at the design gross weight of 395,000 lbs is tabulated below for two temperatures and altitudes:

Runway Altitude	Temperature	CFL, ft.	V_{CEF}, Kts.
S.L.	Std.	5665	108
S.L.	90°F	6377	113
3000 Ft.	Std.	6788	116
3000 Ft.	90°F	7810	123

The critical field length (CFL) is defined as the distance required to accelerate to a critical engine failure speed (V_{CEFS}) and either: (a) abort the takeoff and stop, or (b) continue takeoff on the remaining engines, in the same runway length. The takeoff distance includes only the distance to the unstick velocity.

3.3.3 Landing Performance

Figure 3.3-2 presents the landing distance over a 50-foot obstacle, ground roll, velocity at 50 feet, and touchdown velocity versus gross weight for runway altitudes of sea level and 3000 feet, and for temperatures of standard (59°F) and 90°F. These data are for a wing sweep of 16 degrees with the C.G. at 33.4 percent MAC. The trimmed $C_{L_{MAX}}$ is 3.20 at 50 feet and 3.16 in the presence of the ground. Wing angle of attack during ground roll is 1.0 degrees. The landing distance (sea level, standard day) for the basic landing weight of 149,315 lbs is 3280 ft.

The landing calculation procedure follows MIL-C-5011A rules such that velocity at 50 ft is 1?0 percent of power-off stall speed and touchdown velocity is 110 percent of power-off stall speed.

3.3.4 Climb and Acceleration Performance

Time, distance, and fuel to climb from sea level to the optimum subsonic flight path are presented in Figure 3.3-3 for Configuration 2120. A constant Mach climb of .73, consistent with the cruise Mach number, was selected for the climb. The climb data are presented for military power on the engines.

Time, distance, and fuel to climb and accelerate from the subsonic flight path to the Mach 2.2 flight path are presented in Figure 3.3-4 as a function of initial gross weight. These data are presented for a maximum augmentation power (MIL. power + maximum duct heat) and 72.5 degrees wing sweep. The acceleration path

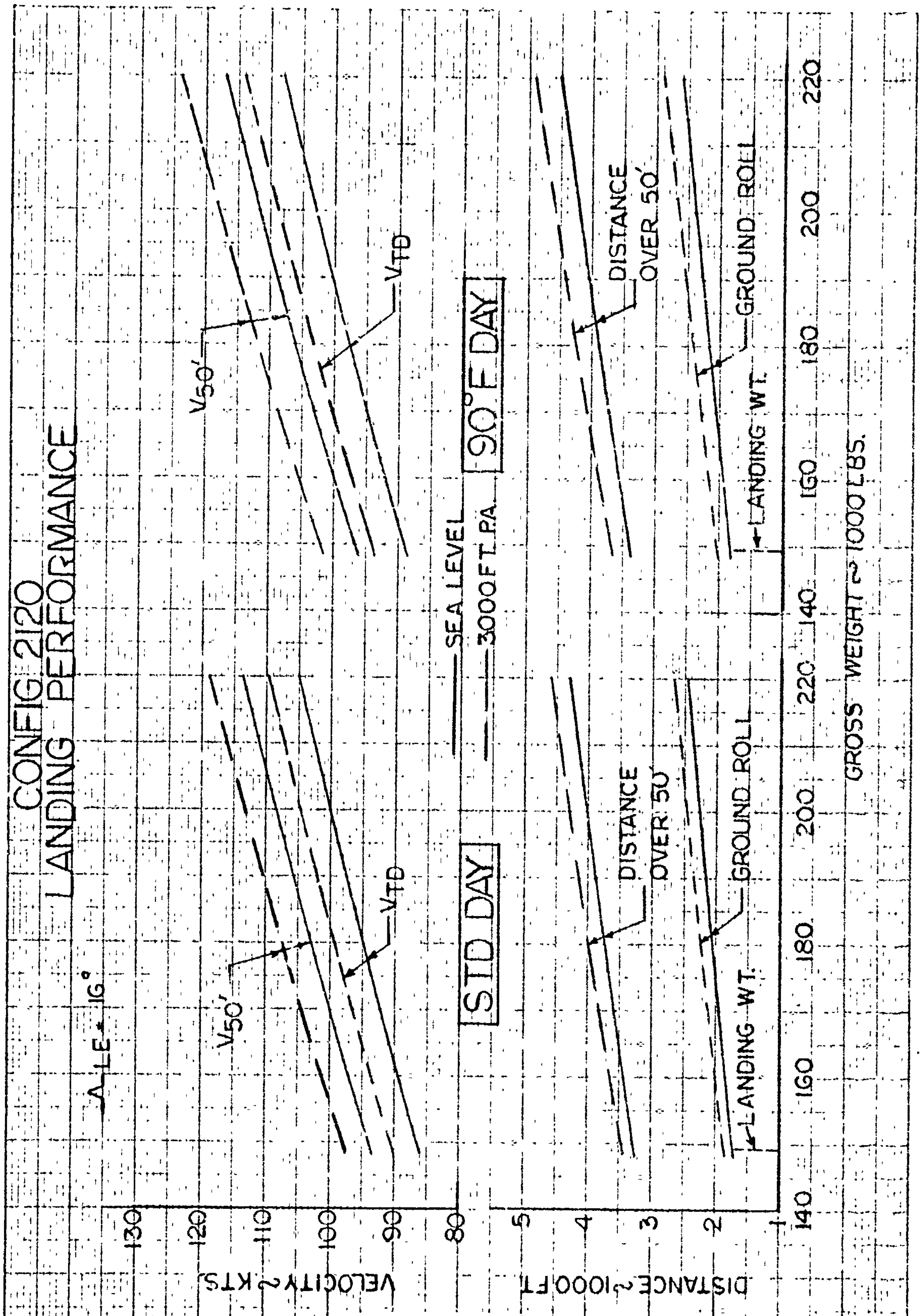

CONFIG 2120
LANDING PERFORMANCE
A LE = 16°
SEA LEVEL
3000 FT PA
STD DAY
90°F DAY
$V_{50'}$
V_{TD}
DISTANCE OVER 50'
GROUND ROLL
LANDING WT.
VELOCITY ~ KTS
DISTANCE ~ 1000 FT
GROSS WEIGHT ~ 1000 LBS.
130
120
110
100
90
80
5
4
3
2
1
140
160
180
200
220

FIGURE 3.3-2

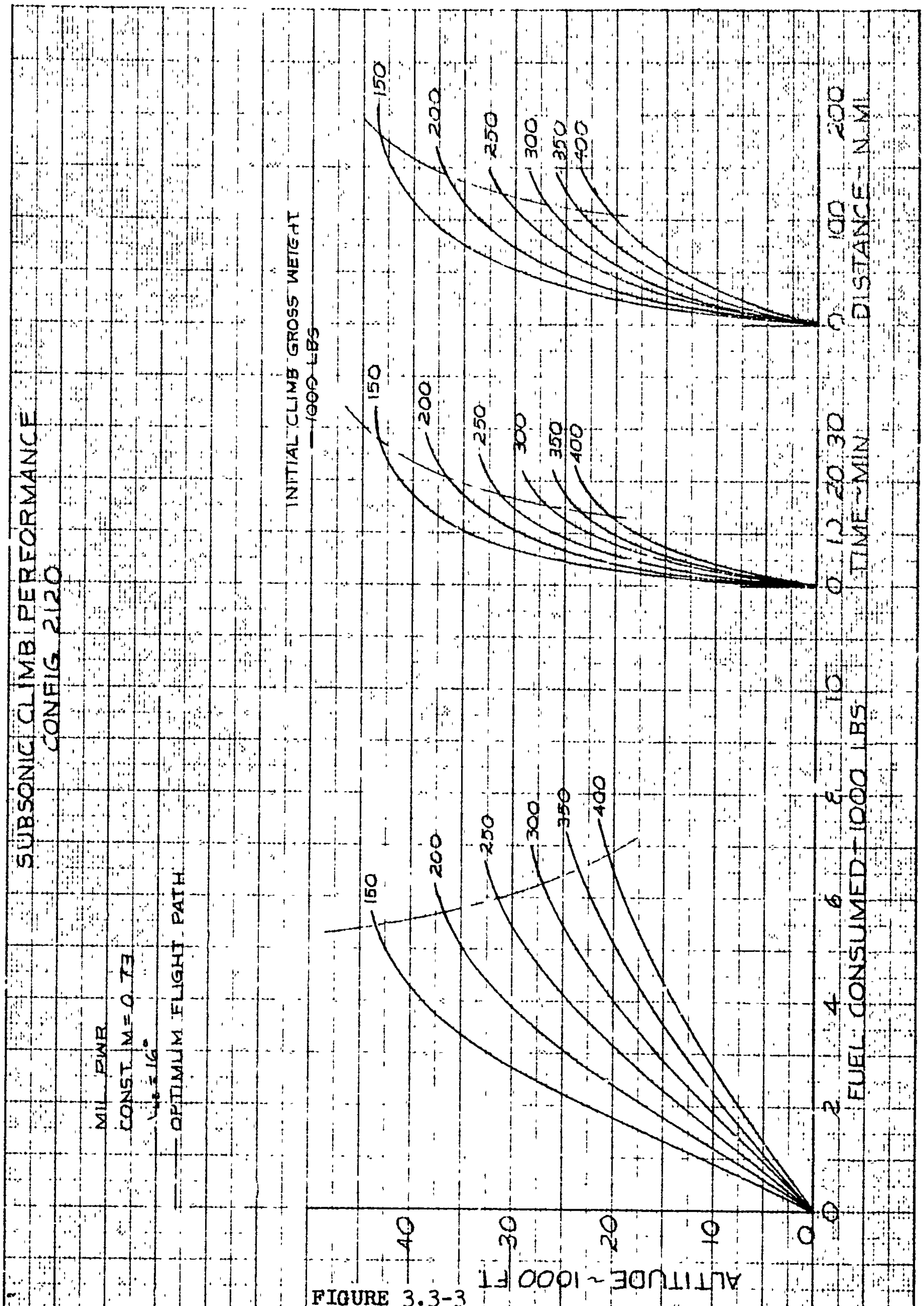

SUBSONIC CLIMB PERFORMANCE
CONFIG 2120
MIL PWR
CONST M=0.73
OPTIMUM FLIGHT PATH
INITIAL CLIMB GROSS WEIGHT
—1000 LBS
150
200
250
300
350
400
FUEL CONSUMED~1000 LBS
TIME~MIN
DISTANCE~N.MI
ALTITUDE~1000 FT

FIGURE 3.3-3

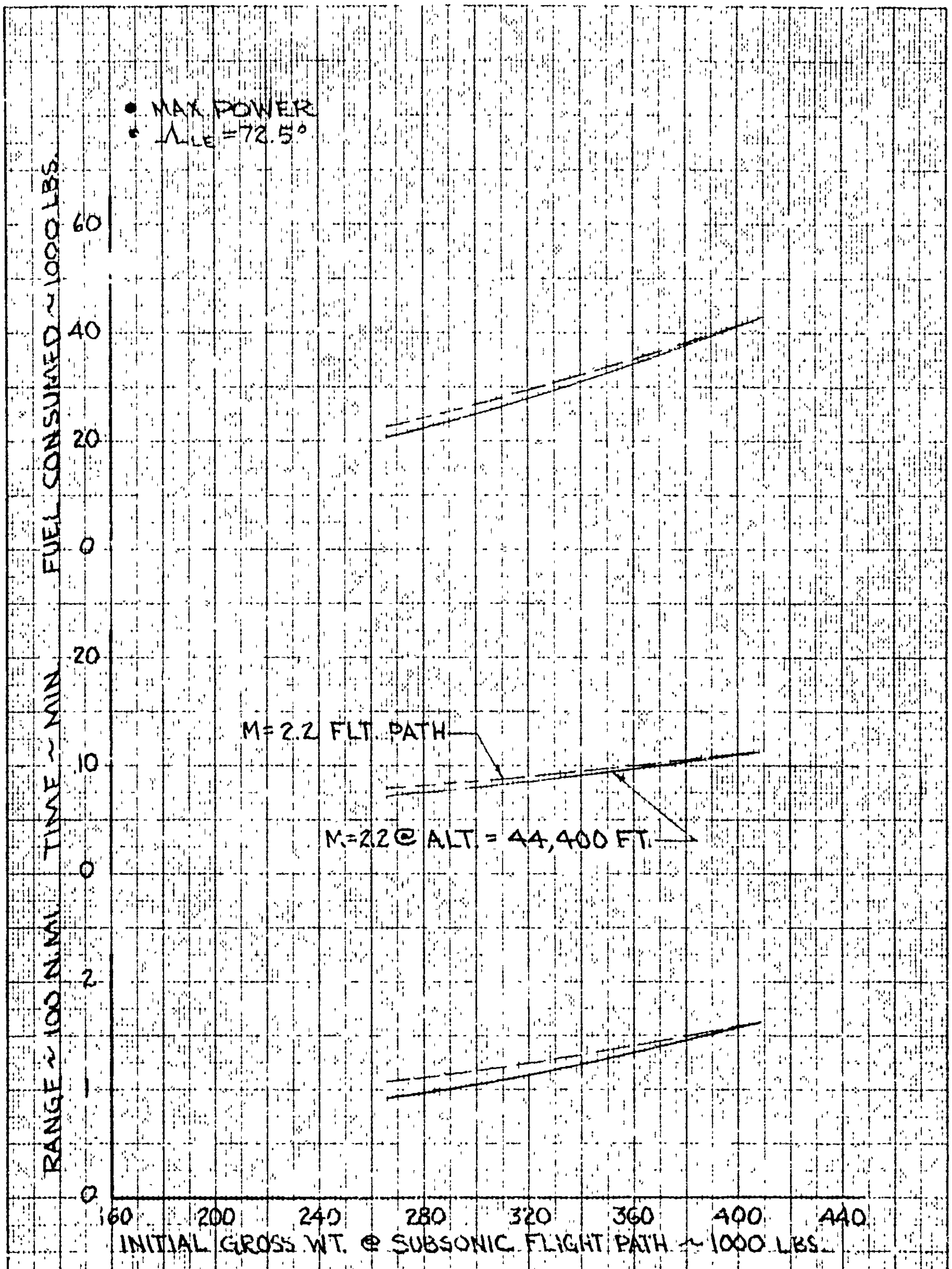


FIGURE 3.3-4 SUPERSONIC ACCELERATION PERFORMANCE FROM SUBSONIC FLIGHT PATH CONFIG. 2120

followed is to accelerate at constant altitude, as determined by the subsonic flight path altitude, to the structural limit speed (see Figure 3.3-31 of FZM 4038-II-1) followed by a climb and acceleration along the structural limit to Mach 2.2 at 44,400 ft. Climb to the Mach 2.2 optimum flight path then follows.

3.3.5 Cruise and Dash Performance

Flight paths were optimized by the same procedure described in Section 3.3.2.5 of FZM-4038-II-1. All installed TSFC values derived from engine manufacturer's data have been increased 5% in accordance with MIL-C-5011A rules. The optimum altitude and specific range data for maximum range subsonic cruise as a function of gross weight are presented in Figure 3.3-5. Specific range data versus gross weight for Mach .85 and Mach 1.2 sea level dash are presented in Figures 3.3-6 and 3.3-7. Optimum altitude and specific range data for maximum Mach 2.2 cruise as a function of gross weight are presented in Figure 3.3-8.

3.3.6 Refuel Compatibility

The speed-altitude compatibility between the KC-135A tanker and Configuration 2120 has not been investigated; however, the degree of compatibility should be improved over that previously presented for Configuration 2010-H in Figure 3.2-26 of FZM 4038-II-1 due to the change in takeoff wing loading from 194 PSF to 175 PSF.

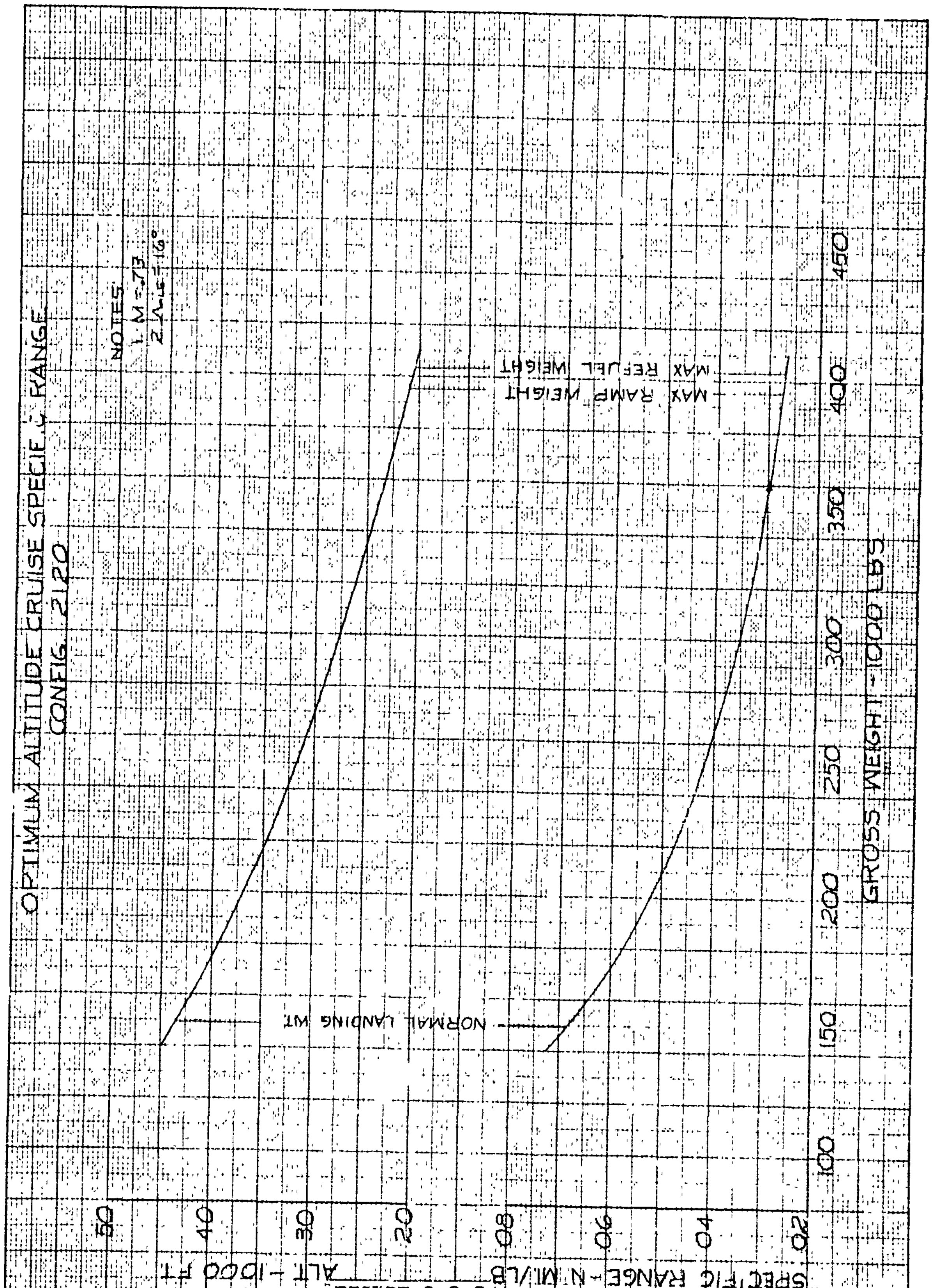

OPTIMUM ALTITUDE CRUISE SPECIFIC RANGE
CONFIG. 2120
NOTES:
1. M=.73
2. Λ_{LE}=16°
MAX REFUEL WEIGHT
MAX RAMP WEIGHT
NORMAL LANDING WT
100
150
200
250
300
350
400
450
GROSS WEIGHT - 1000 LBS
50
40
30
20
ALT - 1000 FT
.08
.06
.04
.02
SPECIFIC RANGE - N MI/LB

FIGURE 3.3-5

SECRET

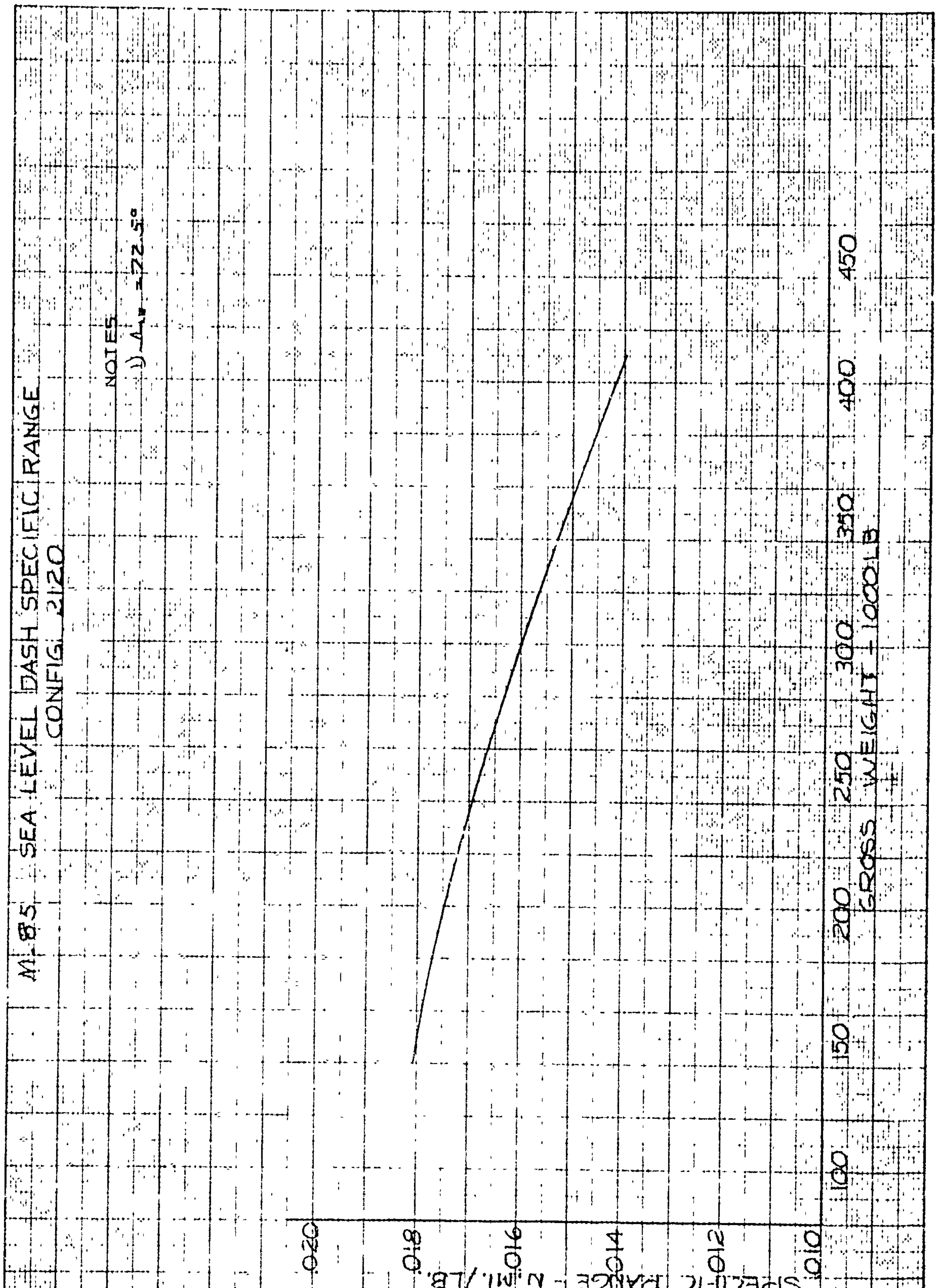


FIGURE 3.3-6

SECRET

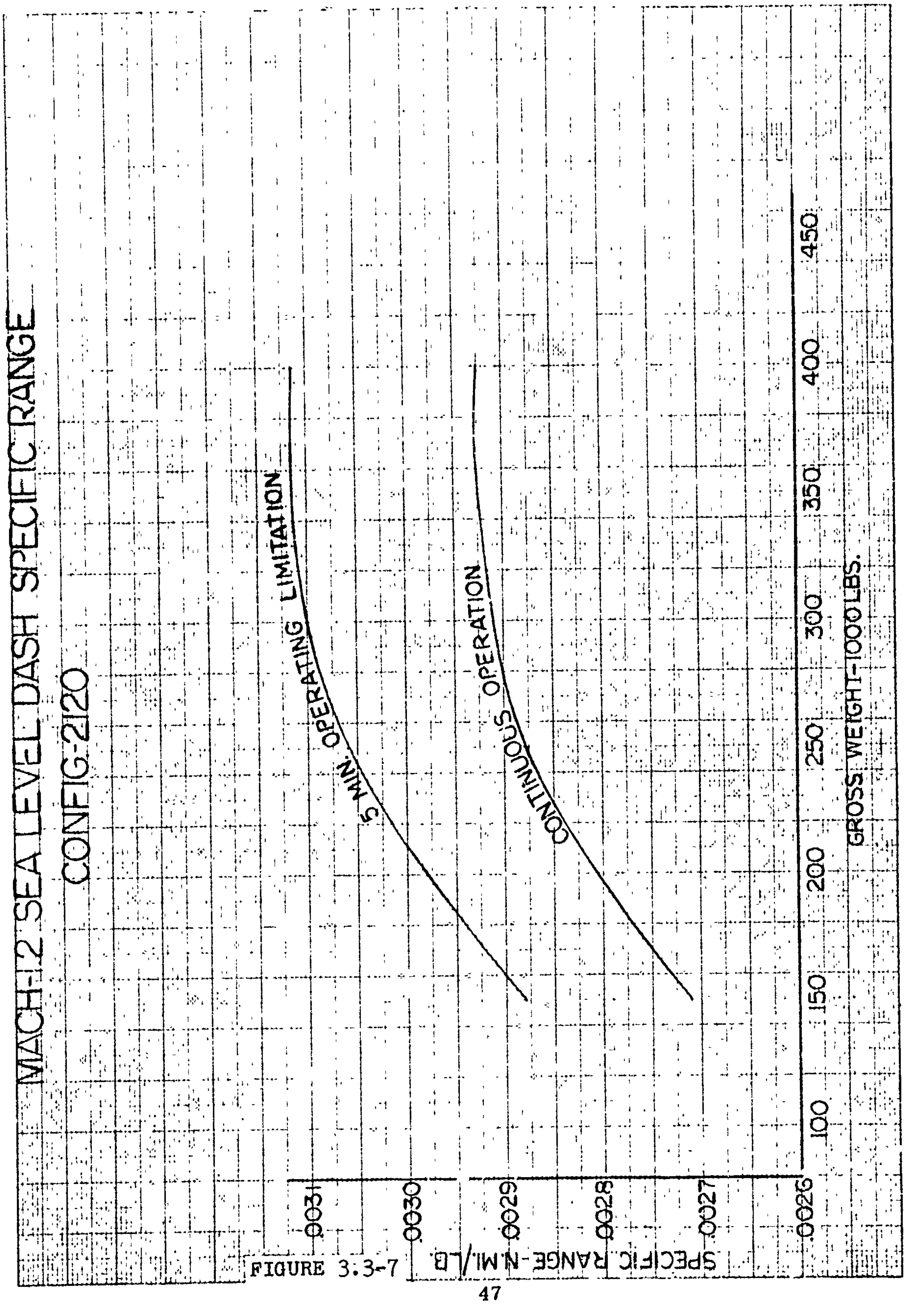

MACH-1.2 SEA LEVEL DASH SPECIFIC RANGE
CONFIG-2120
5 MIN. OPERATING LIMITATION
CONTINUOUS OPERATION
100
150
200
250
300
350
400
450
GROSS WEIGHT-1000 LBS.
.0031
.0030
.0029
.0028
.0027
.0026
SPECIFIC RANGE-N.MI./LB

FIGURE 3.3-7

SECRET

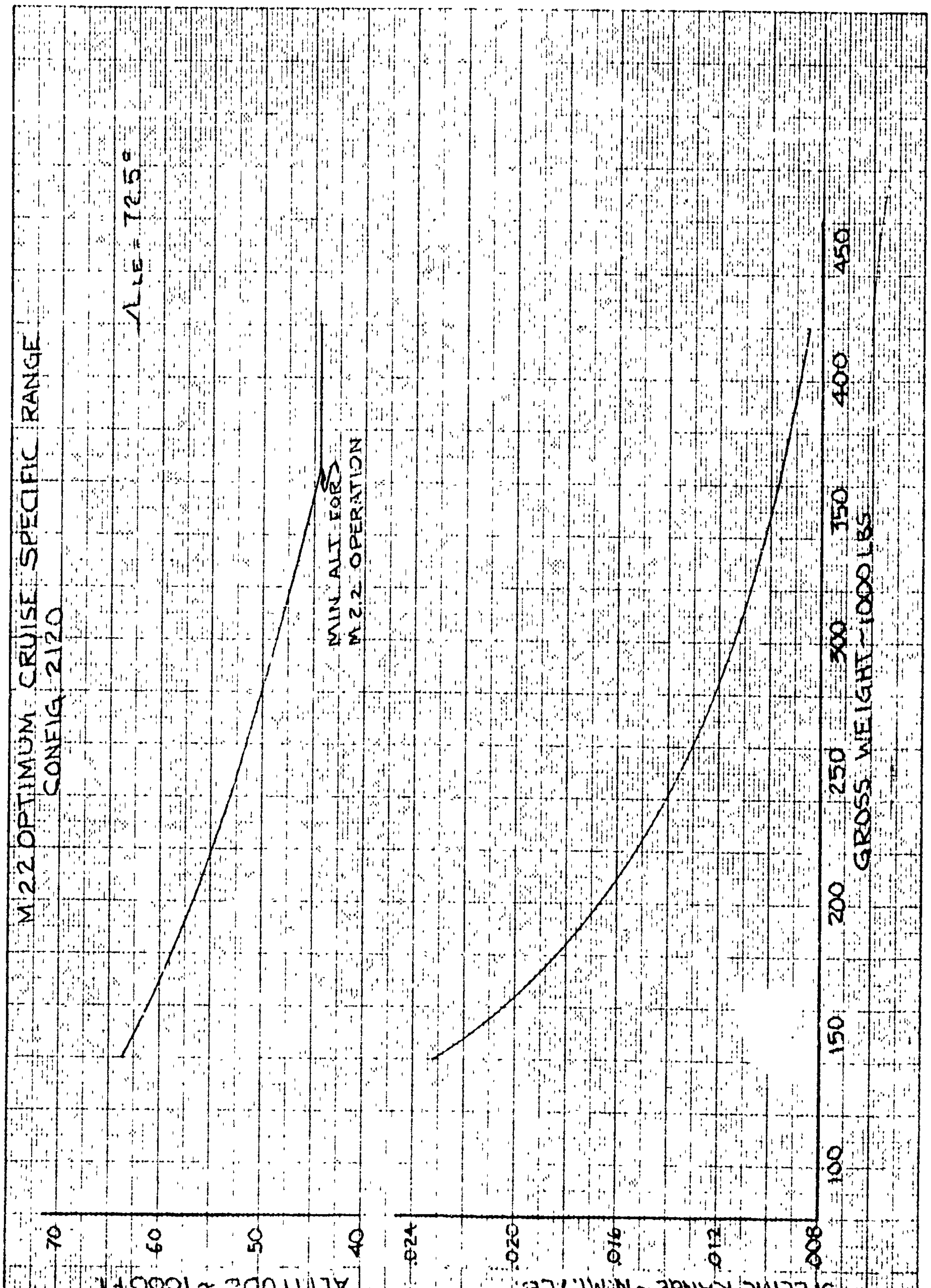


FIGURE 3.3-8

SECRET

3.3.7 Mission Capability

The Mach .85 sea level penetration capability of Configuration 2120 (design gross weight of 395, 000 lb.) is presented in Figure 3.3-9 for symmetrical refueled and non-refueled missions. The refueled missions are presented using both 1000 n. mi. post-refuel stage KC-135A tankers and radius KC-135A tankers. The configuration performance exceeds the design non-refueled mission requirements of 6300 n. mi. total range with 2000 n. mi. Mach .85 sea level dash by 20 n. mi. of high altitude cruise. The design refueled mission of 9000 n. mi. total range with 2000 n. mi. Mach .85 sea level dash is exceeded by 380 n. mi. A design gross weight of 390, 000 lb. is required to meet the design non-refueled mission.

The Mach .85 sea level penetration capability for a mission with the dash occurring just prior to a 500, 1000, or 1500 n. mi. recovery range at altitude is compared to the symmetrical mission capability in Figure 3.3-10. With 2000 n. mi. sea level dash and 500 n. mi. high altitude recovery range, the aircraft can achieve 5710 n. mi. non-refueled and 8350 n. mi. refueled total ranges.

Figure 3.3-11 presents the refueled and non-refueled symmetrical mission capability for a Mach .85 sea level dash mission with 5 minutes of Mach 1.2 dash occurring prior to bomb drop. This is compared to the basic symmetrical Mach .85 sea level dash mission. Acceleration from Mach .85 to Mach 1.2 occurs with maximum reheat power. With 2000 n. mi. sea level dash (5 minutes at Mach 1.2) the aircraft can achieve 5700 n. mi. non-refueled and 8810 n. mi. refueled total ranges.

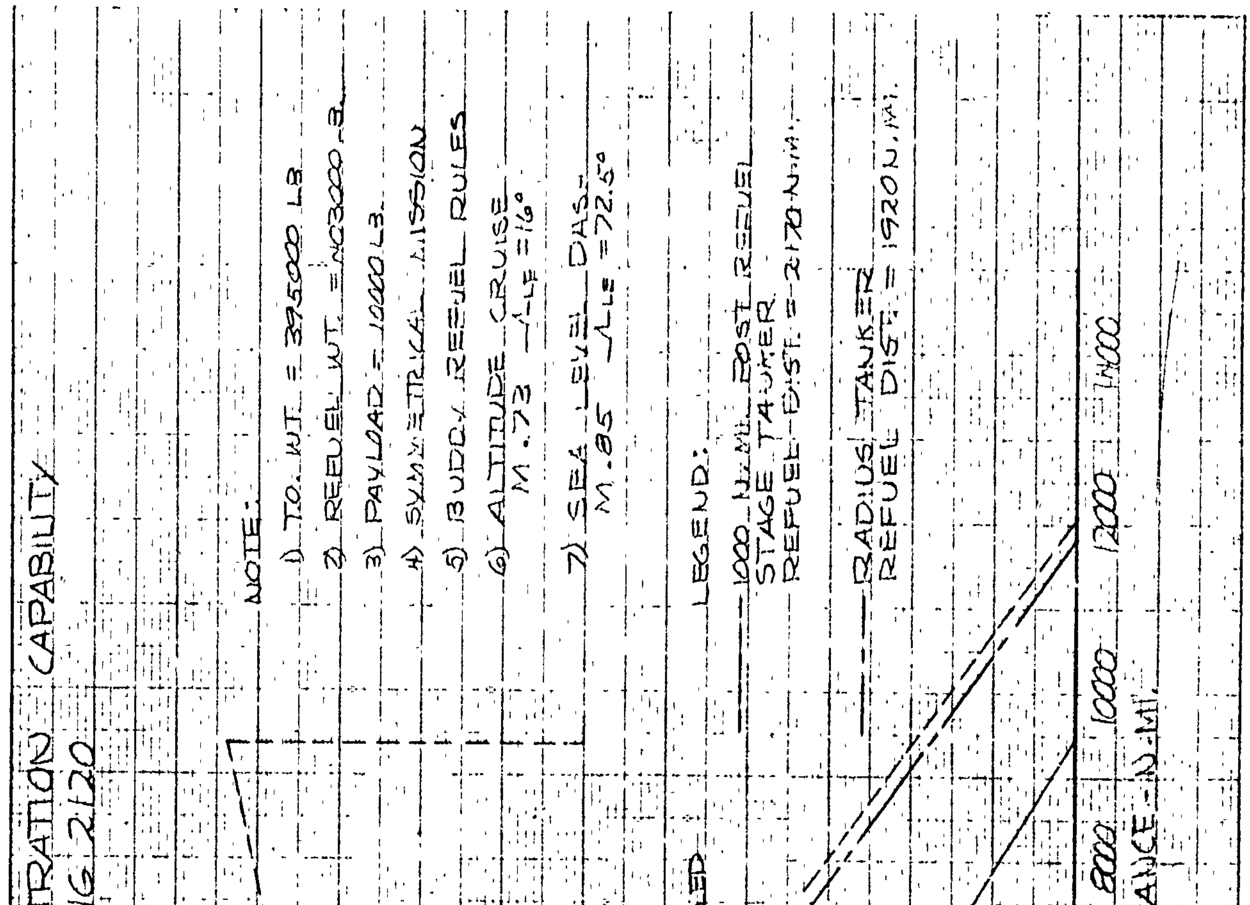

SEA LEVEL PENETRATION CAPABILITY
CONFIG 2120
NOTE:
1) T.O. WT. = 395000 LB.
2) REFUEL WT. = 403000 LB.
3) PAYLOAD = 10000 LB.
4) SYMMETRICAL MISSION
5) BUDDY REFUEL RULES
6) ALTITUDE CRUISE M .73 Λ LE = 16°
7) SEA LEVEL DASH M .85 Λ LE = 72.5°
LEGEND:
1000 N.MI. POST REFUEL STAGE TANKER
REFUEL DIST. = 2170 N.MI.
RADIUS TANKER
REFUEL DIST. = 1920 N.MI.
2600 N.MI. DASH
KC-135A
END REFUEL
NON-REFUELED
REFUELED
ALTITUDE - 1000 FT.
40
20
0
DASH DIST. - N.MI.
M .85 SEA LEVEL
4000
2000
0
0
2000
4000
6000
8000
10000
12000
14000
TOTAL DISTANCE - N.MI.

FIGURE 3.3-9

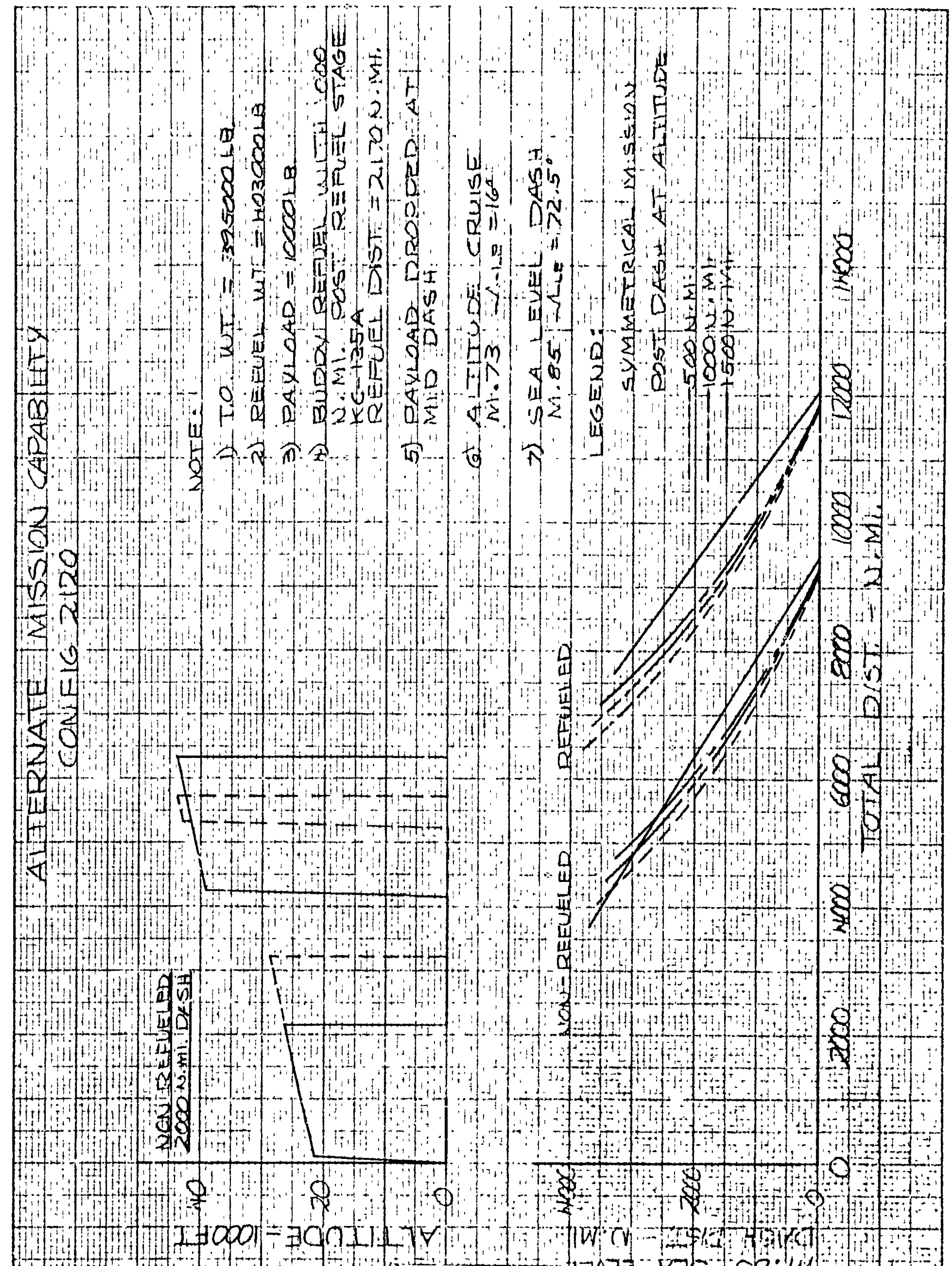

ALTERNATE MISSION CAPABILITY
CONFIG 2120
NOTE:
1) T.O. WT = 395000 LB
2) REFUEL WT = 403000 LB
3) PAYLOAD = 10000 LB
4) BUDDY REFUEL WITH 1000 N.MI. POST REFUEL STAGE KC-135A
REFUEL DIST = 2170 N.MI.
5) PAYLOAD DROPPED AT MID DASH
6) ALTITUDE CRUISE M.73
7) SEA LEVEL DASH M.85
LEGEND:
SYMMETRICAL MISSION
POST DASH AT ALTITUDE
500 N.MI.
1000 N.MI.
1500 N.MI.
NON REFUELED
2000 N.MI. DASH
NON-REFUELED
REFUELED
ALTITUDE - 1000 FT
M.85 SEA LEVEL DASH DIST - N.MI.
TOTAL DIST - N.MI.
0
2000
4000
6000
8000
10000
12000
14000

FIGURE 3.3-10

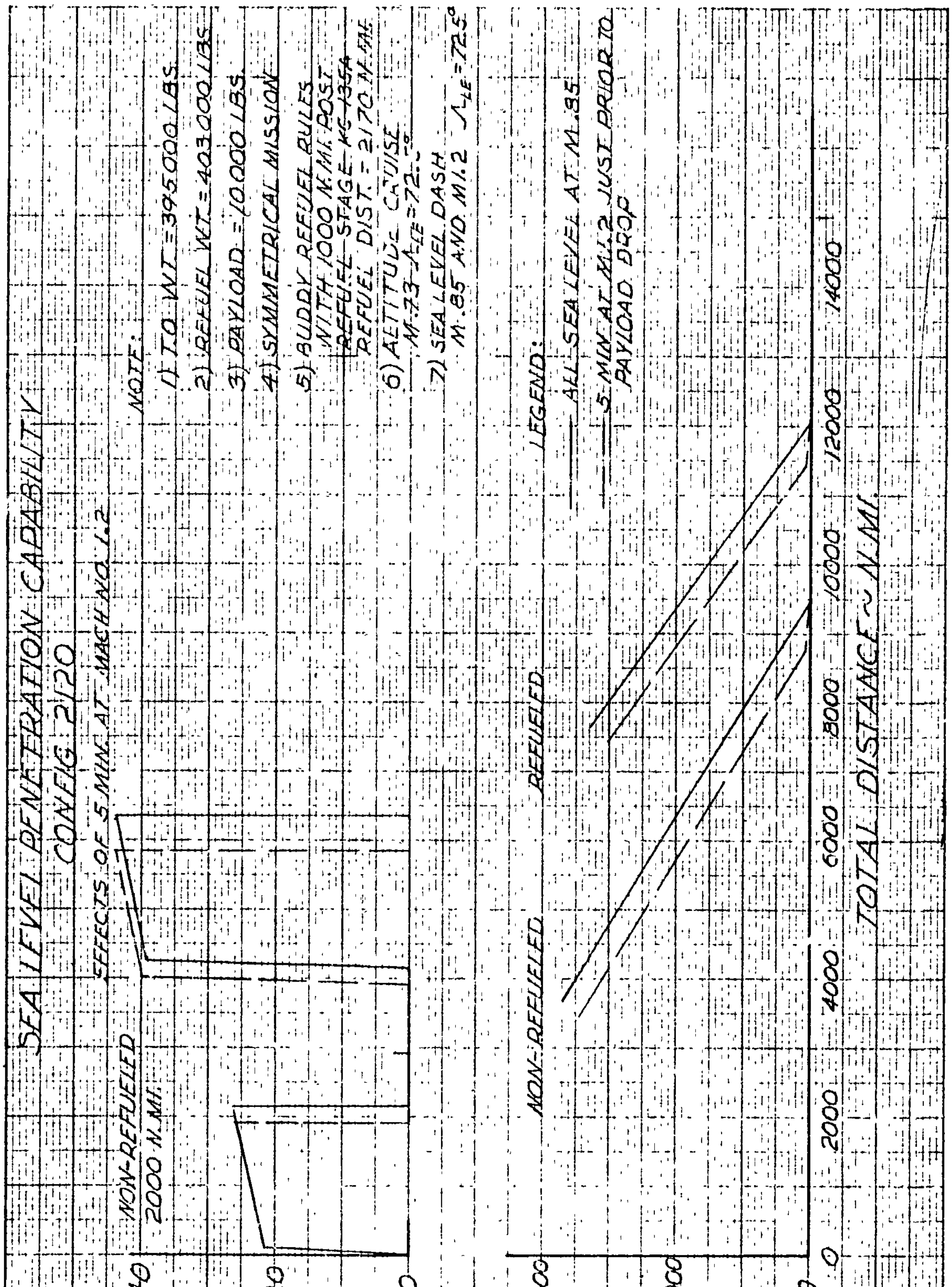

SEA LEVEL PENETRATION CAPABILITY
CONFIG 2120
EFFECTS OF 5 MIN. AT MACH NO. 1.2
NON-REFUELED
2000 N.MI.
NOTE:
1) T.O. WT. = 395,000 LBS.
2) REFUEL WT. = 403,000 LBS.
3) PAYLOAD = 10,000 LBS.
4) SYMMETRICAL MISSION
5) BUDDY REFUEL RULES WITH 1000 N.MI. POST REFUEL STAGE-KC-135A REFUEL DIST. = 2170 N.MI.
6) ALTITUDE CRUISE M.73 Λ_{LE}=72.5°
7) SEA LEVEL DASH M.85 AND M1.2 Λ_{LE}=72.5°
LEGEND:
ALL SEA LEVEL AT M.85
5 MIN AT M1.2 JUST PRIOR TO PAYLOAD DROP
NON-REFUELED
REFUELED
ALTITUDE ~ 1000 FT
40
20
0
SEA LEVEL DASH DIST ~ N.MI.
4000
2000
0
0
2000
4000
6000
8000
10000
12000
14000
TOTAL DISTANCE ~ N.MI.

FIGURE 3.3-11

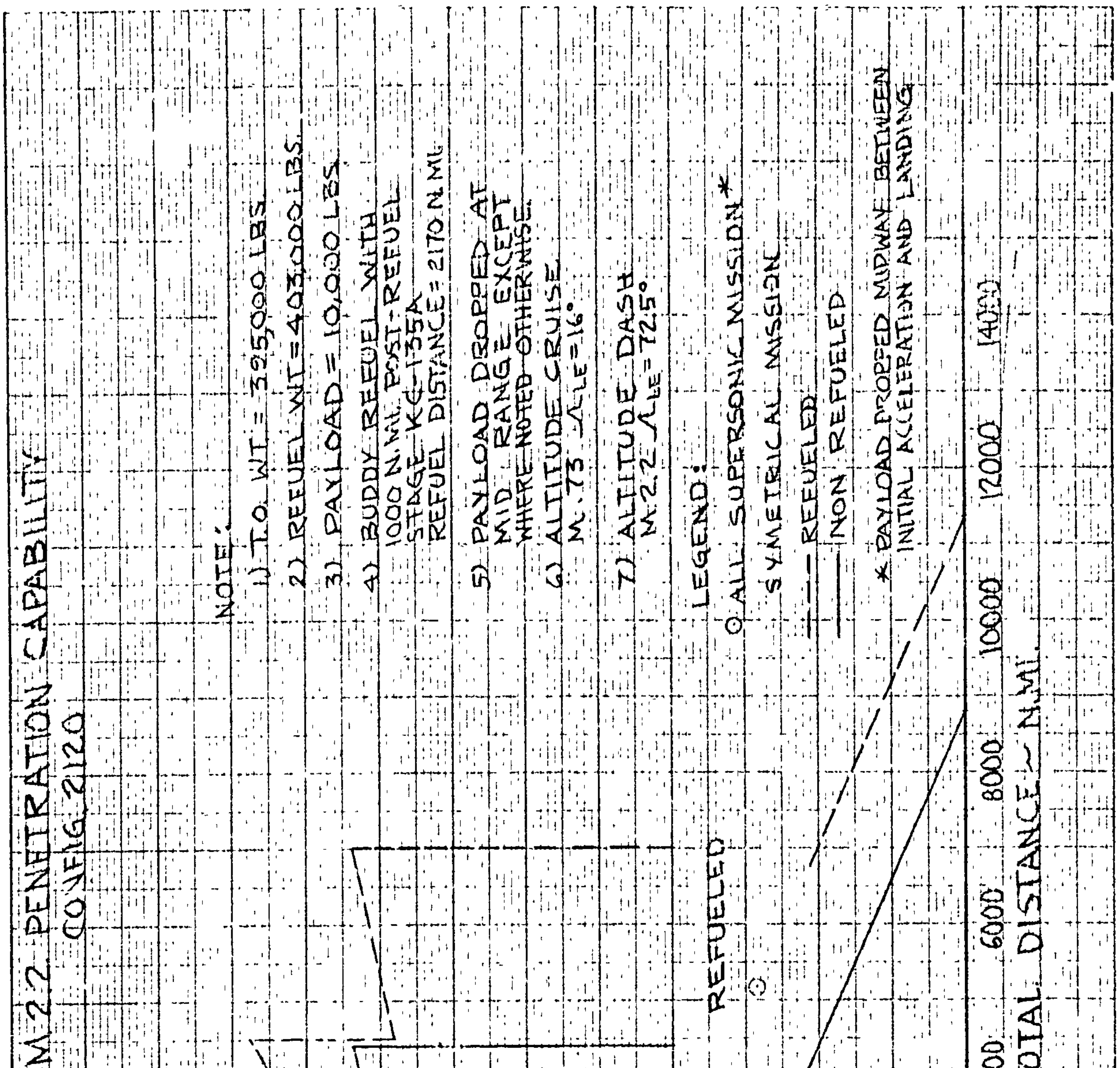

M 2.2 PENETRATION CAPABILITY
CONFIG 2120
NOTE:
1) T.O. WT = 325,000 LBS
2) REFUEL WT = 403,000 LBS
3) PAYLOAD = 10,000 LBS
4) BUDDY REFUEL WITH 1000 N.MI. POST-REFUEL STAGE KC-135A
REFUEL DISTANCE = 2170 N.MI.
5) PAYLOAD DROPPED AT MID RANGE EXCEPT WHERE NOTED OTHERWISE
6) ALTITUDE CRUISE M .73 Λ LE = 16°
7) ALTITUDE DASH M 2.2 Λ LE = 72.5°
LEGEND:
ALL SUPERSONIC MISSION *
SYMETRICAL MISSION
REFUELED
NON REFUELED
* PAYLOAD DROPPED MIDWAY BETWEEN INITIAL ACCELERATION AND LANDING
END REFUEL
NON-REFUELED
REFUELED
ALTITUDE ~ 1000 FT
60
40
20
0
M 2.2 DISTANCE
3000
2000
1000
0
0
2000
4000
6000
8000
10000
12000
14000
TOTAL DISTANCE ~ N.MI.

FIGURE 3.3-12

Figure 3.3-12 presents the Mach 2.2 penetration capability for the refueled and non-refueled symmetrical missions. The refueled mission is based on a 1000 n. mi. post-refuel stage KC-135A. The configuration can achieve a non-refueled total range of 4410 n. mi. (vs. a design requirement of 3300 n. mi.) with 2000 n. mi. of Mach 2.2 dash.

All mission rules and allowances are consistent with MIL-C-5011A with the exception that no fuel allowance is made for combat. The mission rules are described in Section 3.3.2.7 of FZM-4038-II-1.

3.3.8 Minimum Fuel Flow

The minimum fuel flow for maximum endurance of Configuration 2120 is presented in Figure 3.3-13 for sea level and altitude operation along with the optimum Mach numbers and altitudes. The maximum loiter time over base is 25.0 hours, and maximum loiter time between refuels is 26.1 hours. All fuel flows are increased 5% as required by MIL-C-5011A rules.

3.3.9 Effect of Payload Weight

The effect of payload weight on subsonic mission capability is presented in Figure 3.3-14 for Configuration 2120. The basic mission has a 10,000-lb payload in the forward bay and an 18,000-lb capabity fuel tank in the aft bay filled to only 10,000 lb at takeoff. After refuel, the aft bay tank is filled to capacity. For lighter payloads, the takeoff weight remains constant by

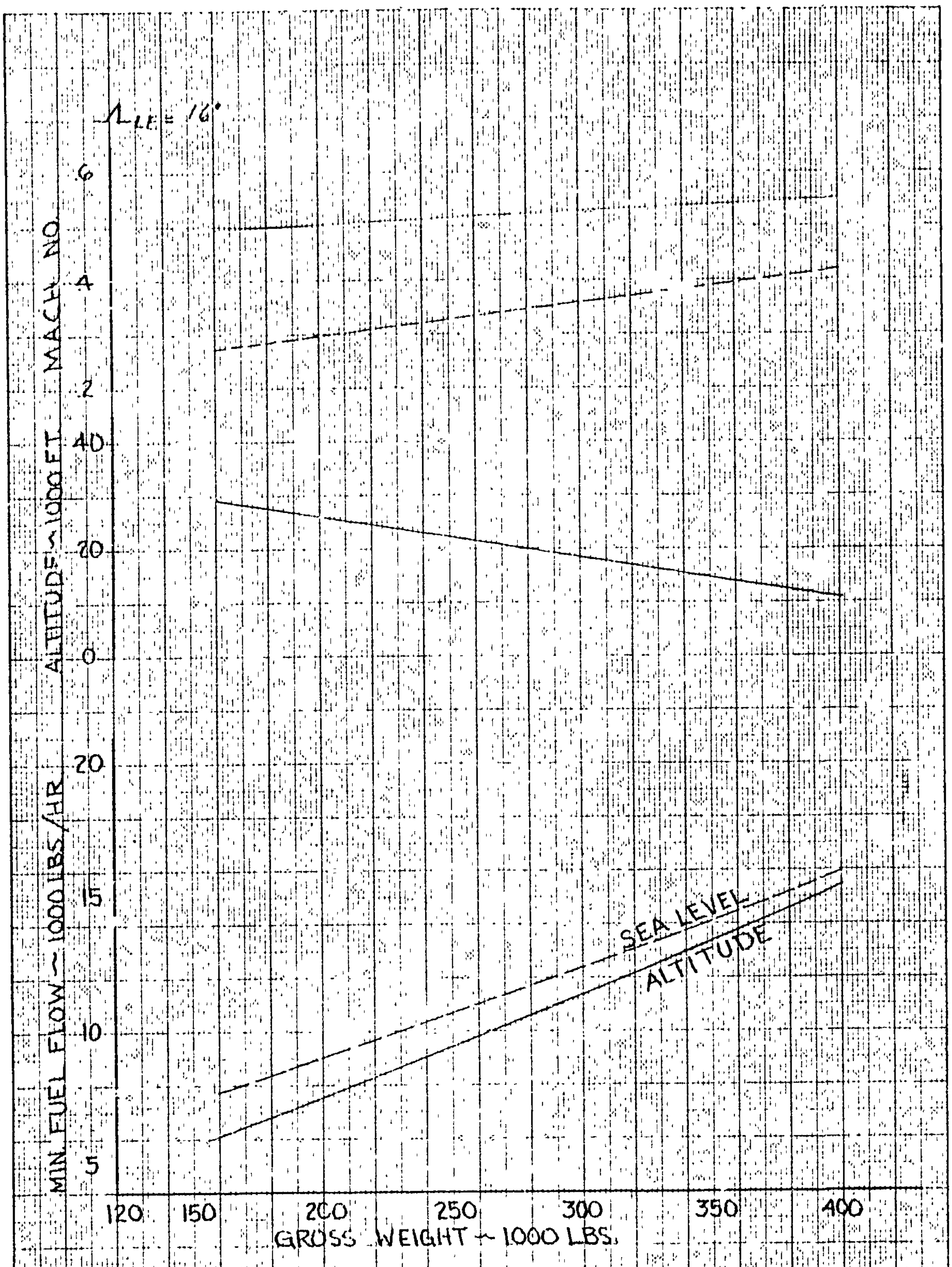
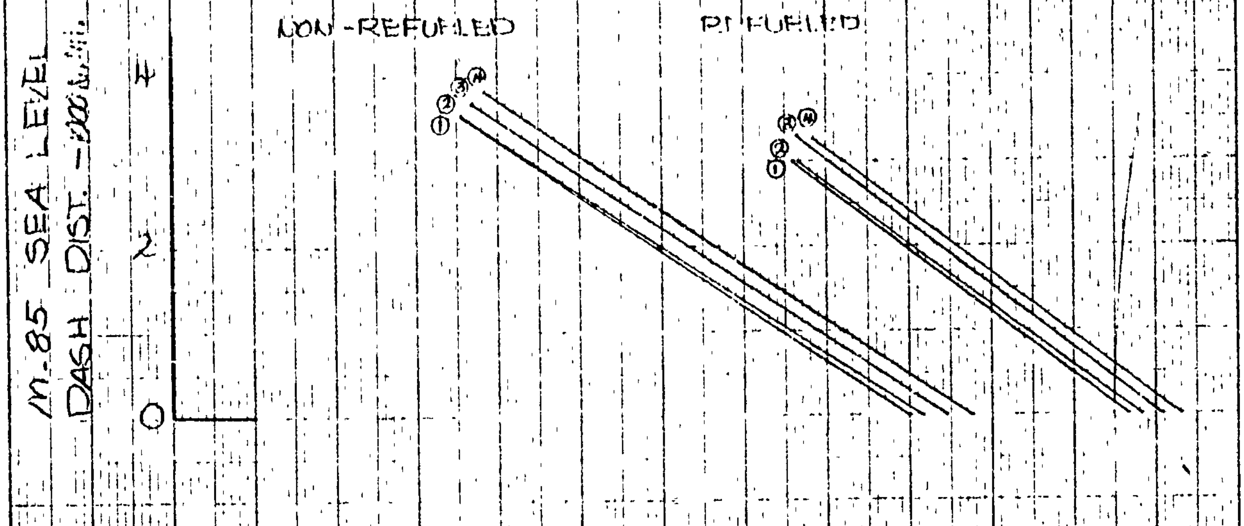


FIGURE 3.3-13 MAXIMUM ENDURANCE CONFIG. 2120

SUBSONIC MISSION CAPABILITY



LEGEND	PAYLOAD	DRY WT. W/O PAYLOAD	NON-REFUELED BOMB BAY FUEL	NON-REFUELED G.W.	REFUELED BOMB BAY FUEL	REFUELED G.W.
1	20000	133925	0	394520	0	394520
2	10000	133925	0	384520	0	384520
3	10000	134405	10000	395000	18000	403000
4	2000	134405	18000	395000	18000	395000

NOTE:
1) ALL PAYLOAD CARRIED INTERNALLY
2) BASED ON 2000 N.MI. M.85 SL DASH

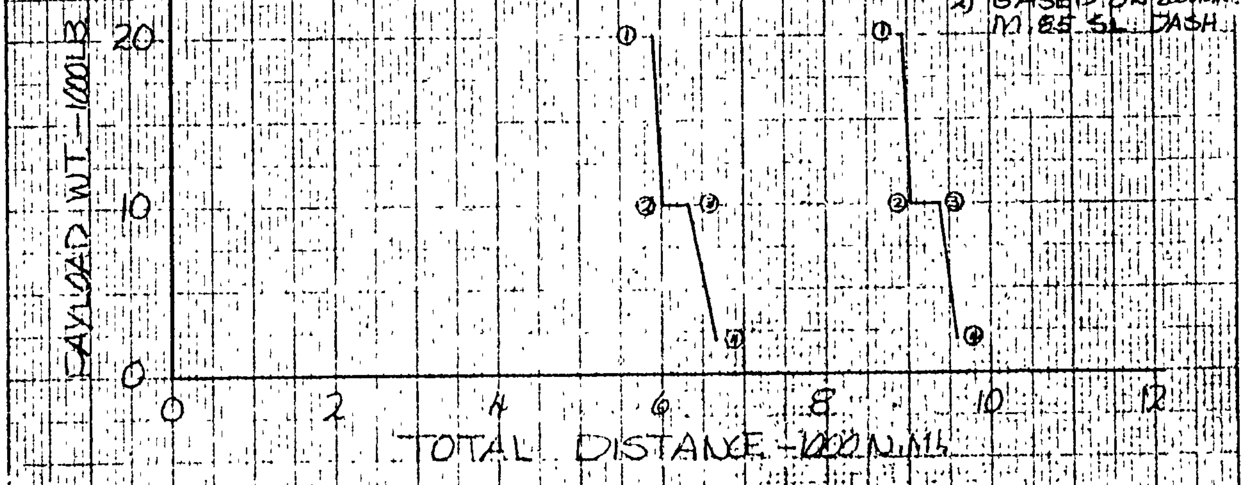


FIGURE 3.3-14 EFFECT OF PAYLOAD ON RANGE
CONFIG 2120

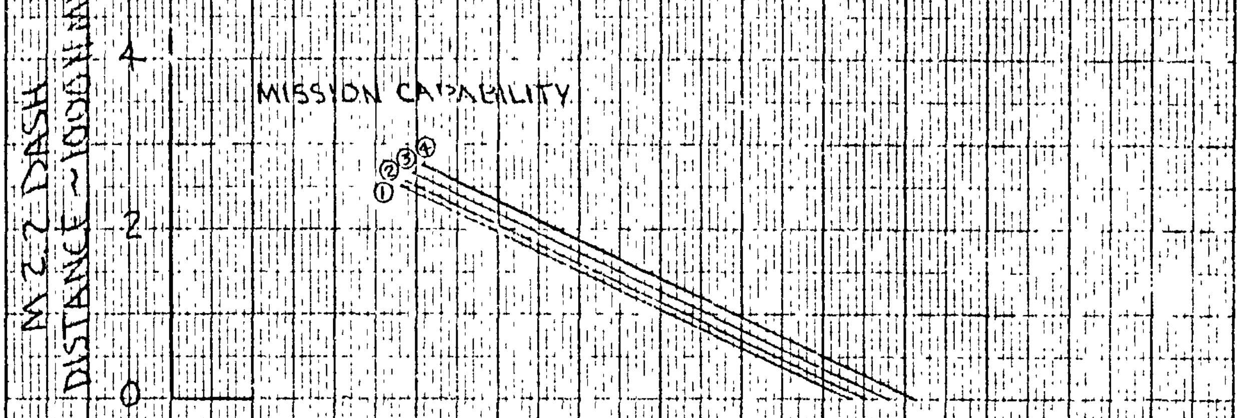
adding more fuel to the aft bay fuel tank. The refuel weight is reduced since the aft bay fuel tank is already filled to capacity on the basic refueled missions. For payloads heavier than 10,000 lbs, the aft bay fuel tank is removed to make room for the additional payload. This tank weighs 1680 lbs. In addition, a 1200-lb bomb rack is installed in the aft bomb bay to accommodate the additional payload. The change in dry weight due to removal of the aft bay fuel tank and inclusion of the bomb rack is -480 lbs. Pertinent weights are tabulated in Figure 3.3-14.

Figure 3.3-15 presents the effect of payload weight on the Mach 2.2 mission capability. Total range is plotted versus payload for a symmetrical non-refueled mission that includes 2000 n. mi. at Mach 2.2.

3.3.10 Sensitivity of Range to Drag, TSFC, and Weight

The sensitivity of subsonic range capability to drag changes is presented in Figure 3.3-16. Data are presented for maximum zone, zero zone and 2000 n. mi. zone using a symmetrical refueled mission with a 1000 n.mi. post-refuel stage KC-135A tanker and the non-refueled symmetrical mission. Thus, a plot of penetration zone versus total distance can be generated from these data for any drag level. This same type of trade data for TSFC changes are presented in Figure 3.3-17. The incremental change is subsonic range due to dry weight changes (holding constant gross weight) is presented in Figure 3.3-18.

SUPERSONIC MISSION CAPABILITY
NON-REFUELED



LEGEND	PAYLOAD	DRY WT W/O PAYLOAD	BOMB BAY FUEL	G. W.
1	20000	133925	0	394,520
2	10000	133925	0	384,520
3	10000	134405	10000	395,000
4	2000	134405	18000	395,000

NOTE:
1) ALL PAYLOAD CARRIED INTERNALLY
2) BASED ON 2000 N.MI. AT MACH 2.2

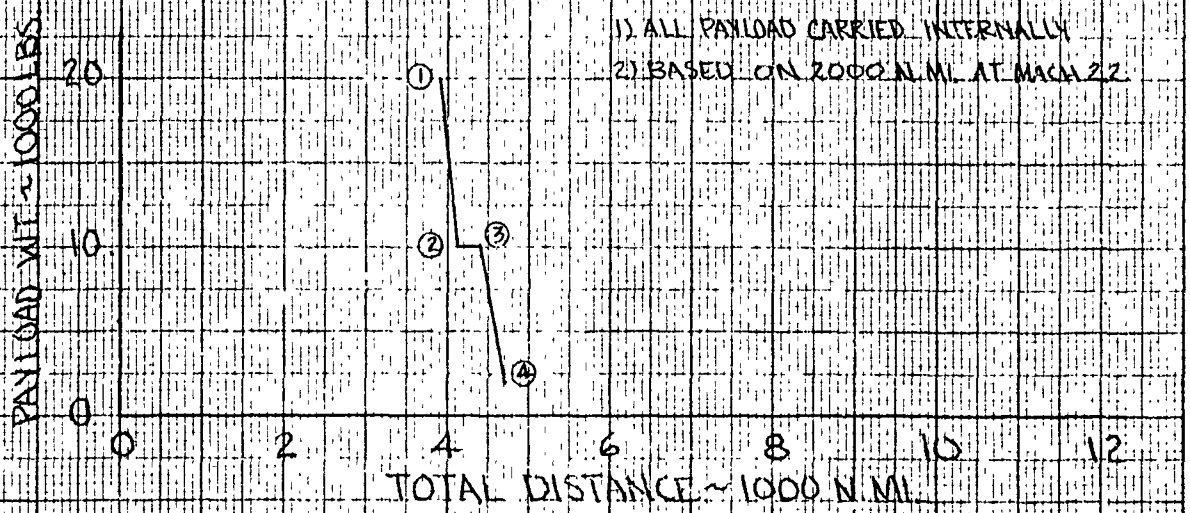


FIGURE 3.3-15

EFFECT OF PAYLOAD ON RANGE
CONFIG 2120

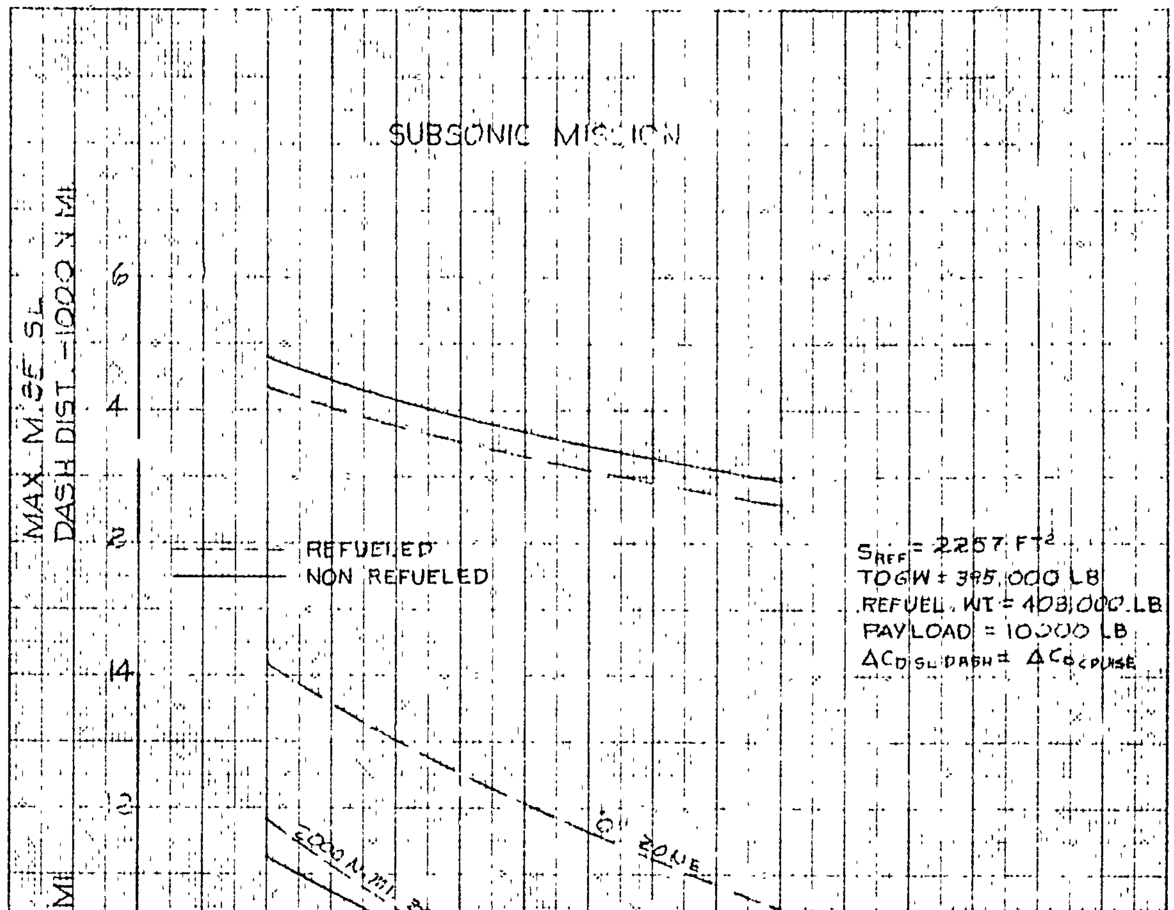


FIGURE 3.3-16 EFFECT OF DRAG CHANGES CONFIG. 2120

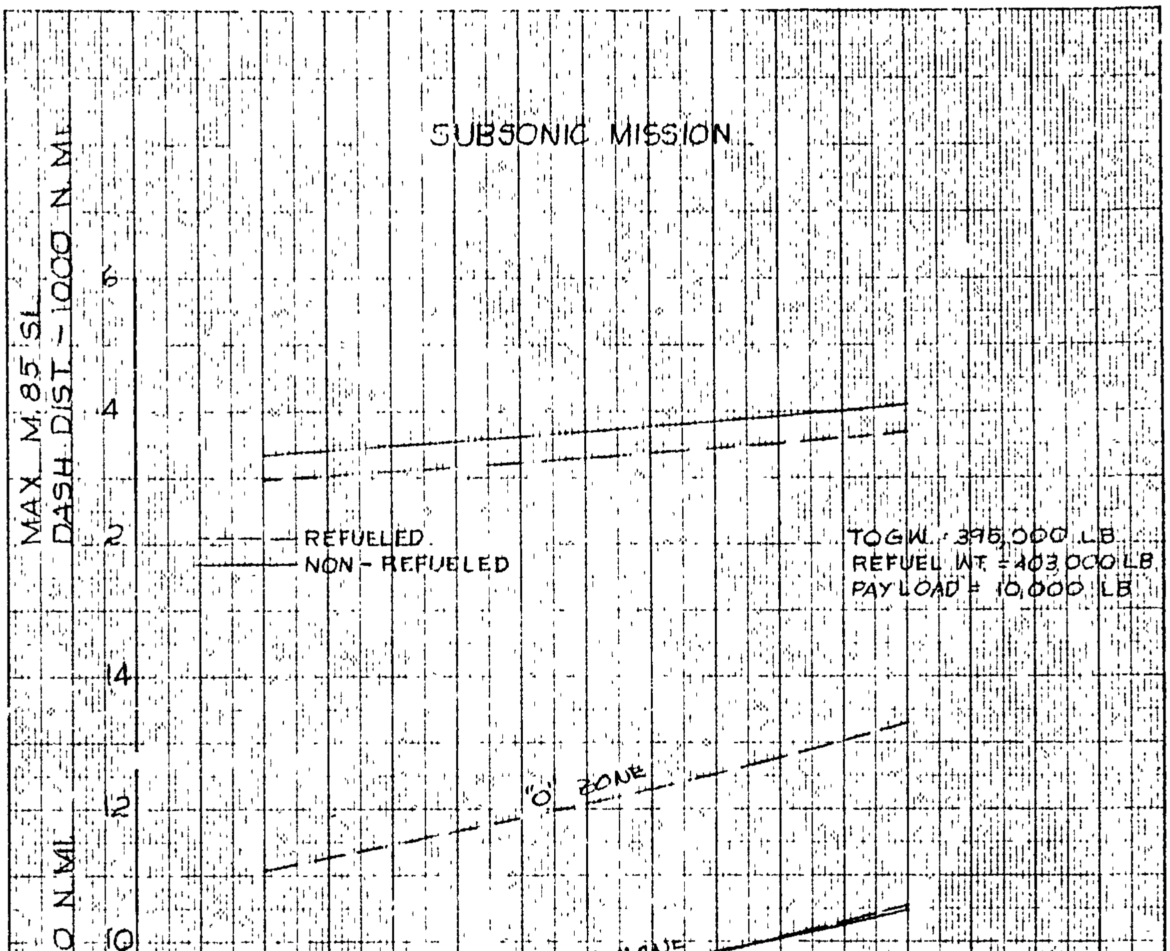


FIGURE 3.3-17 EFFECT OF TSFC CHANGES
CONFIG 2120

SECRET

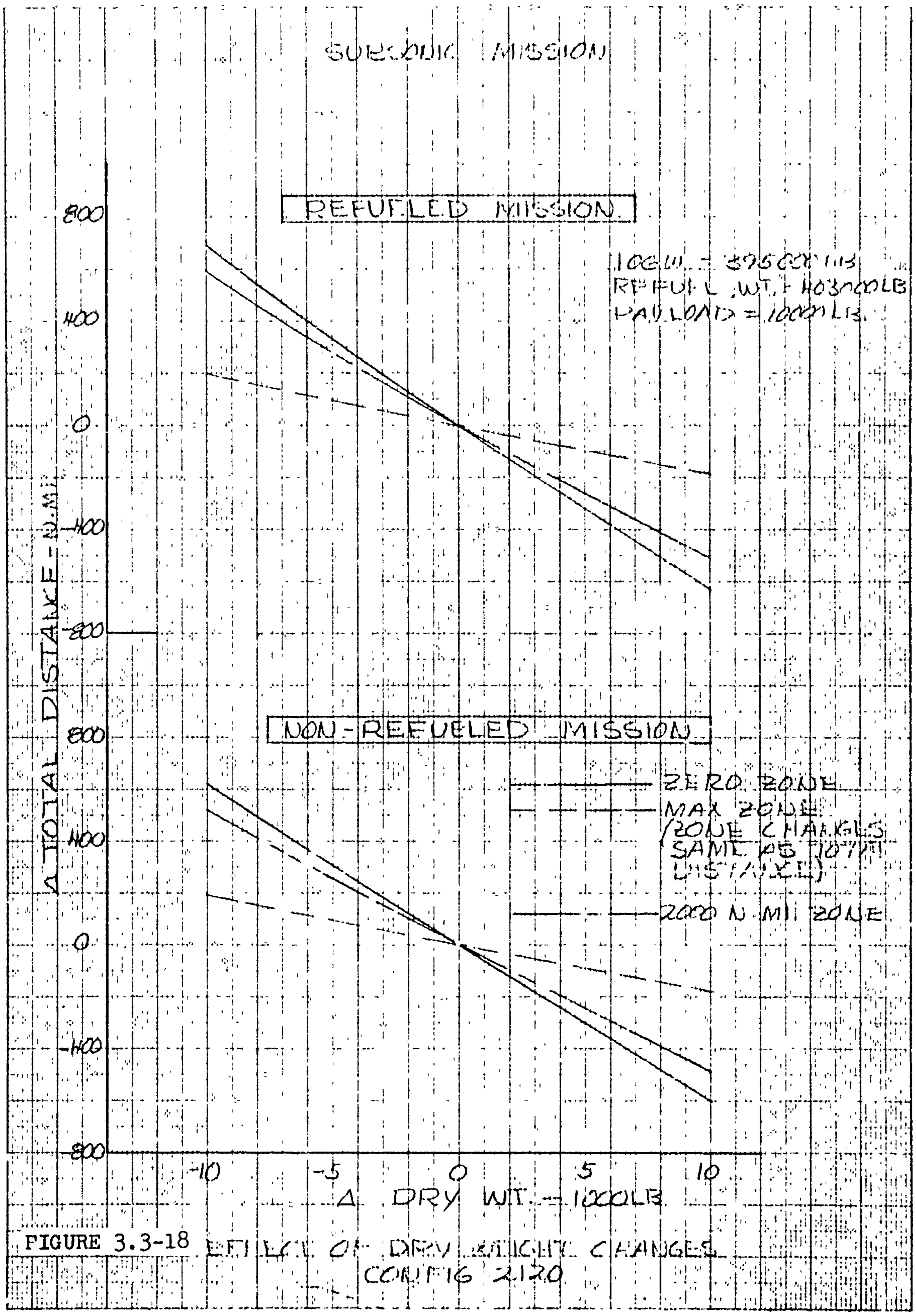


FIGURE 3.3-18 EFFECT OF DRY WEIGHT CHANGES CONFIG 2120

SECRET

3.3.11 Engine - Aircraft Matching

The engine scale for Configuration 2120 is sized by the 6000-ft takeoff distance requirement using a trimmed $C_{L_{MAX}}$ of 3.2. This engine scale of .426 permits adequate sea level dash thrust with optimum cruise at high altitudes and low gross weights approaching normal power (also military power).

Figure 3.3-19 presents the subsonic engine airframe match in terms of thrust, SFC and L/D. A near perfect match exists.

Figure 3.3-20 presents the supersonic engine-airplane match in terms of thrust, SFC, and L/D. The SFC match at Mach 2.2 is relatively good. Improvement in supersonic match through the use of larger engines will not have an appreciable effect on supersonic range due to the higher engine weight displacing fuel load to maintain a fixed wing loading of 175 PSF.

A tabular comparison of the thrust, SFC, and L/D match is presented below in Table 3.3-II.

TABLE 3.3-II ENGINE-AIRFRAME MATCH

CONFIGURATION 2120

	Mach .85 Sea Level	Mach .73 25,000 Ft.	Mach 2.2 55,000 Ft.
THRUST MATCH:			
Operating Thrust	30300 - 36700	16200	43400
Thrust at Min. SFC	47800(normal pwr)	16200	Min A/B 15000
Thrust at L/D MAX.	Above Mil pwr	18100	51500
SFC MATCH:			
Operating SFC	1.05 - 1.019	.85	1.901
MIN SFC	1.001	.85	Min A/B 1.61
SFC at L/D MAX	Above Mil pwr	.852	2.102
L/D MATCH:			
Operating L/D	6.0 - 9.6	20.4	5.05
MAX L/D	11.1	20.6	5.32

SECRET

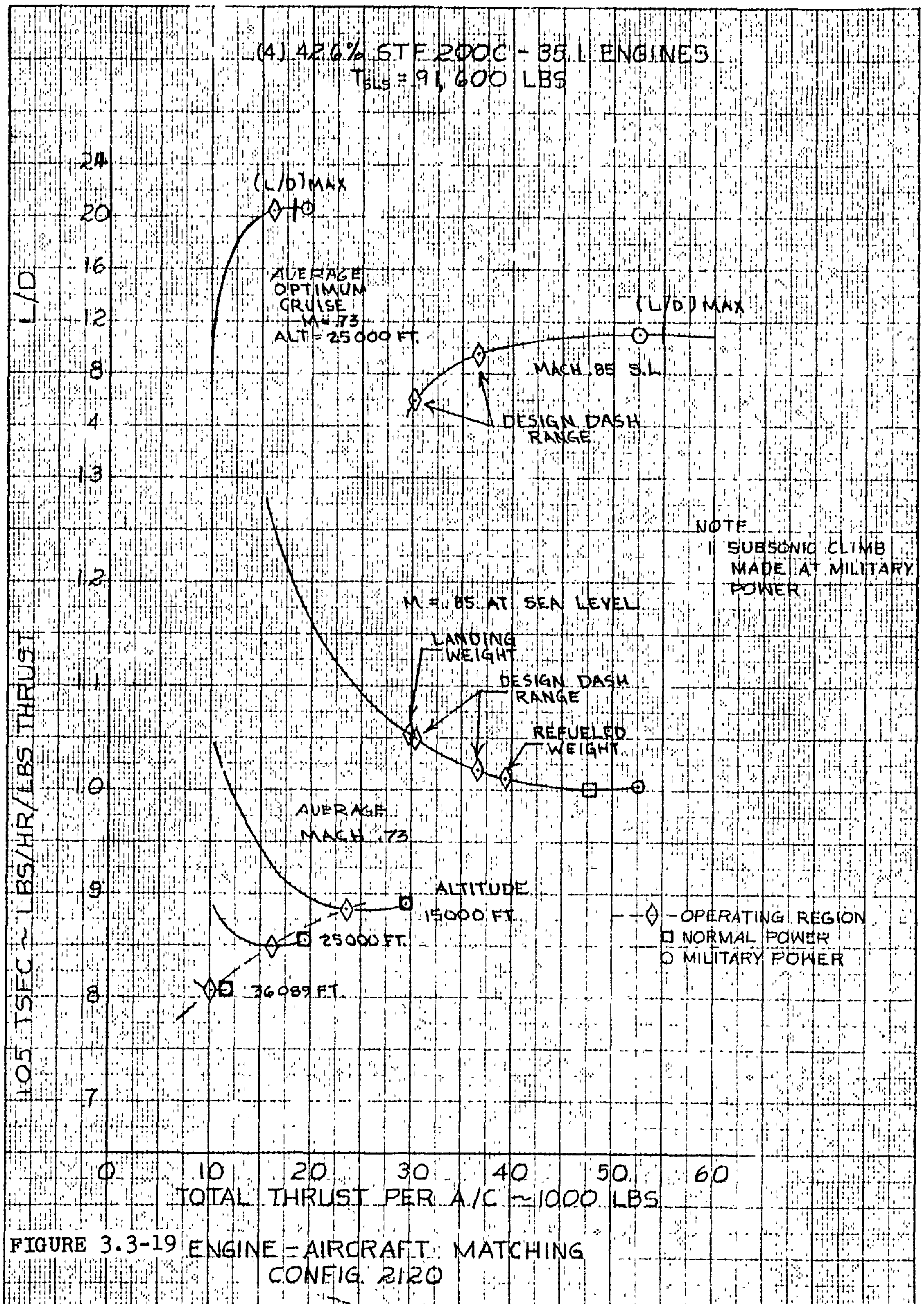


FIGURE 3.3-19 ENGINE - AIRCRAFT MATCHING
CONFIG. 2120

SECRET

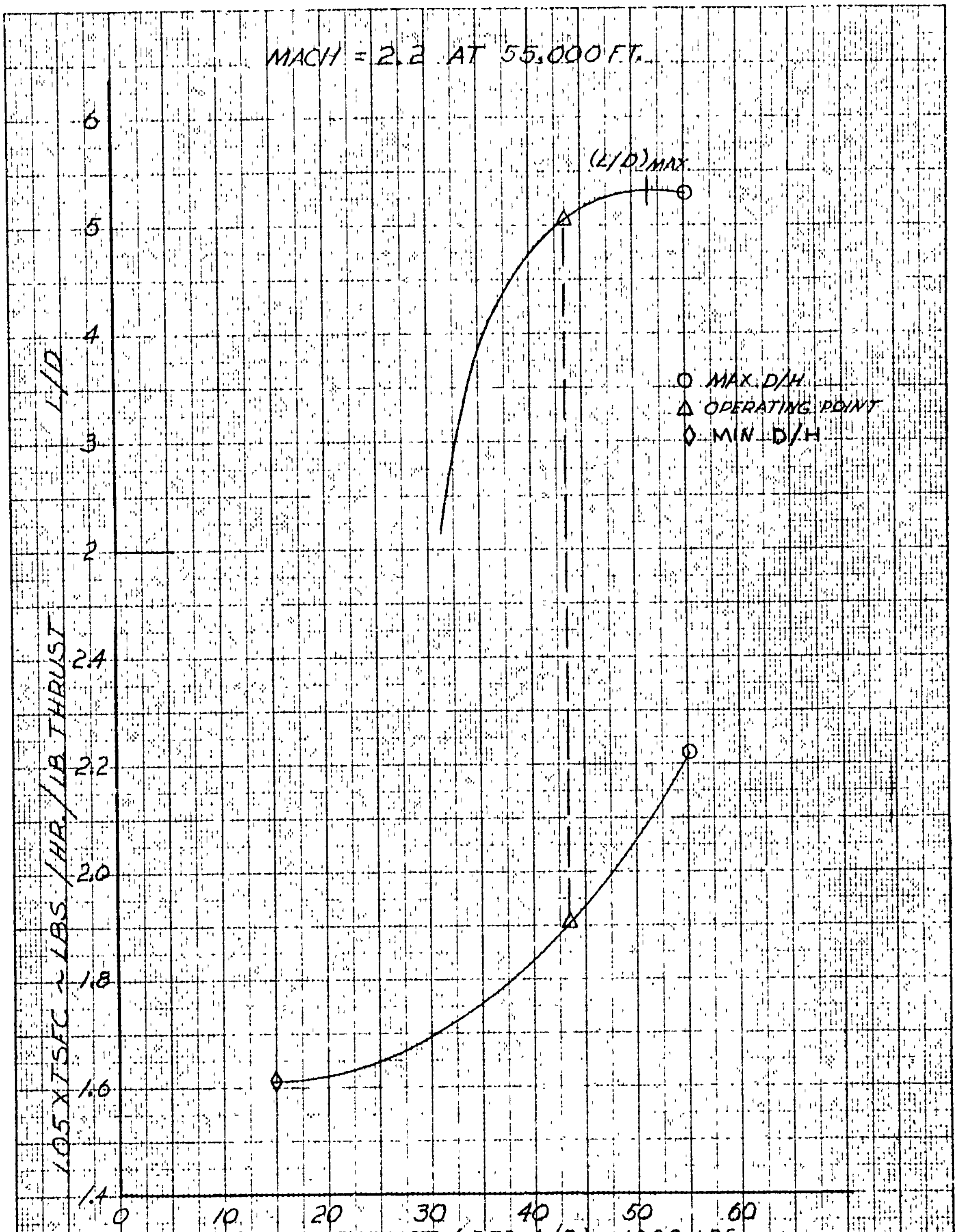


FIGURE 3.3-20 ENGINE-AIRCRAFT MATCHING
SUPERSONIC SPEEDS
CONFIG. 2120

3.4 AERODYNAMICS

3.4.1 Introduction

The total drag coefficient for all configurations evaluated for the AMPSS studies was calculated as follows:

$$C_{D_{Total}} = C_{D_{Min}} + K(C_L - C_L)^2 + \Delta C_{D_{Camber}} + C_{D_{Trim}} + C_{D_{Inlet}} + C_{D_{Nozzle}}$$

where

$C_{D_{Min}}$	basic airplane drag.
K	drag-due-to-lift factor (polar).
C_L	polar displacement along the lift coordinate.
$\Delta C_{D_{Camber}}$	increase in $C_{D_{Min}}$ due to cambered airfoils.
$C_{D_{Trim}}$	drag increment due to horizontal tail deflection to trim.
$C_{D_{Inlet}}$	incremental drag coefficient due to momentum loss of air passing into the inlet up to the engine compressor face (includes additive drag, lip suction pressure drag, boundary layer bleed).
$C_{D_{Nozzle}}$	pressure drag on exterior surfaces of engine nozzle.

3.4.2 Take-off and Landing Aerodynamics

The low speed aerodynamics for configuration 2120 both in the presence of the ground and in free air is shown in Figures 3.4-1 and 3.4-2 at a CG location of 33.4%. Recent wind tunnel tests on the F-111B have exceeded $C_{L_{max}}$ values of 3.15 and have actually reached values greater than 3.6 as shown in section 3.4.11 of this report.

3.4.3 Minimum Drag

Subsonic - Values of minimum drag at subsonic speeds to Mach critical are shown in Figure 3.4-3. These values were computed by the GD/FW minimum drag method which has been programmed for the IBM 7090 as GD/FW procedure T-34. This method employs a component build-up as a function of wetted area, Reynolds number and form factor as reported in paragraph 3.4.4.1 of FZM-4038-II-1.

Supersonic - Minimum drag coefficients at supersonic Mach numbers were generated by adding flat plate friction drag to the wave drag. Skin friction drags were calculated utilizing Eckert's reference enthalpy method as applied to each component. The GD/FW IBM 7090 procedure K-35 was employed to perform wave drag analysis as described in paragraph 3.4.4.1 of FZM-4038-II-1.

The variation of minimum drag with Mach number is shown in Figure 3.4-4 for the wing in the swept position, $\Lambda = 72.5^{\circ}$. The effects of altitude on minimum drag are also shown in this same figure for a Mach number of 2.2.

3.4.4 Drag-Due-to-Lift

Subsonic - The change in drag due to the change in lift coefficient is calculated as a function of the combined airfoil section properties as described in paragraph 3.4.4.2 of FZM-4038-II-1. The drag-due-to-lift with the wing in the 16° and 72.5° sweep position is shown in Figure 3.4-5.

Supersonic - Values of drag-due-to-lift at M = 2.2 were developed for configuration 2120 by interpolating between tail-on and tail-off wind tunnel data for the 1/15th scale model of the F-111B. The AMPSS designs have the same planform and thickness distribution as the F-111B. The resulting polar is shown in Figure 3.4-6.

With the change in engine location from the rear of the fuselage for 2120, a reduction in total planform area results. This is entirely due to the reduction of theoretical horizontal tail area caused by moving the exposed tail inboard as shown in Figure 3.4-7. Polar shape values, K, were developed for configuration 2120 at M = 2.2 by linearly interpolating between tail-on and tail-off values of K for the F-111B as depicted in Figure 3.4-8. This resulted in a value of K = 0.384 for 2120 as compared to the 0.354 level used for configuration 2010-H.

The variation of K with Mach number is shown in Figure 3.4-9 and was generated by fairing a curve through the subsonic value developed in the previous section to the value of K = 0.384 at M = 2.2.

3.4.5 Buffet Onset

The wing planform of 2120 is the same as 2010-H, therefore there would be no change in buffet characteristics. The curve of buffet onset is, however, reproduced in Figure 3.4-10 for convenience.

3.4.6 Critical Mach Number

The technique used to calculate Mach critical for the wing in the 16° sweep position is described in GD/FW FZA-381. Using this method a curve of Mach critical versus C_L is shown in Figure 3.4-11.

3.4.7 Trim Drag

As a result of the work presented in paragraph 3.4.4.6 of FZM-4038-II-1 it was determined that with proper c.g. control at subsonic speeds, little or no trim drag would result. Trim drag increments for a range of c.g. locations are shown in Figure 3.4-12 for $\Lambda = 16^{\circ}$ at $M \leqq M_{cr}$ and $\Lambda = 72.5^{\circ}$ at $M = 0.85$. Trim increments for $M = 1.2$ at sea level and at $M = 2.2$ at cruise altitude are shown in Figure 3.4-13.

Since the completion of AMPSS Phase II, more emphasis has been placed on the supersonic range capabilities of the configuration. For this reason it has been assumed that a supersonic camber will be incorporated in the wing design to yield a positive trimming increment in C_{m_0} at $M = 2.2$. This would reduce horizontal tail deflection to trim and, consequently, reduce trim drag.

Well developed methods of generating and evaluating such cambers are available at GD/FW through IBM procedures S-02 and S-15. These methods have been applied to the AMPSS wing planform to design a cambered surface which gives a positive C_{m_O} increment of .012. Work is currently underway to incorporate this camber into an AMPSS configuration to take advantage of this C_{m_O} shift. Using this incremental value of C_{m_O}, horizontal tail deflection to trim and the resulting trim drags were calculated. This trim drag increment is shown in Figure 3.4-13 and is shown added to the M = 2.2 polar in Figure 3.4-5.

3.4.8 Nacelle Design

Inlet - A staggered, podded, siamese nacelle arrangement similar to that of AMPSS configuration 2110 as reported in FZM-4124 was chosen for the AMPSS configuration 2120. Each inlet was circular and had a translating, variable diameter, double-cone centerbody. The basic characteristics of this inlet are the same as those reported for the AMPSS type B and F configurations described in FZM-4038-II-1.

The same inlet total pressure recovery as reported in FZM-4038-II-1 was retained for the AMPSS configuration 2120. For convenience, this curve is shown in Figure 3.4-14. The Mach 2.2 total pressure recovery of 90% can now be substantiated based upon the following.

a. Recent performed Mach 2.2 1/6th scale F-111 fuselage-inlet composite model tests at AEDC yielded a total pressure recovery of 87% at the F-111 design mass flow ratio,

b. An expected 2% increase in total pressure recovery over the basic 1/6th scale F-111 test results when the data is extrapolated to full scale Reynolds numbers, and

c. The use of a more optimum spike arrangement in the AMPSS configuration 2120 in conjunction with the use of fully circular inlets which is expected to yield another 3% total pressure recovery over that of the F-111.

The inlet drag reported in FZM-4038-II-1 has been retained for the AMPSS configuration 2120 (corrected for engine scale and reference area variations) with the following exceptions:

a. The subsonic additive drag has been re-evaluated based upon inlet test performed in 1959 by Pratt & Whitney.

b. The miscellaneous drag now is 5 counts at subsonic speeds (3 counts airplane mismatch drag, 0.5 count air conditioning drag, 0.5 count oil cooling drag, and 1.0 count for the drag of the remaining accessory systems) and 8 counts at supersonic speeds (same break-down as above but with 6 counts airplane mismatch drag).

The corrected inlet drag curve is given in Figure 3.4-15.

Nozzle - Refer to section 3.4.5.5 of FZM-4038-II-1.

3.4.9 Cruise Polars

Total drag at lift for three (3) different cruise conditions are shown in Figure 3.4-16. These flight conditions are $\Lambda = 16°$, M = 0.72 at 25,000 ft.; $\Lambda = 72.5°$, M = 0.85 at sea level; and $\Lambda = 72.5°$, M = 2.2 at 55,000 ft.

3.4.10 Aerodynamic Data for Parametric Studies

Growth Study - To determine the change in performance as the airplane size is changed, three other airplanes were evaluated. These three airplanes were designated 2111A, 2112A and 2113A and were designed for gross weights of 354,000 lbs, 450,000 lbs and 275,000 lbs respectively. The value of $C_{D_{min}}$ as a function of gross weight is shown in Figure 3.4-17.

Wing Loading Study - The change in $C_{D_{min}}$ as a function of wing loading was established from data for configuration 2111A as a base line airplane. The change in $C_{D_{min}}$ with a change in wing loading is shown in Figure 3.4-18. All other aerodynamic parameters were considered the same as 2111A.

Sweep Study - A parametric study was done to establish the subsonic performance characteristics of 2120 as a function of wing sweep angle. Values of $C_{D_{min}}$, K, and ΔC_L are shown in Figure 3.4-19 for wing sweep angles from 16° to 72.5° at sea level conditions. Performance was computed with gust response characteristics held constant. This was done by reducing dash

Mach number as wing sweep angle was reduced according to the relationship shown in Figure 3.4-20.

Engine Scale - The effect on performance due to changes in engine size was evaluated at subsonic speed only. Changes in nacelle wetted area are shown in Figure 3.4-21 and changes in drag as a function of changes in engine scale are shown in Figure 3.4-22 for the three different configurations evaluated (2111A, 2114, and 2115).

3.4.11 Future Configuration Improvements Through Aerodynamic Refinement

Low Speed Aerodynamics - Through February of 1964, accumulated wind tunnel data on F-111B demonstrated values of $C_{L_{max}} = 3.16$ with a full span single slotted flap and double slotted flap. (GD/FW report FZA-12-013) The present AMPSS design (2120) incorporates essentially this same flap geometry. These tests were run on a 1/12th scale full span model and a 1/6th half span model.

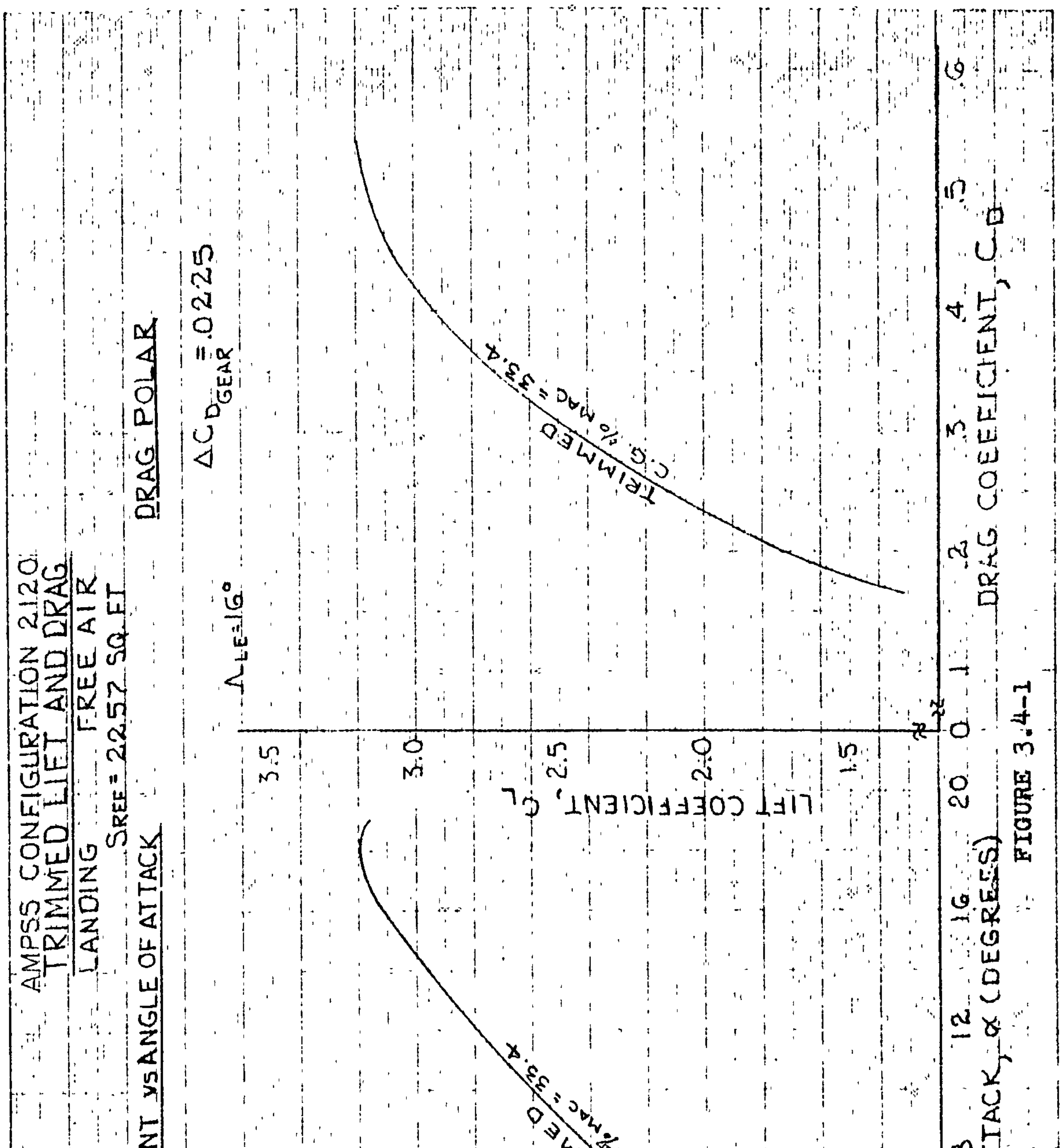
During May and June of this year further testing has been conducted on the F-111B full span model. One of the better configurations utilizing a double slotted slat and tripple slotted flap is shown in Figure 3.4-23 and demonstrates values of $C_{L_{max}}$ greater than 3.6.

Parametric studies reported in Section 3.3 and summarized in Figure 4.6-1 demonstrate that for an AMPSS configuration with a wing loading of 175 lbs./ft.^{2} a decrease in engine size from 42.6% (Configuration 2120) to 33.7% would yield an increase in range of 340 miles. If the growth curve were then followed down to give the same range on 2120 a configuration with a gross weight of 330,000 lbs. would result.

This much reduction in engine scale would require a flap system capable of developing a $C_{L_{max}} = 3.94$ to achieve a 6000-foot take-off distance. Values such as these,of course, have not been attained, however, it is obvious that GD/FW experience and continued development with full span flap systems such as used on the F-111 program will offer continued benefits to the future AMPSS studies.

Supersonic Drag - Throughout the entire AMPSS program considerable effort has been given to the accurate prediction of lift and drag characteristics at supersonic speeds. The major geometry for the AMPSS designs have been dictated by the subsonic mission requirements, however, it has been possible to make minor adjustments to the geometry to obtain the best possible supersonic capability.

It has been found that the most important single criteria for maximum supersonic L/D is the fineness ratio of the configuration expressed as Frontal Area over length squared (A/l^2). A curve of L/D_{max} as a function of A/l^2 is shown in Figure 3.4-24 with various aircraft configurations spotted on the curve. It is apparent that at a particular value of A/l^2 a considerable spread in L/D_{max} has been obtained on different configurations. This is primarily a function of local geometry and future studies will be devoted to refinements of the AMPSS designs. This will be accomplished by using the GD/FW IBM wave drag procedure K-35. From the output of this procedure oblique area distributions are generated for any supersonic Mach number. These oblique area distributions can then be inspected as an aid to proper location and shape of components to obtain minimum wave drag at the design supersonic Mach number.



AMPSS CONFIGURATION 2.120
TRIMMED LIFT AND DRAG
LANDING FREE AIR
S_{REF} = 2257 SQ. FT.
LIFT COEFFICIENT vs ANGLE OF ATTACK
DRAG POLAR
$\Delta C_{D_{GEAR}}$ = .0225
Λ_{LE} = 16°
TRIMMED
C.G. % MAC = 33.4
LIFT COEFFICIENT, C_L
ANGLE OF ATTACK, α (DEGREES)
DRAG COEFFICIENT, C_D
3.5
3.0
2.5
2.0
1.5
0
4
8
12
16
20
.1
.2
.3
.4
.5
.6

FIGURE 3.4-1

AMPSS CONFIGURATION 2120
TRIMMED LIFT AND DRAG
TAKE-OFF GROUND EFFECT
$S_{REF} = 2257$ SQ. FT.

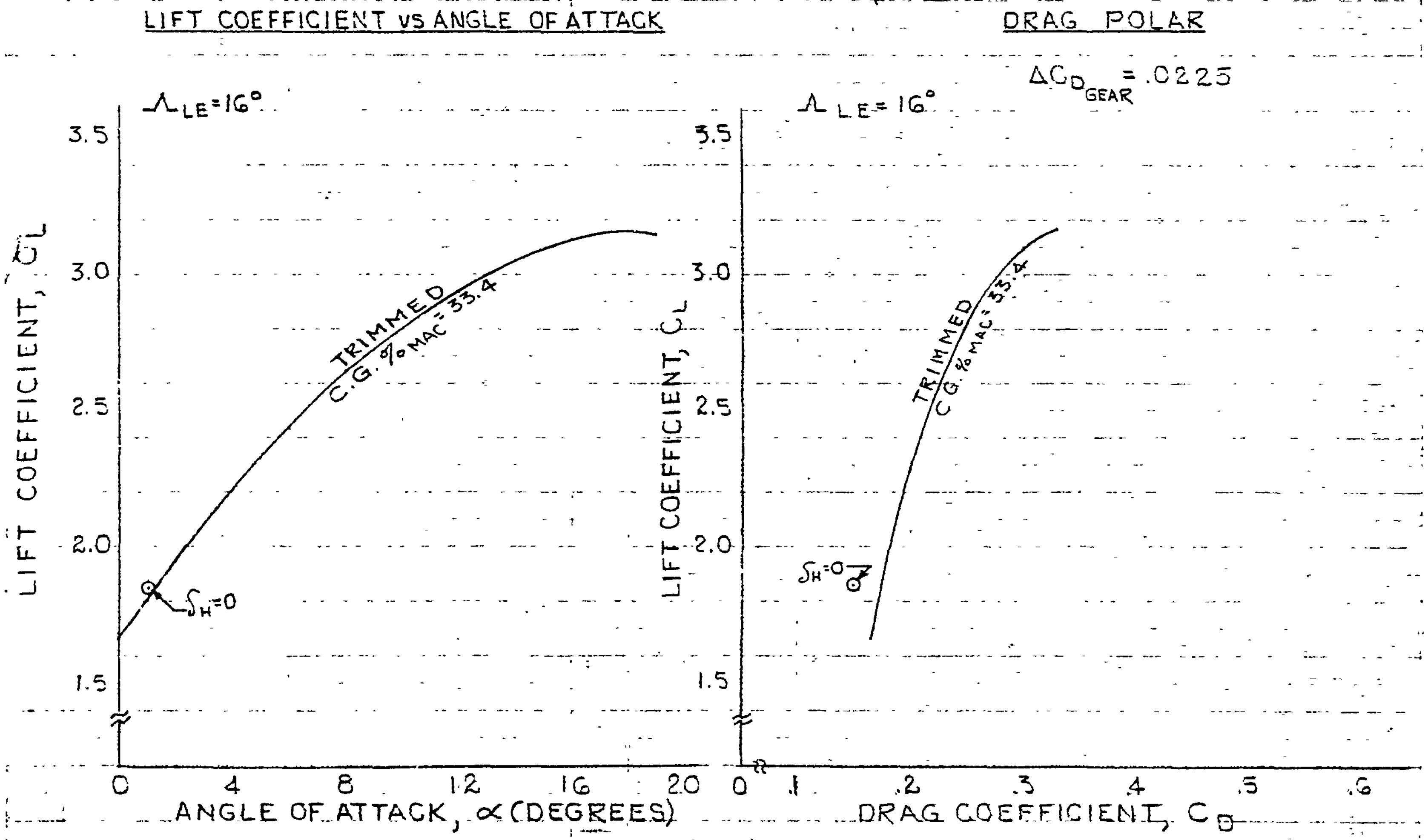


FIGURE 3.4-2

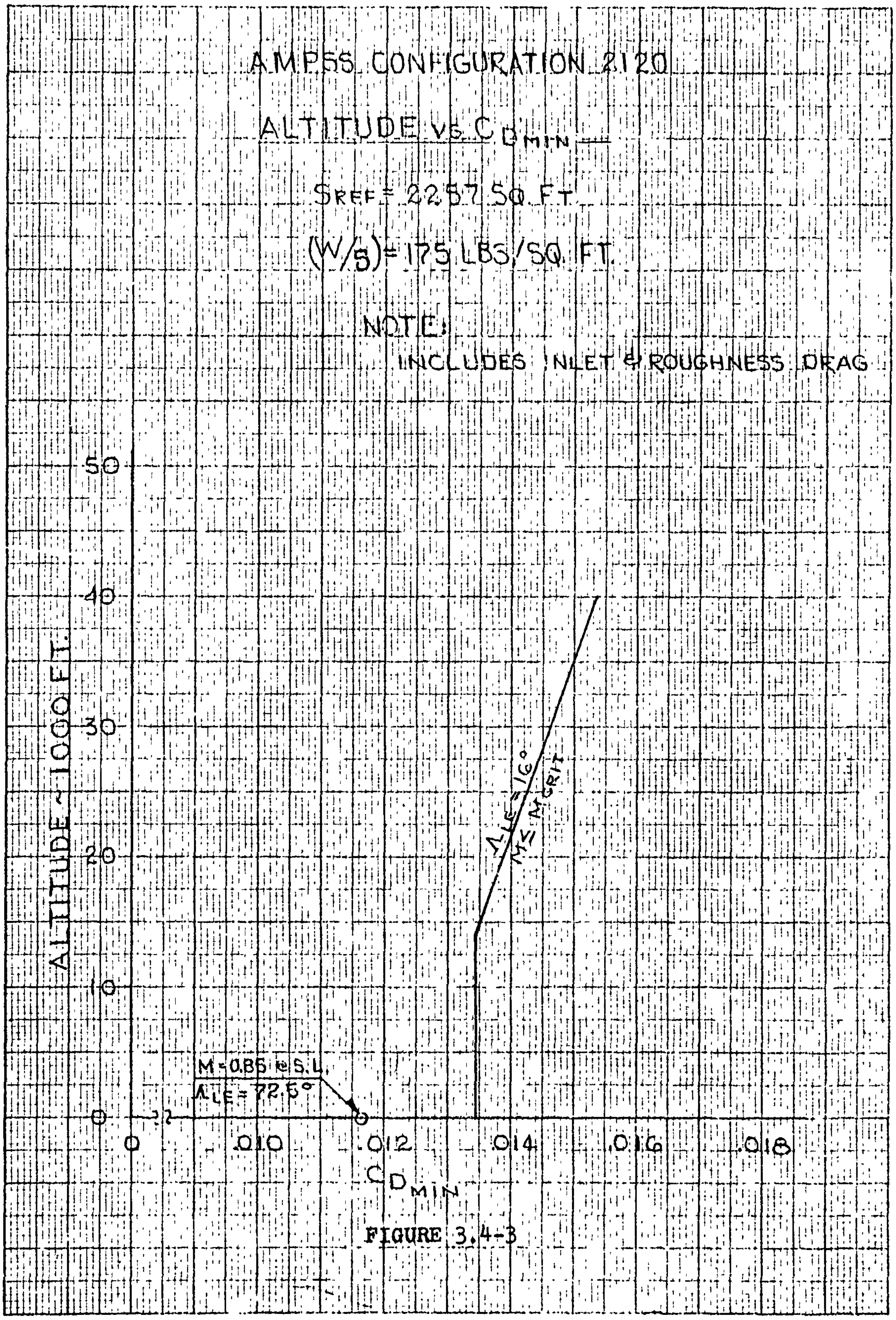

AMP55 CONFIGURATION 2120
ALTITUDE VS $C_{D_{MIN}}$
S_{REF} = 2257 SQ FT
(W/S) = 175 LBS/SQ FT
NOTE:
INCLUDES INLET & ROUGHNESS DRAG
ALTITUDE ~ 1000 FT.
50
40
30
20
10
0
Λ_{LE} = 16°
M ≤ M_{CRIT}
M = 0.85 @ S.L.
Λ_{LE} = 72.5°
0
.010
.012
.014
.016
.018
$C_{D_{MIN}}$

FIGURE 3.4-3

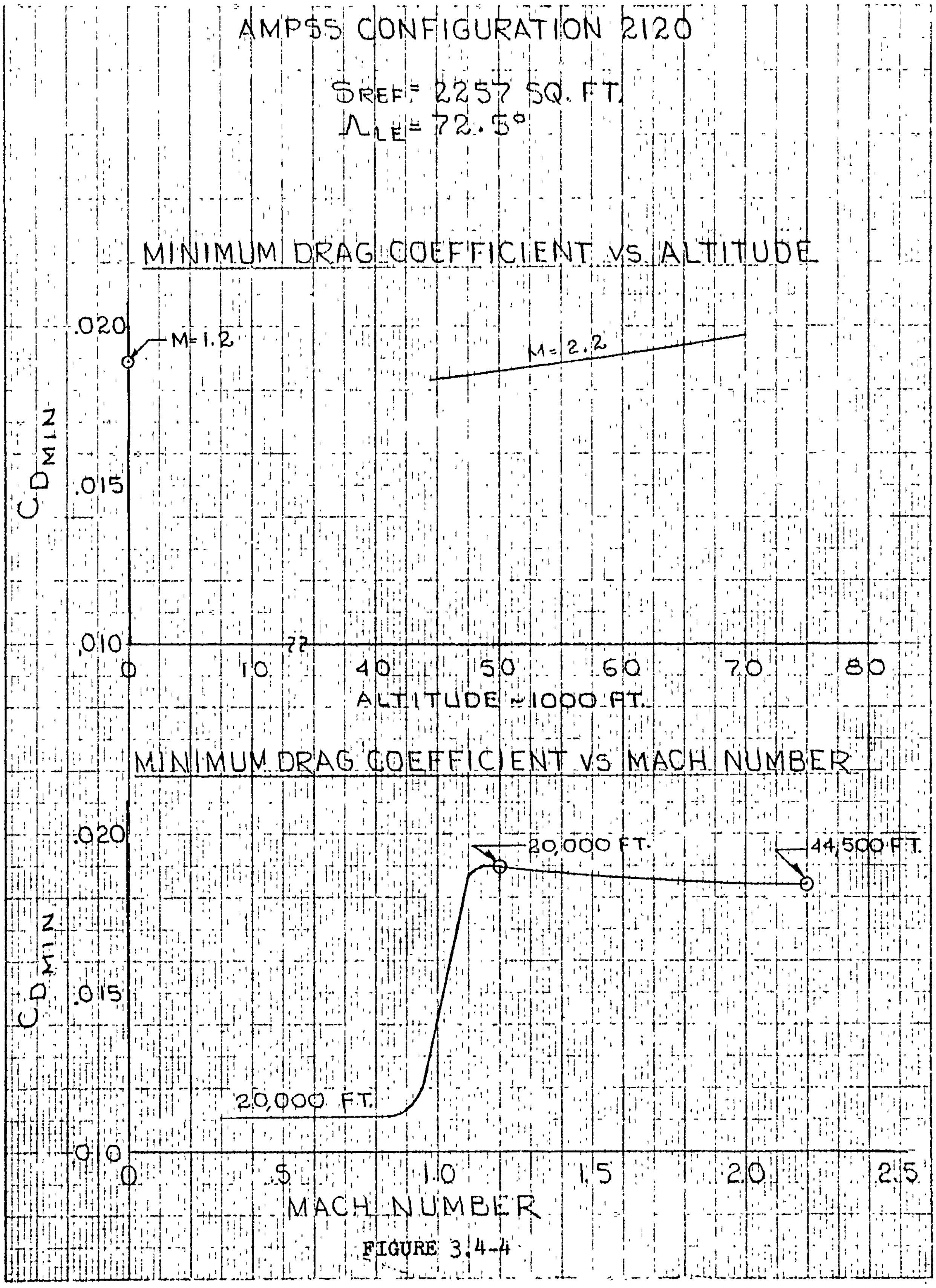

AMPSS CONFIGURATION 2120
SREF = 2257 SQ. FT.
ΛLE = 72.5°
MINIMUM DRAG COEFFICIENT VS. ALTITUDE
.020
M=1.2
M=2.2
CDMIN
.015
.010
0
10
40
50
60
70
80
ALTITUDE ~1000 FT.
MINIMUM DRAG COEFFICIENT VS MACH NUMBER
.020
20,000 FT.
44,500 FT.
CDMIN
.015
20,000 FT.
.010
0
.5
1.0
1.5
2.0
2.5
MACH NUMBER

FIGURE 3.4-4

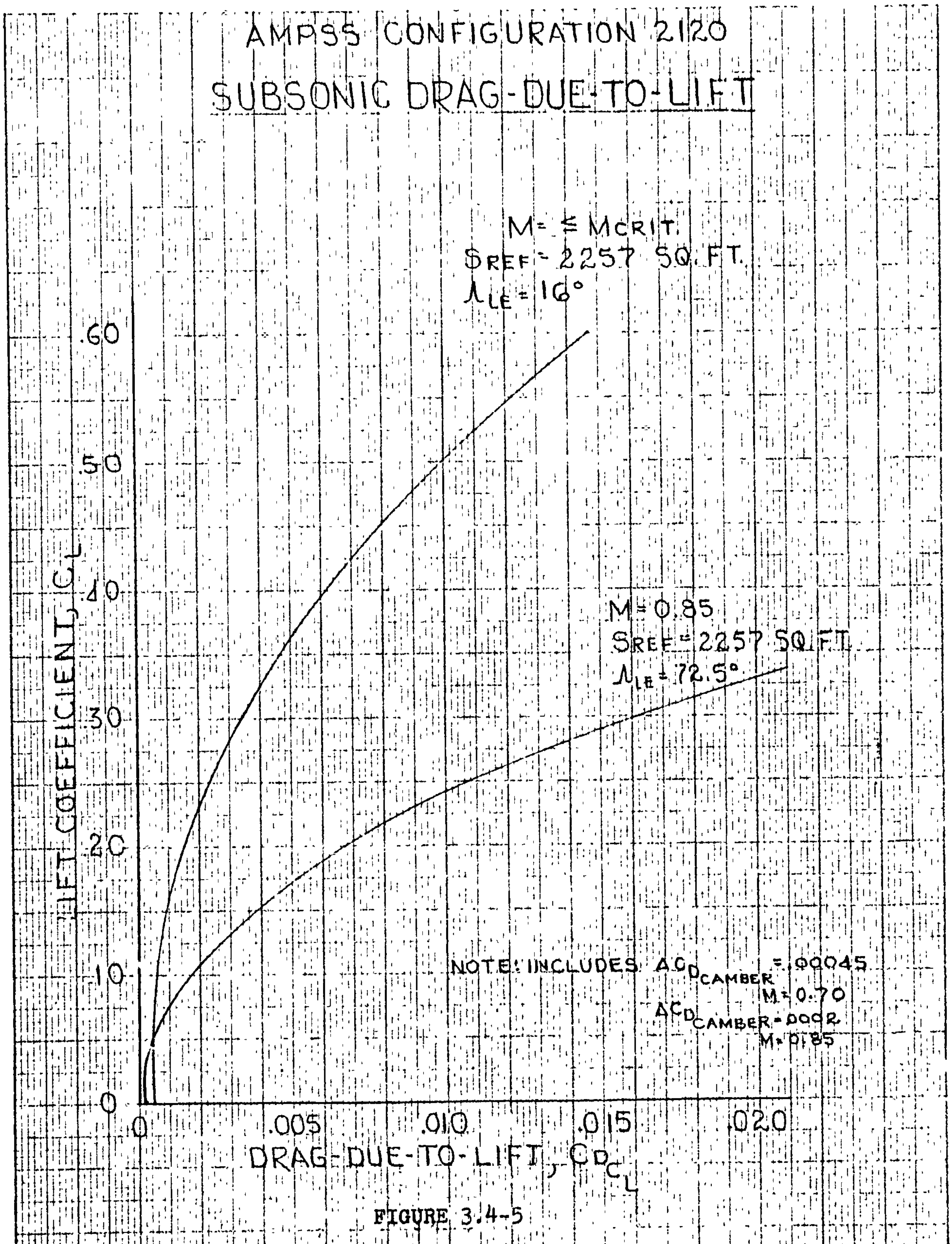

AMPSS CONFIGURATION 2120
SUBSONIC DRAG-DUE-TO-LIFT
M = ≤ MCRIT
SREF = 2257 SQ. FT.
ΛLE = 16°
M = 0.85
SREF = 2257 SQ. FT.
ΛLE = 72.5°
NOTE: INCLUDES ΔCD CAMBER = .00045
M = 0.70
ΔCD CAMBER = .0002
M = 0.85
.60
.50
.40
.30
.20
.10
0
0
.005
.010
.015
.020
LIFT COEFFICIENT, CL
DRAG-DUE-TO-LIFT, CDCL

FIGURE 3.4-5

SECRET

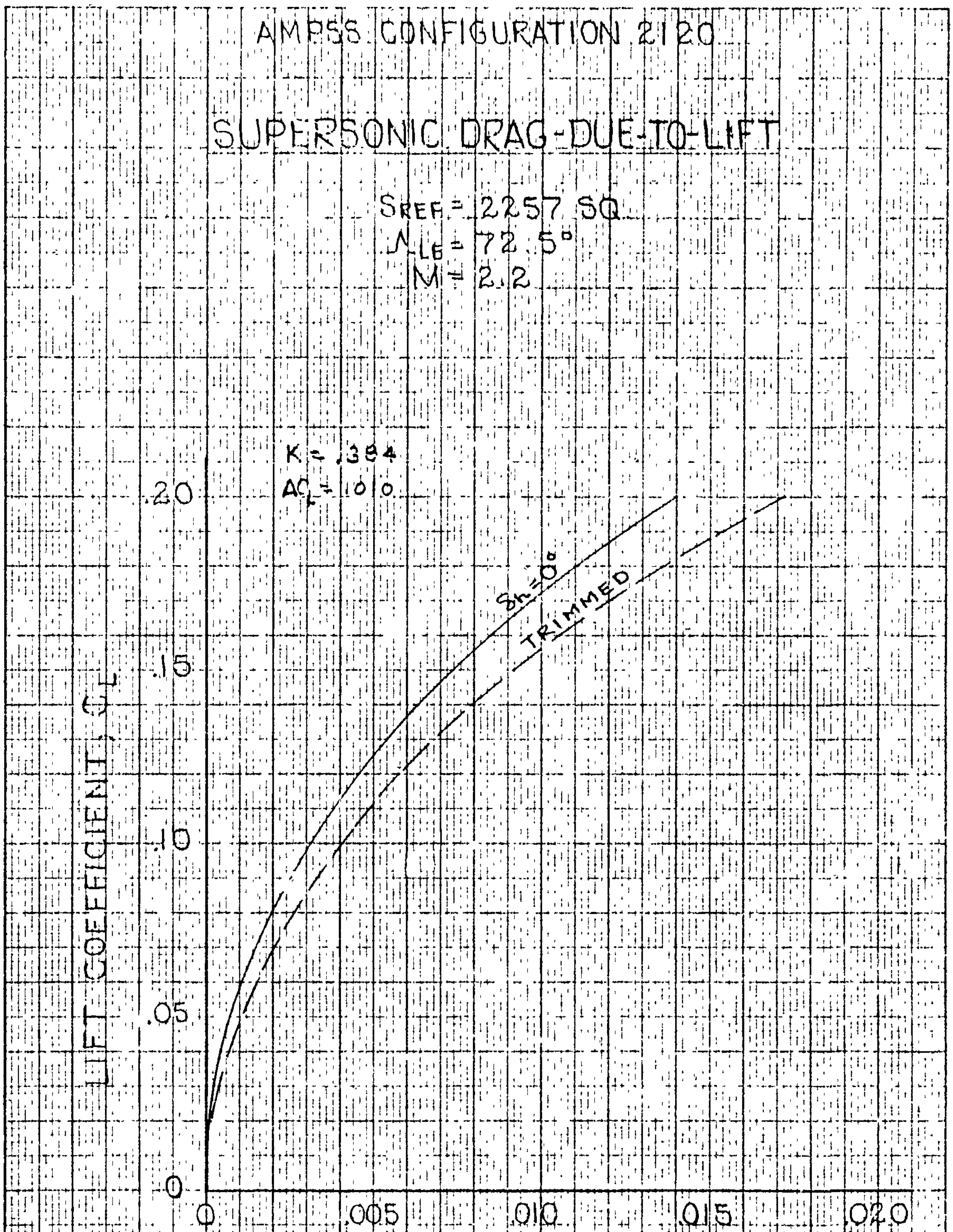


FIGURE 3.4-6

SECRET

AMPSS CONFIGURATION 2120

M=2.2 DRAG-DUE-TO-LIFT DEVELOPMENT

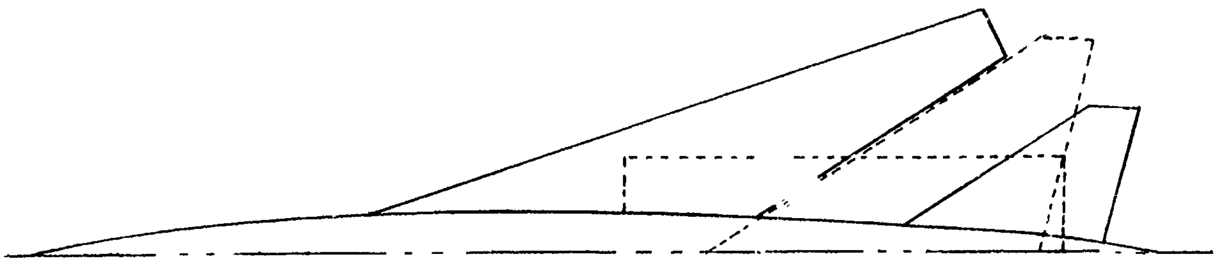


FIGURE 3.4-7

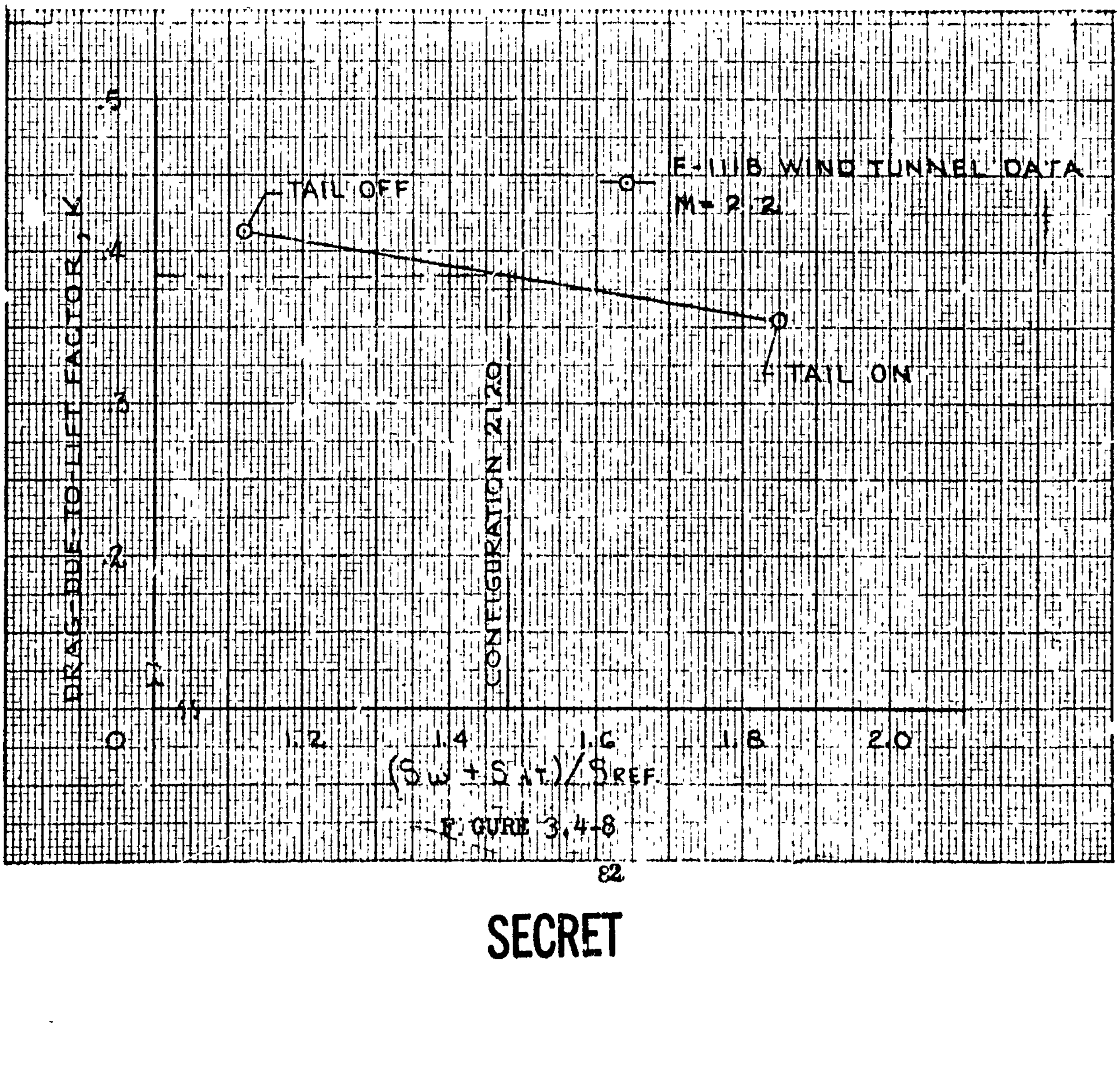


FIGURE 3.4-8

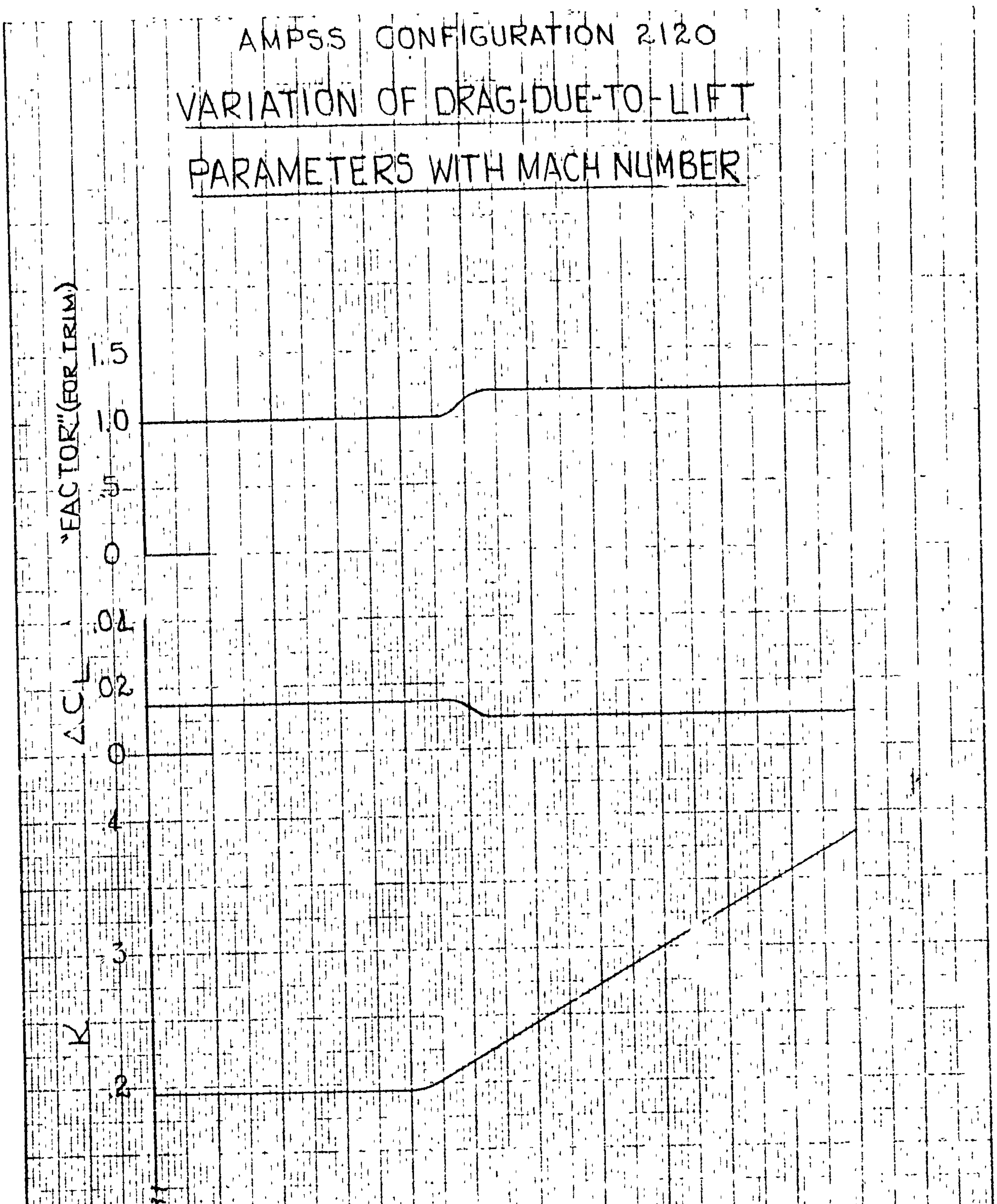

AMPSS CONFIGURATION 2120
VARIATION OF DRAG-DUE-TO-LIFT
PARAMETERS WITH MACH NUMBER
"FACTOR" (FOR TRIM)
1.5
1.0
.5
0
ΔC_L
.04
.02
0
K
.4
.3
.2
.1
0
.5
1.0
1.5
2.0
MACH NUMBER

FIGURE 3.4-9

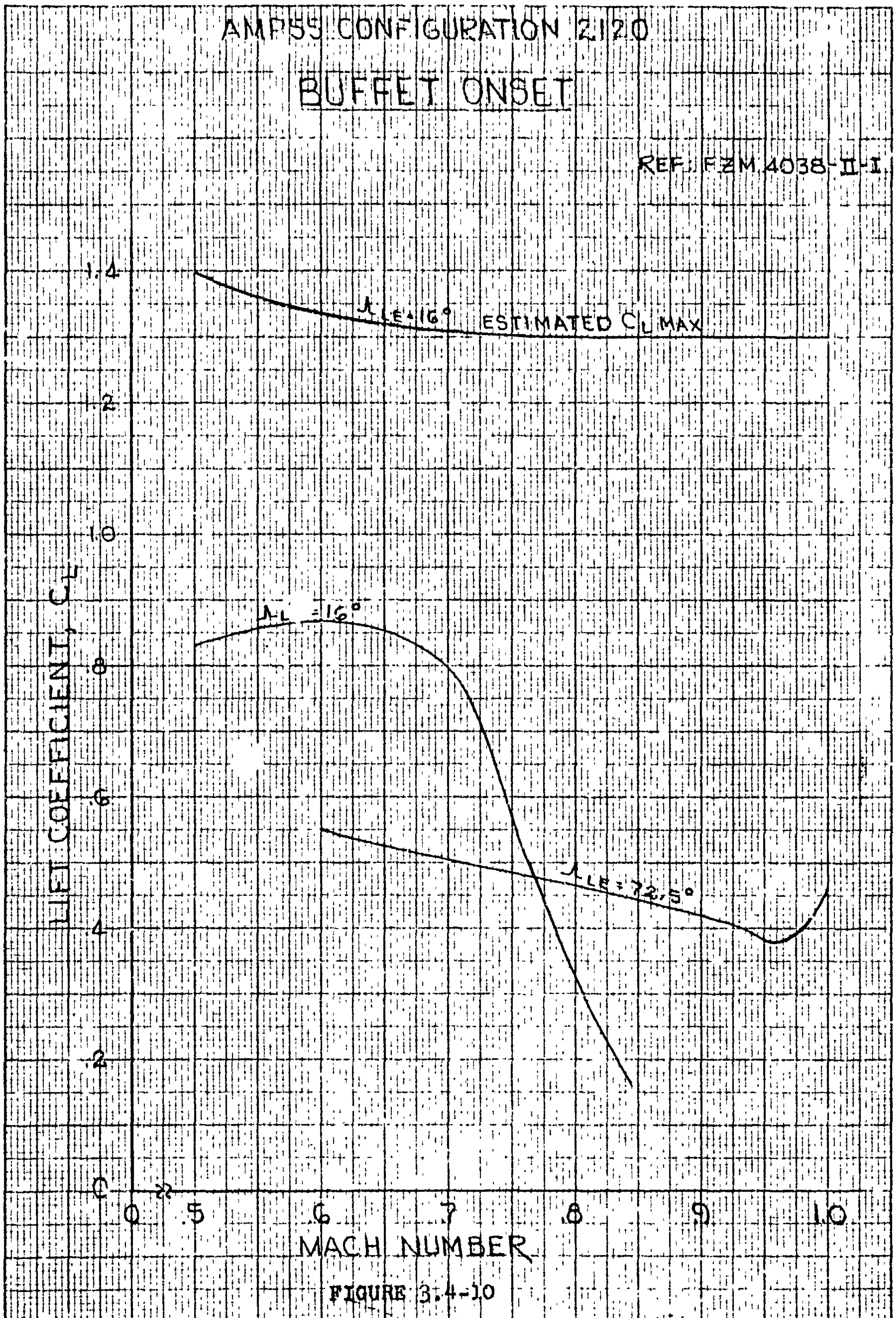
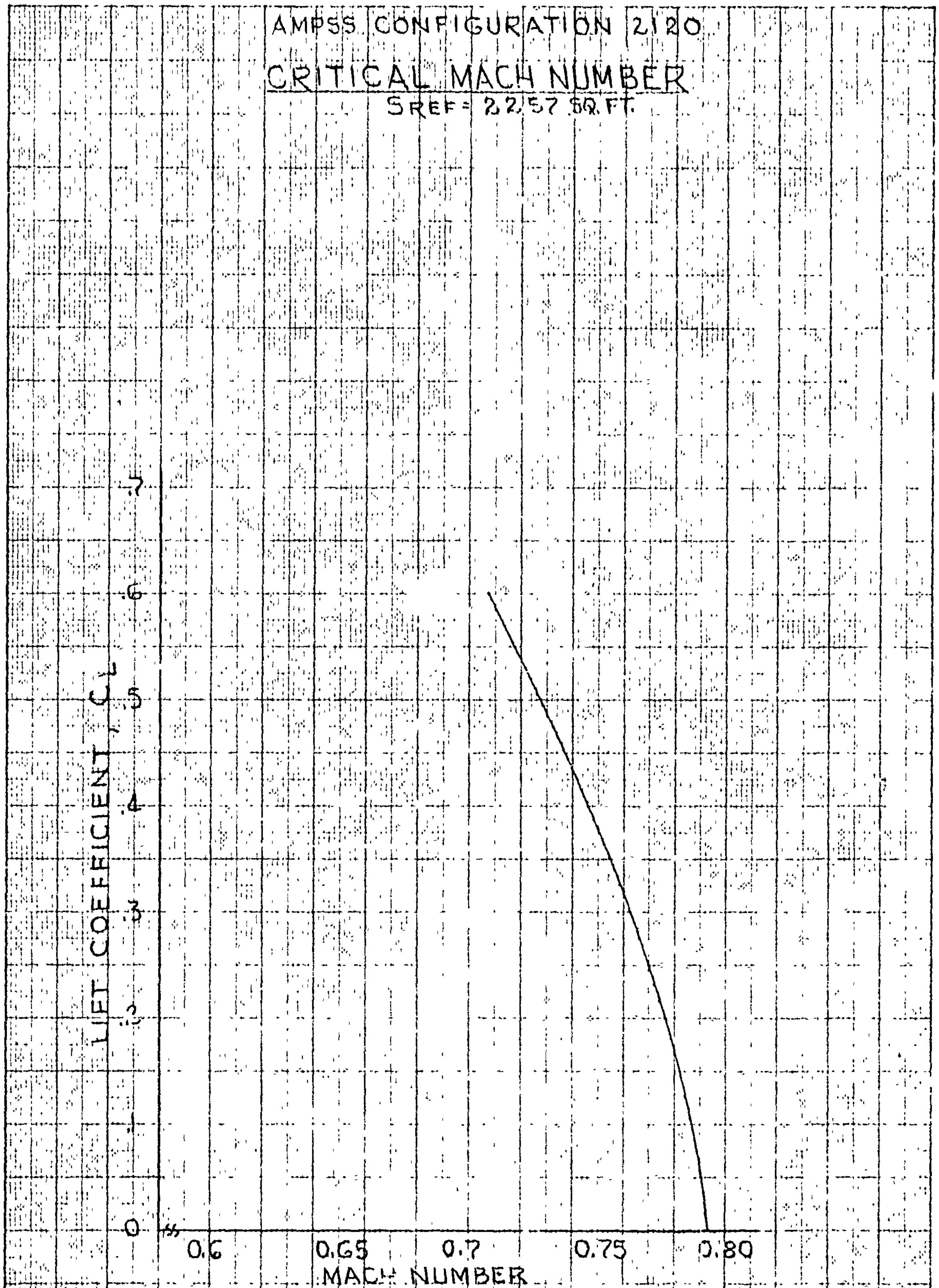

AMP55 CONFIGURATION 2120
BUFFET ONSET
REF: FZM 4038-II-I
Λ_{LE} = 16° ESTIMATED C_L MAX
Λ_L = 16°
Λ_{LE} = 72.5°
LIFT COEFFICIENT, C_L
MACH NUMBER
1.4
1.2
1.0
.8
.6
.4
.2
0
0
.5
.6
.7
.8
.9
1.0

FIGURE 3.4-10



AMPSS CONFIGURATION 2120
CRITICAL MACH NUMBER
SREF = 2257 SQ. FT.
LIFT COEFFICIENT, C_L
.7
.6
.5
.4
.3
.2
.1
0
0.6
0.65
0.7
0.75
0.80
MACH NUMBER

FIGURE 3.4-11

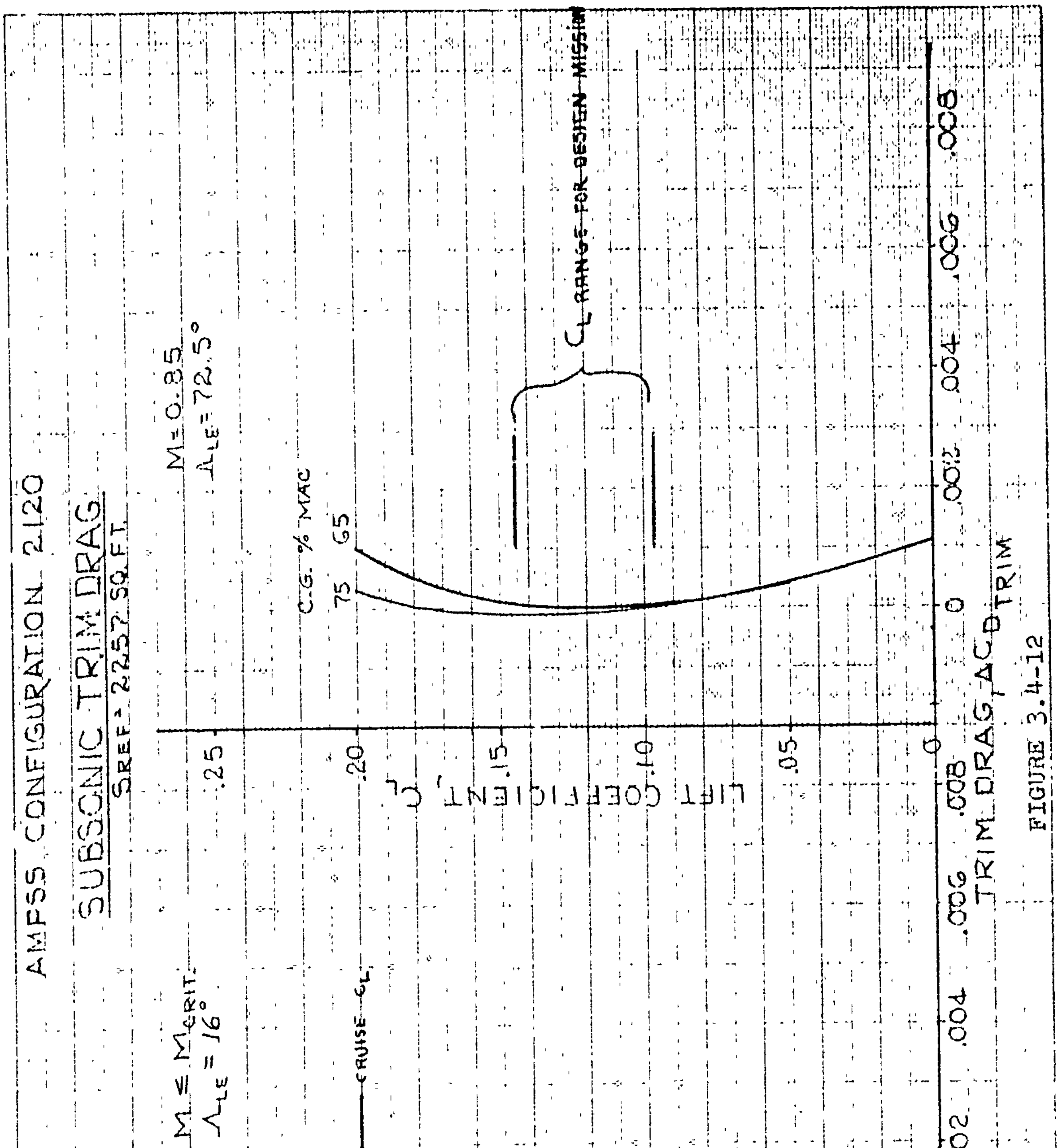

AMPSS CONFIGURATION 2120
SUBSONIC TRIM DRAG
SREF = 2257 SQ.FT.
M ≤ M CRIT
Λ LE = 16°
C.G. % MAC
33.4
25
CRUISE C L
M = 0.85
Λ LE = 72.5°
C.G. % MAC
75
65
C L RANGE FOR DESIGN MISSION
LIFT COEFFICIENT, C L
.6
.5
.4
.3
.2
.1
0
.25
.20
.15
.10
.05
0
.008
.006
.004
.002
0
.002
.004
.006
.008
TRIM DRAG, ΔC D TRIM

FIGURE 3.4-12

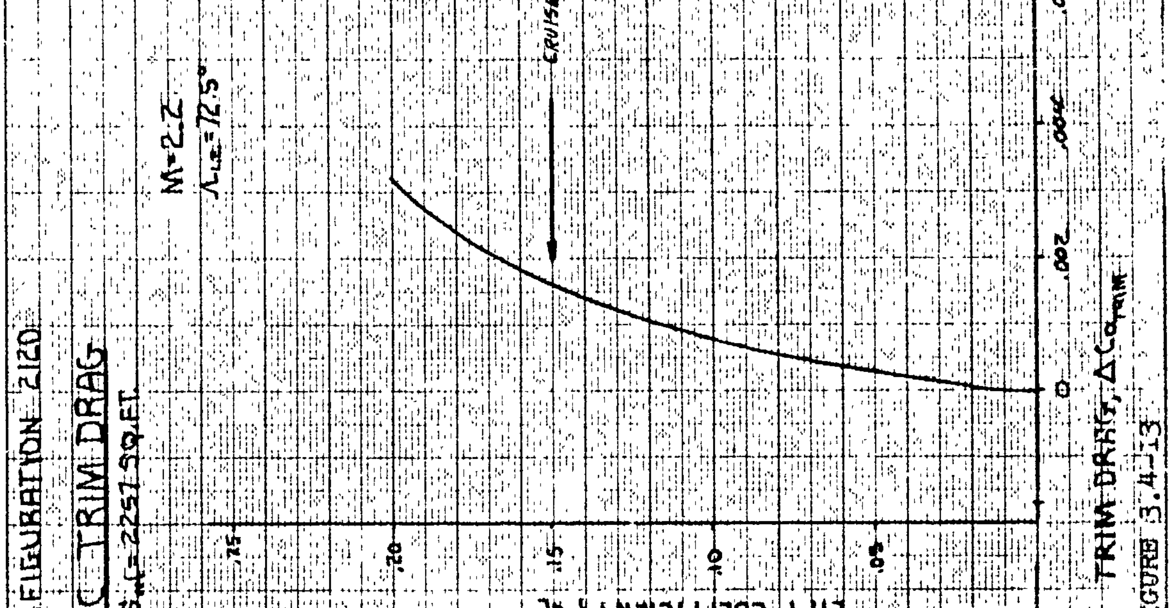

AMPSS CONFIGURATION 2120
SUPERSONIC TRIM DRAG
S_{REF} = 2257 SQ.FT.
M = 1.2
Λ_{LE} = 72.5°
M = 2.2
Λ_{LE} = 72.5°
C.G. % MAC
.25
CRUISE C_L @ ALT.
DESIGN MISSION (Refueled)
DESIGN MISSION (Non-Refueled)
C_L RANGE FOR TRADE
LIFT COEFFICIENT, C_L
.25
.20
.15
.10
.05
0
.002
.004
.006
.008
TRIM DRAG, $\Delta C_{D_{TRIM}}$

FIGURE 3.4-13

SECRET

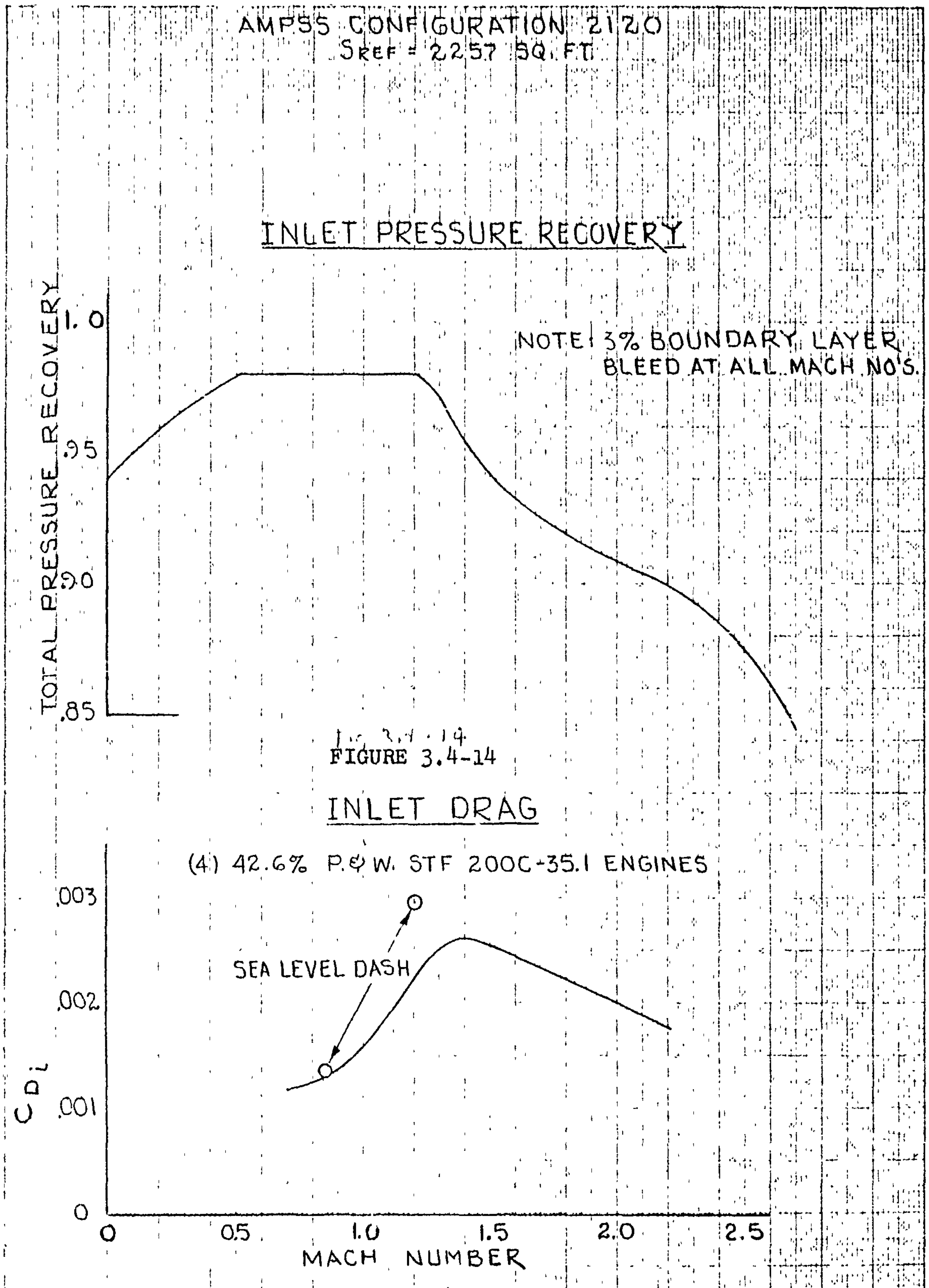


FIGURE 3.4-14

FIGURE 3.4-15

SECRET

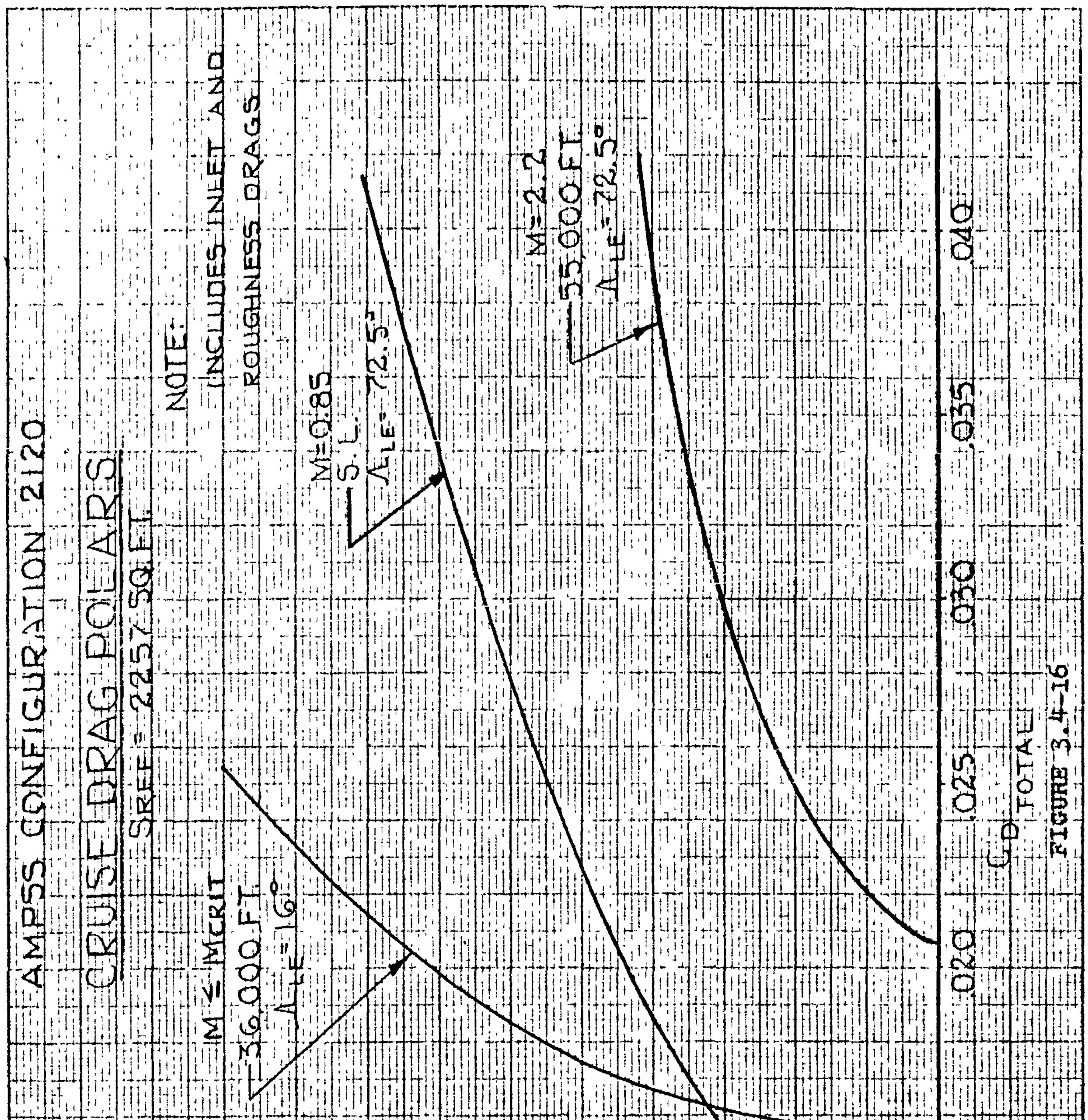

AMPSS CONFIGURATION 2120
CRUISE DRAG POLARS
SREF = 2257 SQ FT
NOTE: INCLUDES INLET AND ROUGHNESS DRAGS
M ≤ MCRIT
36,000 FT
Λ_{LE} = 16°
M=0.85
S.L.
Λ_{LE} = 72.5°
M=2.2
55,000 FT
Λ_{LE} = 72.5°
0
1
2
3
4
5
LIFT COEFFICIENT, C_L
0
.010
.020
.025
.030
.035
.040
C_D TOTAL

FIGURE 3.4-16

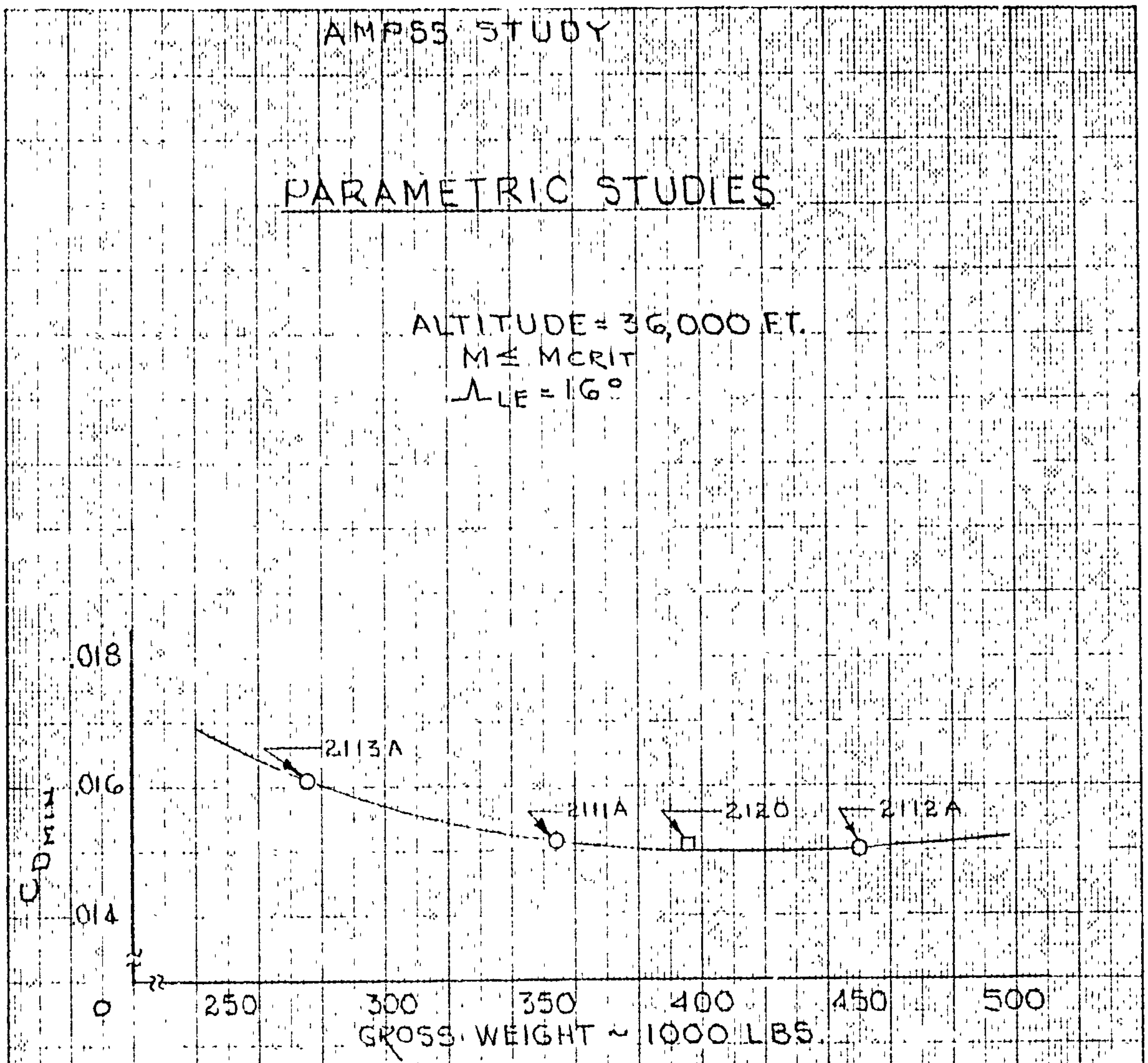


FIGURE 3.4-17

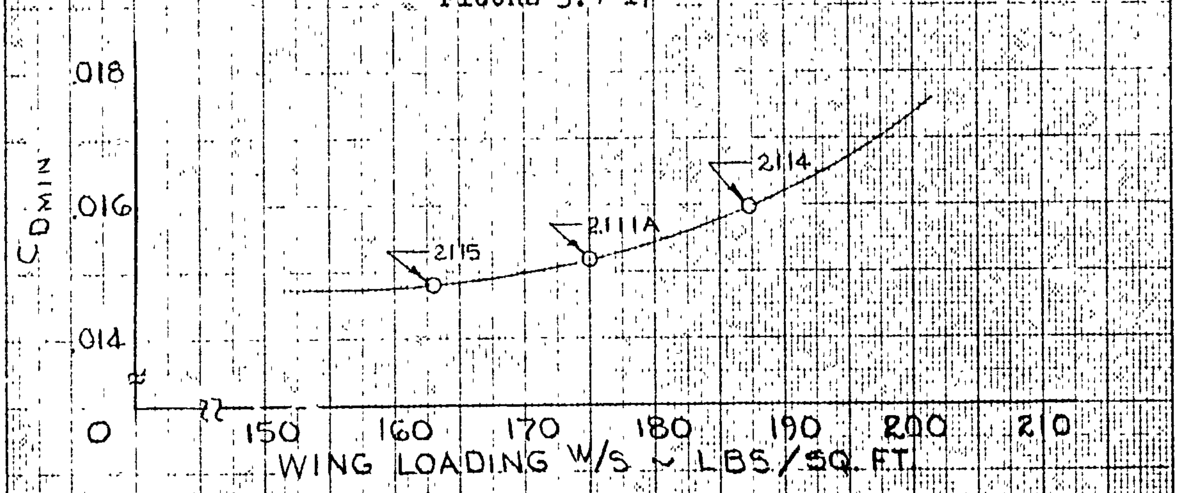


FIGURE 3.4-18

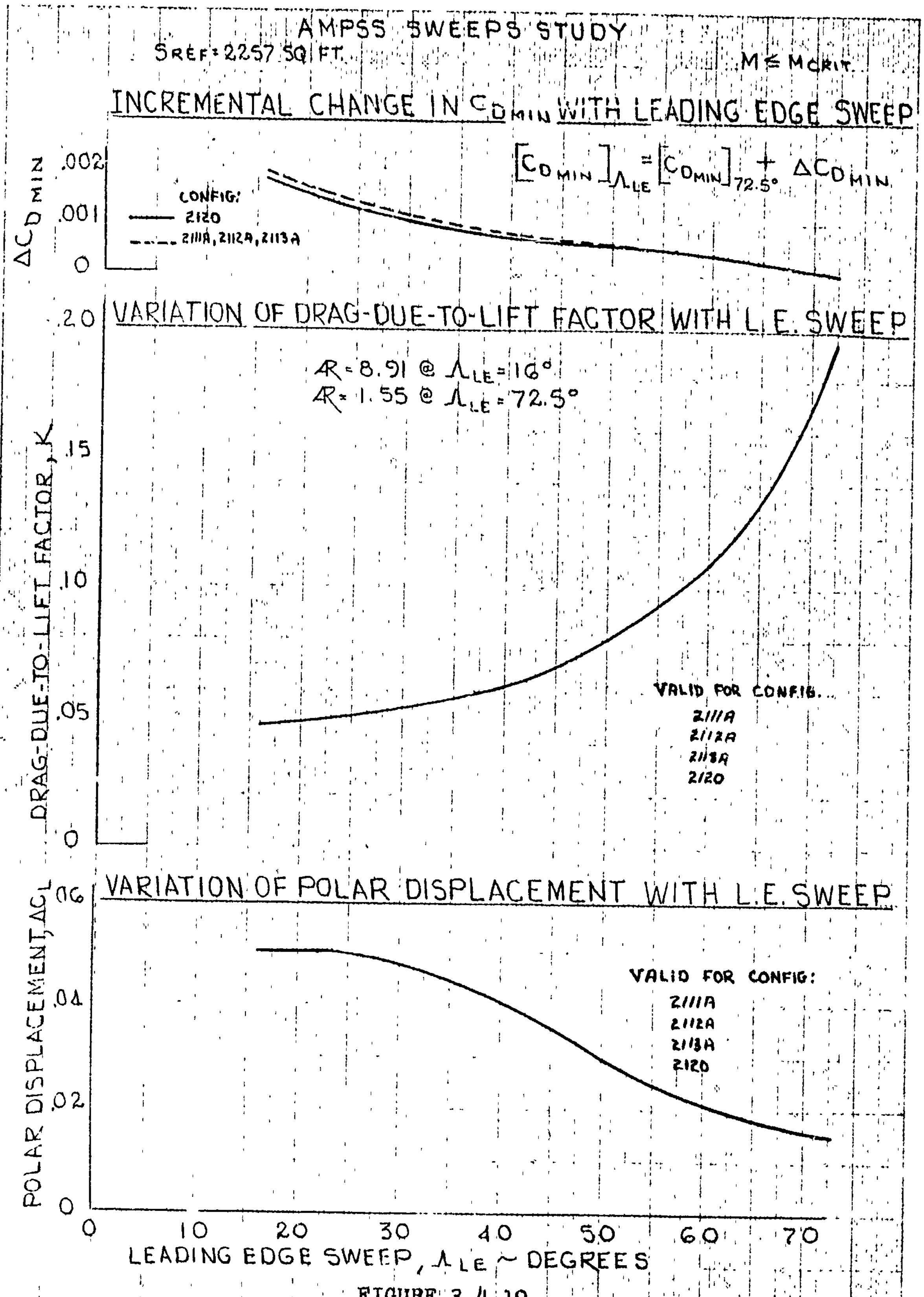

AMPSS SWEEPS STUDY
SREF = 2257 SQ FT.
M ≤ MCRIT
INCREMENTAL CHANGE IN CDMIN WITH LEADING EDGE SWEEP
[CD MIN]ΛLE = [CD MIN]72.5° + ΔCD MIN
ΔCD MIN
.002
.001
0
CONFIG:
2120
2111A, 2112A, 2113A
VARIATION OF DRAG-DUE-TO-LIFT FACTOR WITH L.E. SWEEP
AR = 8.91 @ ΛLE = 16°
AR = 1.55 @ ΛLE = 72.5°
DRAG-DUE-TO-LIFT FACTOR, K
.20
.15
.10
.05
0
VALID FOR CONFIG.
2111A
2112A
2113A
2120
VARIATION OF POLAR DISPLACEMENT WITH L.E. SWEEP
POLAR DISPLACEMENT, ΔCL
.06
.04
.02
0
VALID FOR CONFIG:
2111A
2112A
2113A
2120
0
10
20
30
40
50
60
70
LEADING EDGE SWEEP, ΛLE ~ DEGREES

FIGURE 3.4-19

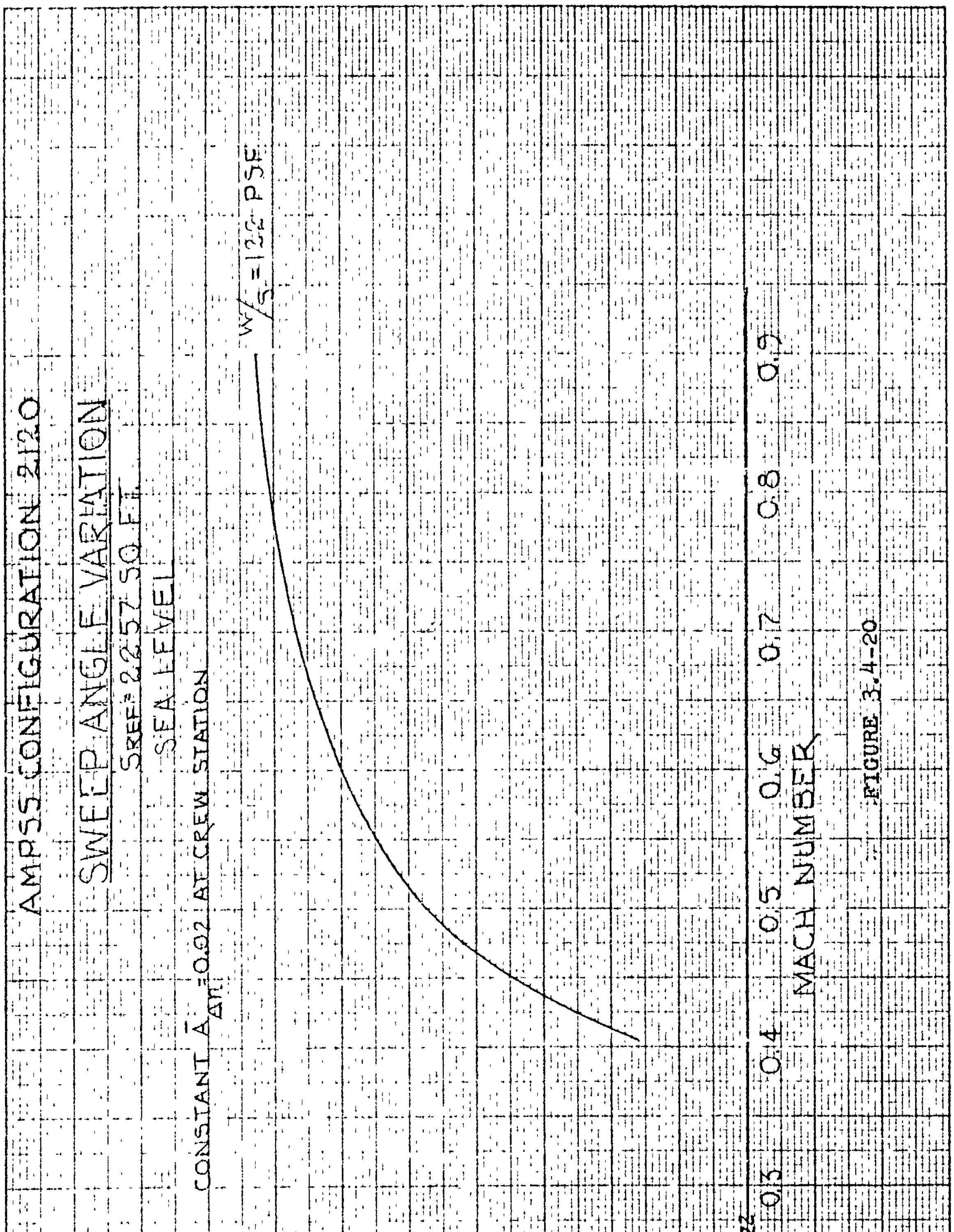

AMPSS CONFIGURATION 2120
SWEEP ANGLE VARIATION
SREF = 2257 SQ FT
SEA LEVEL
CONSTANT A = 0.92 AT CREW STATION
W/S = 122 PSF
LEADING EDGE SWEEP, Λ LE ~ DEGREES
80
60
40
20
0
0.3
0.4
0.5
0.6
0.7
0.8
0.9
MACH NUMBER

FIGURE 3.4-20

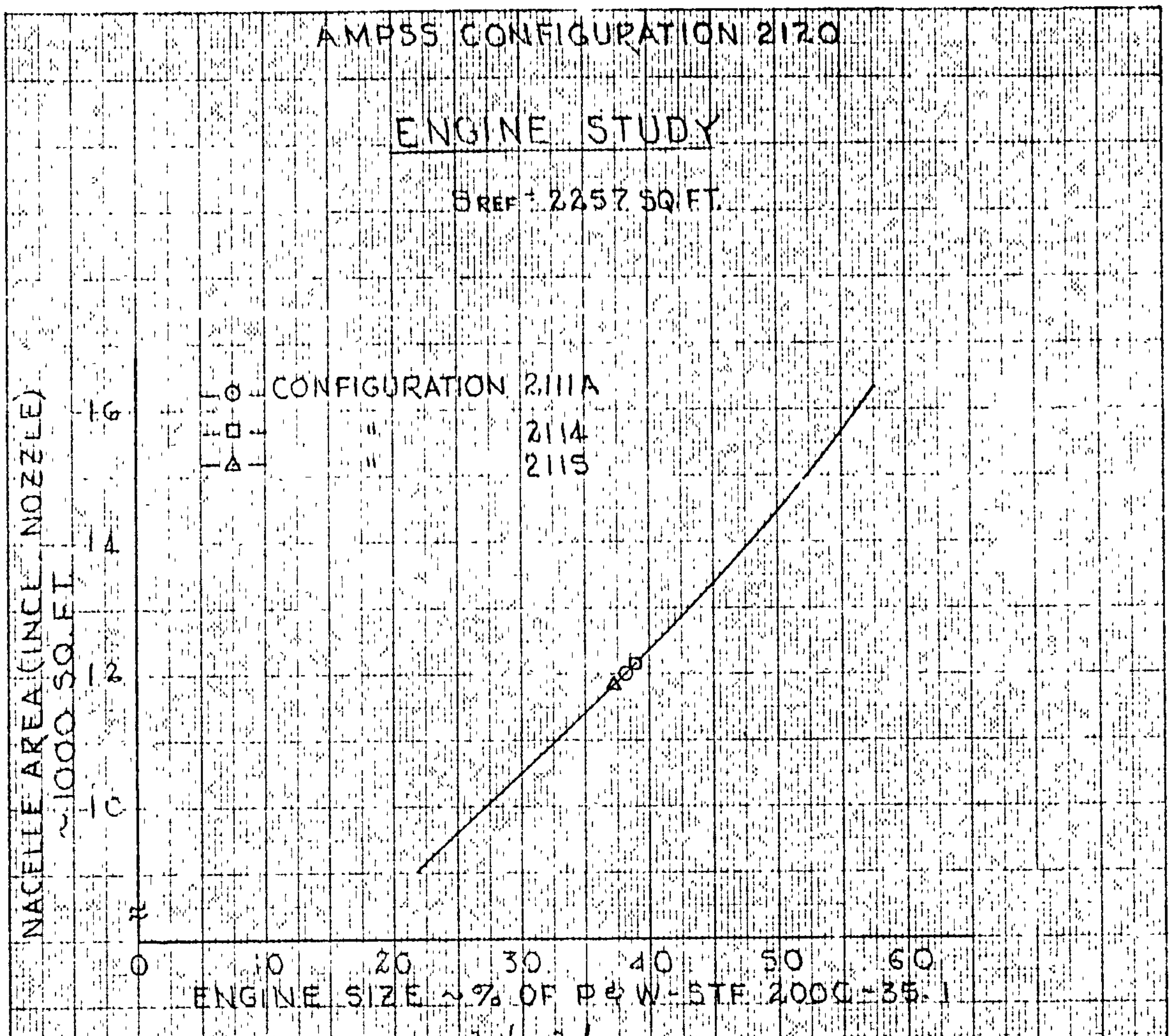


FIGURE 3.4-21

FIGURE 3.4-22

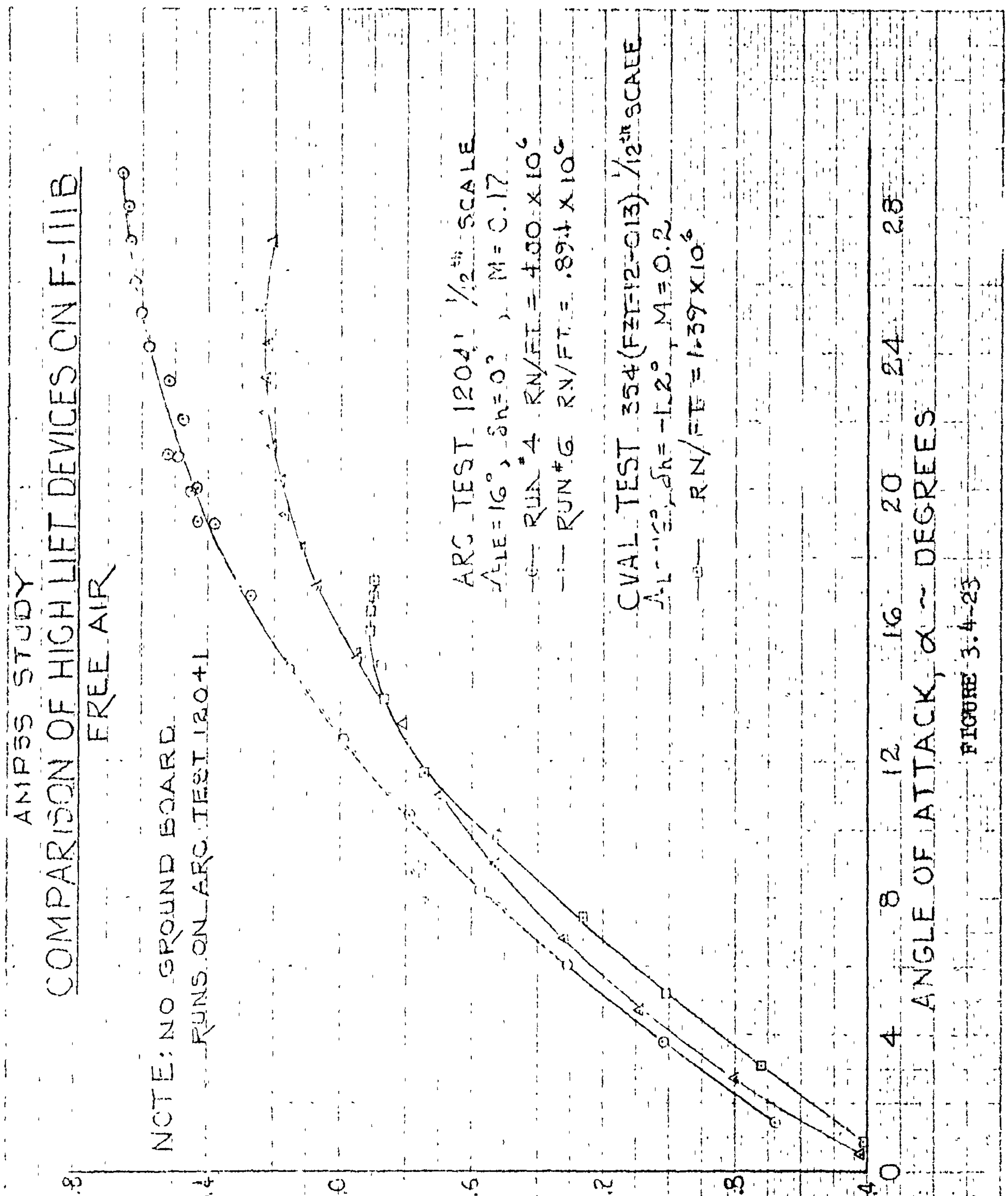

AMP55 STUDY
COMPARISON OF HIGH LIFT DEVICES ON F-111B
FREE AIR
NOTE: NO GROUND BOARD
RUNS ON ARC TEST 1204 L
ARC TEST 1204 1/12th SCALE
RUN #4 RN/FT = 4.30 x 10^6
RUN #6 RN/FT = .894 x 10^6
CVAL TEST 354 (FZT-12-013) 1/12th SCALE
RN/FT = 1.39 x 10^6
LIFT COEFFICIENT, C_L
1.4
1.8
2.2
2.6
3.0
3.4
3.8
ANGLE OF ATTACK, α ~ DEGREES
0
4
8
12
16
20
24
28

FIGURE 3.4-23

SECRET

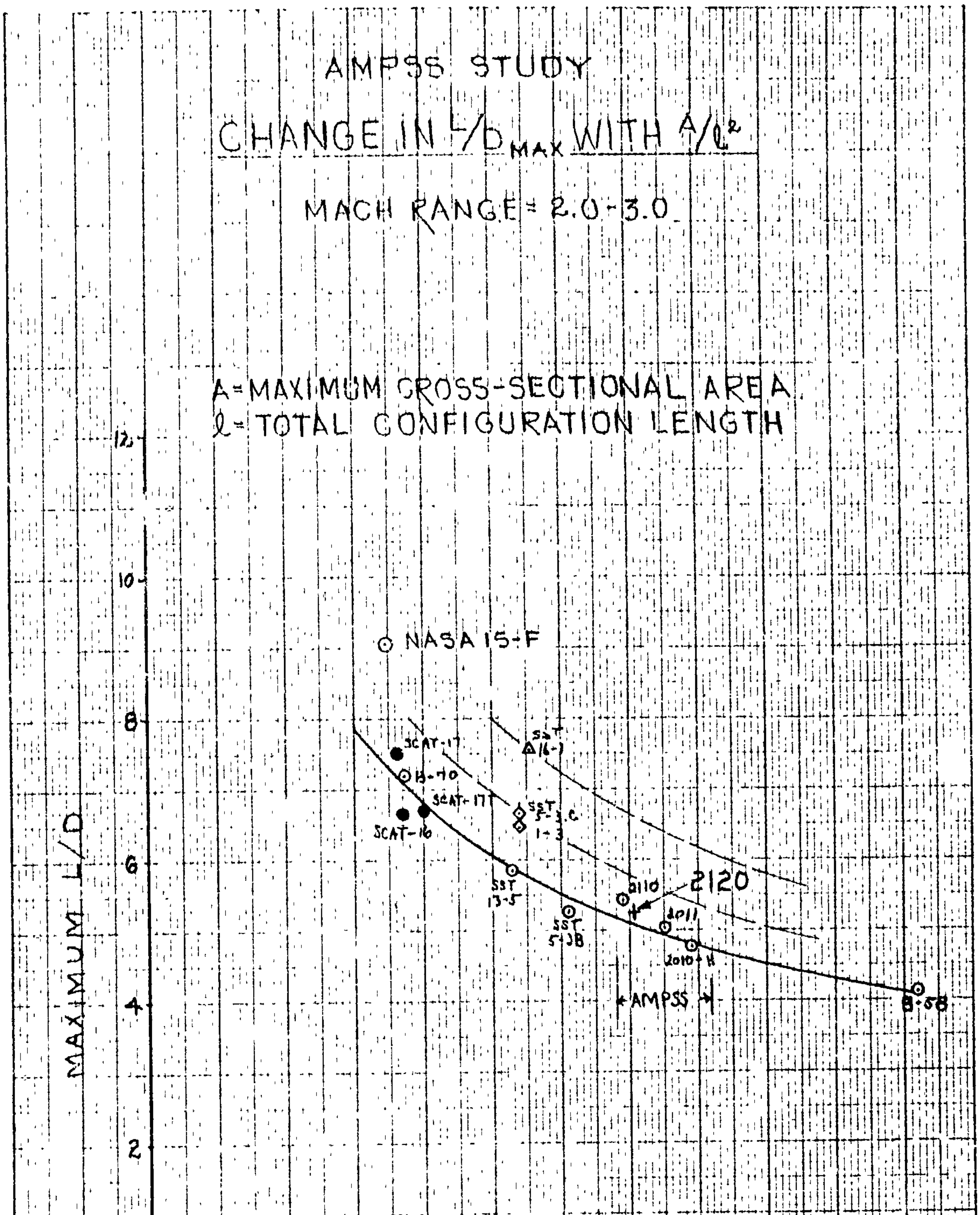


FIGURE 3.4-24

SECRET

3.5 PROPULSION

3.5.1 Data Source

Since publication of the last AMPSS project report (GD/FW Report FZM-4124, SLAMP Configuration 2110, 27 April 1964), Pratt & Whiteney has published new performance data for the STF 200C-35.1 engine (P&W report TDM 1860, Preliminary Performance Estimates of the STF 200C-35.1 Duct Heating Turbofan Engines, 1 May 1964). These engine performance data are essentially the same as those previously published for the STF 200C-35.1 engine with the following two exceptions:

1. Military and normal power thrust at Mach 0.85, sea level are increased.
2. Augmented (duct heater lit) performance is improved.

The increase in normal power thrust at Mach 0.85, sea level, had little effect on Configuration 2120 design because thrust requirements at this flight condition do not determine engine size. However, if P&W had not increased the normal power thrust at this flight condition (approximately 30%), engine size would have been determined by the Mach 0.85, sea level,thrust requirement. The performance improvements during augmented operation had a small effect on airplane design in that takeoff thrust was slightly increased which resulted in a little smaller relative engine scale. Improvements in supersonic augmented performance yielded increased aircraft supersonic range. However, this had no effect on aircraft design (size) since the supersonic performance of the basic design exceeded the supersonic range requirement.

3.5.2 Installation Penalties

The installed data penalties were determined in accordance with the procedures outlined in Section 4.2.2.6, page 4-25 and Section 4.2.2.7, page 4-35 of GD/FW Report FZM-4038-II-1, Advanced Manned Precision Strike System, Technical Report, 3 February 1964. The only difference in current installation penalties and previous penalties are thos resulting from data changes made by P&W.

3.5.3 Installed Data

Installed data based on Pratt & Whitney TDM 1860 are presented in Figure 3.5-1 thru 3.5-11. These data reflect the effects of inlet pressure recovery, bleed, power extraction, and exhaust nozzle performance. P&W nozzle base drags are not included in the subsonic performance, but are included in the supersonic performance.

The nozzle performance data presented by P&W in their report, TDM 1860, is not directly applicable in its entirety to calculate installed engine thrust. Their blow-in-door nozzle performance data, as presented in TDM 1860, defines the nozzle thrust minus drag for a range of Mach numbers and engine powers for a configuration wherein the nozzle is installed behind a cylindrical nacelle forebody. At subsonic speeds with dry engine power, the nozzle thrust minus drag in this installation will be lower than in an airplane installation. At these speeds the drag of a boat-tail surface is higher when this boat-tail surface is located behind a cylinder than when located behind a boat-tailed nacelle. The subsonic installed nozzle performance has thus been approached by comparison of the blow-in-door nozzle with a conventional C-D ejector nozzle and recognizing that the thrust minus drag of these two nozzles will be equal if the blow-in-door nozzle aerodynamic design is done properly. The conventional C-D ejector nozzle configuration permits a clear separation of thrust and drag forces not possible on the blow-in-door nozzle due to the complex internal-external aerodynamic interactions. Thus, the external drag of the conventional C-D ejector has been included in the airplane drag analysis and the nozzle thrust performance has been accounted for in the engine performance. The external pressure drag of a conventional ejector boat-tail is zero at Mach .85 as shown by the B58 J79 engine low base drag nozzle. The internal performance of a conventional C-D ejector has been estimated to result in a nozzle thrust coefficient of 0.985. The drag of two percent nozzle corrected secondary airflow is charged against the engine performance. Also, 300 pounds were added to the full scale engine weight to account for the controls and actuators required for a C-D nozzle. This weight increase is included in the data in Figure 3.5-12.

At supersonic cruise speed, a similar analysis is not required and the blow-in-door nozzle thrust minus drag quoted in P&W TDM 1860 has been incorporated in the engine performance. This is possible since the nozzle boat-tail during A/B operation at supersonic cruise is small. Thus, the difference in nozzle pressure drag between the P&W wind tunnel test (cylindrical nacelle forebody) and the airplane installation is essentially zero.

3.5.4 Scaling Data

Scaling data based on TDM 1860 (Reference 3.5.2) are presented in Figure 3.5-12 and 3.5-13. Data are shown for both weight and overall dimensions for engine sizes down to about 30 percent thrust size.

3.5.5 Operating Limits

Operating limits are shown in Figure 3.5-14. The region bounded by the solid line represents the flight conditions at which the engine will operate normally -- continuously at normal power and 30 minutes at military. At low altitudes, the region bounded by the solid line and the dashed line represent flight conditions at which military power is limited to 5 minutes operation and normal power is limited to 30 minutes operation. In this region, the military power time limit can be raised to 30 minutes by adding 100 pounds of weight per engine; normal power can be made continuous by adding 50 pounds of weight per engine; continuous operation at powers below normal power can be achieved without any weight increase.

SECRET

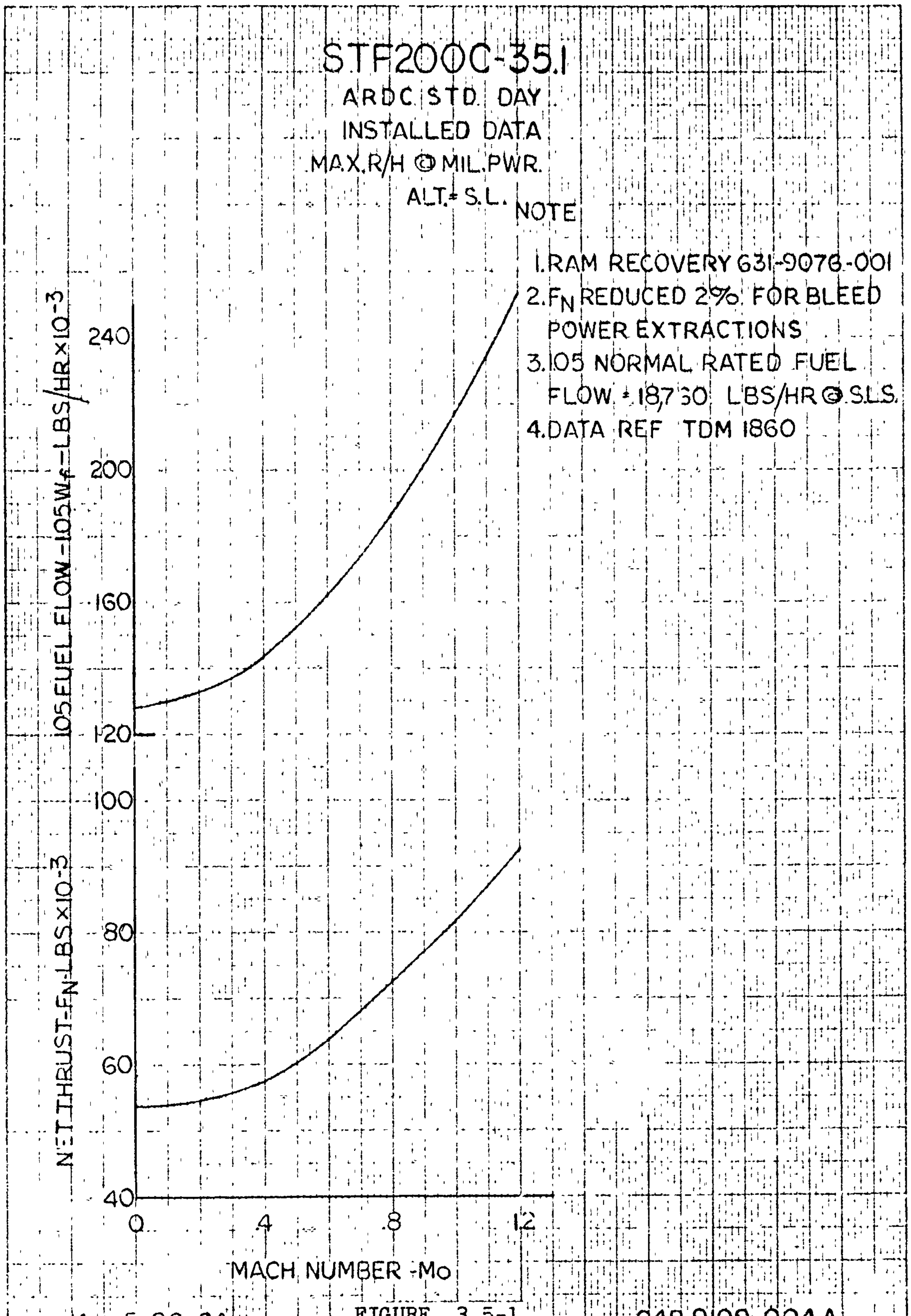


FIGURE 3.5-1

SECRET

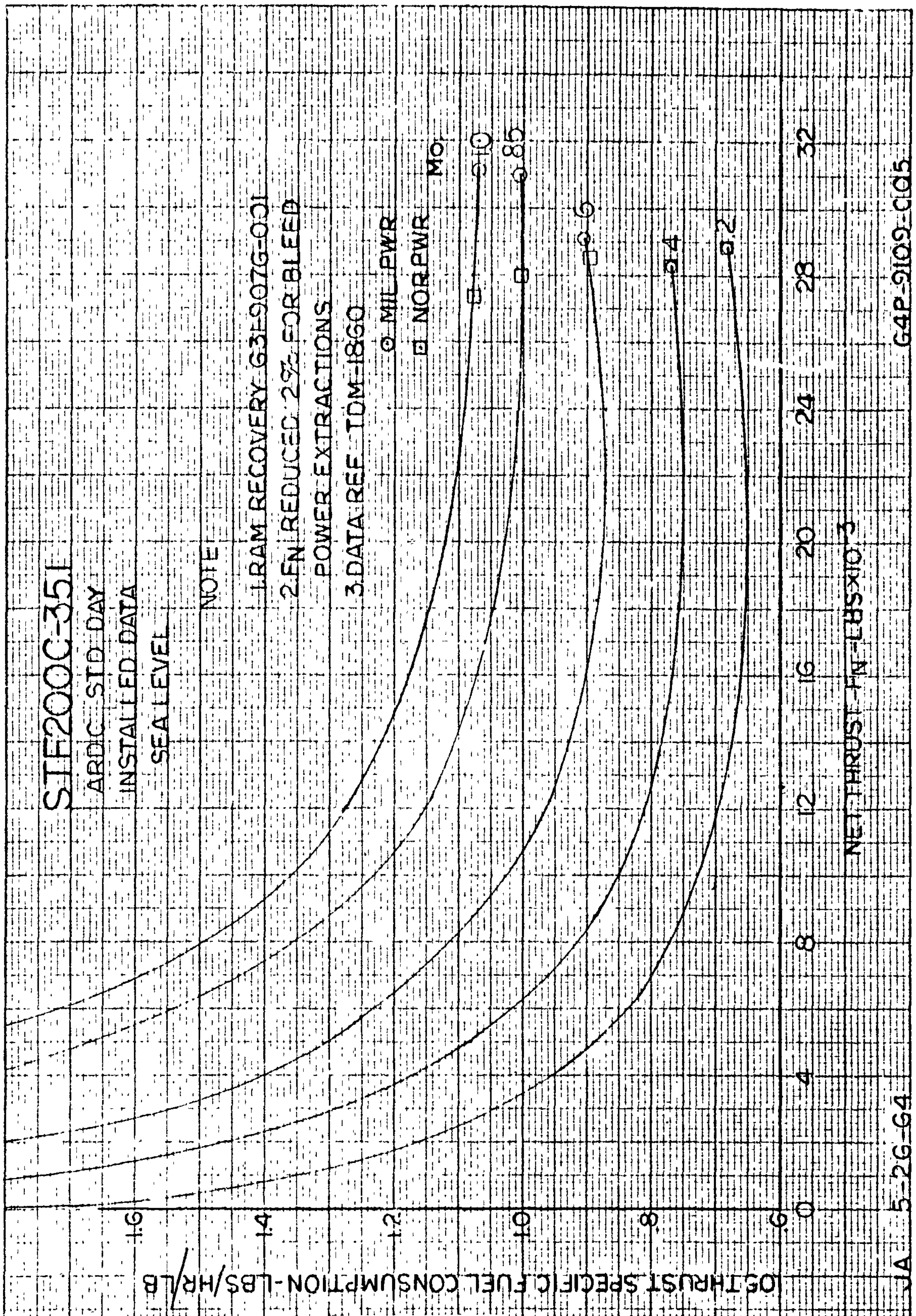
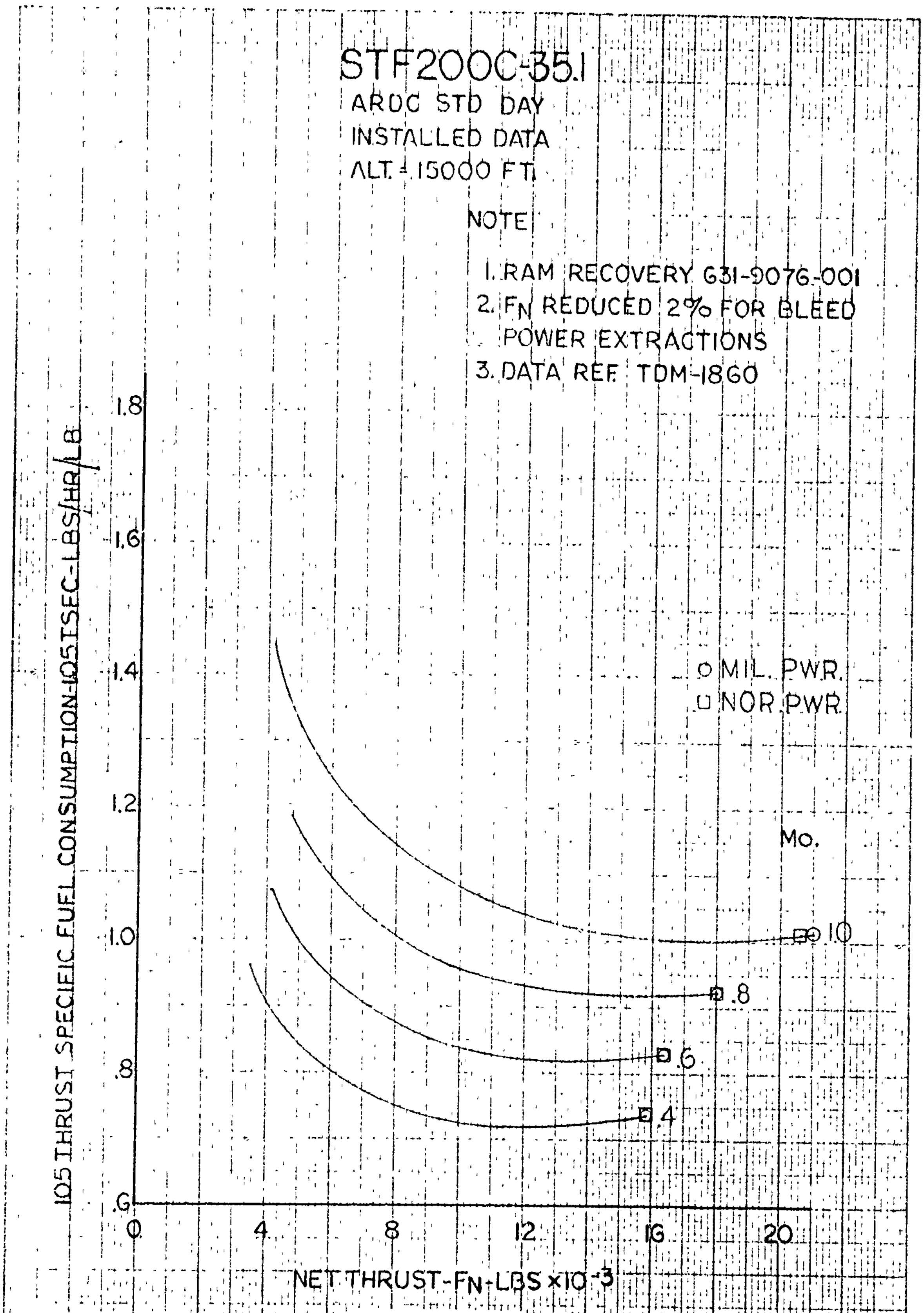
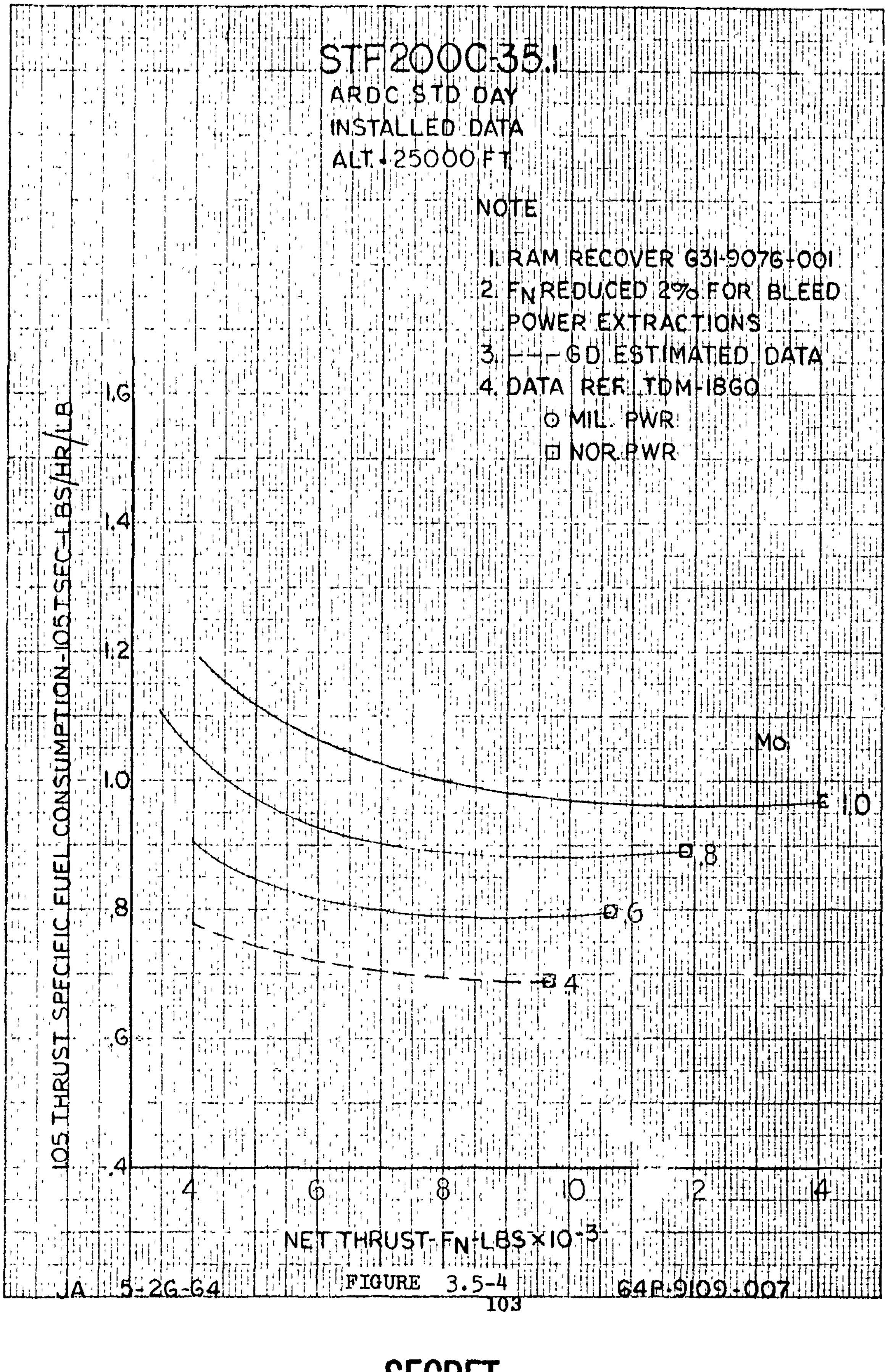

STF200C-35.1
ARDC STD DAY
INSTALLED DATA
SEA LEVEL
NOTE
1.RAM RECOVERY G31-907G-001
2.FN REDUCED 2% FOR BLEED
POWER EXTRACTIONS
3.DATA REF TDM-1860
⊙ MIL PWR
⊡ NOR PWR
Mo
1.0
.85
.6
.4
.2
1.6
1.4
1.2
1.0
.8
.6
0
4
8
12
16
20
24
28
32
NET THRUST-FN-LBS×10^-3
INSTALLED THRUST SPECIFIC FUEL CONSUMPTION-LBS/HR/LB
G4P-9109-C105
JA 5-26-64

FIGURE 3.5-2



JA 5-26-64 FIGURE 3.5-3 64R-9109-006



STF 200C-35.1
ARDC STD DAY
INSTALLED DATA
ALT - 25000 FT
NOTE
1. RAM RECOVER G31-9076-001
2. FN REDUCED 2% FOR BLEED POWER EXTRACTIONS
3. - - - GD ESTIMATED DATA
4. DATA REF. TDM-1860
MIL. PWR
NOR. PWR
105 THRUST SPECIFIC FUEL CONSUMPTION-105 TSEC-LBS/HR/LB
1.6
1.4
1.2
1.0
.8
.6
.4
Mo
1.0
.8
.6
.4
4
6
8
10
12
14
NET THRUST-FN-LBS x 10^-3
JA 5-26-64
64P-9109-007

FIGURE 3.5-4

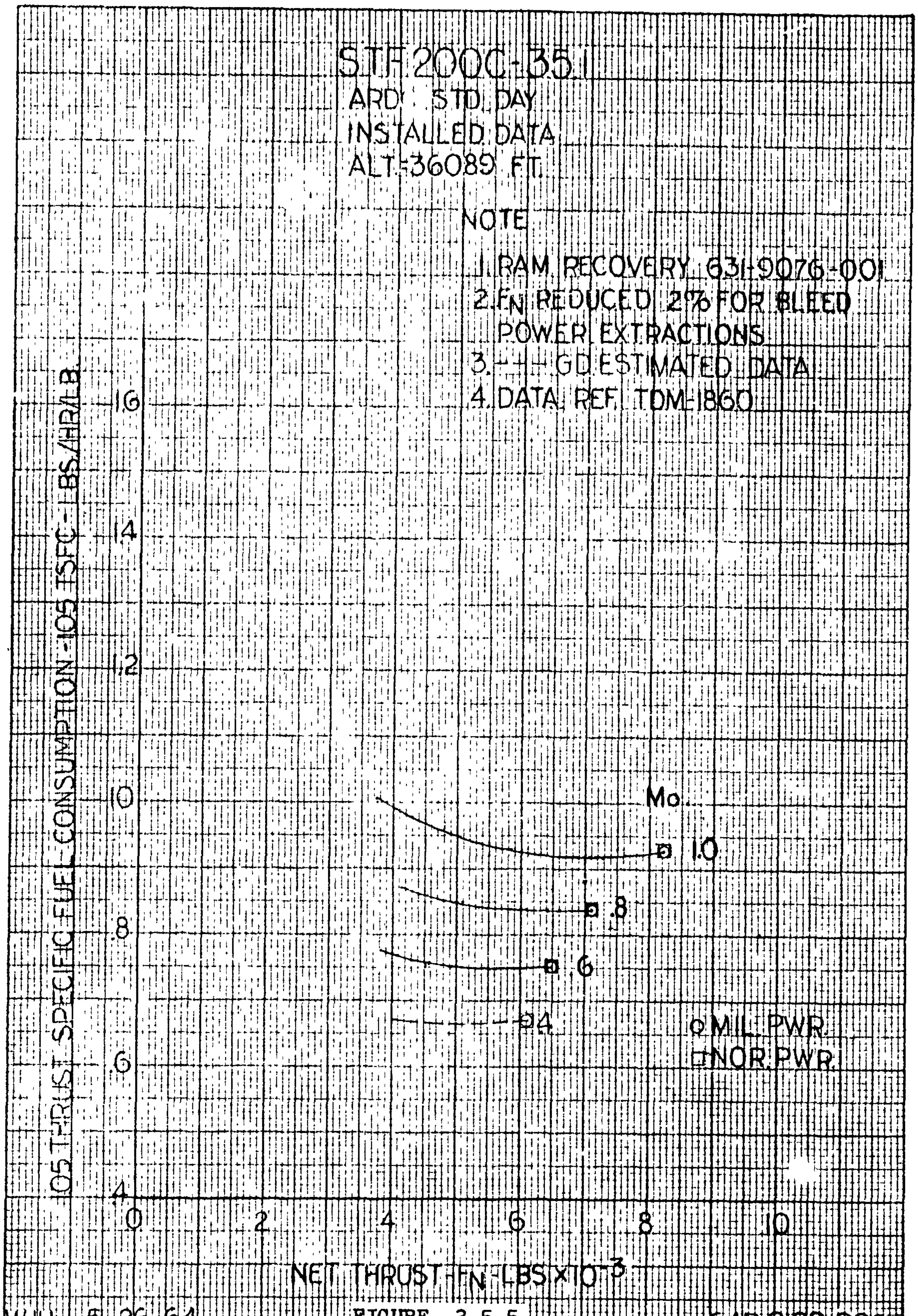

STF 200C-35 I
ARDC STD. DAY
INSTALLED DATA
ALT-36089 FT.
NOTE
1. RAM RECOVERY 631-9076-001
2. FN REDUCED 2% FOR BLEED POWER EXTRACTIONS
3. --- GD ESTIMATED DATA
4. DATA REF TDM-1860
105 THRUST SPECIFIC FUEL CONSUMPTION -105 TSFC - LBS/HR/LB
16
14
12
10
8
6
4
Mo
1.0
.8
.6
.4
○ MIL PWR
□ NOR PWR
0
2
4
6
8
10
NET THRUST - FN - LBS x 10^-3
WJH 5-26-64
64P-9109-008

FIGURE 3.5-5

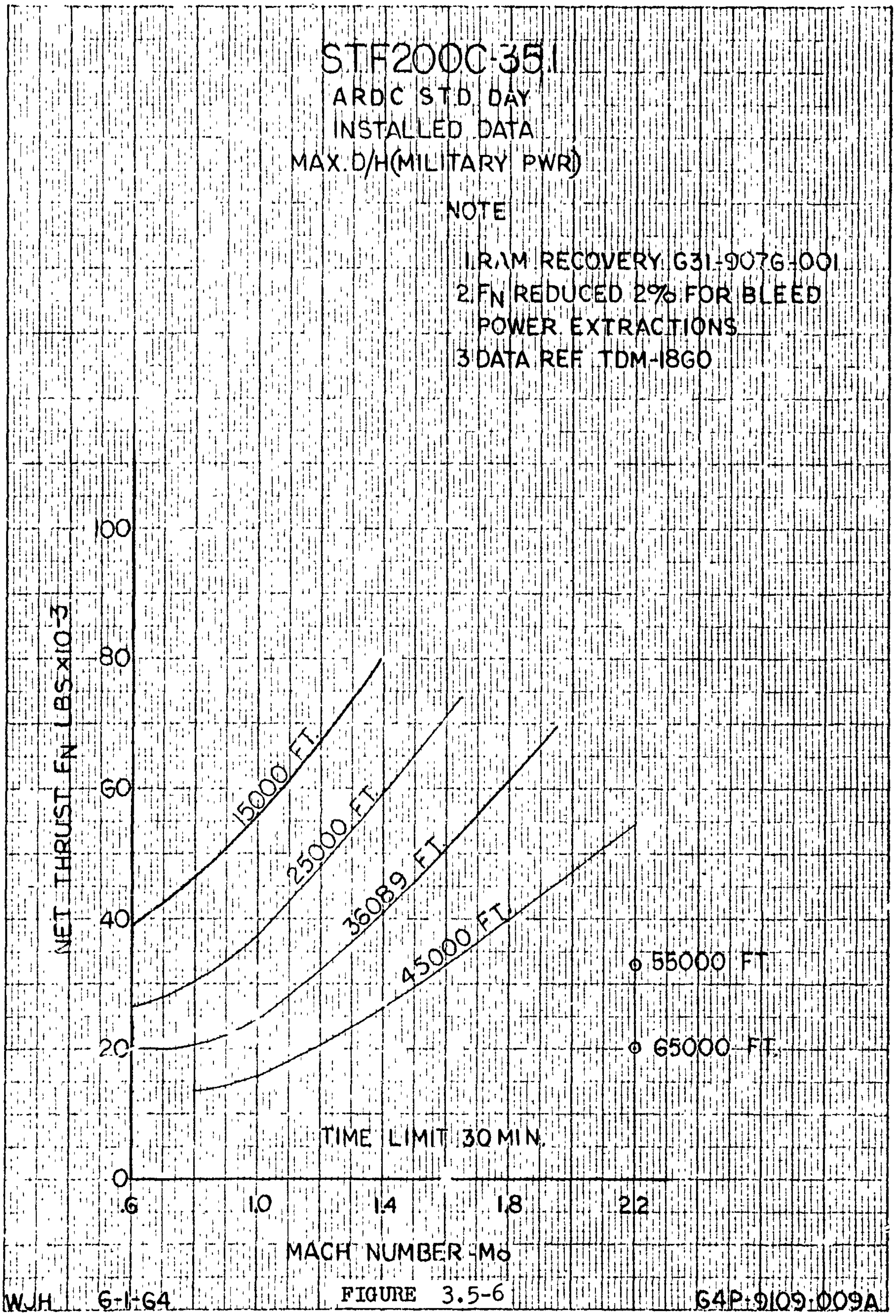
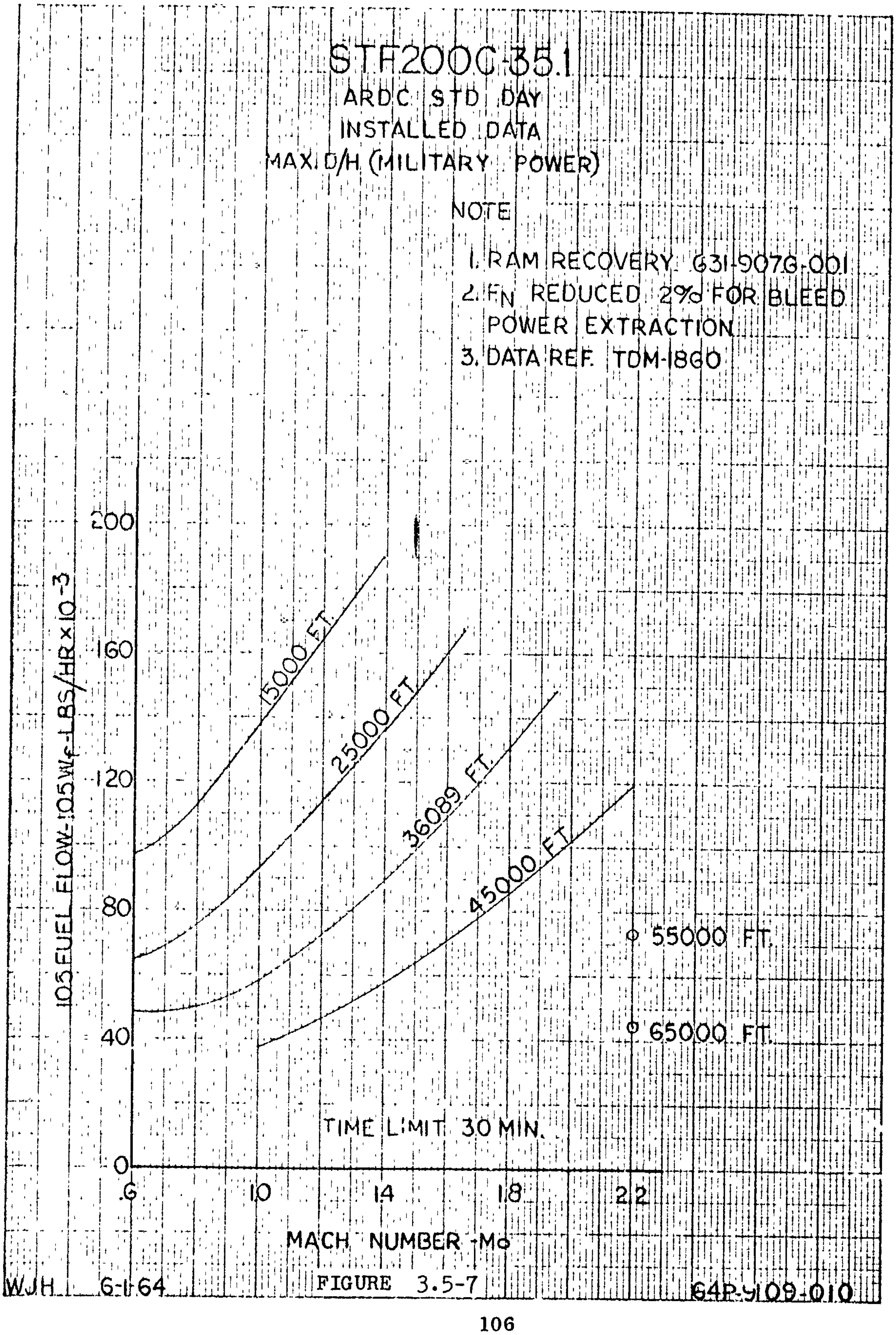

STF200C-35.1
ARDC STD DAY
INSTALLED DATA
MAX. D/H (MILITARY PWR)
NOTE
1. RAM RECOVERY G31-9076-001
2. FN REDUCED 2% FOR BLEED POWER EXTRACTIONS
3 DATA REF TDM-1860
15000 FT.
25000 FT.
36089 FT.
45000 FT.
55000 FT.
65000 FT.
TIME LIMIT 30 MIN.
NET THRUST FN LBS x10-3
MACH NUMBER - Mo
0
20
40
60
80
100
.6
1.0
1.4
1.8
2.2
WJH 6-1-64
64P-9109-009A

FIGURE 3.5-6


STF200C-35.1
ARDC STD DAY
INSTALLED DATA
MAX. D/H (MILITARY POWER)
NOTE
1. RAM RECOVERY 631-9076-001
2. FN REDUCED 2% FOR BLEED POWER EXTRACTION
3. DATA REF. TDM-1860
15000 FT.
25000 FT.
36089 FT.
45000 FT.
55000 FT.
65000 FT.
TIME LIMIT 30 MIN.
105 FUEL FLOW-105 Wf-LBS/HR x 10-3
0
40
80
120
160
200
.6
1.0
1.4
1.8
2.2
MACH NUMBER-Mo
WJH 6-1-64
64P-9109-010
FIGURE 3.5-7

SECRET

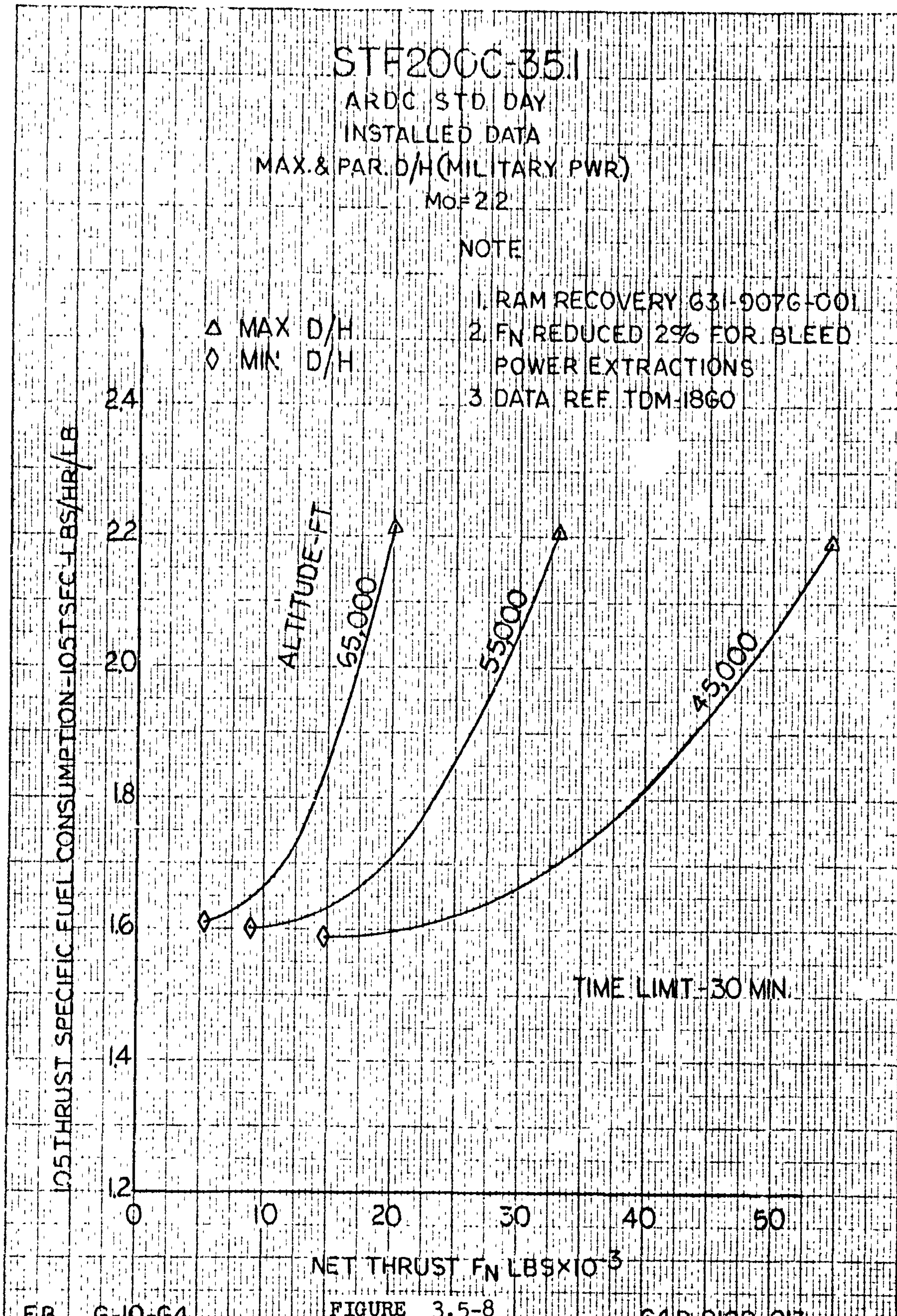


FIGURE 3.5-8

SECRET

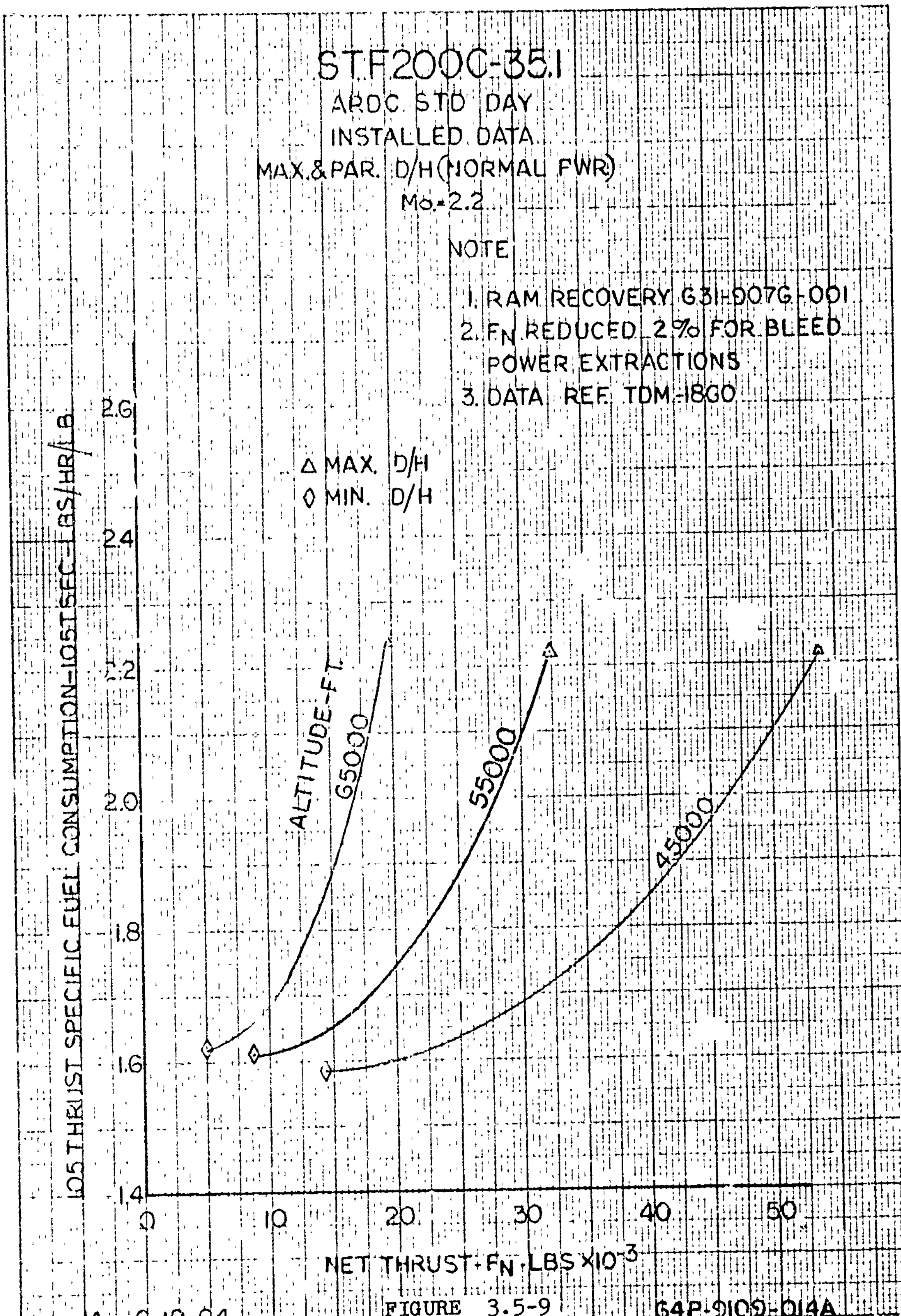
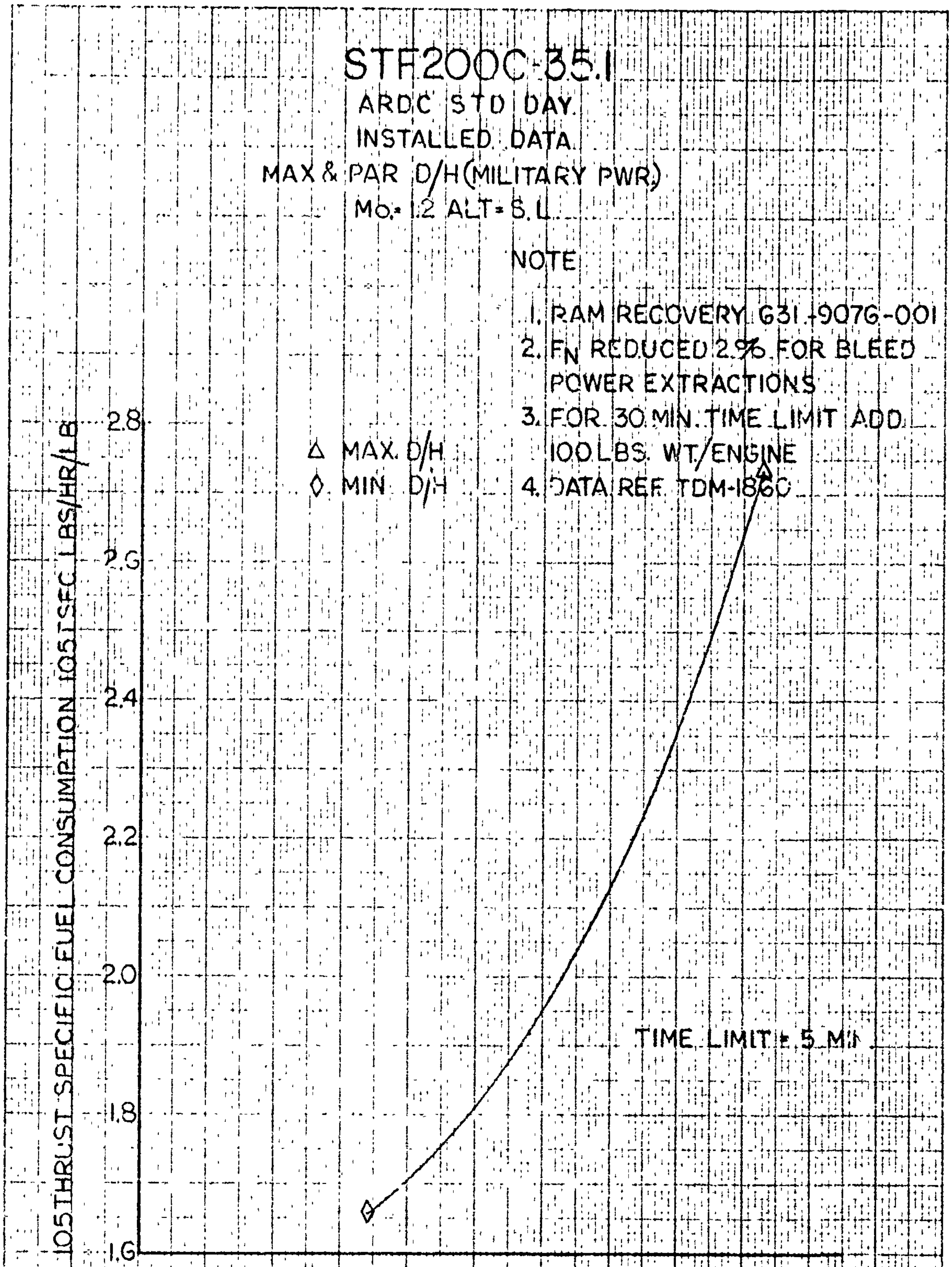
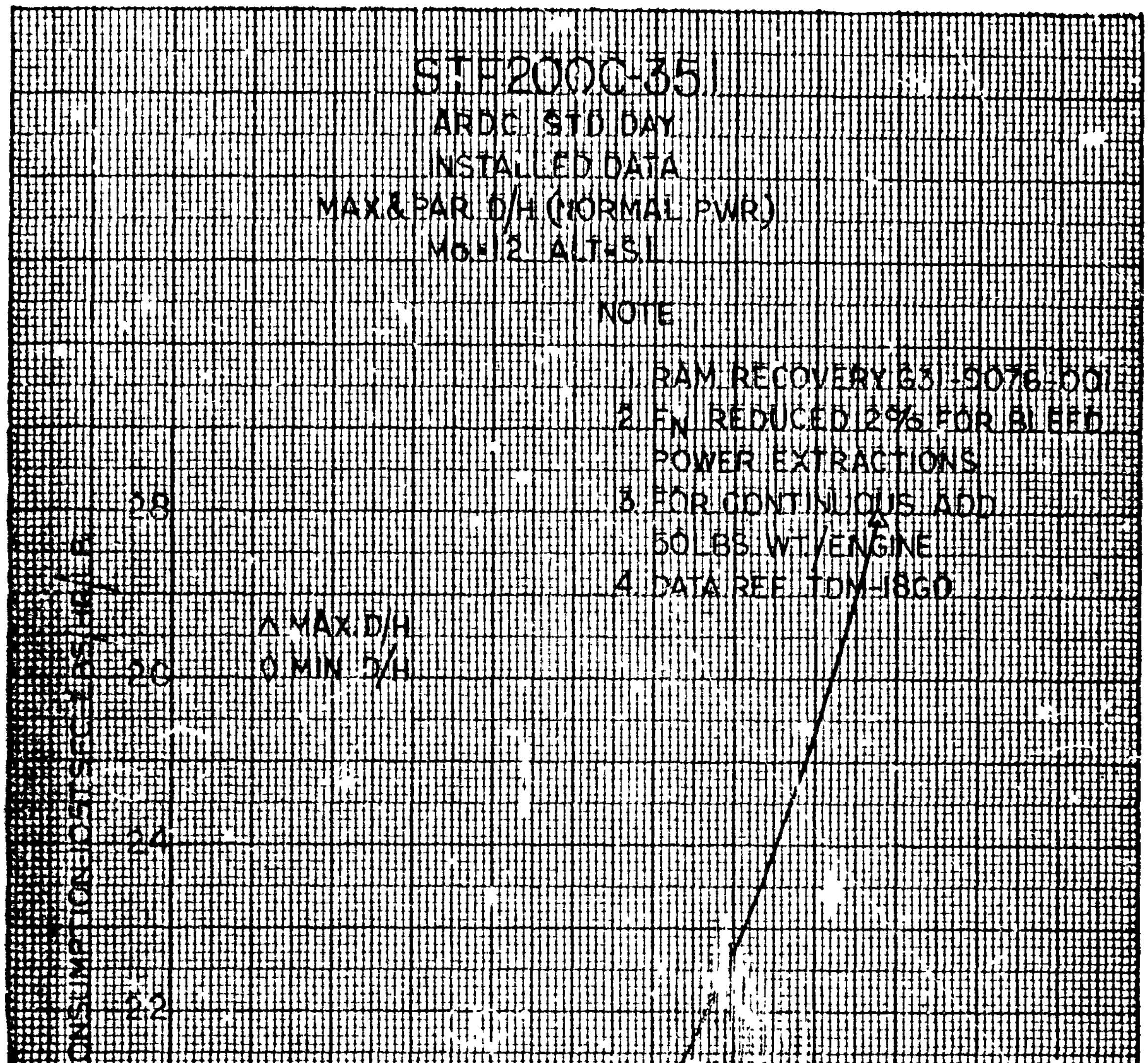

STF200C-35.1
ARDC STD DAY
INSTALLED DATA
MAX.&PAR. D/H (NORMAL PWR)
Mo.=2.2
NOTE
1. RAM RECOVERY 631-0076-001
2. F_N REDUCED 2% FOR BLEED POWER EXTRACTIONS
3. DATA REF. TDM-1860
△ MAX. D/H
◊ MIN. D/H
ALTITUDE-FT.
65000
55000
45000
105 THRUST SPECIFIC FUEL CONSUMPTION-105TSFC-LBS/HR/LB
2.6
2.4
2.2
2.0
1.8
1.6
1.4
0
10
20
30
40
50
NET THRUST-F_N-LBS $\times 10^{-3}$
JA 6-10-64
64P-9102-014A

FIGURE 3.5-9


STF200C-35.1
ARDC STD DAY
INSTALLED DATA
MAX & PAR D/H (MILITARY PWR)
Mo = 1.2 ALT = S.L.
NOTE
1. RAM RECOVERY 631-9076-001
2. FN REDUCED 2% FOR BLEED POWER EXTRACTIONS
3. FOR 30 MIN. TIME LIMIT ADD 100 LBS. WT/ENGINE
4. DATA REF. TDM-1860
△ MAX. D/H
◊ MIN. D/H
TIME LIMIT = 5 MIN
1.6
1.8
2.0
2.2
2.4
2.6
2.8
0
20
40
60
80
100
1.05 THRUST SPECIFIC FUEL CONSUMPTION 1.05 TSFC LBS/HR/LB
NET THRUST FN LBS x 10-3
JA 6-10-64
64P-0109-011A
FIGURE 3.5-10


STF200C-35
ARDC STD DAY
INSTALLED DATA
MAX & PAR. D/H (NORMAL PWR)
Mo=2 ALT=S.L.
NOTE
1. RAM RECOVERY 63-5076 DO
2. FN REDUCED 2% FOR BLEED POWER EXTRACTIONS
3. FOR CONTINUOUS ADD 50LBS. WT/ENGINE
4. DATA REF. TDM-18GO
△ MAX. D/H
◇ MIN. D/H
TIME LIMIT = 30 MIN.
NET THRUST SPECIFIC FUEL CONSUMPTION-TSFC-LBS/HR/LB
1.6
1.8
2.0
2.2
2.4
2.6
2.8
0
20
40
60
80
100
NET THRUST-FN-LBS×10⁻³
JA 6-10-64
G4P-5109-012A
FIGURE 3.5-11

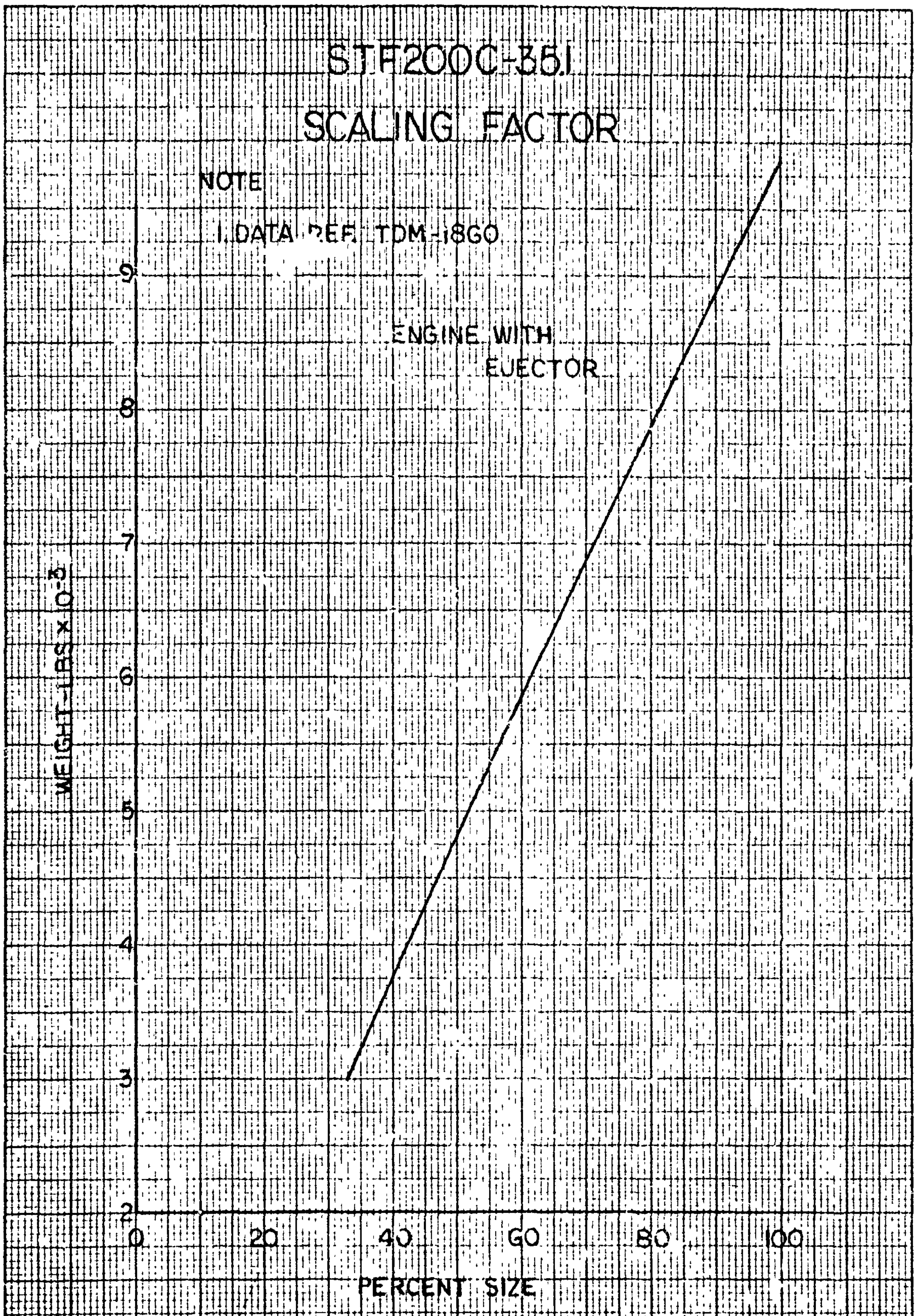

STF200C-35I
SCALING FACTOR
NOTE
1. DATA REF. TDM-1860
ENGINE WITH
EJECTOR
WEIGHT-LBS x 10^-3
9
8
7
6
5
4
3
2
0
20
40
60
80
100
PERCENT SIZE
JA 6-25-64
G4P-9109-001A

FIGURE 3.5-12

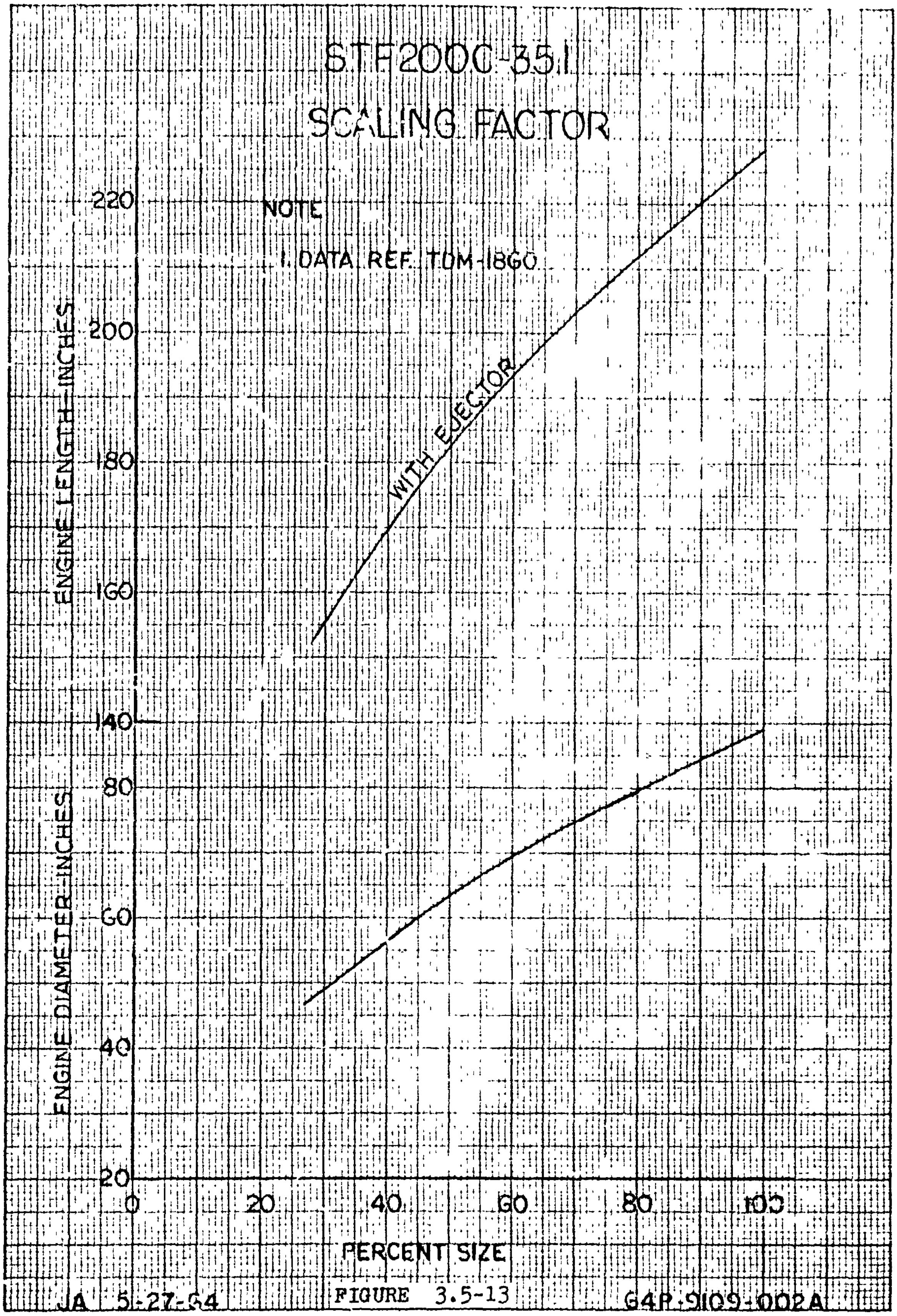

STF200C-35.1
SCALING FACTOR
NOTE
1. DATA REF. TDM-1860
WITH EJECTOR
ENGINE LENGTH–INCHES
ENGINE DIAMETER–INCHES
220
200
180
160
140
80
60
40
20
0
20
40
60
80
100
PERCENT SIZE
JA 5-27-54
G4P-9109-002A

FIGURE 3.5-13

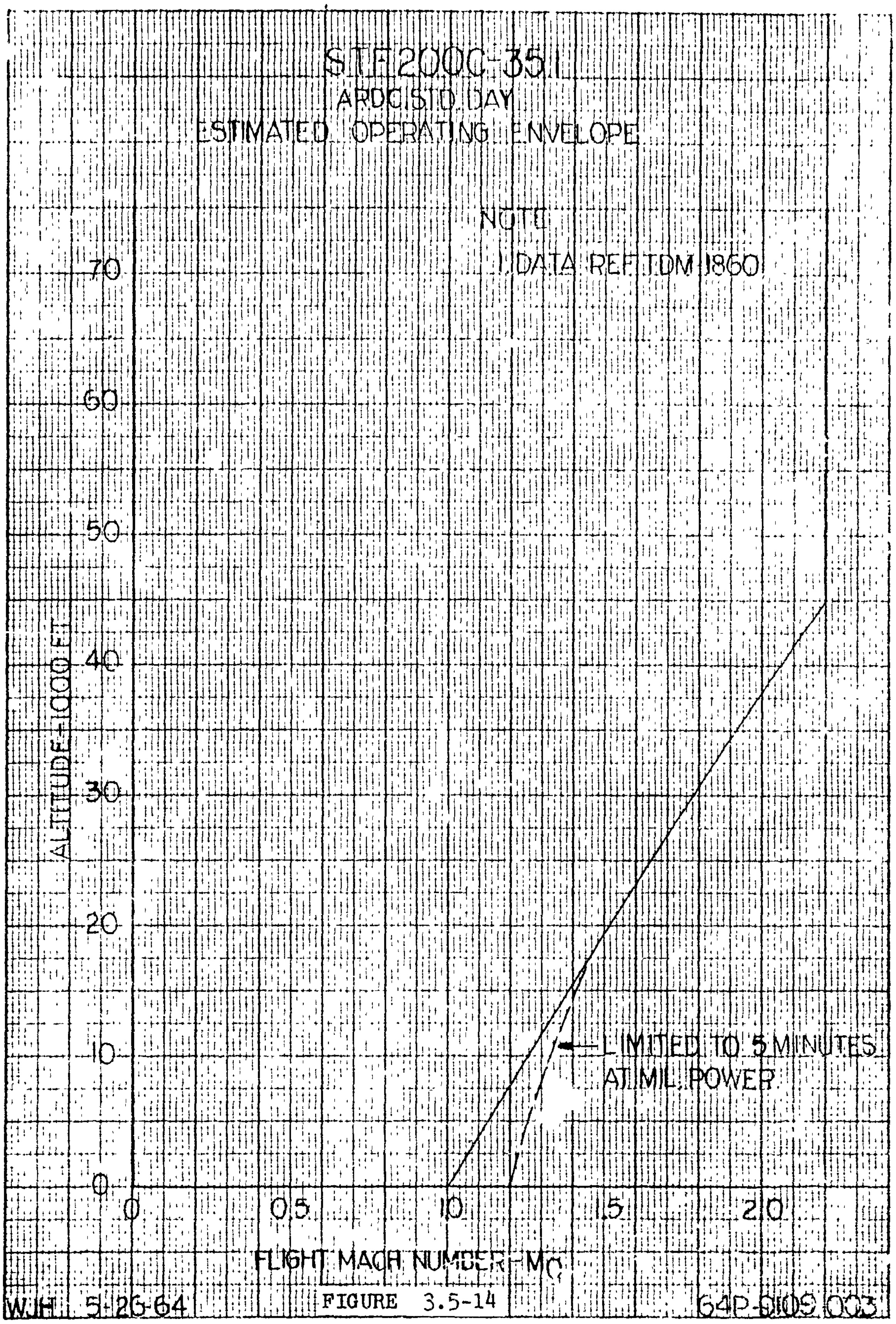

STF 200C-351
ARDC STD DAY
ESTIMATED OPERATING ENVELOPE
NOTE
1. DATA REF TDM 1860
ALTITUDE-1000 FT
70
60
50
40
30
20
10
0
0
0.5
1.0
1.5
2.0
FLIGHT MACH NUMBER - M_0
LIMITED TO 5 MINUTES AT MIL POWER
WJH 5-26-64
64P-0109.003

FIGURE 3.5-14

3.6 STABILITY AND FLIGHT CONTROL

The stability and flight controls characteristics for configuration 2120 are in general, similar to those for configuration 2010H presented previously in Section 7 of FZM-4038-II-2. Because of this similarity, only the most significant characteristics affected by modifications to configuration geometry and physical loadings are presented herein for configuration 2120. Design criteria and flight control system descriptions remain as presented in Sections 7.1 and 7.4 of FZM-4030-II-2. It should be noted that reference geometry for 2120 is based upon the planform of the wing in the extended position (neglecting the glove area forward of the wing panel leading edge) similar to the reference area used for 2010H.

3.6.1 Static Stability

Low Speed longitudinal stability for the extreme loading conditions is indicated in Table 3.6-I by the static margins for each condition. As was the case for 2010H, an intermediate wing sweep of 26° is required for landing to maintain at least the minimum desired static margin of 5 per cent MAC. The variation of the aft center of gravity limit (5 per cent static margin) with wing sweep is presented in Figure 3.6-1.

Table 3.6-I Low Speed Balance

Loading Condition	Λ_{LE} Degrees	Center of Gravity % MAC	Static Margin % MAC
Maximum Weight	16	31.4	7.0
No Fuel with payload	26	39.3	18.8
No Fuel without payload	26	51.4	6.7

The vertical tail size has been selected to provide adequate directional stability throughout the ranges of flight-loading conditions including the extremes of low speed and 2.2 Mach number at 50,000 feet. Effects of angle of attack up to ten degrees and aeroelastic losses, where applicable, have been included in the analysis. For 2.2 Mach number at 50,000 a minimum directional stability level of +.0005 per degree is provided at the aft loading of 75 per cent MAC. The minimum directional stability levels for low speed flight which occur at the most aft loading for landing of 51.4 per cent MAC are +.00128 and +.00276 per degree for the clean and high lift configurations respectively.

3.6.2 Primary Control Surfaces

The surfaces used for control are of the same types as employed on 2010H. These are

1. All-movable horizontal tail for longitudinal control, trim, and stability augmentation
2. Conventional rudder for directional control, trim and stability augmentation

3. Differential horizontal tail deflection for roll trim, stability augmentation, and roll control at the high wing sweeps

4. Spoiler deflection in addition to differential horizontal tail motion for roll control at low wing sweeps.

Analysis of control capabilities for configuration 2120 has been limited to evaluations of longitudinal control for low spsed and trim at the basic flight conditions as shown in paragraphs 3.6.3 and 3.6.4. Evaluations of lateral and directional control for 2010H and subsequent configurations have indicated that sufficient control will be available on configuration 2120.

3.6.3 Low Speed Control

The adequacy of longitudinal control for low speed is indicated in Figure 3.6-2 which presents nose gear unstick capability in the form of gross weight divided by the product of reference wing area and dynamic pressure (corresponding to minimum-unstick speed). The trimmed maximum lift coefficient is also presented in this figure for direct comparison with this nose gear unstick capability. Thus, at the center of gravity positions where $\frac{GW}{qS}$ for unstick is greater than trimmed $C_{L_{max}}$, longitudinal control is more than adequate to unstick at stall speed. The forward loading limit is based upon nose gear unstick at the stall speed associated with $C_{L_{max}}$ in full ground effect.

3.6.4 Longitudinal Trim

The horizontal tail deflections for trim at the basic flight conditions are presented in Figures 3.6-3 through 3.6-6 as a function of trimmed lift coefficient. Two representative center of gravity locations have been utilized for each flight condition and the aerodynamic center is noted on the figures to facilitate comparison with these typical loadings. Estimated effects of aeroelasticity are included in the predictions for trim deflection and aerodynamic center. The effects of wing camber on trim deflections are also presented in these figures. The zero lift moment change is based upon analytical predictions which have been verified with wind tunnel results.

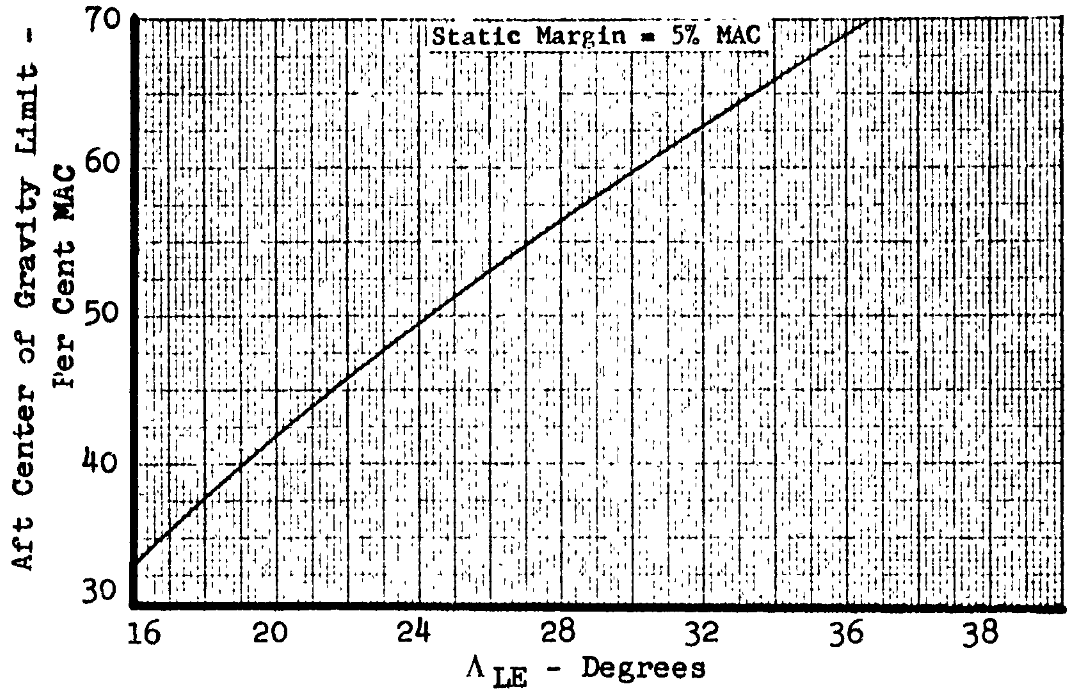


Figure 3.6-1 EFFECT OF WING SWEEP ON LOW SPEED AFT CENTER OF GRAVITY LIMIT

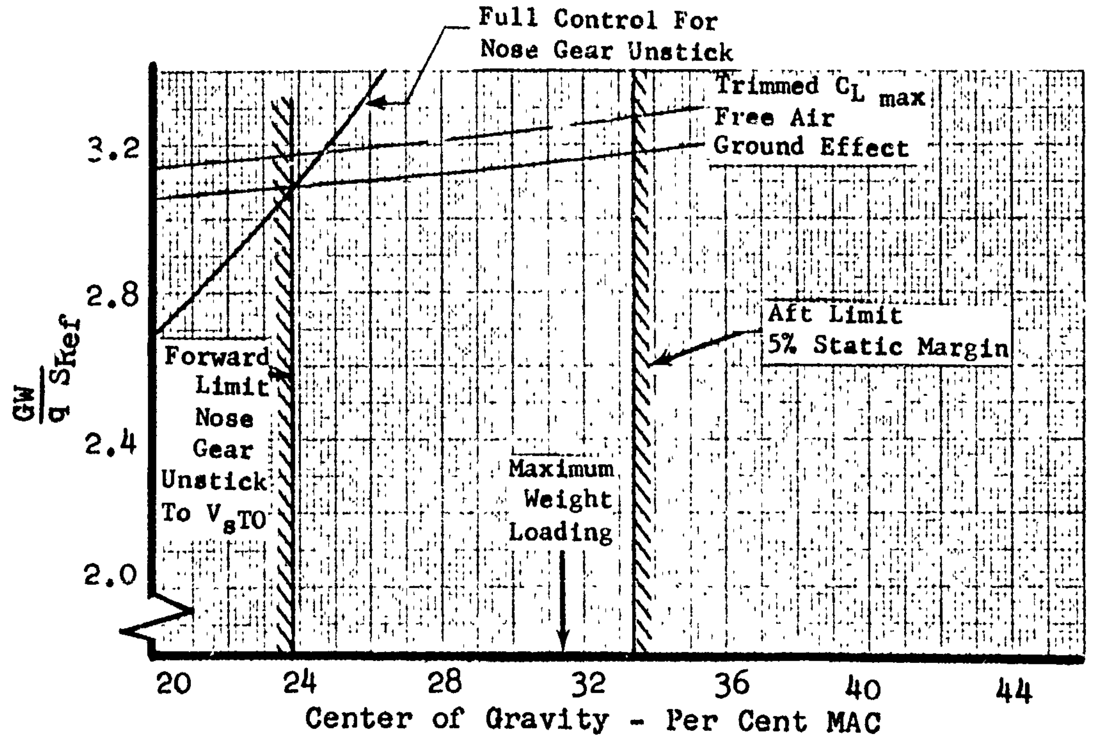
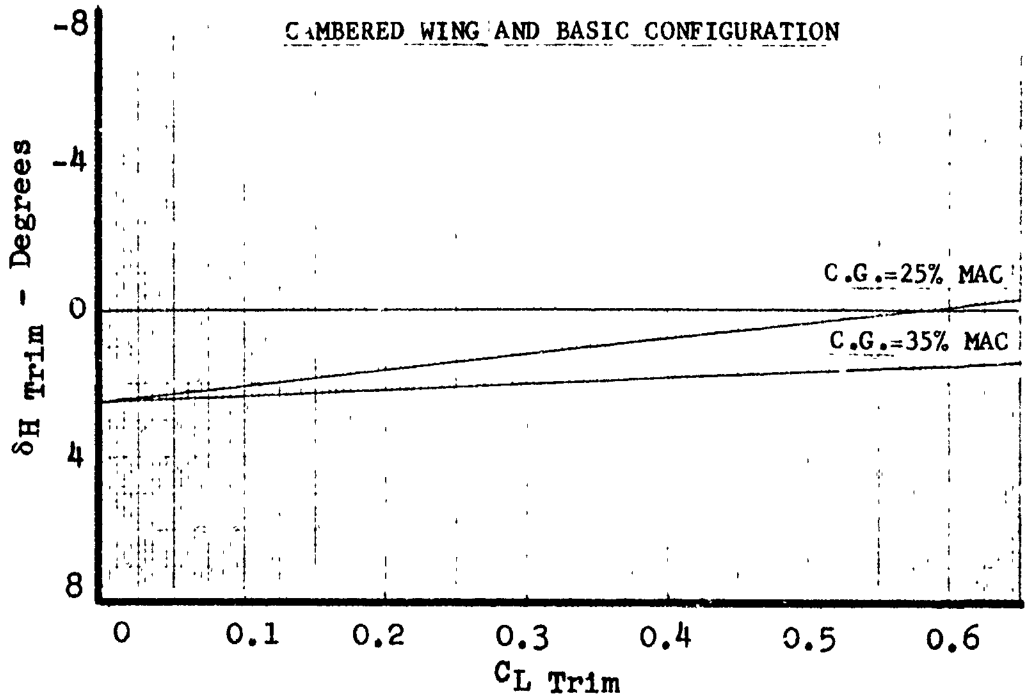


Figure 3.6-2 LOW SPEED LONGITUDINAL CONTROL AND BALANCE

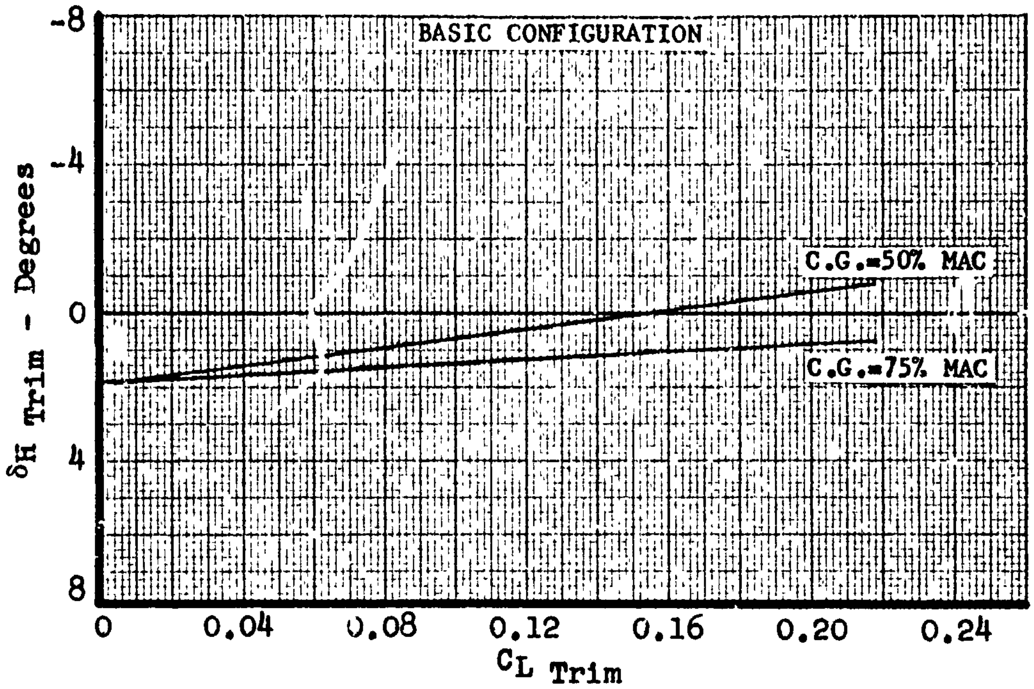
Λ_{LE} = 16 Degrees



Mach 0.7 30,000 Feet Altitude

$\Lambda_{LE}=16^{o}$ ac=41.0 Per Cent MAC

Figure 3.6-3 LONGITUDINAL TRIM - SUBSONIC CRUISE



Mach 0.85 Sea Level Altitude
$\Lambda_{LE}=72.5^{o}$ ac= 90.2 Per Cent MAC

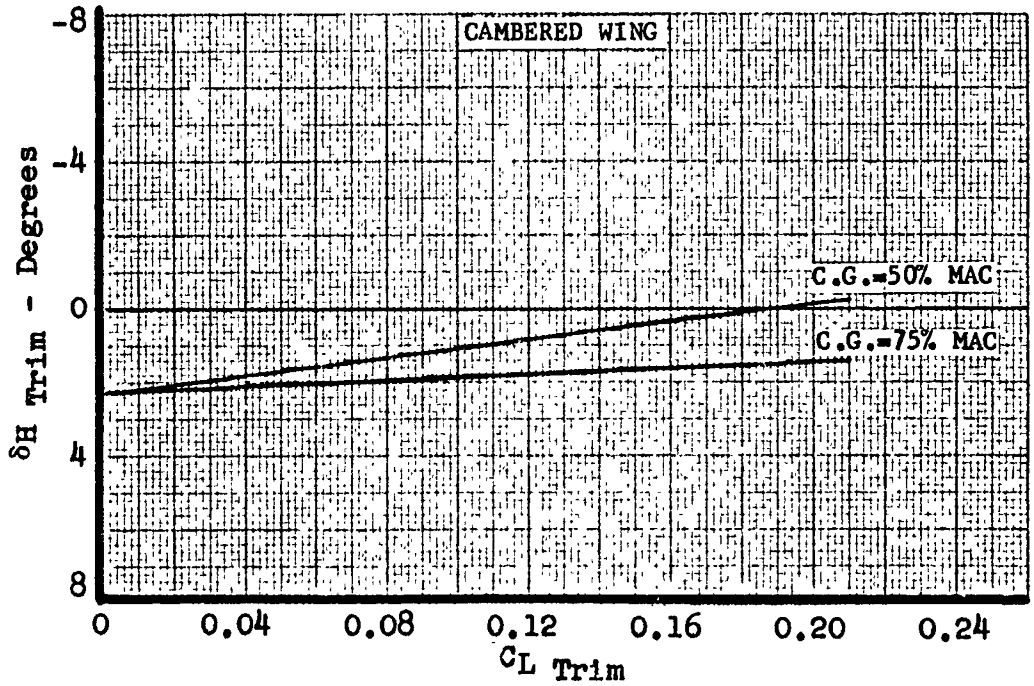
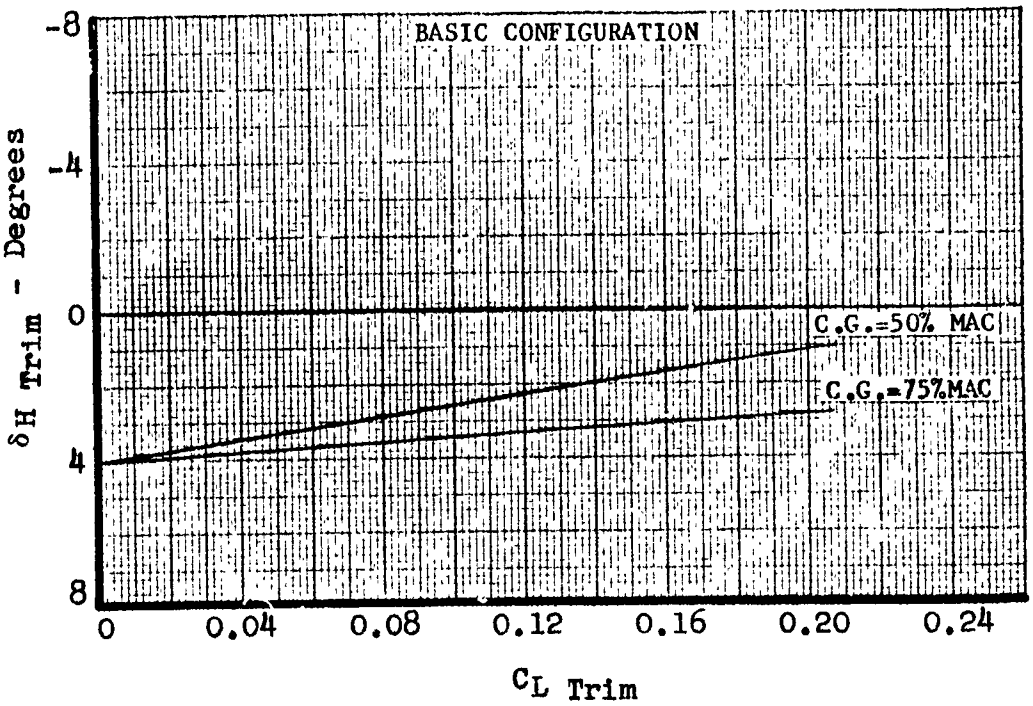


Figure 3.6-4 LONGITUDINAL TRIM - SUBSONIC PENETRATION



Mach 1.2 Sea Level Altitude
$\Lambda_{LE}=72.5^{\circ}$ ac=91.8 Per Cent MAC

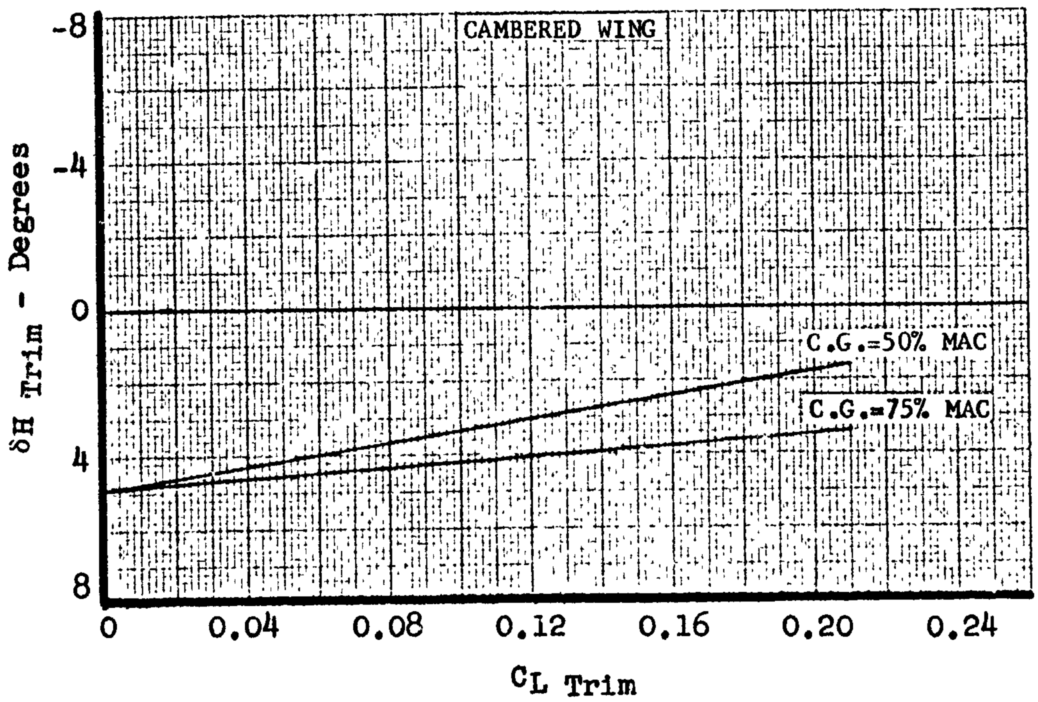
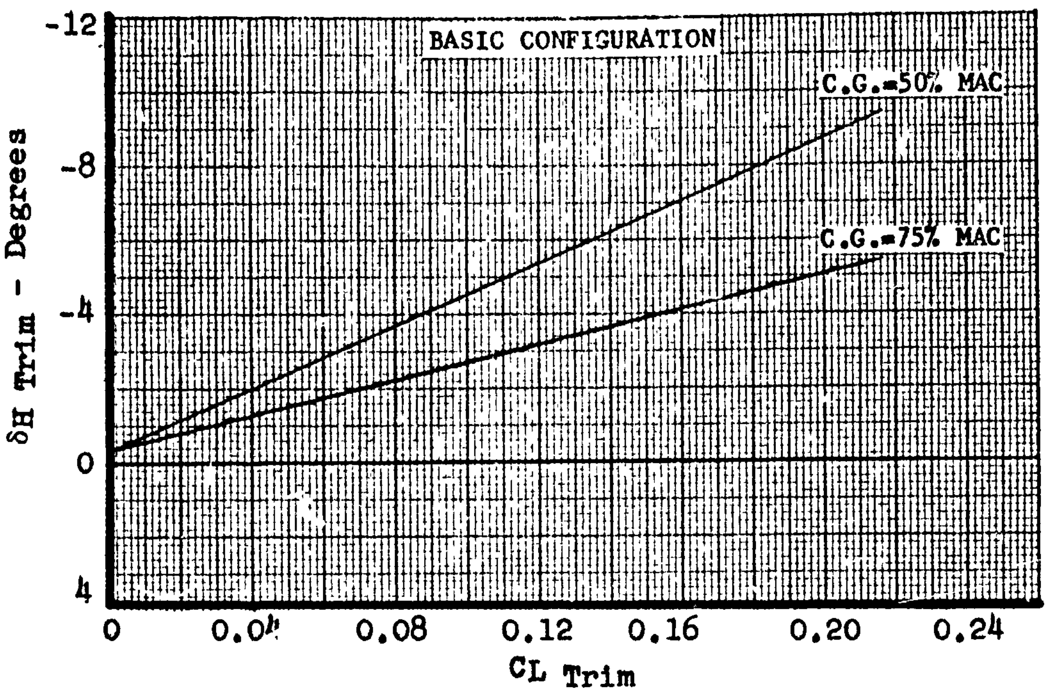


Figure 3.6-5 LONGITUDINAL TRIM - SUPERSONIC PENETRATION



Mach 2.2   55,000 Feet Altitude

Λ_{LE}=72.5°   ac=106.0 Per Cent MAC

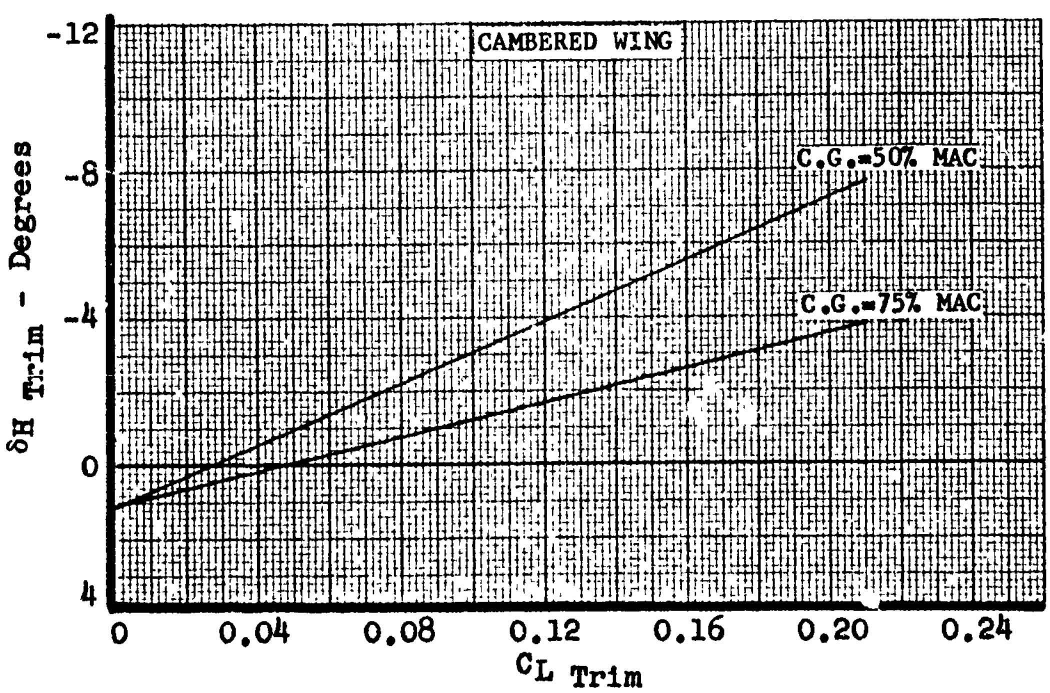


Figure 3.6-6 LONGITUDINAL TRIM - SUPERSONIC CRUISE

3.7 STRUCTURAL CRITERIA AND ANALYSIS

3.7.1 Structural Criteria

The structural criteria for the design point aircraft, AMPSS Configuration 2120, is essentially the same as that for the supersonic AMPSS study aircraft in Reference FZM-4038-II-2 and SLAMP Configuration 2110 in Reference FZM-4124 with the exception of the allowable sea level dash capabilities. The sea level Mach number has been increased from Mach .85 to Mach 1.2. The design speed altitude envelope for Configuration 2120 is presented in Figure 3.7-1.

The design gross weights for the present configurations are:

1.	Basic Flight Design Gross Weight	395,000 lbs.
2.	Maximum Refueled Gross Weight	403,000 lbs.
3.	Unrefueled Start of Dash Gross Weight	315,300 lbs.
4.	Refueled Start of Dash Gross Weight	349,000 lbs.
5.	Landing Design Gross Weight	271,000 lbs.
6.	Minimum Flying Weight	165,115 lbs.

The critical flight and ground loads conditions used for Configuration 2120 occur at essentially the same points in the flight regime as for the AMPSS Configuration 2010-H. The increase in Mach number at sea level did not affect the selection of design conditions. This is evident by examination of Figure 3.7-2 which illustrates the variation in maximum wing bending moments at the pivot station for various speeds and wing sweep positions. This figure shows the relative load levels as compared to the critical flaps down condition.

Table 3.7-I lists the critical flight loads conditions and the critical component(s) for each condition. No change was made to

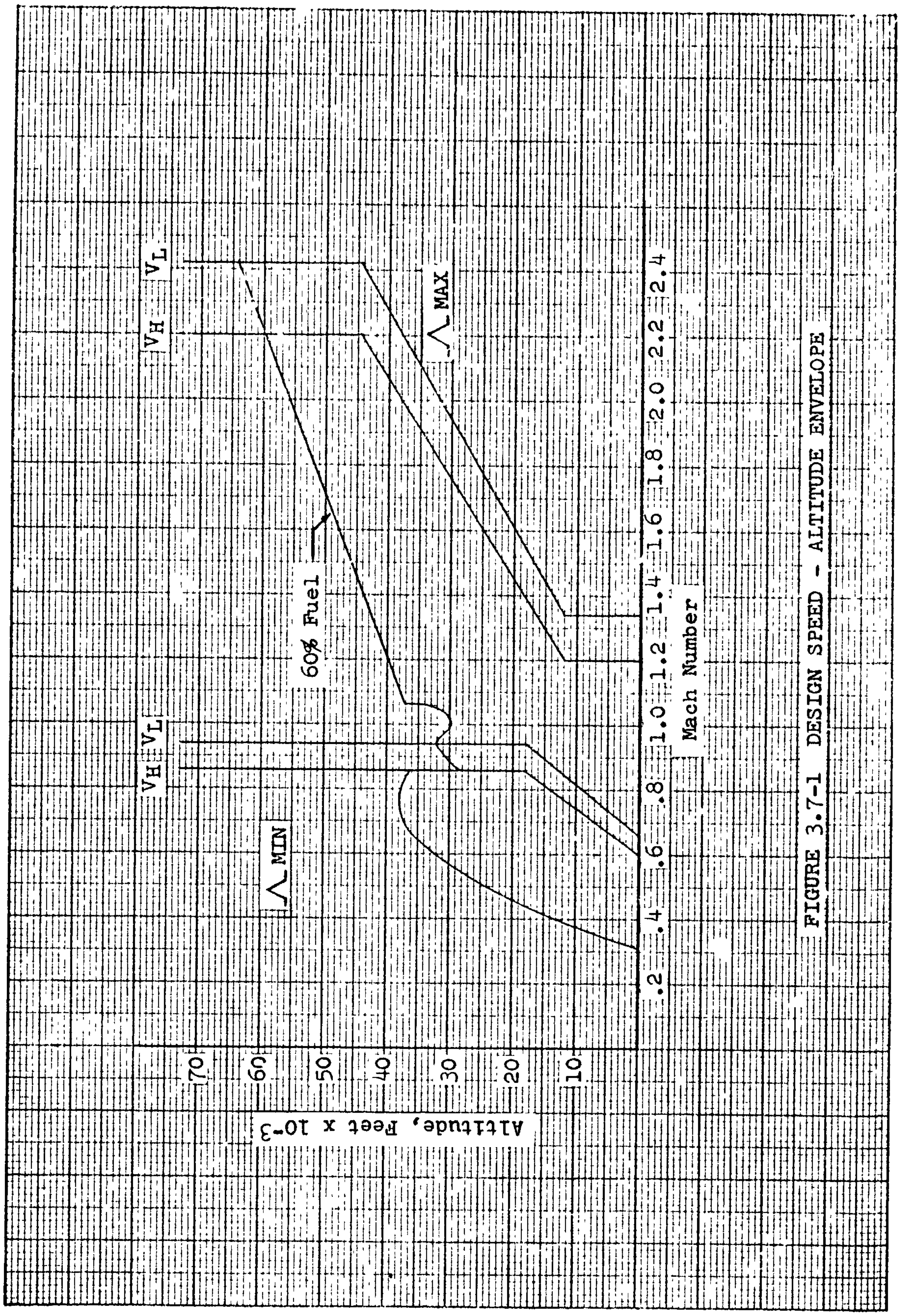


FIGURE 3.7-1 DESIGN SPEED - ALTITUDE ENVELOPE

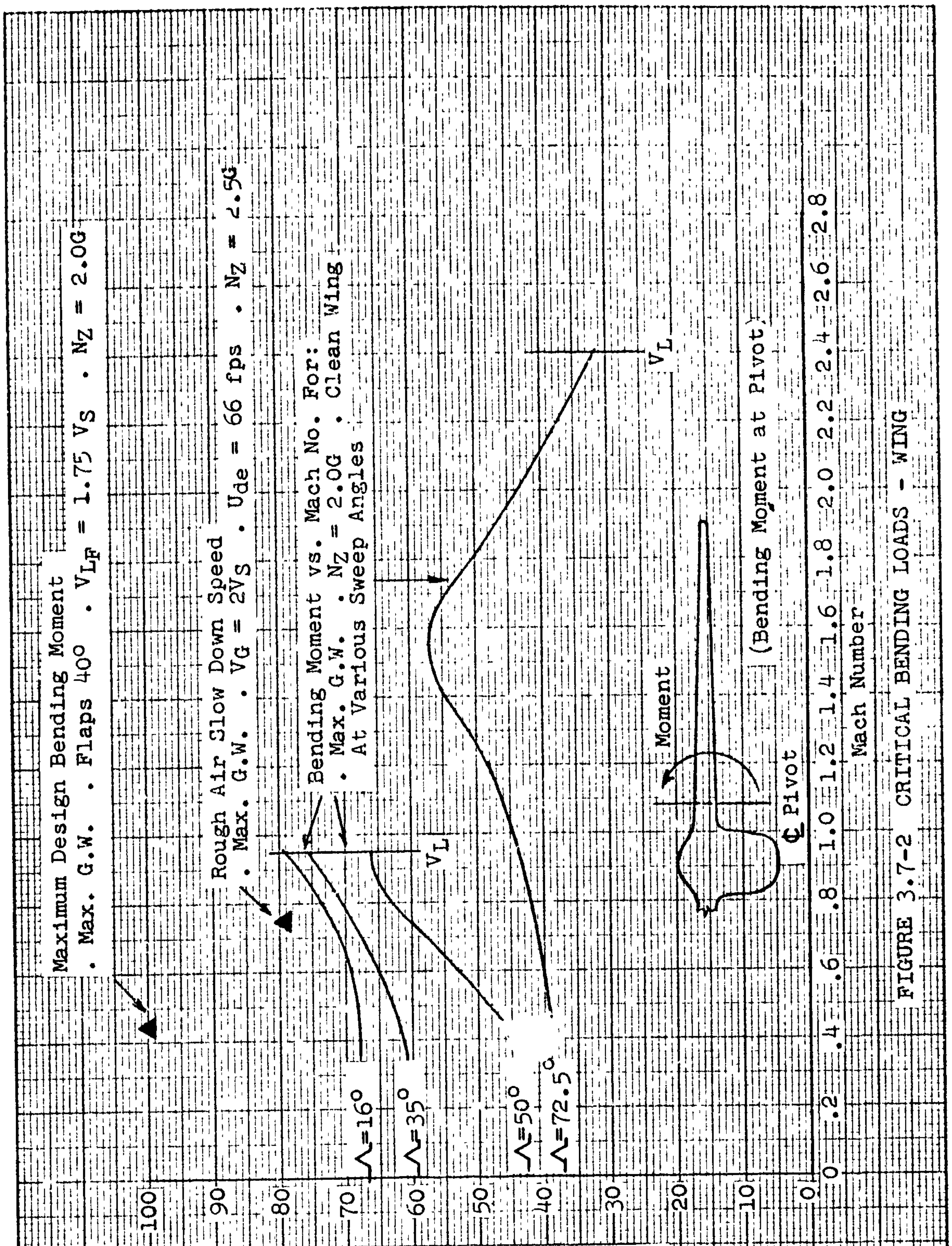


FIGURE 3.7-2 CRITICAL BENDING LOADS - WING

TABLE 3.7-I CRITICAL FLIGHT LOADS CONDITION SUMMARY
CONFIGURATION 2120

Critical Component	Description	G.W.	Λ_W	h	M	V_e	n_Z
Wing Shear, Bending Moment and Torsion Aft Fuselage Shear & Bndg.	Balanced Maneuver 40° Flap @ V_{LF}	395,000	16°	Sea Level	.46	301	2.0
Fwd Fuselage Shear & Bndg.	1 g Balanced Flt + 66 fps Gust @ V_G	395,000	16°	18,000	.74	344	2.50
Fwd Fuselage Shear & Bndg.	1 g Balanced Flt + 50 fps Gust @ V_H	165,115	16°	18,000	.85	397	4.2
Aft Fuselage Reversed Bending	1 g Balanced Flt + 50 fps Gust on Hori. Tail	395,000	72.5°	11,700	1.2	636	1.0
Horizontal Tail Shear & Bndg.	Balanced Maneuver @ $C_{N_{max}}$	395,000	50°	20,000	.63	282	2.0
Vertical Tail Shear & Bndg. Aft Fuselage Torsion & Side Bending	5° Sideslip Zero Rudder @ V_H	-	72.5°	11,700	1.2	636	1.0

the condition selected for the vertical tail loads because the selected arbitrary condition is considered adequate. The magnitude of the load was increased from 133,000 lbs. to 168,000 lbs. due to an increase in vertical tail area. The other principal difference in the two configurations is the lower wing loading of 175 psf of Configuration 2120.

The lower wing loading of 175 psf will affect the fatigue damage accumulation. This is primarily due to the fact that the lower wing loading makes the aircraft more sensitive to gust. The previous studies on Configuration 2010-H indicate that the forward fuselage was the only component of the aircraft in which fatigue controlled the allowable stresses. The studies of Configuration 2120 indicate that the relationship of static allowables to fatigue allowables that was found on Configuration 2010-H still exists. The basic 1 g allowables for the static design conditions and for fatigue are given below for aluminum alloy 2024, (Configuration 2120).

	Wing	Forward Fuselage	Aft Fuselage
Static 1 g stress @ basic TOGW	14,030	13,030	12,650
Fatigue tension 1 g allowable @ basic TOGW	19,400	12,800	13,700

It should be noted that a conservative stress concentration factor of four (KT = 4.0) and a scatter factor of four were used in determining the fatigue allowable and the methods of analysis used were described in Section 5.1.+, Page 5-75 thru 5-121 of FZM-4038-II-2.

3.7.2 Materials

The materials considered for this design are the same as those selected in Report FZM-4038-II-2, Section 5.2, for the supersonic airplane. These materials and the components in which they will be used are as follows:

COMPONENT	MATERIAL
Wing	
Center Section Box	D6ac steel @ 220 KSI heat treat
Pivot Assembly	D6ac steel @ 220 KSI heat treat
Outboard Panels	
Skins	2024-T81 aluminum
Bulkheads & Longerons	7079-T851 aluminum
Horizontal Tail	
Skins	2024-T81 aluminum
Spindles & Carry-Through	D6ac steel @ 220 KSI heat treat
Vertical Tail	
Fin	2024-T81 aluminum
Rudder	2024-T81 aluminum
Ventrals	2024-T81 aluminum
Fuselage	
Skins	2024-T81 aluminum
Bulkheads & Frames	7079-T851 aluminum
Substructure	2024-T81 aluminum
Landing Gear	D6ac steel @ 220 KSI heat treat
Nacelles	
Cover Panels	2024-T81 aluminum & Ti 8AL - 1Mo - 1V titanium
Bulkheads & Frames	2024-T81 aluminum
Ducts	2024-T81 aluminum & Ti 8AL - 1Mo - 1V titanium

3.7.3 Structural Weights

Structural weights for all the major airframe components except the horizontal tail and nacelles were determined using the methods presented in FZM-4038-II-2, Section 6.4. The weight of the structure for the horizontal tail was derived on a unit weight basis with allowances for carry-through structure. Nacelle structural weights were calculated using the equations on Page 81 of General Dynamics Report MR-SS-006. Minor modifications to the equations were made to allow for mounting two engines per side rather than one, and weight penalties for staggering the nacelles were included.

The structural weights calculated for the three airplane configurations in the growth study are plotted in Figure 3.7-3. For the gross weight selected for Configuration 2120, the structural weights used in the performance part of the study were obtained from this figure. In addition, the structural weights were calculated for Configuration 2120 and are itemized in Table 3.7-III. For comparison, the structural weights of the major airframe components in Configurations 2010-H and 2120 are listed in Table 3.7-II.

TABLE 3.7-II STRUCTURAL WEIGHT COMPARISONS

	2010-H	2120-Figure 3.7-1	2120 Calc.
Wing	34,627	36,800	36,705
Fuselage	19,410	21,550	21,468
Horizontal Tail	8,563	5,100	5,114
Vertical Tail	2,600	3,120	3,092
Ventrals	662	-	-
Nacelles	6,059	6,130	6,130
Landing Gear	12,830	11,850	11,850
Total Structure	84,751	84,550	84,359
Struct. Weight / Gross Weight	.212	.214	.214

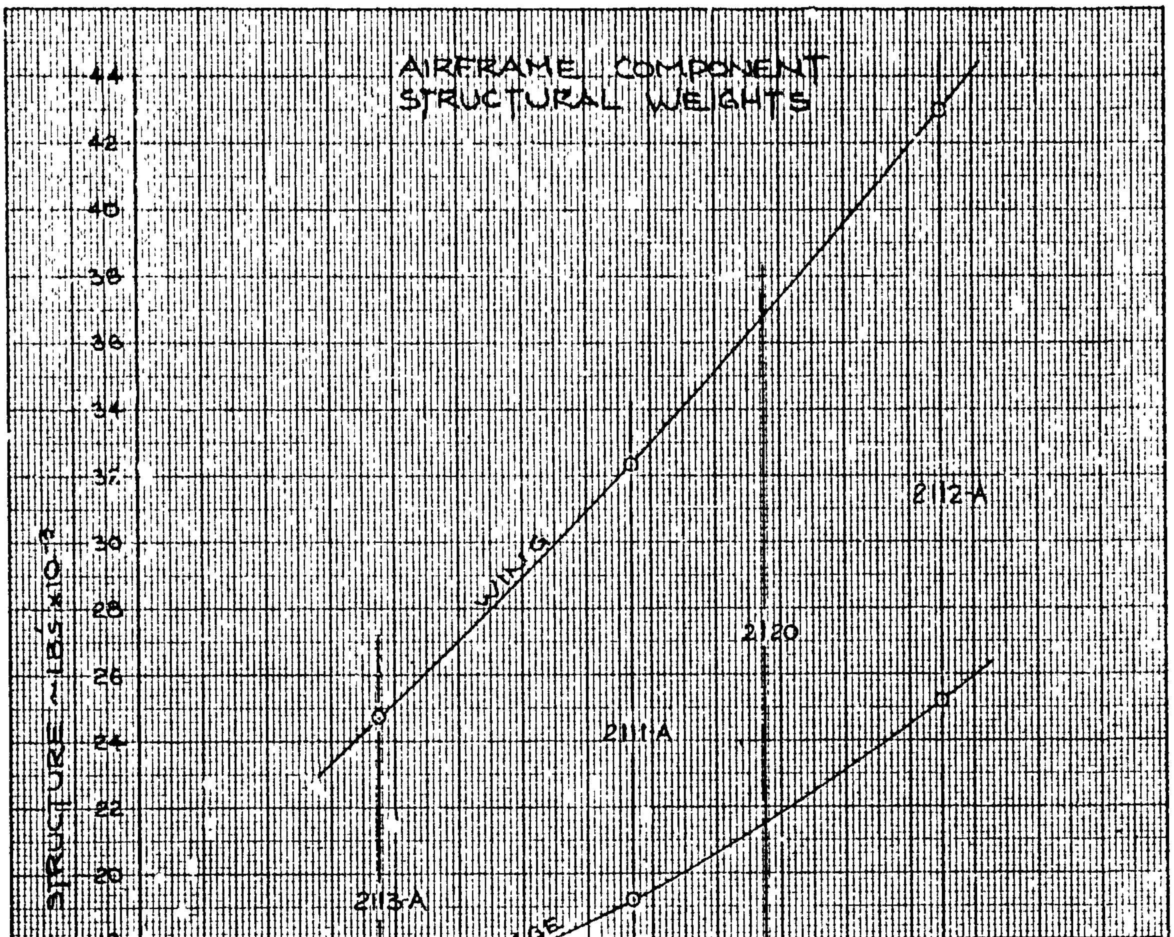

AIRFRAME COMPONENT
STRUCTURAL WEIGHTS
WEIGHT OF STRUCTURE ~ LBS x10^-3
GROSS WEIGHT ~ LBS x10^-3
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
32
34
36
38
40
42
44
200
250
300
350
400
450
WING
FUSELAGE
LANDING GEAR
NACELLES
HOR TAIL
VERT TAIL
2113-A
2111-A
2120
2112-A

FIGURE 3.7-3

TABLE 3.7-III CONFIGURATION 2120 STRUCTURAL WEIGHT SUMMARY

WING	
Structural Box	(26,332)
Basic Box	21,200
Sweep Kick Penalty	728
Pivot Penalty	3,970
Fuel Provisions	434
Secondary Structure	(9,862)
Glove	2,035
Pivot Fairing	1,375
Fixed Leading Edges	428
Fixed Trailing Edges	250
Spoilers	479
Trailing Edge Flaps	3,150
Leading Edge Slats	2,145
Temperature Penalty	(185)
Miscellaneous	(326)
TOTAL	36,705

VERTICAL TAIL	
Basic Box	804
Bending Material	1,042
Sweep Kick Penalty	73
Rudder	615
Rudder Back-up	154
Temperature Penalty	404
TOTAL	3,092

HORIZONTAL TAIL	
Exposed Portion	4,219
Carry-Through & Pivots	612
Blast Penalty	283
TOTAL	5,114

NACELLES	
Cowling	4,908
Pylons	1,222
TOTAL	6,130

FUSELAGE	
Base Weight	(7,886)
Basic Sheel	4,878
Cockpit Provisions	303
N.L.G. Provisions	335
Wing Reaction	1,360
Windshield & Canopy	803
Tail Provisions	270
Flight Loads Material	(5,303)
Fwd Vertical Inertia	324
Aft Vertical Inertia	489
Fwd Side Bending	630
Aft Side Bending	1,335
Fwd Fuel Inertia	563
Aft Fuel Inertia	644
Engine Bending	148
Hori. Tail Bending	1,170
Configuration Penalties	(7,321)
Fuel Provisions	3,540
M.L.G. Cutout & Load	987
M.L.G. Doors	585
Fwd Wpn's Bay Cutout	284
Fwd Wpn's Bay Doors	471
Aft Wpn's Bay Cutout	260
Aft Wpn's Bay Doors	394
Engine Mntg. Provisions	650
Capsule Provisions	150
Blast & Temp. Penalties	(508)
Fatigue	(450)
TOTAL	21,468

LANDING GEAR	
Rolling Stock	(5,374)
Tires	1,200
Wheels	1,734
Brakes	2,440
Structure	(6,476)
TOTAL	11,850

Although there is little difference in the total structural weights for the two configurations, there are significant differences in the structural weights of some of the airframe components. The major difference is in the structural weight of the horizontal tail where 3,500 lbs. of carry-through structure and rings have been eliminated by the change from an aft fuselage buried engine installation in Configuration 2010-H to a Caravelle type engine installation on 2120. The higher fuselage weight for Configuration 2120 is due to minor configuration changes, higher vertical tail design loads, and increased dynamic pressure accompanying the increase in Mach number to 1.2 at sea level. The change in wing weight resulted from the increased dynamic loading and the increased wing area due to the change in wing loading.

3.7.4 Panel Heating from Nuclear Weapon Detonation

A parametric study was made of the surface temperature rise of selected areas of the AMPSS vehicle to determine the effects of both 1 KT and 50 MT weapon detonations on the minimum allowable thickness of both fiberglass radomes and aluminum skin panels. This thermal analysis was based on a total energy input to the vehicle surface areas of 20 cal/cm^2. This total energy was considered to be distributed over time periods representative of the thermal pulse of weapon detonations of both 1 KT and 50 MT. To facilitate this analysis, it was assumed that the energy input could be represented by a square pulse of a time duration equal to three times the time from detonation to the peak energy flux of the actual energy flux distribution curve.

This study was made only for the high altitude, Mach 2.2 flight condition since it represents the worst heating condition. This is due to the lower convective thermal dissipation characteristics at high altitude. Due to the fact that the total energy from a 1 KT weapon detonation was considered to be distributed over a time period of 1.0 sec., whereas the 50 MT weapon detonation total energy was considered to be distributed over 21.5 seconds, the heat flux, and thus the surface temperature rise, of the 1 KT weapon detonation represents the most severe heating condition.

The temperature rises to be expected in the aluminum skin panels, for different surface emittances, for the 1 KT weapon detonation are shown in Figure 3.7-4. For an aluminum skin panel maximum allowable short time temperature of 450°F, the minimum skin thickness, for a surface emittance of 0.3, is approximately 0.032 inches. Since low emittance coatings cannot generally be maintained, for appreciable periods of time, below 0.3, this panel thickness represents the minimum thickness which should be used for panels exposed to the specified total energy input from a 1 KT weapon detonation.

Analysis of the heating of radomes by 1 KT weapon detonation indicated that, even with the use of low emittance coatings, the fiberglass radomes will experience charring and ablation. A more detailed analysis is required to determine the depth of the char layer. This was not performed, since any appreciable radome charring is considered unacceptable.

Figures 3.7-5 and 3.7-6 indicate the maximum surface temperature for the 50 MT detonation expected for aluminum skin panels and radomes respectively. The results of the aluminum skin panels evaluation, Figure 3.7-5, indicate that low emittance coatings will be required. Also, even with the use of coatings, the minimum allowable panel thickness will be 0.0275 inches.

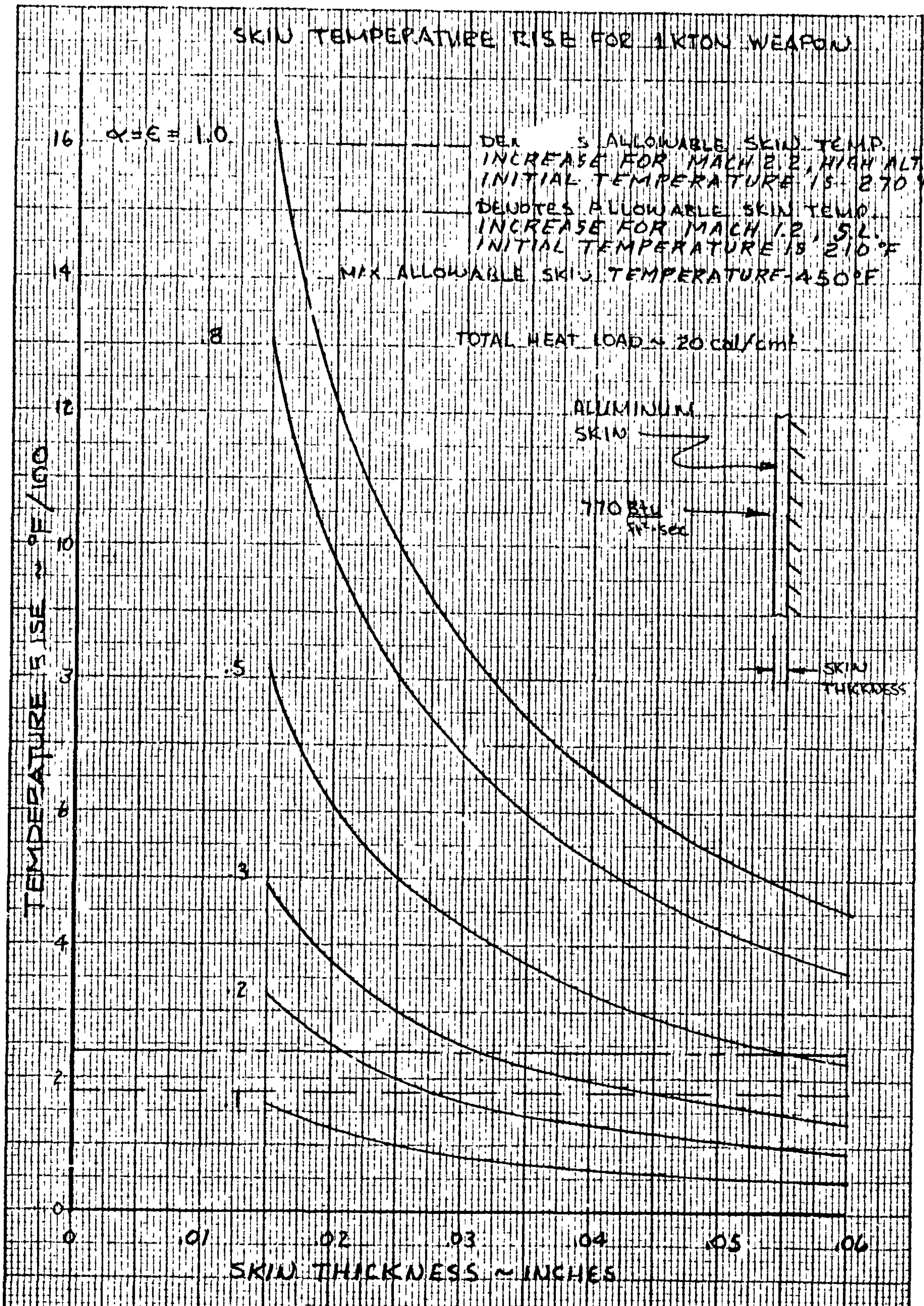

SKIN TEMPERATURE RISE FOR 1KTON WEAPON
α = ε = 1.0
DE S ALLOWABLE SKIN TEMP. INCREASE FOR MACH 2.2, HIGH ALT. INITIAL TEMPERATURE IS 270 °F.
DENOTES ALLOWABLE SKIN TEMP. INCREASE FOR MACH 1.2, S.L. INITIAL TEMPERATURE IS 210 °F
MAX ALLOWABLE SKIN TEMPERATURE 450 °F
TOTAL HEAT LOAD ~ 20 cal/cm²
ALUMINUM SKIN
770 Btu/ft²·sec
SKIN THICKNESS
1.0
.8
.5
.3
.2
.1
16
14
12
10
8
6
4
2
0
.01
.02
.03
.04
.05
.06
TEMPERATURE RISE ~ °F/100
SKIN THICKNESS ~ INCHES

FIGURE 3.7-4

SECRET

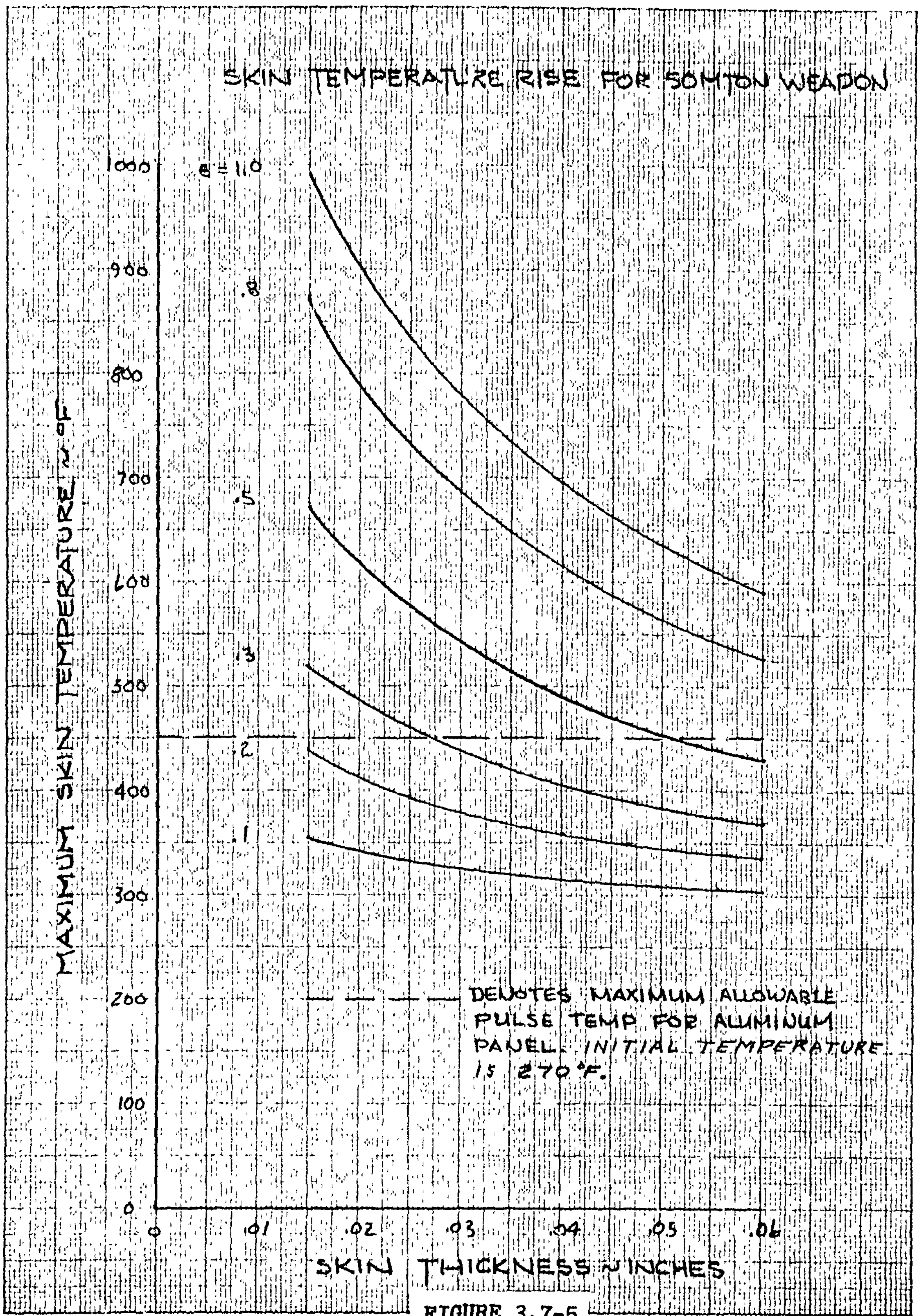


FIGURE 3.7-5

SECRET

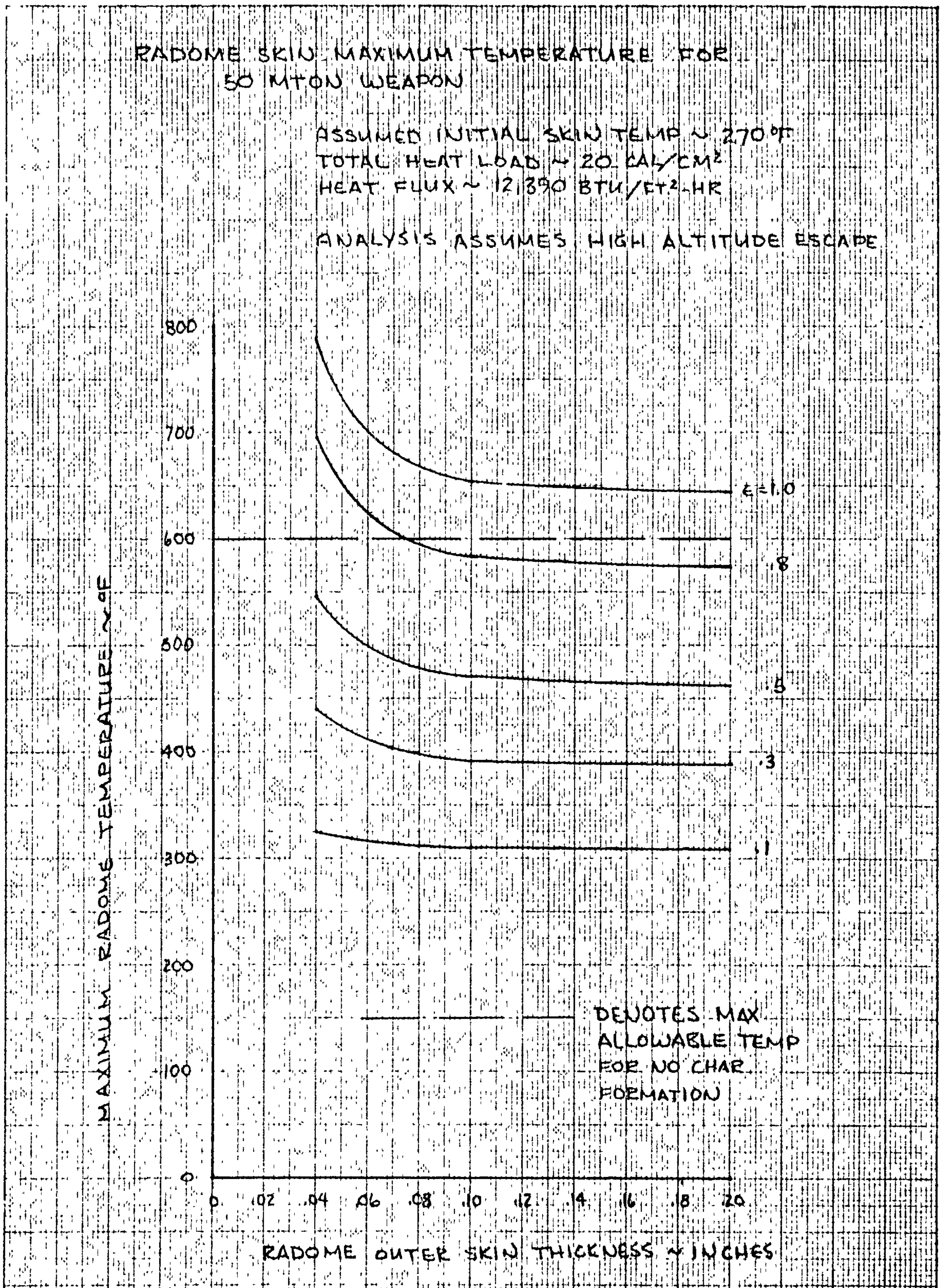

RADOME SKIN MAXIMUM TEMPERATURE FOR 50 MTON WEAPON
ASSUMED INITIAL SKIN TEMP ~ 270°F
TOTAL HEAT LOAD ~ 20 CAL/CM²
HEAT FLUX ~ 12,390 BTU/FT²-HR
ANALYSIS ASSUMES HIGH ALTITUDE ESCAPE
MAXIMUM RADOME TEMPERATURE ~ °F
800
700
600
500
400
300
200
100
0
ε=1.0
.8
.5
.3
.1
DENOTES MAX ALLOWABLE TEMP FOR NO CHAR FORMATION
0
.02
.04
.06
.08
.10
.12
.14
.16
.18
.20
RADOME OUTER SKIN THICKNESS ~ INCHES

FIGURE 3.7-6

The analysis of the radome heating, as shown in Figure 3.7-6, indicates that radome panel thicknesses down to .075 inches can be used for surface emittances of 0.8 without sustaining damage. However, if radome panels are of thin skin fiberglass sandwich honeycomb construction, lower surface emittances will be required if charring is to be avoided. This evaluation is based on the criteria of char formation beginning at 600°F.

The only airframe components which need to be considered for panel heating are the fuselage sides, fuselage lower surface, and portions of the wing and tail secondary structure. In the case of the wing and tail, the flaps, slats, and structural box will have skin gages heavier than .032, and the remainder will have gages below .032 in only a few small areas.

Previous analysis of the fuselage structure showed that a skin thickness of .038 to .040, or greater, was required for primary bending and shear stresses except in a few areas, where the skin thickness was .030. When provisions for fatigue, engine blast, shop handling, etc., are included the gages will increase to a minimum of approximately .045. Of this total gage, 60 to 75% will be on the outer surface of the sandwich panels and the remainder on the inner surface. Therefore, only a small area of the fuselage is expected to have outer surface gages less than .032. The weight penalty associated with the panel heating problem is estimated to be less than 100 pounds.

3.8 WEIGHTS

3.8.1 Introduction

Configuration 2120 is a 395,000 pound gross weight aircraft with a basic design mission payload of 10,000 pounds. It is powered by four Pratt & Whitney STF200C-35.1, .426 scale turbofan engines. All fuel is carried internally and is controlled by an automatic fuel management system.

3.8.2 Weight and Balance

Systems and equipment weights have been derived using the methods described in FZM-4038-II-2, Section 6.4.2. The engine weights are based on data furnished by the manufacturer.

The center of gravity envelope is presented in Figure 3.8-1. This plot of center of gravity versus weight shows the extreme conditions achievable with fuel loadings. With the automatic fuel management system, the aircraft will remain within limits at all times. The wing leading edge sweep for take-off is 16°. For normal landing the wing leading edge sweep is also 16°. However, for the extreme landing condition with zero fuel, the wing is swept to 26°.

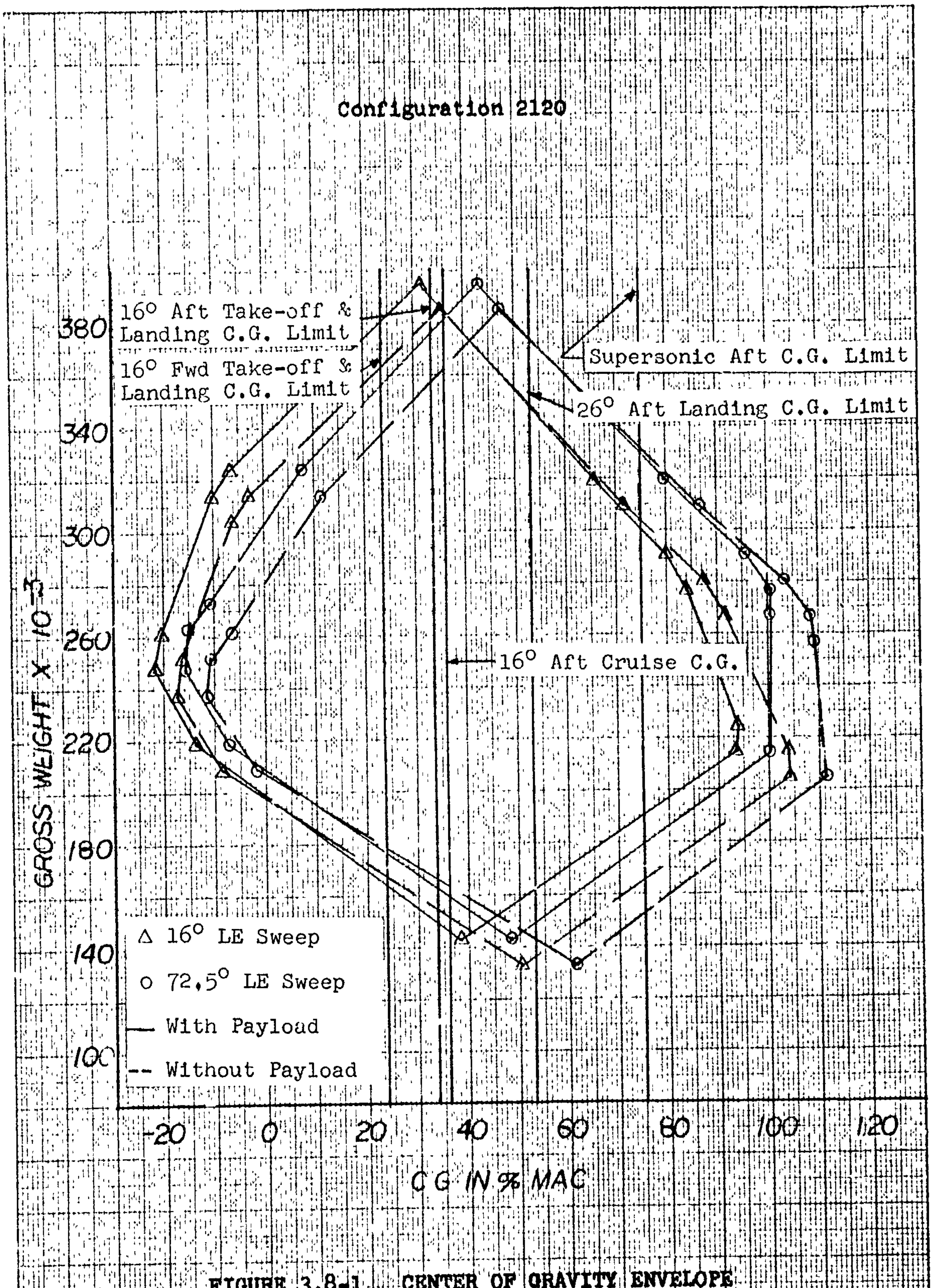


FIGURE 3.8-1 CENTER OF GRAVITY ENVELOPE

SECRET

NAME ________
DATE ________

PAGE ________
MODEL ________
REPORT ________

3.8.3 GROUP WEIGHT STATEMENT

ESTIMATED - [redacted]

(Cross out those not applicable)

Configuration 2120

CONTRACT NO. ________
AIRPLANE, GOVERNMENT NO. ________
AIRPLANE, CONTRACTOR NO. ________
MANUFACTURED BY ________

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY	P&W	
	MODEL	STF200C-35.1	.426 scale
	NO.		
PROPELLER	MANUFACTURED BY		
	DESIGN NO.		
	NO.		

AN-9103-D **SECRET**

NAME

DATE

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE
MODEL
REPORT

1	WING GROUP		30,800
2	CENTER SECTION - BASIC STRUCTURE		
3	INTERMEDIATE PANEL - BASIC STRUCTURE		
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS LBS.)		
5			
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)		
7	AILERONS (INCL. BALANCE WEIGHT LBS.)		
8	FLAPS - TRAILING EDGE		
9	- LEADING EDGE		
10	SLATS		
11	SPOILERS		
12	SPEED BRAKES		
13			
14			
15	TAIL GROUP		8,220
16	STABILIZER - BASIC STRUCTURE	5,100	
17	FINS - BASIC STRUCTURE (INCL. [redacted]Rudder[redacted])	3,120	
18	SECONDARY STRUCTURE (STAB. & FINS)		
19	ELEVATOR (INCL. BALANCE WEIGHT LBS.)		
20	RUDDERS (INCL. BALANCE WEIGHT LBS.)		
21			
22			
23	BODY GROUP		21,[illegible]0
24	FUSELAGE OR HULL - BASIC STRUCTURE		
25	BOOMS - BASIC STRUCTURE		
26	SECONDARY STRUCTURE - FUSELAGE OR HULL		
27	- BOOMS		
28	- SPEEDBRAKES		
29	- DOORS, PANELS & MISC.		
30			
31	ALIGHTING GEAR GROUP - LAND (TYPE:)		13,200

	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS	
34	Main	4,370	6,290	1,080	11,740
35	Nose	330	860	270	1,460
36					
37					
38					
39					

40 ALIGHTING GEAR GROUP - WATER

	LOCATION	FLOATS	STRUTS	CONTROLS
42				
43				
44				
45				

46	SURFACE CONTROLS GROUP		3,760
47	COCKPIT CONTROLS	80	
48	AUTOMATIC PILOT	170	
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS LBS.)	2,440	
50	Variable Sweep Mechanism	1,070	
51	ENGINE SECTION OR NACELLE GROUP		6,130
52	INBOARD		
53	CENTER		
54	OUTBOARD		
55	DOORS, PANELS & MISC.		
56			
57	TOTAL (TO BE BROUGHT FORWARD)		89,660

AN-9103-D

SECRET

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

NAME__________ PAGE__________
DATE__________ MODEL__________
REPORT__________

		Auxiliary	Main	
1	PROPULSION GROUP			21,060
2		AUXILIARY	MAIN	
3	ENGINE INSTALLATION		16,100	
4	AFTERBURNERS (IF FURN. SEPARATELY)			
5	ACCESSORY GEAR BOXES & DRIVES			
6	SUPERCHARGERS (FOR TURBO TYPES)			
7	AIR INDUCTION SYSTEM		630	
8	EXHAUST SYSTEM			
9	COOLING SYSTEM			
10	LUBRICATING SYSTEM			
11	TANKS			
12	COOLING INSTALLATION			
13	DUCTS, PLUMBING, ETC.			
14	FUEL SYSTEM		3,670	
15	TANKS - PROTECTED			
16	- UNPROTECTED			
17	PLUMBING, ETC.			
18	WATER INJECTION SYSTEM			
19	ENGINE CONTROLS		380	
20	STARTING SYSTEM		280	
21	PROPELLER INSTALLATION			
22				
23				
24	AUXILIARY POWER PLANT GROUP			
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP			770
26	HYDRAULIC & PNEUMATIC GROUP			2,180
27				
28				
29	ELECTRICAL GROUP			2,600
30				
31				
32	ELECTRONICS GROUP			7,400
33	EQUIPMENT		5,888	
34	INSTALLATION		1,512	
35				
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)			100
37	FURNISHINGS & EQUIPMENT GROUP			2,840
38	ACCOMMODATIONS FOR PERSONNEL		630	
39	MISCELLANEOUS EQUIPMENT		305	
40	FURNISHINGS		475	
41	EMERGENCY EQUIPMENT		1,430	
42				
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP			1,650
44	AIR CONDITIONING			
45	ANTI-ICING			
46				
47	PHOTOGRAPHIC GROUP			
48	AUXILIARY GEAR GROUP			400
49	HANDLING GEAR			
50	ARRESTING GEAR			
51	CATAPULTING GEAR			
52	ATO GEAR			
53				
54				
55	MANUFACTURING VARIATION			
56	TOTAL FROM PG. 2			89,660
57	WEIGHT EMPTY			128,660

AN-9103-D

NAME ______

DATE ______

SECRET

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

PAGE ______

MODEL ______

REPORT ______

					Design Mission	Alternate Mission	Inflight Refuel
1, 2	LOAD CONDITION				Design Mission	Alternate Mission	Inflight Refuel
3	CREW (NO. 4)				860	860	860
4	PASSENGERS (NO.)						
5	FUEL	Type	Gals.				
6	UNUSABLE				1,290	1,290	1,290
7	INTERNAL	JP-4 @ 6.5	37,015		240,595	240,595	240,595
8							
9							
10	EXTERNAL						
11							
12	BOMB BAY	JP-4 @ 6.5	1,540		10,000		18,000
13							
14	OIL						
15	TRAPPED						
16	ENGINE				450	450	450
17							
18	FUEL TANKS (LOCATION Aft Weapon Bay)				1,680		1,680
19	WATER INJECTION FLUID (GALS)						
20							
21	BAGGAGE						
22	CARGO						
23							
24	ARMAMENT						
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.			
26							
27							
28							
29							
30							
31							
32	AMMUNITION						
33							
34							
35							
36							
37							
38							
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)						
*40	BOMB OR TORPEDO RACKS						
41	Payload Racks				1,200	2,400	1,200
42	Payload				10,000	20,000	10,000
43							
44							
45							
46	EQUIPMENT						
47	PYROTECHNICS						
48	PHOTOGRAPHIC						
49							
*50	OXYGEN				125	125	125
51	Food and Water				140	140	140
52	MISCELLANEOUS						
53							
54							
55	USEFUL LOAD				266,340	265,860	274,340
56	WEIGHT EMPTY				128,660	128,660	128,660
57	GROSS WEIGHT				395,000	394,520	403,000

*If not specified as weight empty.

SECRET

AI-9103-D **SECRET** **GROUP WEIGHT STATEMENT** PAGE ______

NAME ______ **DIMENSIONAL & STRUCTURAL DATA** MODEL ______

DATE ______ REPORT ______

1	LENGTH - OVERALL (FT.) 149.25				HEIGHT - OVERALL - STATIC (FT.)	33.33		
2		Main Floats	Aux. Floats	Booms	Fuse or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
3	LENGTH - MAX. (FT.)				148.58			
4	DEPTH - MAX. (FT.)				11.40			
5	WIDTH - MAX. (FT.)				8.92			
6	WETTED AREA (SQ. FT.)							
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							
8	FUSELAGE VOLUME (CU. FT.)		PRESSURIZED			TOTAL		

		Wing	H. Tail	V. Tail
9		Wing	H. Tail	V. Tail
10	GROSS AREA (SQ. FT.) 16° Wing Sweep	2257	565	421
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)			
12	SPAN (FT.) 16° Wing Sweep	149.8	40.3	17.7
13	FOLDED SPAN (FT.)			
14				
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES) 16° Wing Sweep	12° 15'	40°	48° 35'
16	- AT % CHORD LINE (DEGREES)			
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES)	305.64	334	408
18	- MAX. THICKNESS (INCHES)	41.9	16.7	16.3
***19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)			
20	- MAX. THICKNESS (INCHES)			
***21	THEORETICAL TIP CHORD - LENGTH (INCHES)	76.41	65	165
22	- MAX. THICKNESS (INCHES)	6.87	3.25	6.60
23	DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)			
24	TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.) 57.5			

25	AREAS (SQ. FT.) Flaps	L.E.	T.E. 502	
26	Lateral Controls	Slots 218	Spoilers 298	Ailerons
27	Speed Brakes	Wing	Fuse. or Hull	
28				
29				

30	ALIGHTING GEAR (LOCATION)	Main	Nose		
31	LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)	140.0	87.4		
32	OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)	13	13		
33	FLOAT OR SKI STRUT LENGTH (INCHES)				
34	ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)				
35	HYDRAULIC SYSTEM CAPACITY (GALS.)				

		Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
36	FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
37	Fuel - Internal	Wing			2	8,108
38		Fuse. or Hull			7	28,907
39	- External					
40	- Bomb Bay - Aft				1	1,540
41						
42	Oil					
43						
44						

		Fuel in Wings (Lbs.)	Stress Gross Weight	Ult. L.F.
45	STRUCTURAL DATA - CONDITION	Fuel in Wings (Lbs.)	Stress Gross Weight	Ult. L.F.
46	FLIGHT	52,700	395,000/403,000	3.0
47	LANDING			
48				
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL			
50	CATAPULTING			
51	MIN. FLYING WEIGHT		145,255	
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)			
53	WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)			
54	STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)			
55	PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)			
56				
57	AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)	98,780		

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

***Parallel to ℄ airplane.
****Total usable capacity.

SECRET

3.9 COST AND SCHEDULE SUMMARY

3.9.1 Costs

The cost data shown in Figure 3.9-1 for Configuration 2120 (G.W. = 395,000 lbs.) is the same as that shown for Configuration 2010-H in General Dynamics/Fort Worth report FZM-4039-I and III, dated February 1964.

Since the fabrication techniques, materials and AMPR weights are almost identical it was impossible to show any differences in the cost figures.

The cost summary reflects fiscal year procurement including RDT&E, production costs, together with engines and engine spares, support, AGE, subsystems, training and MTU's, publications, handbooks and spares. Weapons costs are not included.

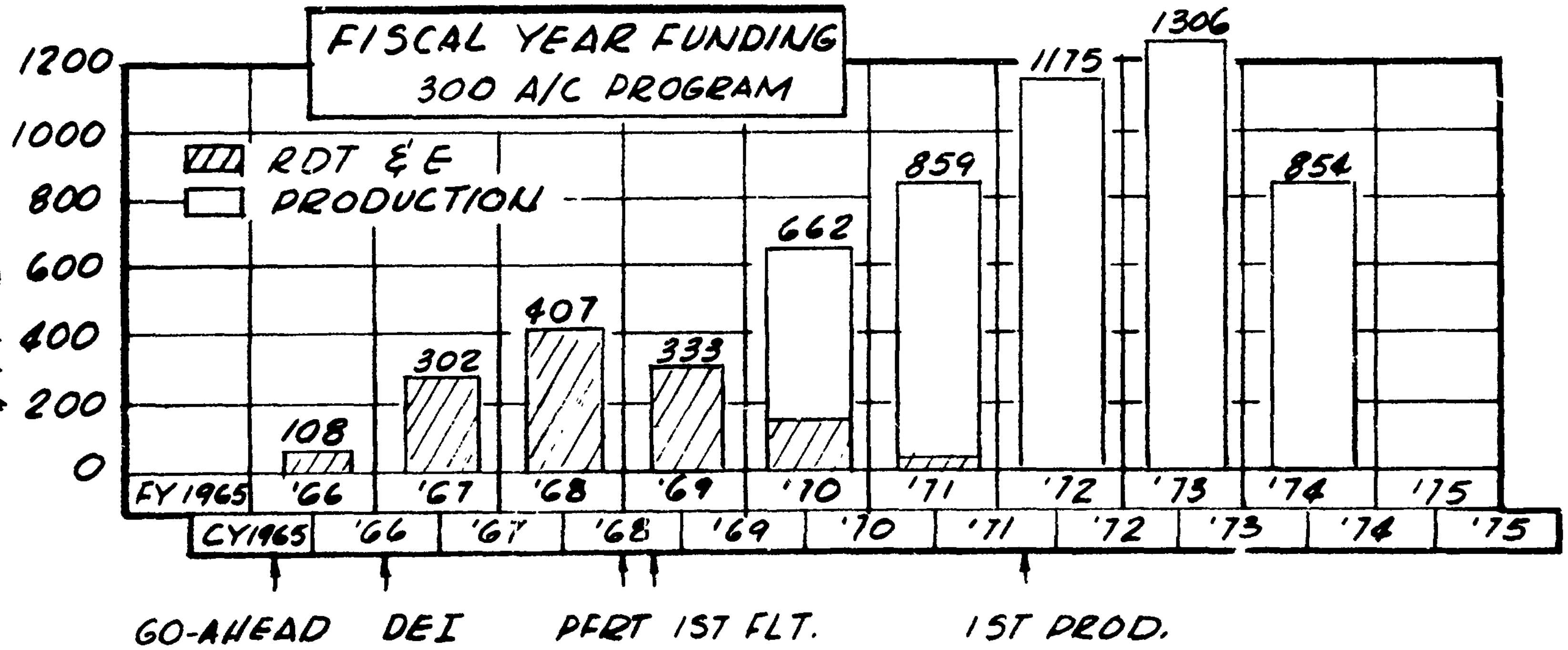
3.9.2 Schedules

The schedules remain unchanged from those shown in FZM-4038-I and III. Since the engines remain the pacing items, the first flight would occur in late 1968 with the first production aircraft available in late 1971.

COST - SUPERSONIC MODEL 2120

FIGURE 3.9-1

	RDT & E	PRODUCTION 100 A/C	PRODUCTION 300 A/C
AIRFRAME	640	893	1932
ENGINES	276	180	474
SUBSYSTEMS	294	427	1152
OTHER (SPARES, AGE, MTU, ETC.)	141	436	1097
TOTAL ($ MILLIONS)	1351	1936	4655



4.0 PARAMETRIC STUDIES

This section presents the parametric study results for improved heavy weight Type "B" aircraft designs. The effect on performance of the various trades are presented for symmetrical non-refueled missions. Performance evaluation for symmetrical refueled missions are not presented since earlier aircraft sizing studies indicated the symmetrical non-refueled mission requirements were more critical than the refueled mission requirements. All airplanes in this trade section have a sea level capability of 5 minutes operation at Mach 1.2 and continuous operation at Mach .9.

The trades discussed in this section were conducted on two different baseline designs. This resulted in two different gross weights being required to make the design mission. The first baseline is that discussed in Section 3.1; namely, those baseline designs utilized to develope the growth curve. This baseline requires a 385,000 pound gross weight airplane to meet the non-refueled design mission. The second baseline is the point design airplane, Configuration 2120 (Section 3.3). This airplane at a gross weight of 395,000 pounds is 20 n. mi. shy of the performance level of the growth curve designs, Section 3.1. The gross weight required by this design level to meet the non-refueled design mission is 390,000 pounds. Despite the two different baselines, the trade studies discussed herein are valid and use the same growth curve shape.

All trade studies contained herein were conducted using a maximum density airplane,i.e., the effect of dry weight changes also caused takeoff gross weight to change by an equivalent amount.

4.1 PENETRATION MACH NUMBER - VARYING RANGE

Figures 4.1-2 through 4.1-6 present non-refueled symmetrical mission growth curves (gross weight vs total distance) for sea level dash Mach numbers from .5 to .9 and dash distances from 500 n. mi. to 2500 n. mi. Maximum range high altitude cruise occurs at Mach .73 with a wing leading edge sweep of 16°. These curves were generated for a 175 PSF wing loading design with engines sized by the 6000 feet takeoff requirement (T/W_{SLS}=.2318). The curves incorporate a 10,000-lb payload which is dropped at mid-dash. Growth curves for the refueled mission are not presented since earlier aircraft sizing studies indicated the symmetrical non-refueled mission requirements were more critical than the symmetrical refueled mission requirements.

In an effort to determine maximum range during sea level dash, an investigation of leading edge sweep (see Section 3.4.9 for aerodynamic discussion) indicated maximum range occurred at sweep angles less than those defined by crew station ride quality considerations. Therefore, leading edge sweep angles during sea level dash were limited to those shown in Figure 4.1-1. This variation of leading edge sweep as a function of Mach number results in a constant gust sensitivity, $\overline{A}_{\Delta n}$, of 0.02 $\frac{\text{RMS g's}}{\text{RMS Ft/Sec}}$ at the crew station.

Performance analysis of the growth curves consisted of investigating three gross weights, namely, 275,000 lbs, 354,000 lbs and 450,000 lbs. Performance methods used are identical to those previously described in Section 3.3.2 of GD/FW report FZM-4038-II-1 dated 3 February 1964. Dash trade plots (dash distance versus total distance) were constructed for the above three gross weights at sea level dash Mach numbers from .5 to .9. The growth curves of this section were then generated from these dash trade plots.

Dash trade plots for any given gross weight can be reconstructed by plotting dash distance versus total distance from the growth curves. Zero zone and maximum zone can be extrapolated, with maximum zone occurring at the total distance equal to dash distance point.

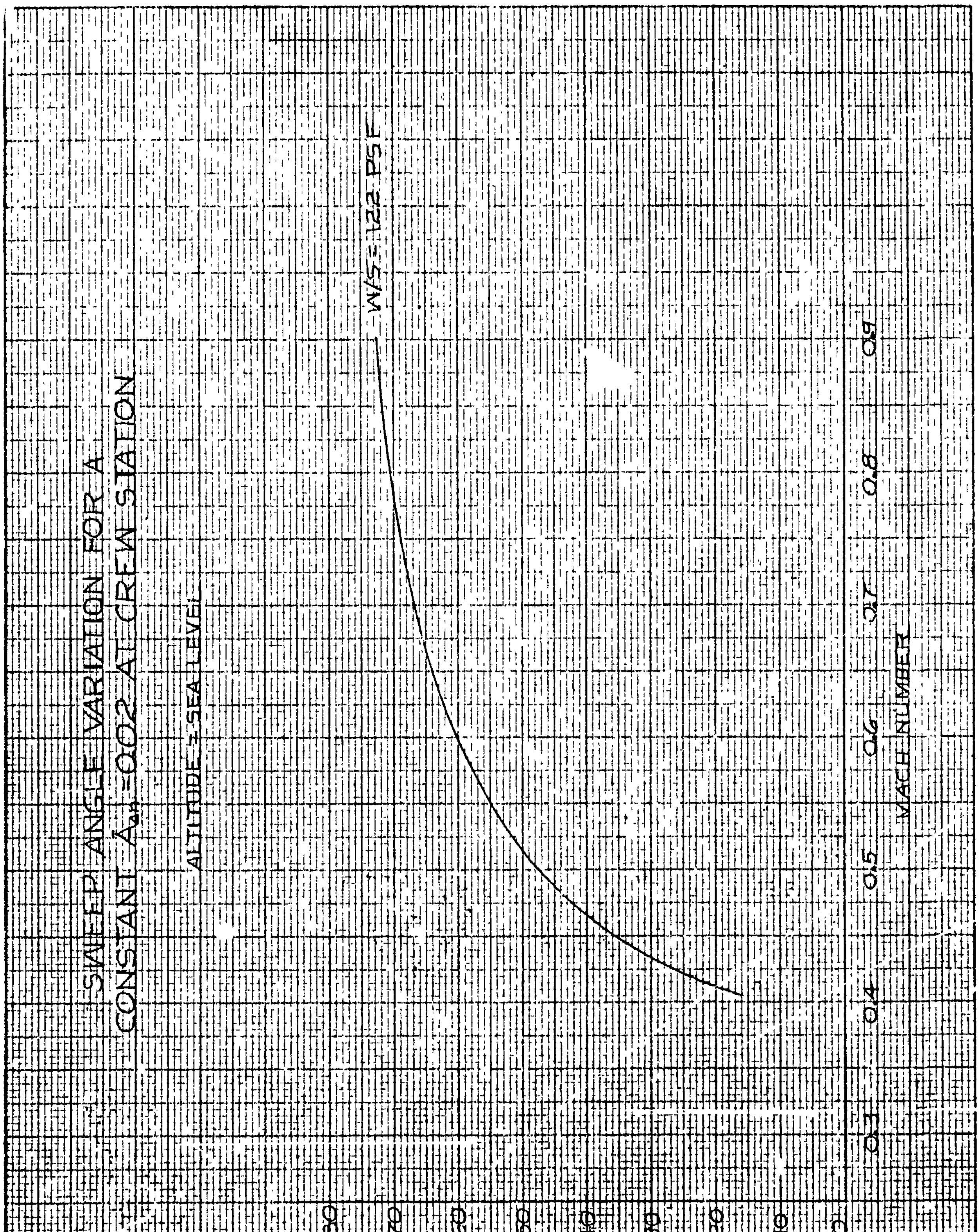

SWEEP ANGLE VARIATION FOR A
CONSTANT Δn = .002 AT CREW STATION
ALTITUDE = SEA LEVEL
W/S = 122 PSF
0.3
0.4
0.5
0.6
0.7
0.8
0.9
MACH NUMBER
0
10
20
30
40
50
60
70
80
Λ LE ~ DEGREES

FIGURE 4.1-1

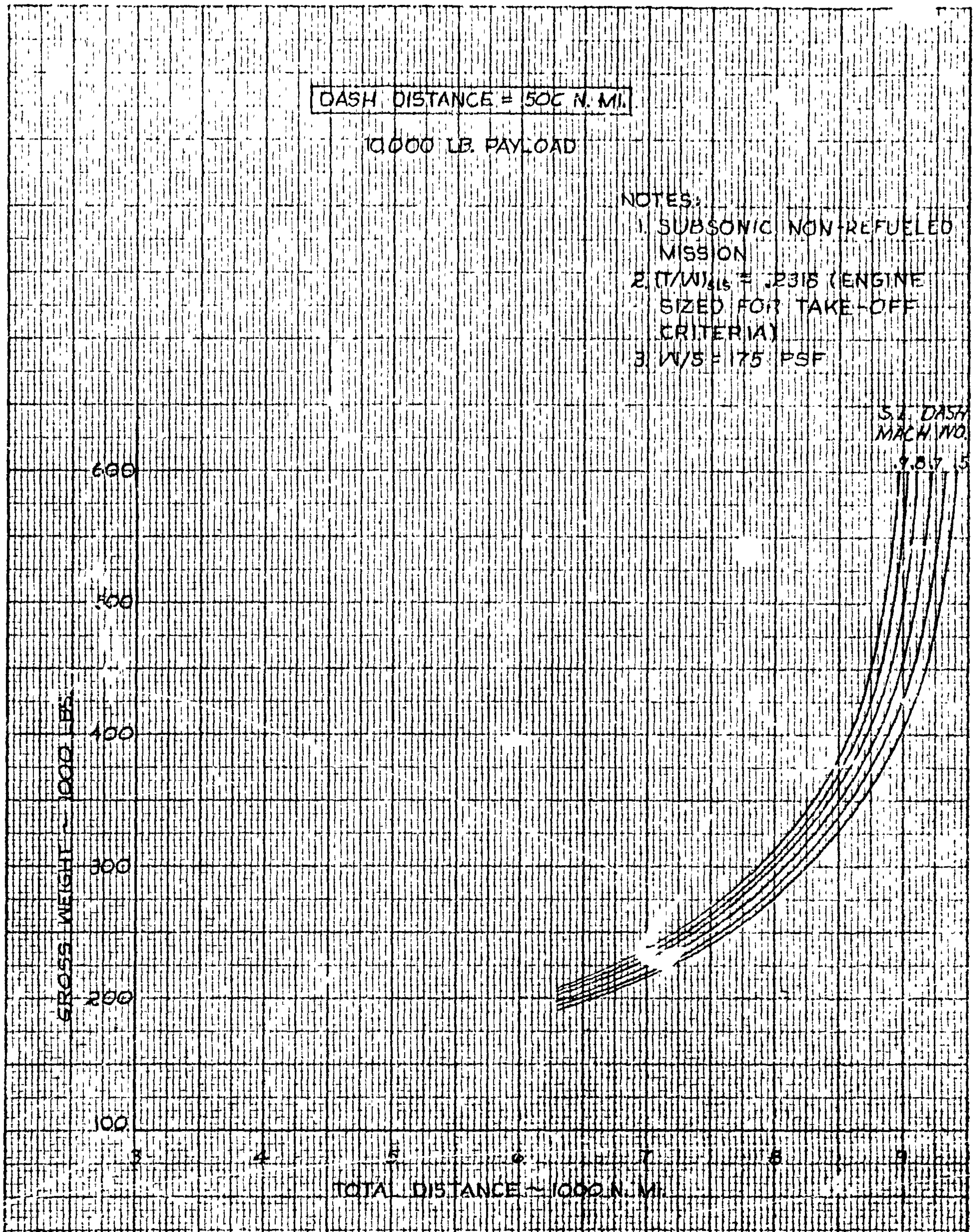


FIGURE 4.1-2 GROSS WEIGHT EFFECT ON RANGE

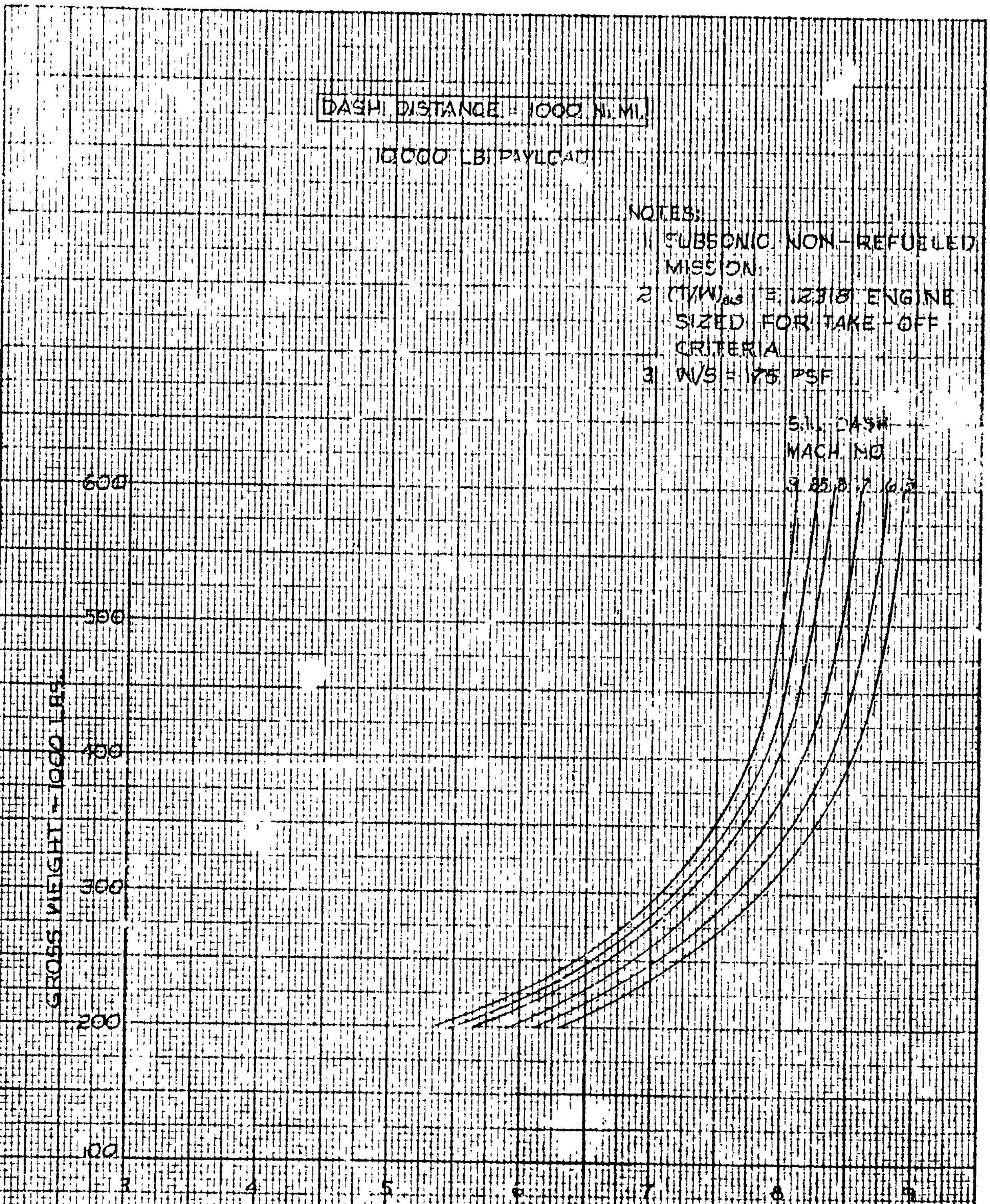


FIGURE 4.1-3 GROSS WEIGHT EFFECT ON RANGE

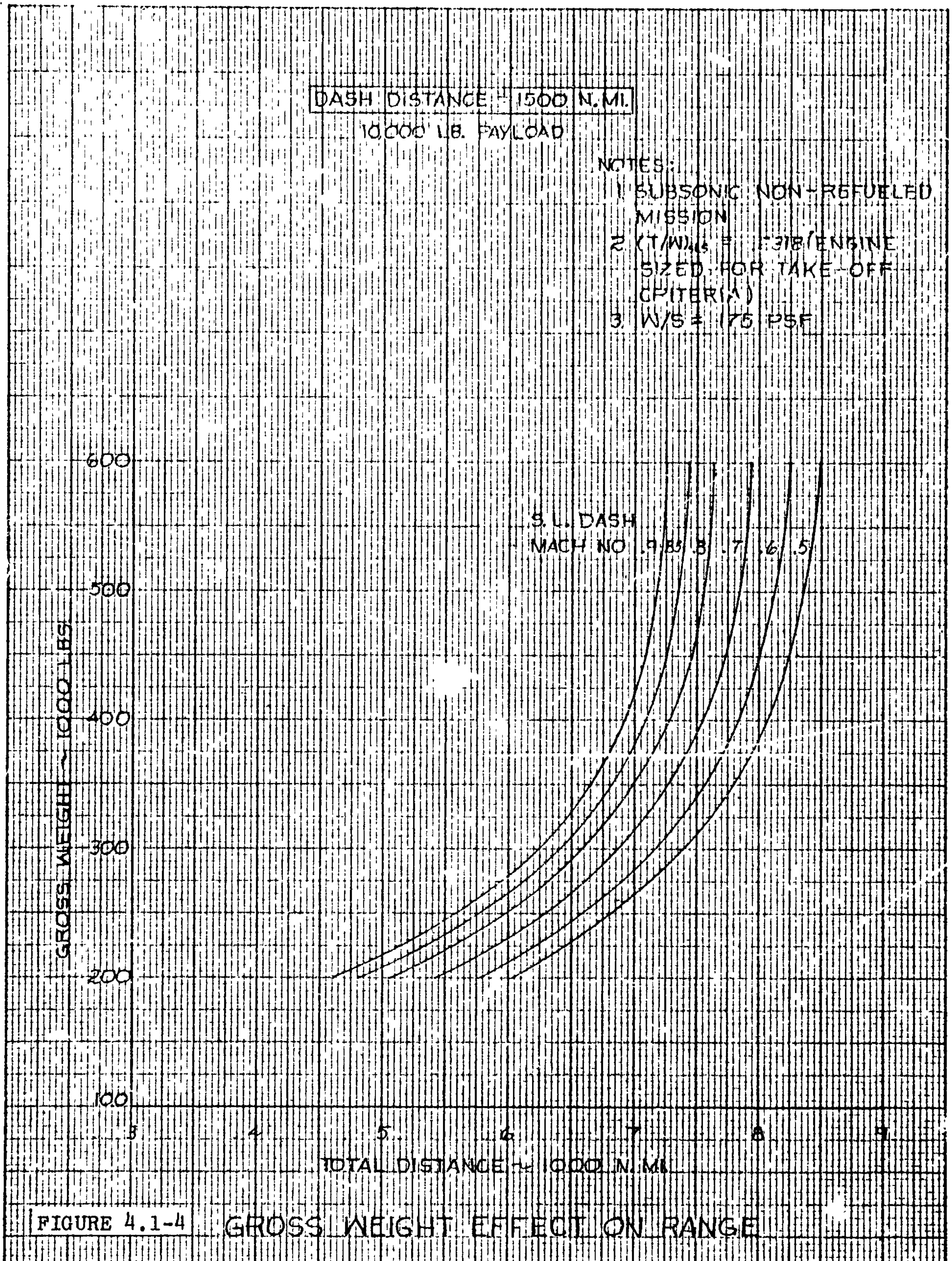


FIGURE 4.1-4 GROSS WEIGHT EFFECT ON RANGE

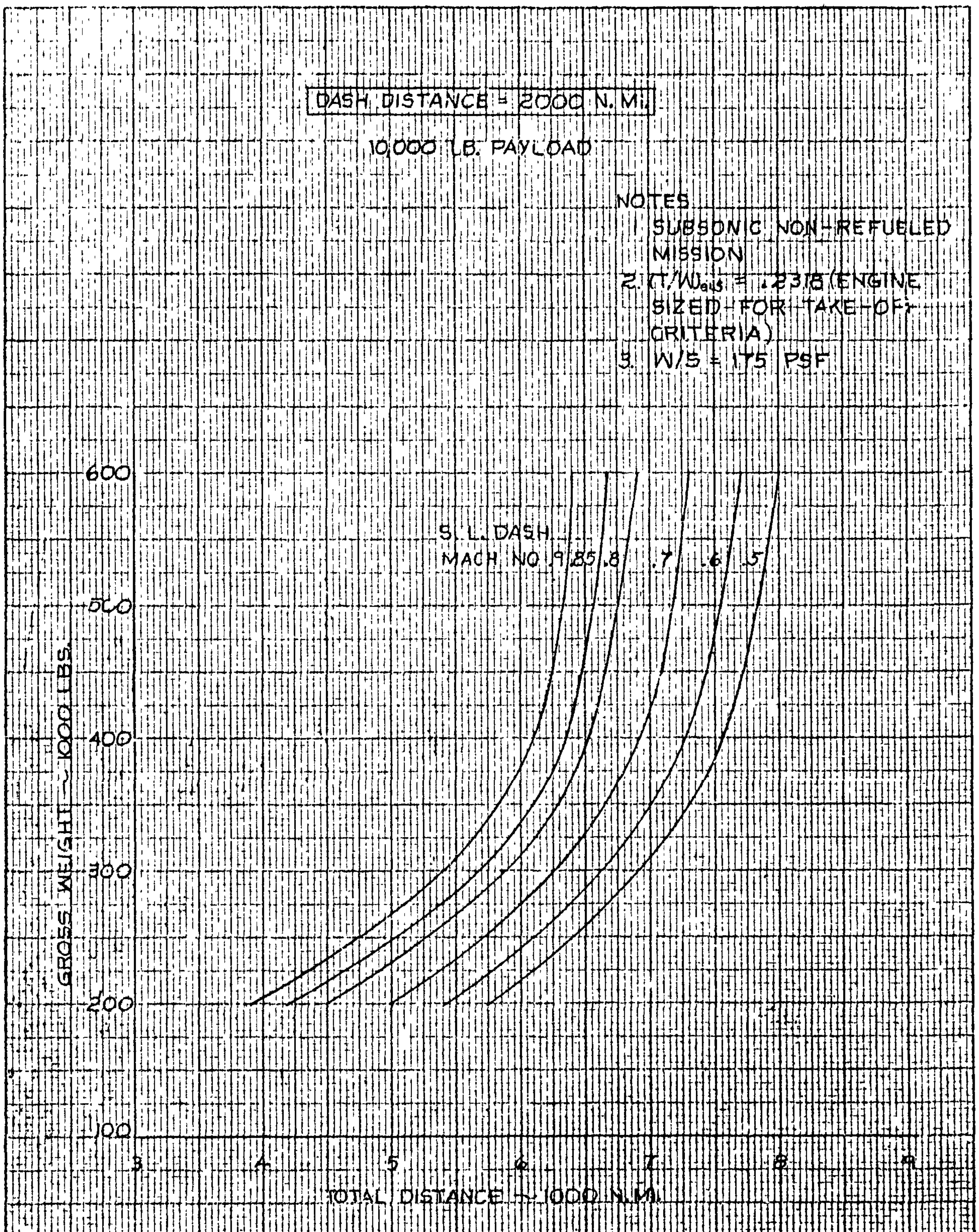


FIGURE 4.1-5 GROSS WEIGHT EFFECT ON RANGE

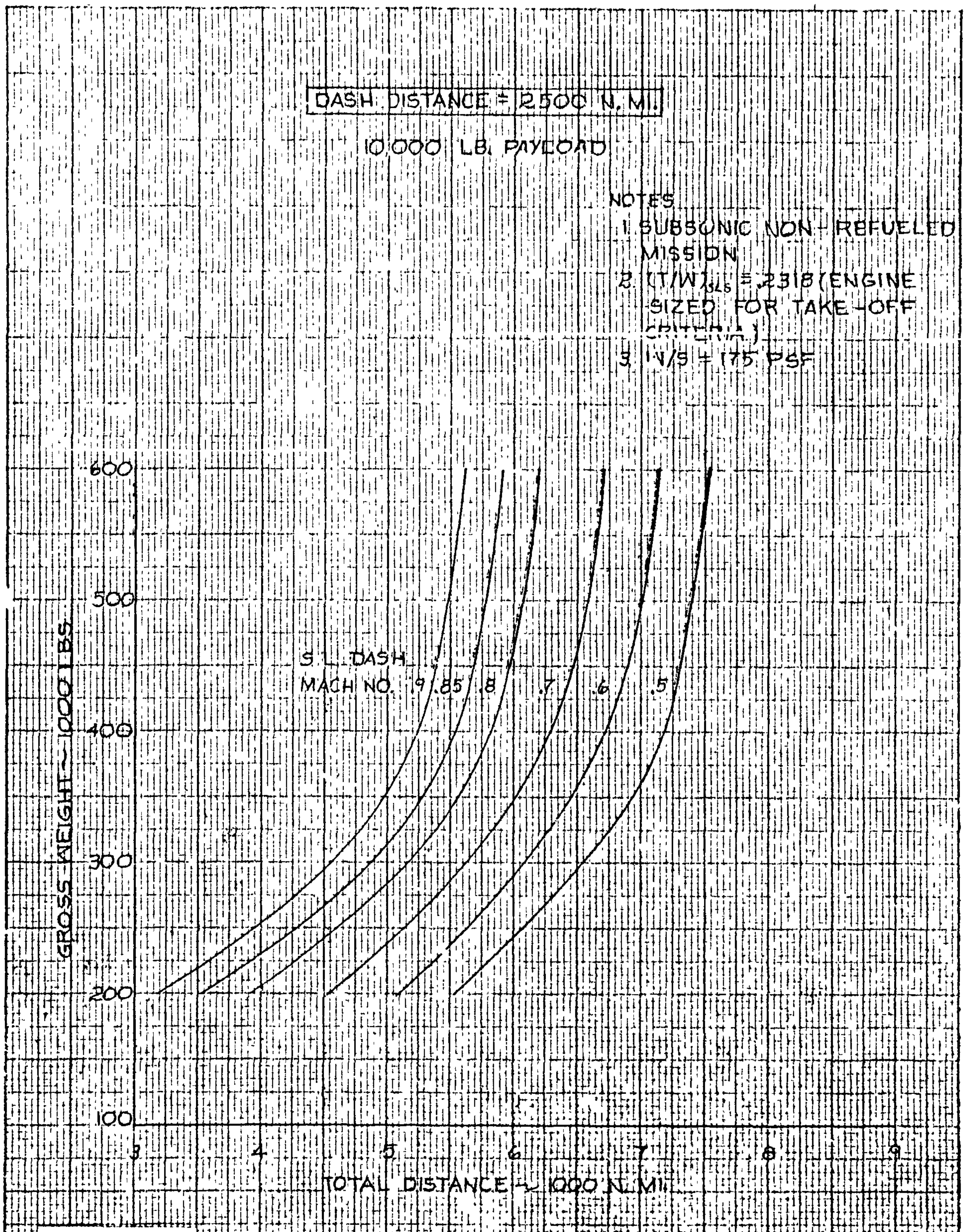


FIGURE 4.1-6 GROSS WEIGHT EFFECT ON RANGE

4.2 PENETRATION MACH NUMBER - CONSTANT RANGE

Figure 4.2-1 indicates the effect of dash Mach number on takeoff gross weight for dash speeds from Mach .5 to Mach .9 and sea level dash distances from 500 n. mi. to 2500 n. mi. with a total distance of 6300 n. mi. These data are presented for a 175 PSF wing loading with engines sized by 6000 feet takeoff criteria as discussed in Section 4.1. The 10,000-lb payload is dropped at mid-dash. These data were generated from data presented in Section 4.1.

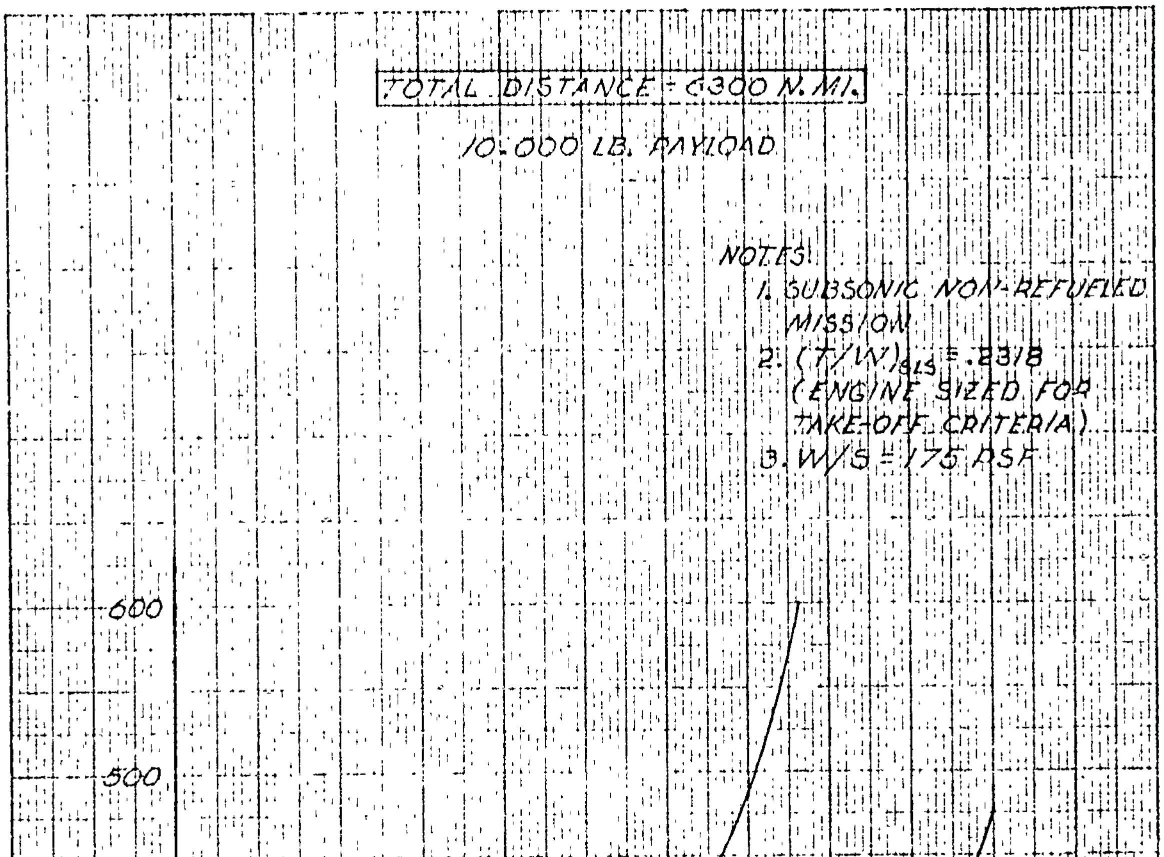


FIGURE 4.2-1 GROSS WEIGHT VS. MACH NUMBER

4.3 HIGH GUST EFFECTS

4.3.1 Structural Criteria and Weights

The effect of encountering discrete vertical and lateral gusts of up to 120 fps superimposed upon the incremental load factor programmed for the automatic terrain following equipment is shown in Table 4.3-I.

Incremental load factors are shown for two different gross weights. Those in Column 3 produce the maximum nW and those in Column 4 produce the maximum value of n. Column 5 gives the vertical tail load while Columns 6 and 7 show the horizontal tail incremental and balancing loads, respectively. Column 8 gives the structural weight penalty associated with each of the Mach numbers.

In the weight penalty calculations, the Δn values of Columns 3 and 4 were superimposed on a basic 2.0g condition, (1.0g balanced flight + 1.0g for the automatic terrain following system load factor).

Although the load factors are increased considerably there is no weight penalty in the wing because the basic design condition with flaps extended produces higher loads. The basic design criteria conditions give a vertical tail load of 168,000 pounds and a horizontal tail load of 137,000 pounds (Reference Section 3.7.1); therefore, there are no weight penalties due to the gust conditions on the tail structure.

However, since the fuselage and nacelle weights are greatly influenced by the load factor, any increase in load factor will cause an increase in structural weight. The resulting weight penalties of Column 8 are entirely the result of the fuselage and nacelle weights.

TABLE 4.3-I

1	2	3	4	5	6	7	8
M	Λ_{LE}	Δn_G G.W. = 349,000#	Δn_G G.W. = 186,200#	Vertical Tail Load (Pounds)	Incremental Horizontal Tail Load (Pounds)	Balancing Horizontal Tail Load (Pounds)	Structural Weight Penalty (Pounds)
.5	47°	1.64	2.76	75,100	± 59,600	+36,500	4,840
.6	60°	1.65	2.80	91,000	± 70,400	+ 3,000	4,908
.7	67°	1.67	2.85	108,000	± 83,400	- 6,750	4,993
.8	70.5°	1.72	2.92	126,000	± 98,600	- 3,600	5,134
.9	72.5°	1.82	3.10	145,000	±116,000	+11,500	5,468

4.3.2 Effects On Performance

The effect of encountering discrete vertical and lateral gusts of up to 120 feet per second on Configuration 2120 decreased total distance as is indicated in Table 4.3-II. The data are presented for a non-refueled symmetrical mission with 2000 n. mi. dash at Mach numbers from .5 to .9. This range reduction results from:

1. Incorporation of larger engines sized for takeoff instead of the previous 42.6% engine scale. This increased engine scale results from the increased gross weight (Section 4.3.1) due to considering 120 feet per second gust effects.

2. Increased nacelle and engine weights due to the larger engine scale.

3. Increased structural weight due to consideration of 120 feet per second gusts.

The following table indicates performance results between the basic Configuration 2120 and that of Configuration 2120 designed to encounter vertical and lateral gusts of up to 120 feet per second. Through usage of the growth curves of Section 4.1, gross weights required for a non-refueled symmetrical mission with 6300 n. mi. total distance and 2000 n. mi. sea level dash distance are also indicated.

TABLE 4.3-II

M	Configuration	Point Design Airplane Gross Weight -Lb.	Point Design Airplane Range - N.Mi.	Gross Weight For 6300 N. Mi. Range - Lb
.5	2120	395,000	7600	238,000
	2120*	400,310	7450	251,000
.6	2120	395,000	7260	269,500
	2120*	400,378	7085	286,000
.7	2120	395,000	6910	307,000
	2120*	400,463	6720	329,000
.8	2120	395,000	6535	356,000
	2120*	400,604	6360	390,000
.85	2120	395,000	6320	390,000
	2120*	400,771	6180	440,000
.9	2120	395,000	6080	480,000
	2120*	400,938	5880	In excess of 500,000

* Indicates incorporating 120 FPS gust restrictions

4.4 TAKEOFF DISTANCE REDUCTION

Figure 4.4-1 indicates the range penalties associated with decreasing the takeoff distance to 5000 feet instead of 6000 feet, using a trimmed $C_{L_{max}}$ of 3.2 (free air). The reduction in takeoff distance can be obtained from either a decrease in wing loading, an increase in engine size (T/W), or a combination of both. Decreases in wing loading will have an adverse effect on ride quality. This study is for a maximum density airplane with a gross weight of 354,000 pounds and is for the non-refueled subsonic mission with 2000 n. mi., Mach .85 sea level dash distance. As indicated, a wing loading of approximately 172.5 PSF would give maximum performance for a 5000-foot takeoff requirement.

Reductions in takeoff distance will also be obtained through improvements in $C_{L_{max}}$ (See Section 3.4.11).

The following table indicates maximum range at 354,000 pounds gross weight with engines sized for 5000-foot and 6000-foot takeoff distances along with the wing loading and gross weights required to satisfy the design nonrefueled symmetrical mission total range requirement of 6300 n. mi.

T.O. Distance	W/S-PSF	Range at 354,000 Lbs.-N.Mi.	Gross Weight Required for 6300 N.Mi.- Lbs.
5000 ft.	172.5	5615	In excess of 500,000 pounds
6000 ft.	175	6140	385,000

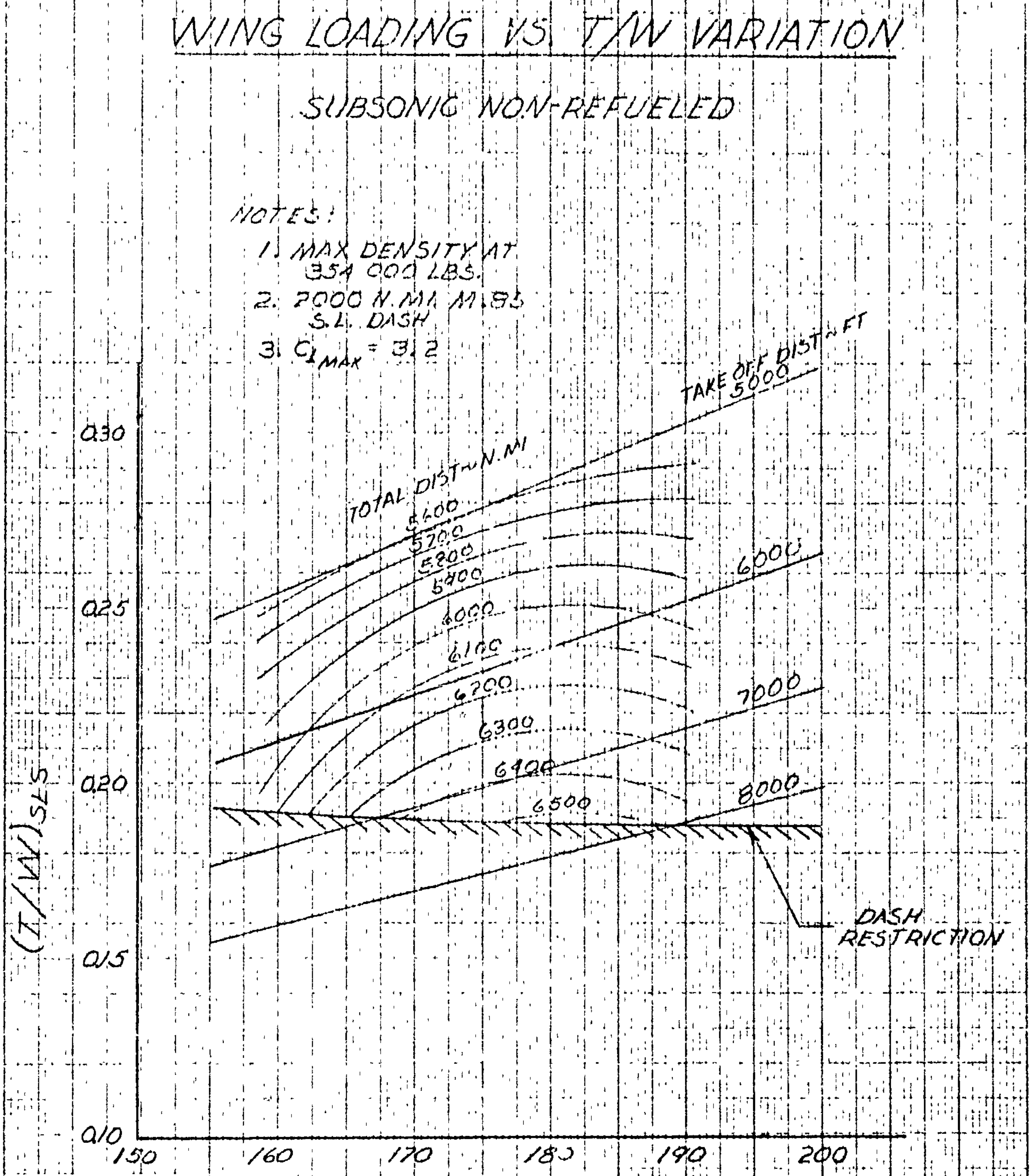

WING LOADING VS T/W VARIATION
SUBSONIC NON-REFUELED
NOTES:
1. MAX DENSITY AT 354 000 LBS.
2. 2000 N.MI. M.85 S.L. DASH
3. $C_{L_{MAX}}$ = 3.2
TAKE OFF DIST ~ FT
5000
6000
7000
8000
TOTAL DIST ~ N.MI
5600
5700
5800
5900
6000
6100
6200
6300
6400
6500
DASH RESTRICTION
$(T/W)_{SLS}$
0.30
0.25
0.20
0.15
0.10
150
160
170
180
190
200
WING LOADING ~ PSF

FIGURE 4.4-1

4.5 EFFECT OF PAYLOAD WEIGHT

Figures 4.5-1 through 4.5-5 present the effect of payload weight on takeoff gross weight for sea level dash Mach numbers from .5 to .9 and dash distances from 500 n. mi. to 2500 n. mi. These data are presented for the non-refueled symmetrical mission. Performance analysis is similar to that for the 10,000-lb payload as discussed in Sections 4.1 and 4.2.

These aircraft were designed with a forward and aft bomb bay area, each capable of carrying a capacity of 10,000 lbs payload. Payloads less than 10,000 lbs are carried in the forward bomb bay with a fuel tank placed in the aft bomb bay. For payloads other than 10,000 lbs, the fuel capacity in the aft bomb bay was changed to maintain a constant takeoff gross weight. With the 20,000 lb payload, the aft bomb bay fuel tank is removed to permit incorporating the additional 10,000 lbs payload in this bomb bay. For payloads less than 10,000 lbs, the aft bomb bay fuel is increased such that for a 2000 lb payload the aft bomb bay area contains a tank filled to a capacity of 18,000 lbs.

SECRET

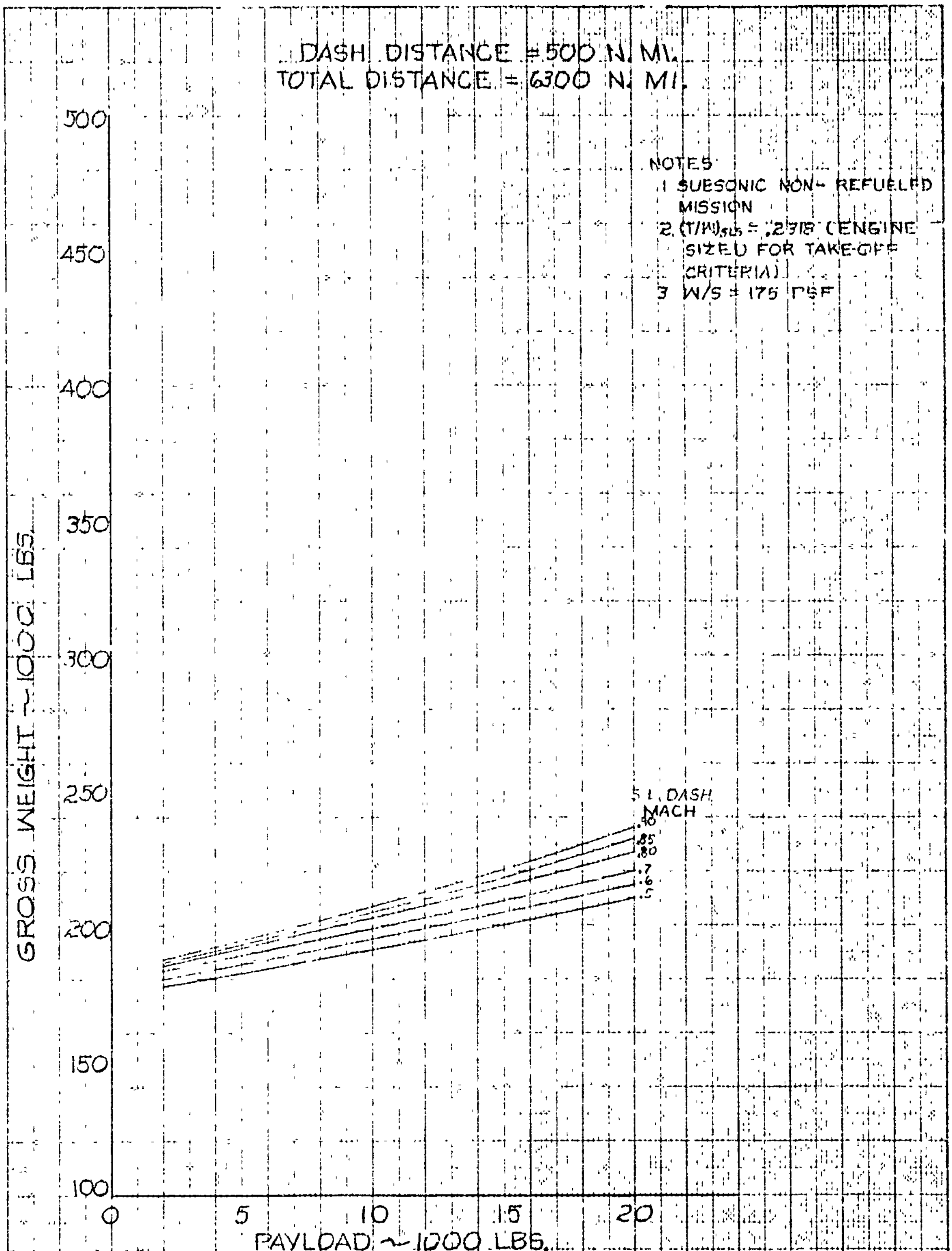


FIGURE 4.5-1 GROSS WEIGHT VS. PAYLOAD

SECRET

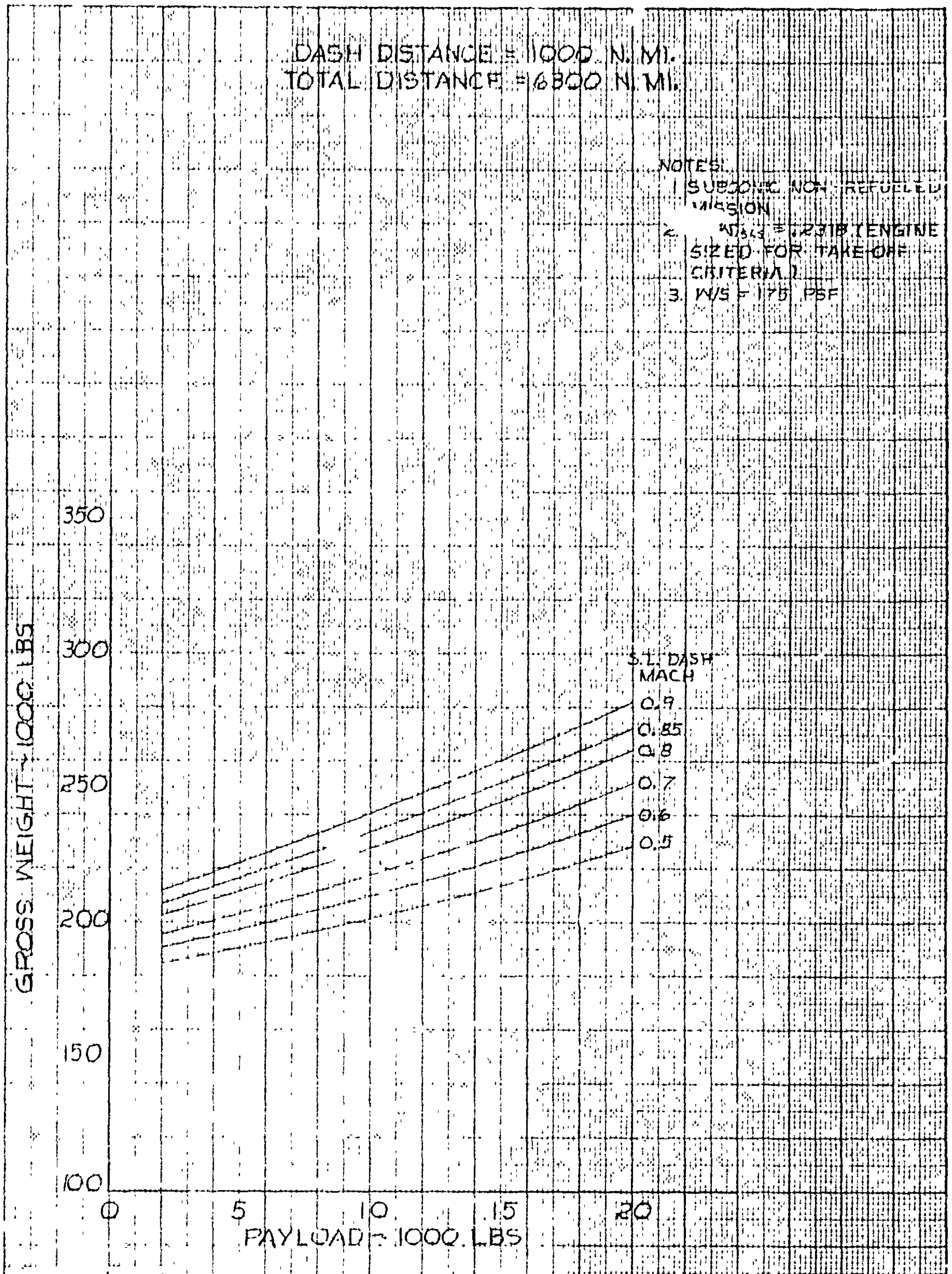


FIGURE 4.5-2 GROSS WEIGHT VS. PAYLOAD

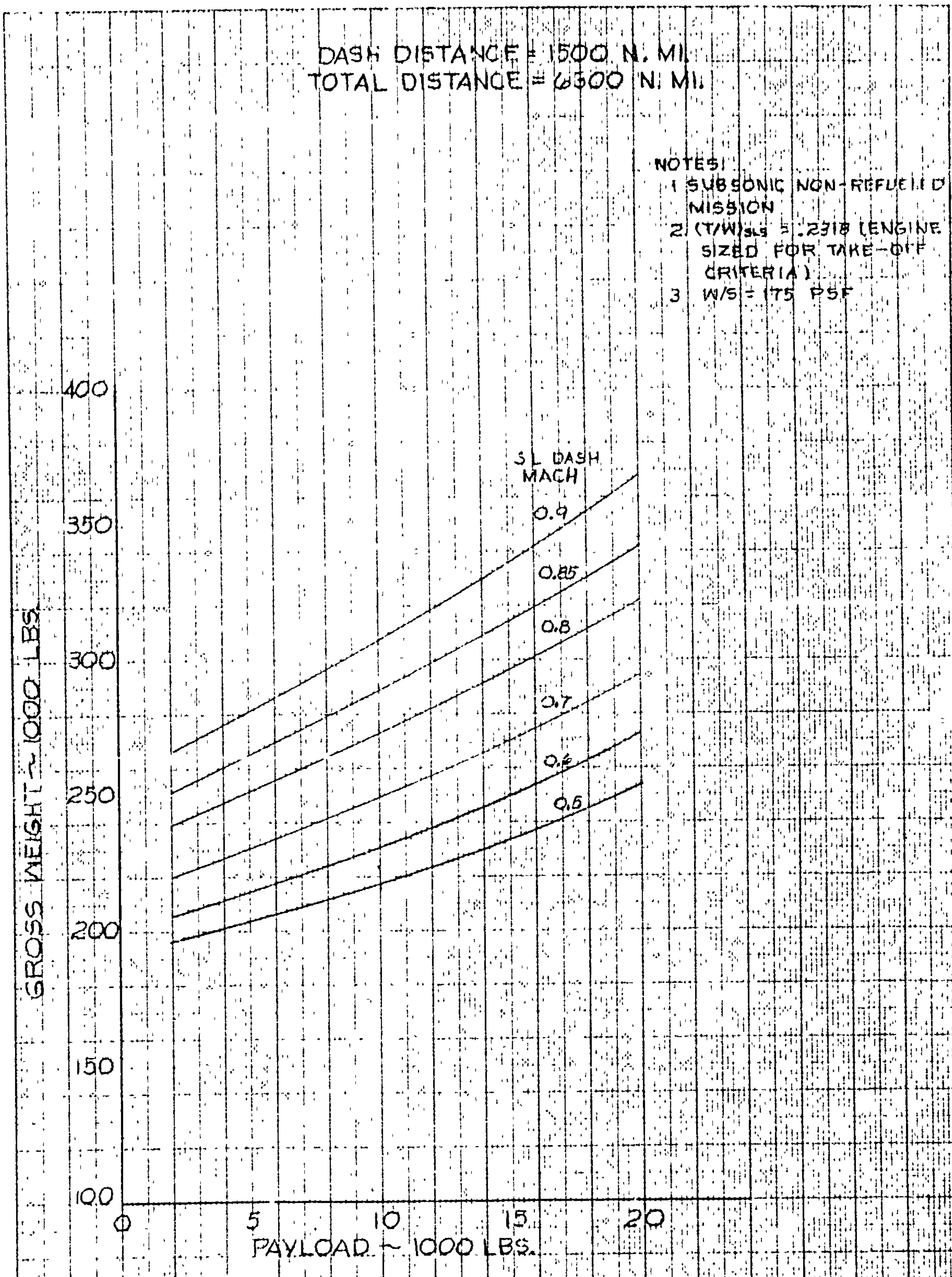


FIGURE 4.5-3 GROSS WEIGHT VS. PAYLOAD

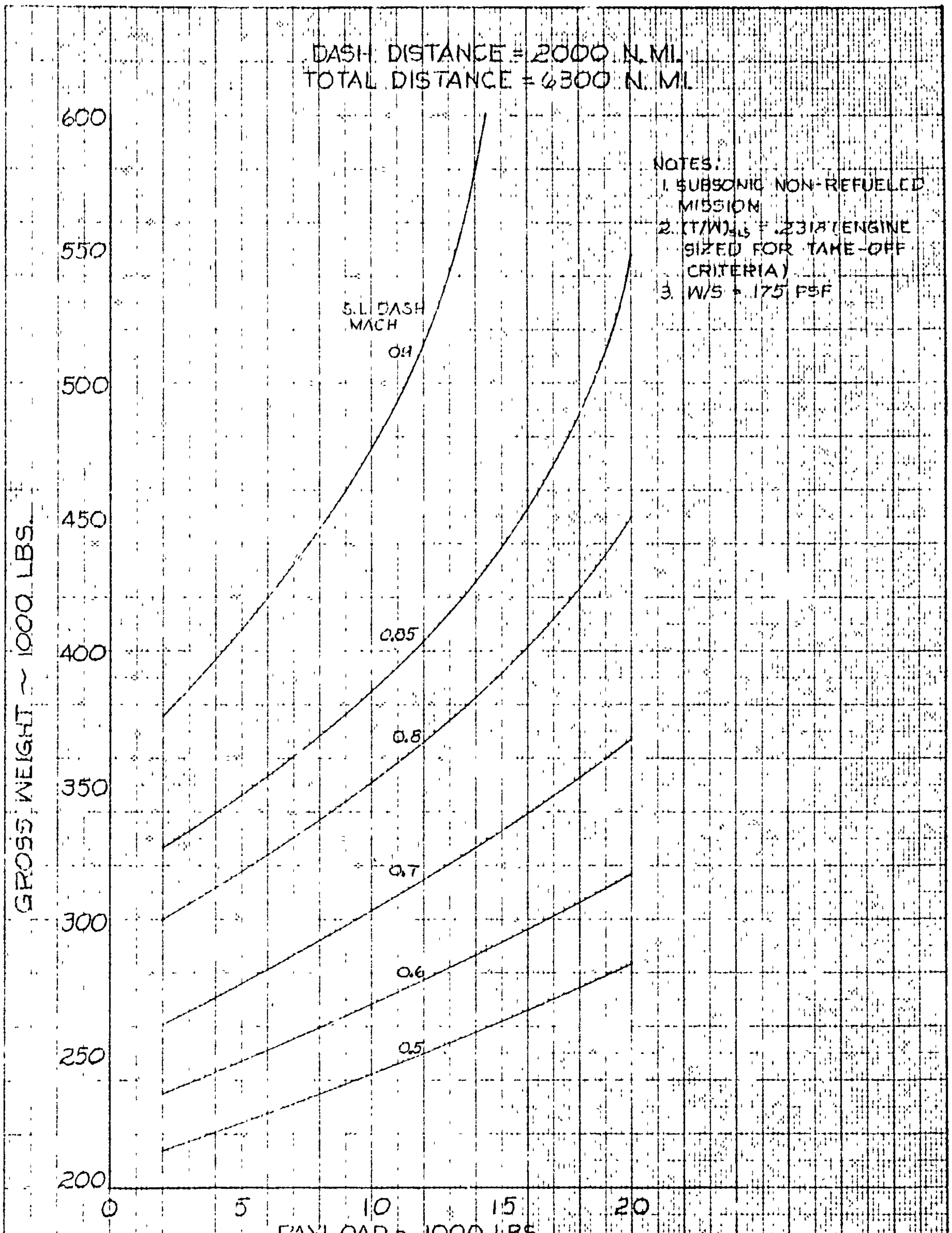


FIGURE 4.5-4 GROSS WEIGHT VS PAYLOAD

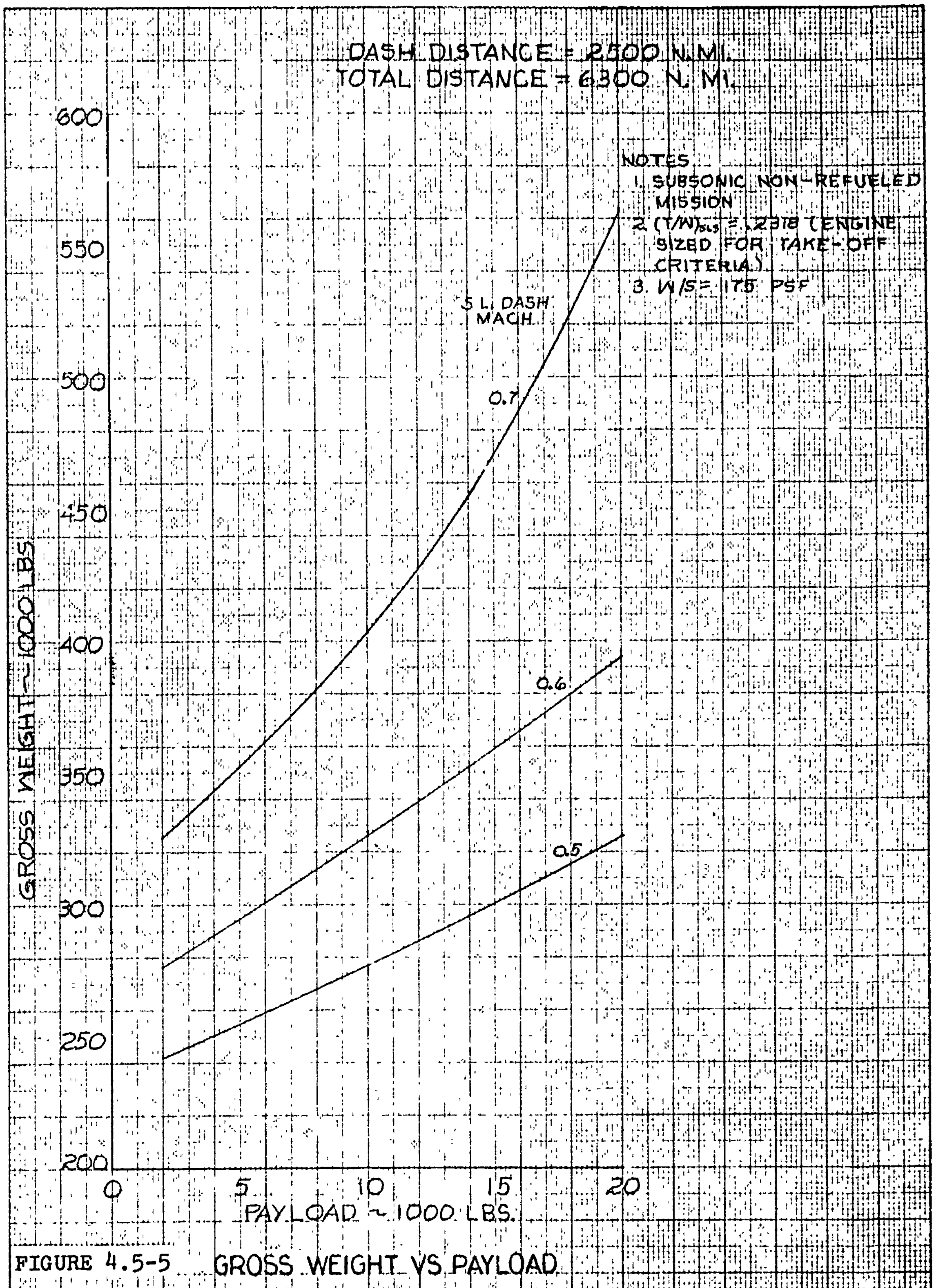


FIGURE 4.5-5 GROSS WEIGHT VS PAYLOAD

4.6 EFFECT OF ATO

Use of RATO during takeoff will permit the aircraft engine scale to be reduced to that required to perform Mach .85 sea level dash immediately after refuel and sustained Mach .90 sea level dash on the design mission. On Configuration 2120(point design airplane) a total range improvement of 340 n. mi. for the non-refueled symmetrical mission with 2000 n. mi. Mach .85 sea level dash distance results when the engine is sized for dash. Use of the growth cruve(Section 4.1 and Figure 4.6-1) indicates an airplane weighing 329,000 lb. without the RATO unit installed is required to perform the design non-refueled symmetrical mission of 6300 n. mi. total range with 2000 n. mi. Mach .85 sea level dash distance. This aircraft will use (4) 0.289 STF 200C-35.1 engines. Takeoff distance without RATO assist can be accomplished in 7380 ft.

A solid propellant RATO unit having 15,250 lbs thrust and weighing 5140 lbs with a time duration of 60 seconds will be required to perform takeoff in 6000 feet. The RATO unit is ignited at brake release and dropped at 50 ft altitude. A total installation weight of 360 lbs was used of which 80 lbs is carried throughout the mission, the remaining 280 lbs is dropped with the RATO unit.

The airplane weight plus solid propellant RATO weight plus installation weight will result in the airplane havinga ramp weight of 334,500 lbs and 175 PSF wing loading at takeoff. The basic airplane without the RATO unit installed would have a 173.1 PSF wing loading.

The following assumptions were used in consideration of the solid propellant RATO unit.

Propellant: Sea Level Delivered ISP = $\frac{lb_f - SEC}{lbm}$

Burning Rate (1000 psi 77°F) = .29 in/sec

Density = 0.064 lb/in^3

Type: Internal burning

Chamber Pressure = 1000 PSI (expanded to 14.7 PSI)

Nominal Motor Diameter = 60 in.

Burning Time = 60 sec.

A storable liquid propellant RATO unit having 15,250 lbs thrust and weighing 4564 lbs with a time duration of 60 seconds is also applicable. This unit would be operated in the same manner as the solid propellant unit, i.e., ignited at brakes releases and dropped at 50 ft. altitude. Installation weights (280 lbs dropped and 80 lbs retained) are the same.

Use of the storable liquid propellant RATO unit will yield an airplane with a ramp weight of 333,824 lbs and 174.9 PSF wing loading at takeoff. The basic airplane without the storable liquid propellant RATO unit would still have a 173.1 PSF wing loading.

The following assumptions and weights were used for consideration of the storable liquid propellant RATO unit.

Assumptions

Propellant: Oxidizer - N_2O_4

Fuel - 50:50 mix. UDMH Hydrazine

O/F - 2/1

Thrust = 15250 lb_f

Burning Time = 60 sec.

Sea Level Isp = 260 $\frac{lb_f - sec}{lbm}$

Chamber Pressure = 300 PSI

Approximate Weight Table

Propellant Weight	3520 lb
Oxidizer Tank (stainless steel, 1/16" thick)	113 lb
Fuel Tank (stainless steel, 1/16" thick)	55 lb
Thrust Chamber and Associated Accessories	250 lb
Feed Pump or Pressure System	80 lb
Plumbing	40 lb
	4058 lb
10% Contingency	406 lb
Total Weight Storable Liquid	4464 lb

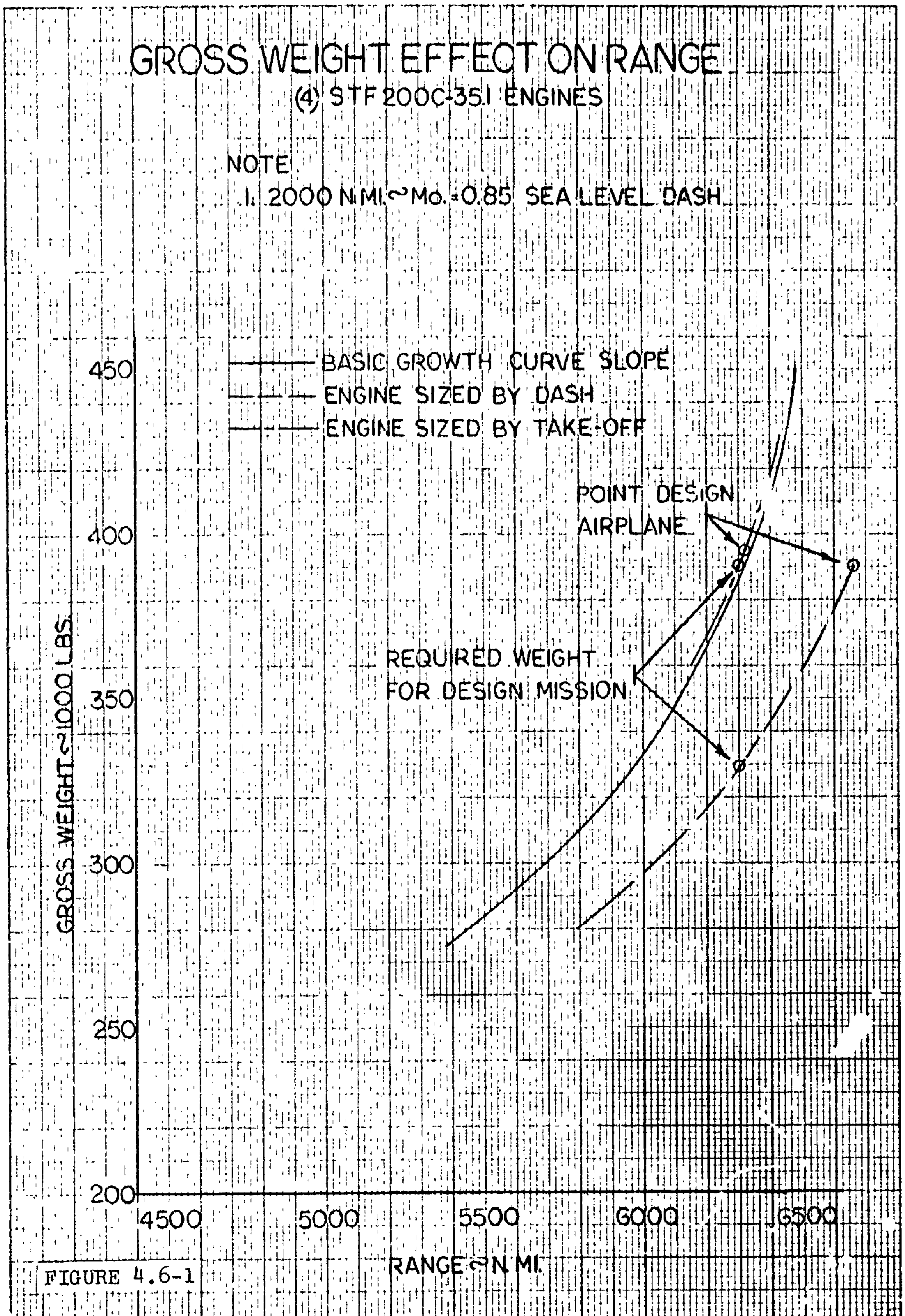

GROSS WEIGHT EFFECT ON RANGE
(4) STF 200C-35.1 ENGINES
NOTE
1. 2000 N.MI. ~ Mo. = 0.85 SEA LEVEL DASH
BASIC GROWTH CURVE SLOPE
ENGINE SIZED BY DASH
ENGINE SIZED BY TAKE-OFF
POINT DESIGN AIRPLANE
REQUIRED WEIGHT FOR DESIGN MISSION
GROSS WEIGHT ~ 1000 LBS.
450
400
350
300
250
200
4500
5000
5500
6000
6500
RANGE ~ N.MI.

FIGURE 4.6-1

4.7 WEIGHT EFFECT OF TITANIUM

4.7.1 Structural Considerations

GD/FW Report FZM-4156, "Comparison Study of Aluminum and Titanium Alloy Structural Airframes," was prepared to obtain weight data for the 2010 configuration airframe. A comprehensive study was made on various portions of the airframe, namely:

1. a typical section of the forward fuselage
2. Item (1) at increased load levels comparable to those existing in the aft fuselage
3. structural wing box outboard of the wing pivot fitting
4. wing outer panel secondary structure
5. projection of the results of Items (1), (2), (3), and (4) to other portions of the airframe.

In view of the similarity between the 2010 and 2120 configurations and loadings it is possible to make an accurate determination of the weight saved by using titanium alloys. The following weight savings can be obtained on Configuration 2120 if titanium alloys are used:

1. Basic fuselage structure	2500 lbs.
2. Wing structure box (outboard of pivot fitting	2030 lbs.
3. Wing trailing edge structure, auxiliary flaps or vanes	180 lbs.
TOTAL Weight Savings	4710 lbs.

This weight saving reflects the optimum mix of aluminum and titanium for maximum weight savings.

4.7.2 Effect on Performance

The effect of increasing the percent of titanium in the aircraft structure to an optimum amount (minimum structural weight) for Configuration 2120 increases total distance for a nonrefueled symmetrical mission with 2000 n. mi. Mach .85 sea level dash distance from 6320 n. mi. to 6440 n. mi. at the maximum density gross weights of 395,000 and 389,790 pounds, respectively. This range increase results from:

1. Incorporation of 41.7 percent engines sized for takeoff instead of the previous 42.6 percent engines. This reduced engine scale results from the reduced takeoff gross weight due to the use of titanium.
2. Reduced structural weight of 4710 pounds due to using an optimum percent of titanium.
3. Reduced nacelle and engine weights of 500 pounds due to the smaller engine scale.

A takeoff gross weight of 361,000 pounds is required by the titanium airplane to meet the nonrefueled design mission.

The following table indicates performance results between the basic Configuration 2120 aluminum structure and that of

Configuration 2120 with optimum percent of titanium. Gross weights required for a nonrefueled symmetrical mission with 6300 n. mi. total distance and 2000 n. mi. Mach .85 sea level dash distance are also indicated.

Configuration	Point Design Airplane		Gross Weight for 6300 N.Mi. Range -Lbs.
	Gross Weight - Lbs.	Range -N.Mi.	
2120 (Basic)	395,000	6320	390,000
2120 incorporating titanium	389,790	6440	361,000

4.8 EFFECT OF MACH 1.2 FOR EXTENDED PERIOD

4.8.1 Structural Considerations

The effect on structural weight of increasing the Mach 1.2 dash duration is shown in Table 4.8-I. The accumulative sea level dash life at Mach 1.2 has been assumed to be 500 hours. The weight penalties of Table 4.8-I are based on this 500-hour time. The weight penalties have been calculated on the basis of the differences in the 1.0 g tension fatigue allowables for the basic design criteria and the additional Mach 1.2 sea level dash. These allowables are conservatively based on a Kt of 4.0 and are shown in Table 4.8-I.

Table 4.8-I

Dash Condition	1G Tension Fatigue Allowable (psi)			Weight Penalty
	Wing	Fwd Fuselage	Aft Fuselage	(Lbs.)
Basic Design Criteria 2500 Hrs < 500' @ Mach .85	19,400	12,800	13,700	-
Supersonic Sea Level Dash for Extended Time (500 Hrs.)	19,400	12,000	12,700	847

4.8.2 Effect on Performance

Figure 4.8-1 indicates performance penalties associated with the subsonic mission due to incorporating a 500-hour accumulative sea level dash life at Mach 1.2 for Configuration 2120. The penalty results from 847 pounds structural weight increase and 315 pounds engine and nacelle weight increase causing dry weight and ramp weight to be increased 1158 pounds. No weight increases in air conditioning system weight were included since this system is sized by Mach 2.2 operation at altitude. As is indicated, the total range for the subsonic nonrefueled symmetrical mission with 2000 n. mi. Mach .85 sea level dash has decreased 50 n. mi.

The following table indicates performance results between the basic Configuration 2120 and that of Configuration 2120 designed to incorporate a 500-hour Mach 1.2 capability. Through usage of the growth curves of Section 4.1, gross weights required for a nonrefueled symmetrical mission with 6300 n. mi. total distance and 2000 n. mi. Mach .85 sea level dash distance are also indicated.

Configuration	Point Design Airplane		Gross Weight for 6300 N.Mi. Range-Lbs.
	Gross Weight Lbs.	Range-N.Mi.	
2120	395,000	6320	390,000
2120*	396,158	6270	404,000

*Incorporating 500 hours at Mach 1.2.

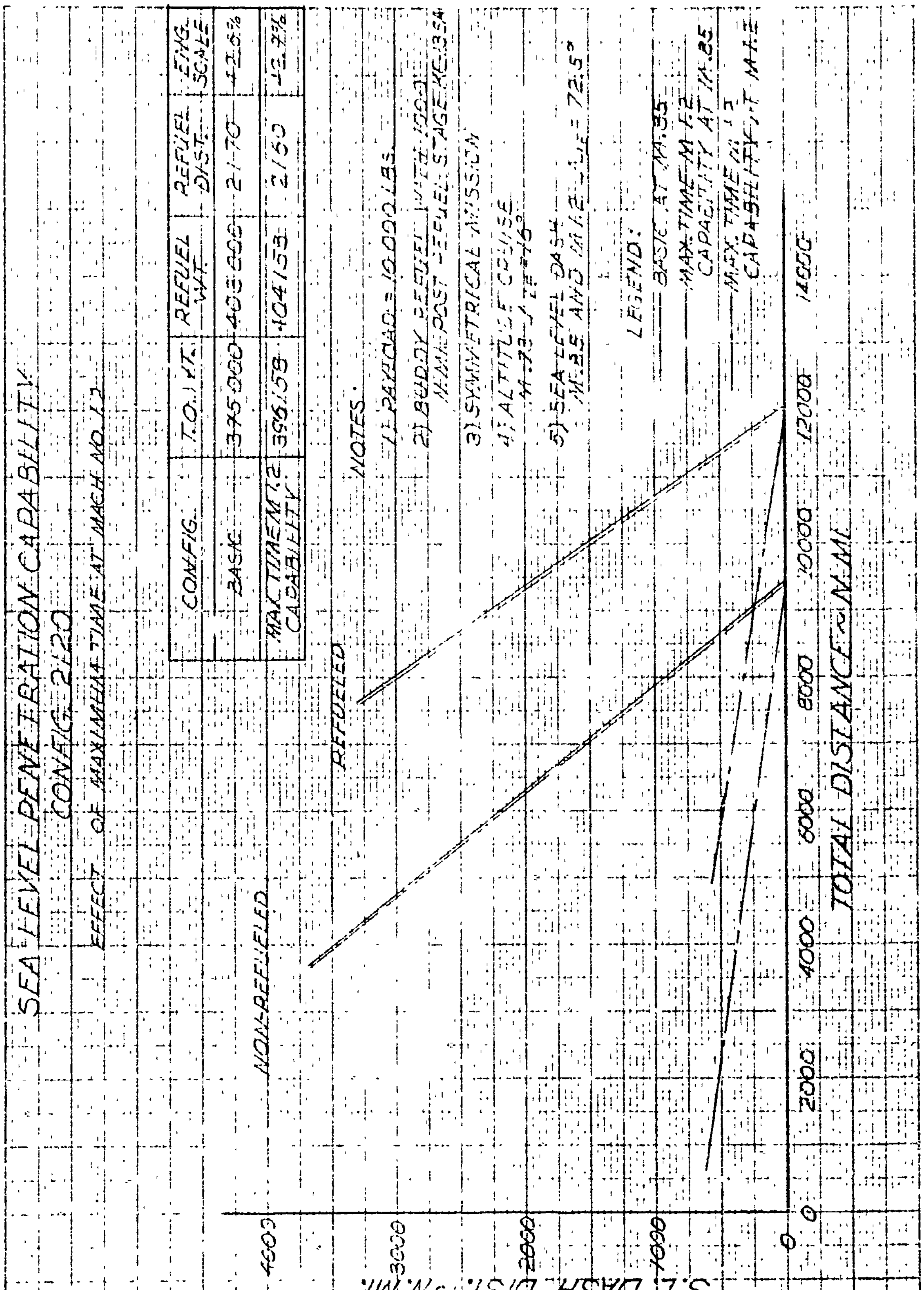

SEA LEVEL PENETRATION CAPABILITY
CONFIG. 2120
EFFECT OF MAXIMUM TIME AT MACH NO. 1.2
CONFIG.
T.O. WT.
REFUEL WT.
REFUEL DIST.
ENG. SCALE
BASIC
375000
MAX TIME M 1.2 CAPABILITY
396153
404153
NOTES:
1) PAYLOAD = 10,000 LBS.
3) SYMMETRICAL MISSION
4) ALTITUDE CRUISE
5) SEA LEVEL DASH
LEGEND:
BASIC AT M.85
MAX TIME M 1.2 CAPACITY AT M.85
NON-REFUELED
REFUELED
TOTAL DISTANCE ~ N.MI.
S.L. DASH DIST. ~ N.MI.
0
2000
4000
6000
8000
10000
12000
14000
1000
2000
3000
4000

FIGURE 4.8-1

4.9 DECREASED TIME BELOW 500 FEET

4.9.1 Structural Considerations

The effect on the structural weights have been determined for two alternate low level usages of the aircraft. These usages are shown in Table 4.9-I together with the 1g tension fatigue allowables associated with each low level usage. The weight savings shown have been calculated on the basis of the higher tension fatigue allowables due to the usage change. The fatigue allowables have been calculated for a stress concentration factor of four(Kt=4.0).

Table 4.9-I

Condition	Basic 1G Tension Fatigue Allowables - psi			Weight Savings in Forward Fuselage (Lbs)
	Wing	Fwd Fuselage	Aft Fuselage	
Basic Condition 2500 hrs < 500'@ Mach .85	19,400	12,800	13,700	-
Alternates @ Mach .85 1500 hrs. < 500' 1000 hours < 4000'	19,400	12,900	14,000	37
500 hours < 500' 2000 hours < 4000'	19,400	13,100	14,200	110

No weight savings occur in the wing and aft fuselage because they are not fatigue critical.

4.9.2 Effect on Performance

As indicated in Section 4.9.1, the weight savings realized by decreasing time below 500 feet is considered to yield negligible performance gains. Previous experience has indicated that 1 pound savings in dry weight will reduce gross weight 8 pounds. This is based on the growth curve slope at 395,000 pounds gross weight. This trade is based on a symmetrical mission incorporating 2000 n. mi. M .85 sea level dash distance with a total range of 6300 n. mi. Based on these results, the following gross weight savings can be realized.

Condition	Dry Weight Savings - Lb.	Gross Weight Savings - Lb.
1500 hrs 500' 1000 hrs 4000'	37	300
500 hrs 500' 2000 hrs 4000'	110	880

5.0 RUNWAY AND GROUND FLOTATION STUDIES

5.1 INTRODUCTION

Landing gear truck assemblies were designed to criteria specified in the AMPSS Statement of Work (64ASZXS-32), Paragraphs 1.1.3.1 and 1.2.10 summarized as follows:

Airfield Type	Passes Required
Rear Area T.O.	300
Medium Load Z.I.	300
Heavy Load Z.I.	Unlimited (5000)
Light Load Z.I.*	300

* Trade Study

Analysis was performed in accordance with SETFL Report 164. For Light Load Z.I. Airfields, further analysis was performed using WES Instruction Report No. 4 and the Portland Cement Association Handbook, Design of Concrete Airport Pavement.

5.2 SUMMARY

Truck assemblies were derived for airplanes from 200,000 lbs. to 450,000 lbs. gross weight. In all instances, the Rear Area T.O. Airfield was the determinant design condition, and analysis was performed to establish adequacy for the other basic criteria. A separate analysis was performed for the Light Load Z.I. trade study.

Arrangements considered included twin tandem, twin triple tandem, and dual twin tandem. Each was analyzed for retracted volume, influence on fuselage shape and size, and mechanical design. An arrangement was then selected for incorporation into the airplane design.

Tire sizes were derived using the formula, tire contact area (A) = $2.52\delta\sqrt{wd}$ where δ = tire deflection, w = tire width, and d = tire diameter. This formula is from the NASA Technical Report R-64, except that the 2.52 factor, representing F-111 state-of-the-art, has been used in lieu of the 2.3 published.

Tire spacing and size were varied to produce the optimum tire pressure and arrangement for the configuration considered. The effect of tire size and spacing variation on the number of allowable passes was investigated and curves drawn to reflect parameters for increments of pass capability.

5.3 CONCLUSIONS

1. For low gross weight (200,000 lbs.) the landing gear does not determine the fuselage cross-section. A twin tandem arrangement was used to provide maximum fuel volume and design simplicity. At high gross weights (to 450,000 lbs.) a dual twin tandem was employed to provide minimum fuselage cross-section and minimum gear stowed volume. A twin triple tandem arrangement was considered, and shows promise at in-between weights but was not seriously considered because of the lack of official eveluation procedures and because of the increased mechanical complexity.
2. The flotation criteria results in better gear volume to gross weight ratios for the lighter airplanes. Gear volumes increase at a greater rate than gross weights for a given truck arrangement.
3. For Light Load Airfields, gear volume increases at a much faster rate than for the basic requirements for equal gross weight increases. It is felt that this criteria, while accurate for the

defined capability of Light Load Airfields, can be relaxed to obtain more acceptable results when the limits of these airfields are better defined.

5.4 DICSUSSION

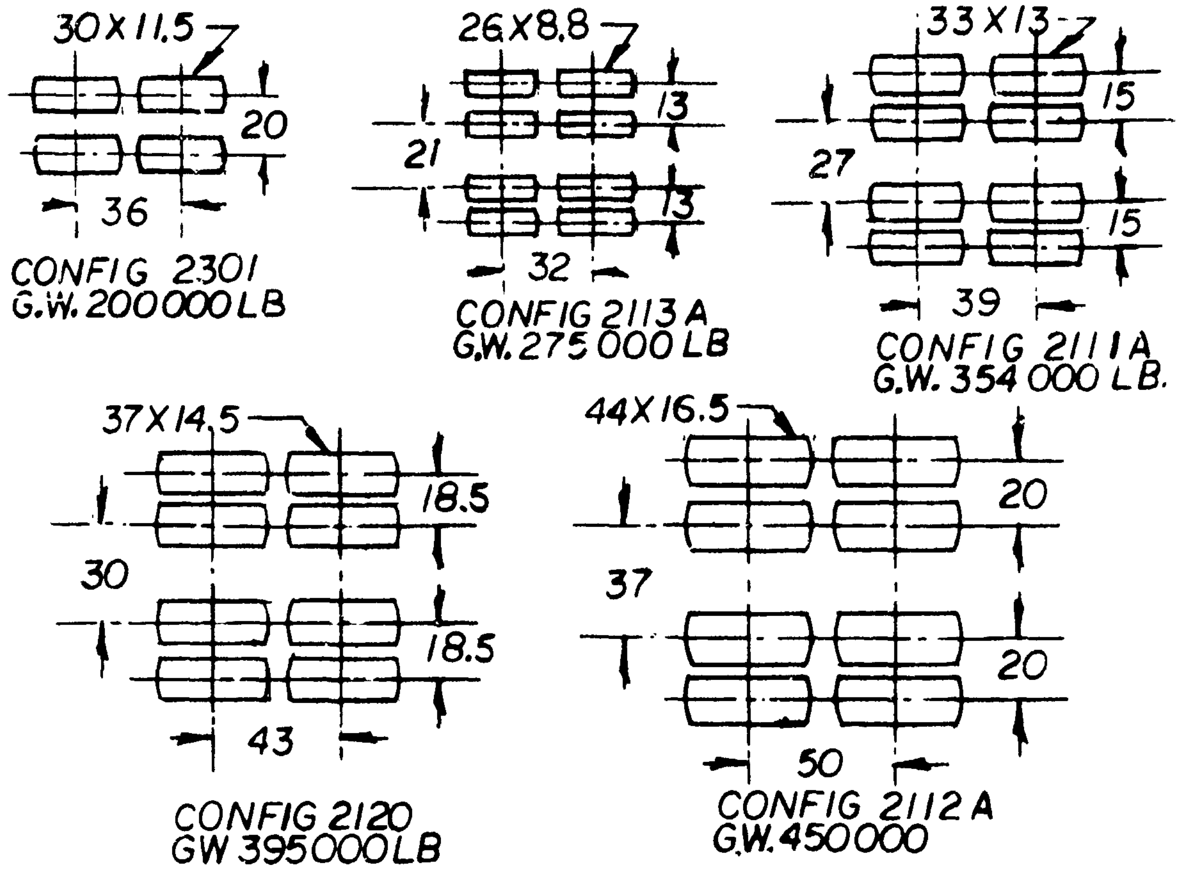
5.4.1 Truck Arrangement-Gross Weight Relationship

The truck sizes and types used on the airplane configurations studied are shown in Figure 5.0-1. The table shows the actual pass capability on a Rear Area Airfield and a gear capability index for ZI bases. This index is derived by dividing the airfield allowable load by the actual gear load, i.e., an index of 1.5 indicates that the airfield is capable of supporting 1.5 times the load actually placed on that gear for the required number of passes. The UCI is shown for each gear so that a comparison can be made with previous requirements.

Figure 5.0-2 illustrates the variation in tire diameter, tire pressure and truck volume with gross weight. From these curves, gear characteristics for airplanes other than those studied can be estimated. Truck volumes are based on minimum tandem clearance and do not necessarily reflect actual installed volumes, i.e., clearances necessary for shock struts and other mechanical components are not included. This allows formulation of a smooth curve necessary to establish design trends. Likewise, the volume shown for 200,000 lbs. is that of a dual twin tandem truck rather than the twin tandem actually used. The major point of interest on this curve is the rapid rate of increase of truck volume beginning in the 300,000 to 350,000 lbs. gross weight range.

5.4.2 Configuration 2120 Flotation Study

Three truck arrangements were considered for this "point design" airplane. The types, relative sizes, and their effect on fuselage



CONF.	PASSES ON REAR AREA A.F.	Z.I. CAPABILITY INDEX HEAVY	MEDIUM	LIGHT	UCI
2301	430	2.08	1.70	.68	70
2113A	332	1.97	1.63	.64	69
2111A	300	1.57	1.36	.53	60
2120	304	1.60	1.29	.51	54
2112A	332	1.62	1.25	.50	49

FIGURE 5.0-1

LANDING GEAR TRUCK SIZES AND FLOTATION CAPABILITIES

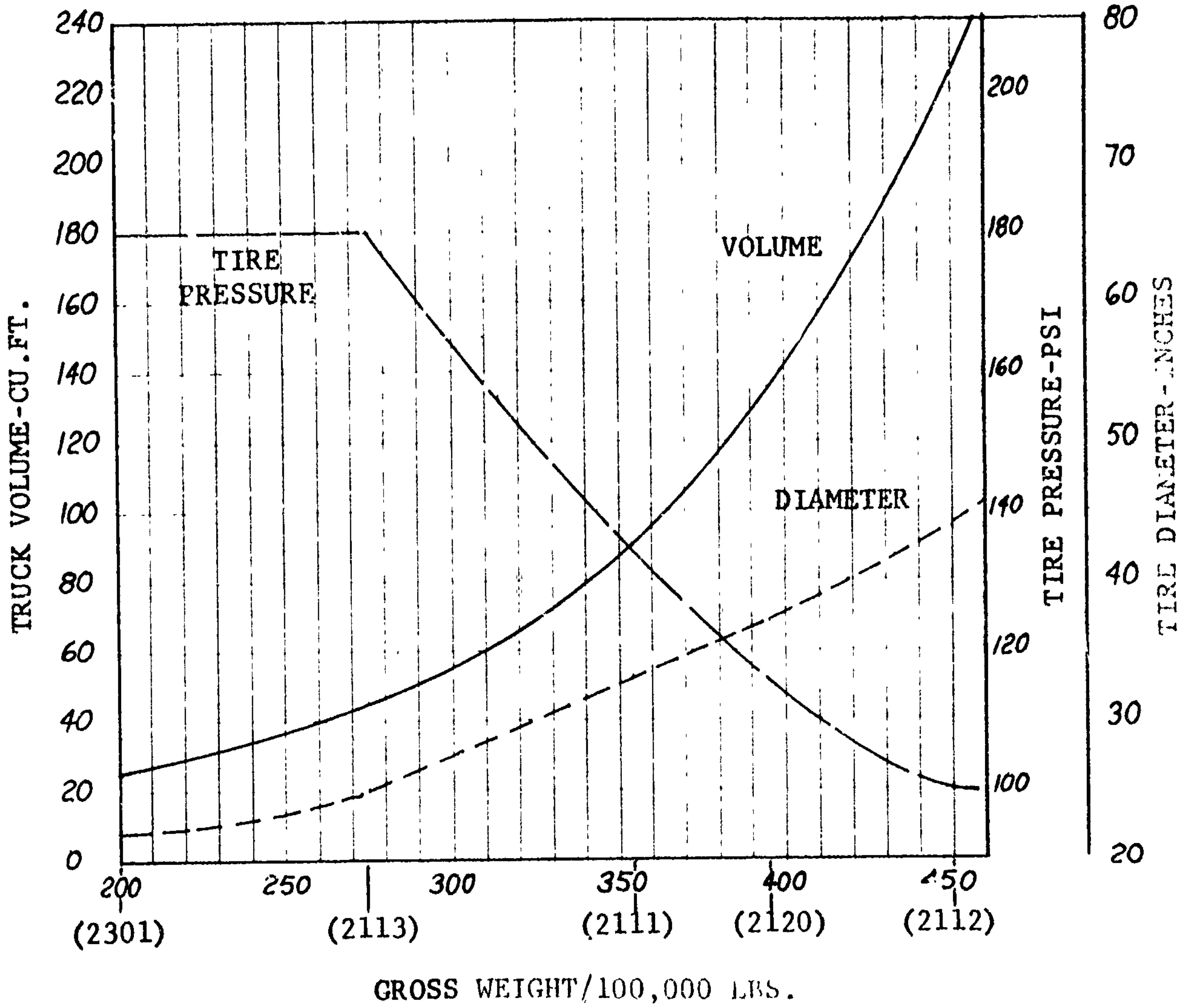


FIGURE 5.0-2
TRUCK VOLUME-TIRE DIAMETER-TIRE PRESSURE
VERSUS GROSS WEIGHT

cross-section are shown in Figure 5.0-3. All have approximately equal pass capability on Rear Area Airfields. The twin tandem was eliminated because of the fuselage cross-section penalty. The twin triple tandem was not used because:

1. Methods shown in SETFL-RPT-164 do not allow analysis of this configuration for ZI airfields.
2. There would probably be a weight penalty caused by the decrease in fuselage depth to width ratio.

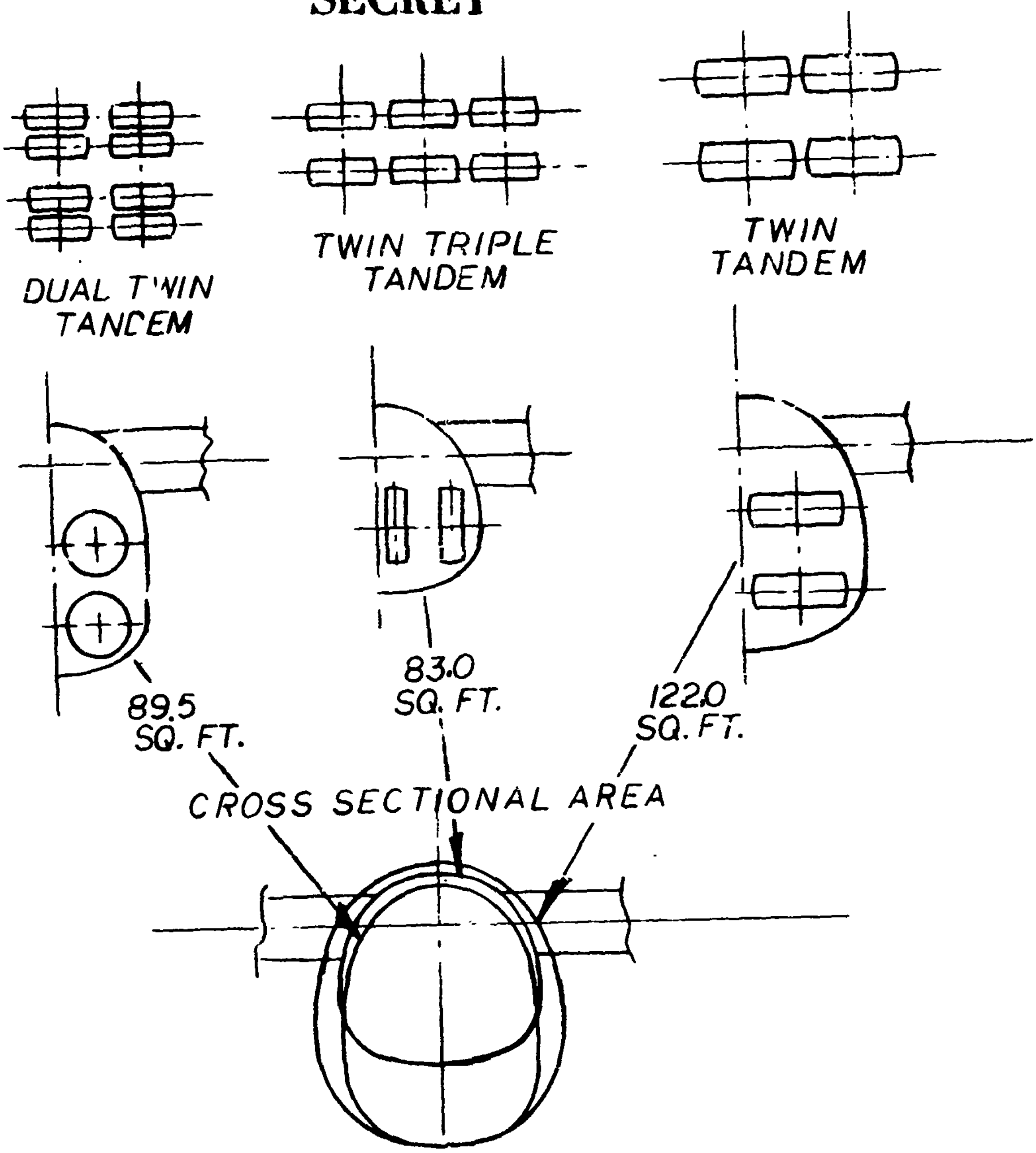


FIG 5.0-3

TRUCK ARRANGEMENTS CONSIDERED
FOR CONFIGURATION 2120
(395000 LB GW)

3. The mechanical linkage or structural rigidity necessary to provide equal wheel loading would result in increased gear weight.

4. The gear structure to counteract the increased tire scrub loads during turning would result in increased gear weight.

The dual twin tandem arrangement shown in Figure 5.0-1 was used because of its compatibility with the airplane fuselage cross-section as dictated by fuel, armament, and other requirements. The 37 x 14.5 tires have a gross contact area of 185 square inches, which results in a tire pressure of 125 psi at the maximum gross wheel loading of 23,000 lbs. Pass capability on Rear Area Airfields and adequacy on Heavy and Medium Load ZI Fields are shown in Figure 5.0-1.

The effect of variation in dual spacing is shown in Figure 5.0-4. This spacing was optimized at 18.5 inches to meet the 300 pass requirement. Because of the retraction method used (see Figure Dwg. #FW6401084), increased pass capability can be attained by adding the dual spacing difference to the gear well length.

Figure 5.0-5 shows the change in tire diameter (constant spacing) for variations in pass capability. Tire size increases will require corresponding increases in fuselage depth and width.

If, for example, it is desired to increase the airplane pass capability to 400, tire diameter would have to be increased to 39.5 inches. This would result in a gear well height increase of 5 inches (2 x (39.5-37), and an overall width increase of 5 inches. If dual spacing only is changed, the gear well height and width remain constant and the length is increased 3.75 inches (22.25-18.5). A more thorough analysis should be performed, but penalties for small capability increases can be estimated from Figures 5.0-4 and 5.0-5.

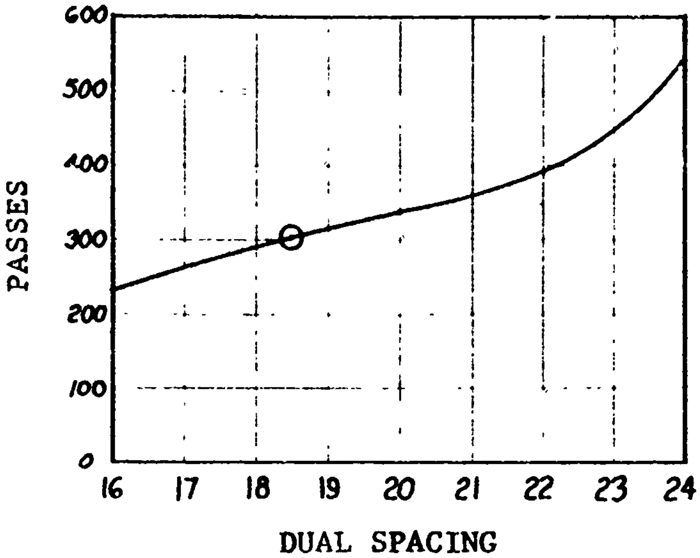


FIGURE 5.0-4

DUAL TIRE SPACING VERSUS PASSES ON REAR AREA AIRFIELD - CONFIGURATION 2120

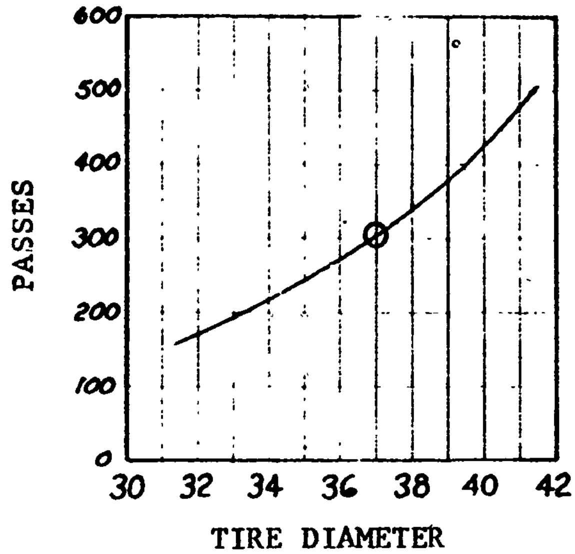


FIGURE 5.0-5

TIRE DIAMETER VERSUS PASSES ON REAR AREA AIRFIELD - CONFIGURATION 2120

The nose landing gear employs two 22 x 8 tires spaced on 18 inch centers. The tire area of 65 square inches results in a contact pressure of 180 psi for the 11,850 lbs. tire load at maximum gross weight. Capability on a Rear Area Airfield is 7300 passes and the flotation indeces on ZI bases are: Heavy - 4.5, Medium (Intermittent) - 3.77, and Light (Intermittent) - 1.76.

5.4.3 Configuration 2301

This study is included to show the reason for selection of the twin tandem gear for 200,000 lbs. gross weight. Figure 5.0-6 illustrates the influence of twin tandem and dual twin tandem trucks on the fuselage cross-section. The twin tandem was selected for its relative mechanical simplicity and for the fuel stowage volume above the gear well.

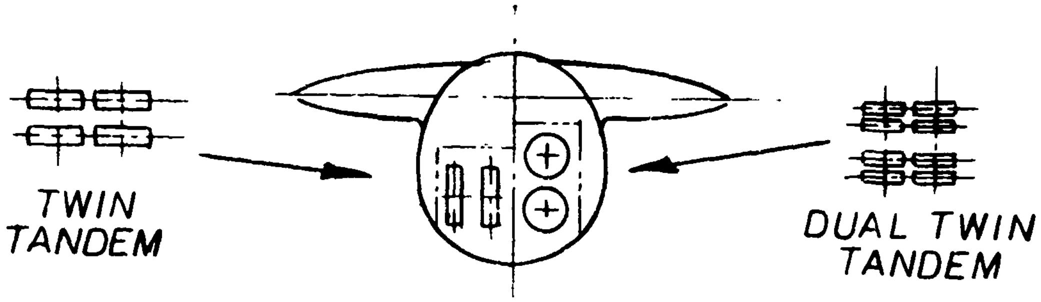


FIG 5.0-6
TWIN TANDEM VS DUAL TWIN TANDEM GEAR FOR CONFIG 2301 (200 000 LB)

5.4.4 Light Load ZI Airfield

5.4.4.1 Analysis per Contractual Criteria

Landing gear truck assemblies were developed for each gross weight studied in accordance with the procedures of SETFL-RPT-164. For Configuration 2301 (200,000 lbs.), the dual twin tandem arrangement developed for Rear Area T.O. Airfields, with the twins respaced to 30 inches (rather than 20 inches) is adequate. For Configuration 2120, however, a much larger truck is necessary. This arrangement is compared with the basic truck in Figure 5.0-7. The substantial increase in gear size precludes estimation of its effect on performance using normal procedures, so this trade study was not carried into configuration design.

The truck volumes required for 300 passes on a Light Load ZI Airfield and on a Rear Area T.O. Airfield are plotted against gross weight in Figure 5.0-8. This comparison shows the severe penalty for light load ZI operation as gross weight is increased, and indicates that further analysis should be performed to establish the validity of the criteria used.

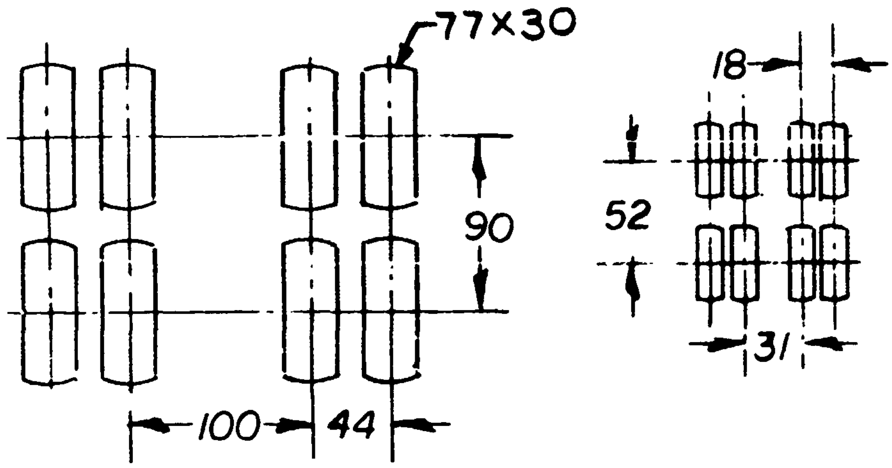


FIG 5.0-7

TRUCK CONFIG FOR LIGHT LOAD ZONE OF INTERIOR AIRFIELDS AND REAR AREA T.O. AIRFIELDS FOR CONFIG 2120 (395 000 LB GW)

5.4.4.2 Criteria Investigation

The criteria for Zone of Interior Airfields in SETFL-RPT-164 is digested from the WES 4-459 report. The WES 4-459 curves for gear evaluation are derived from representative gear analyses on flexible and rigid pavements, with the more critical pavement type determining the load allowables. The 4-459 curves are based on 5000 coverages. SETFL-RPT-164 provides for interpretation to 300 passes by multiplying the curve allowable load by 1.5.

To achieve a better understanding of the criteria and to determine actual airplane capabilities, Configuration 2120 was analyzed in detail for operation from flexible and rigid pavement Light Load Airfields.

Rigid Pavement Analysis

Paragraph 3.4 of SETFL-RPT-164 and Paragraph 3.c. of 4-459 define a Light Load Airfield as one with a load carrying capacity

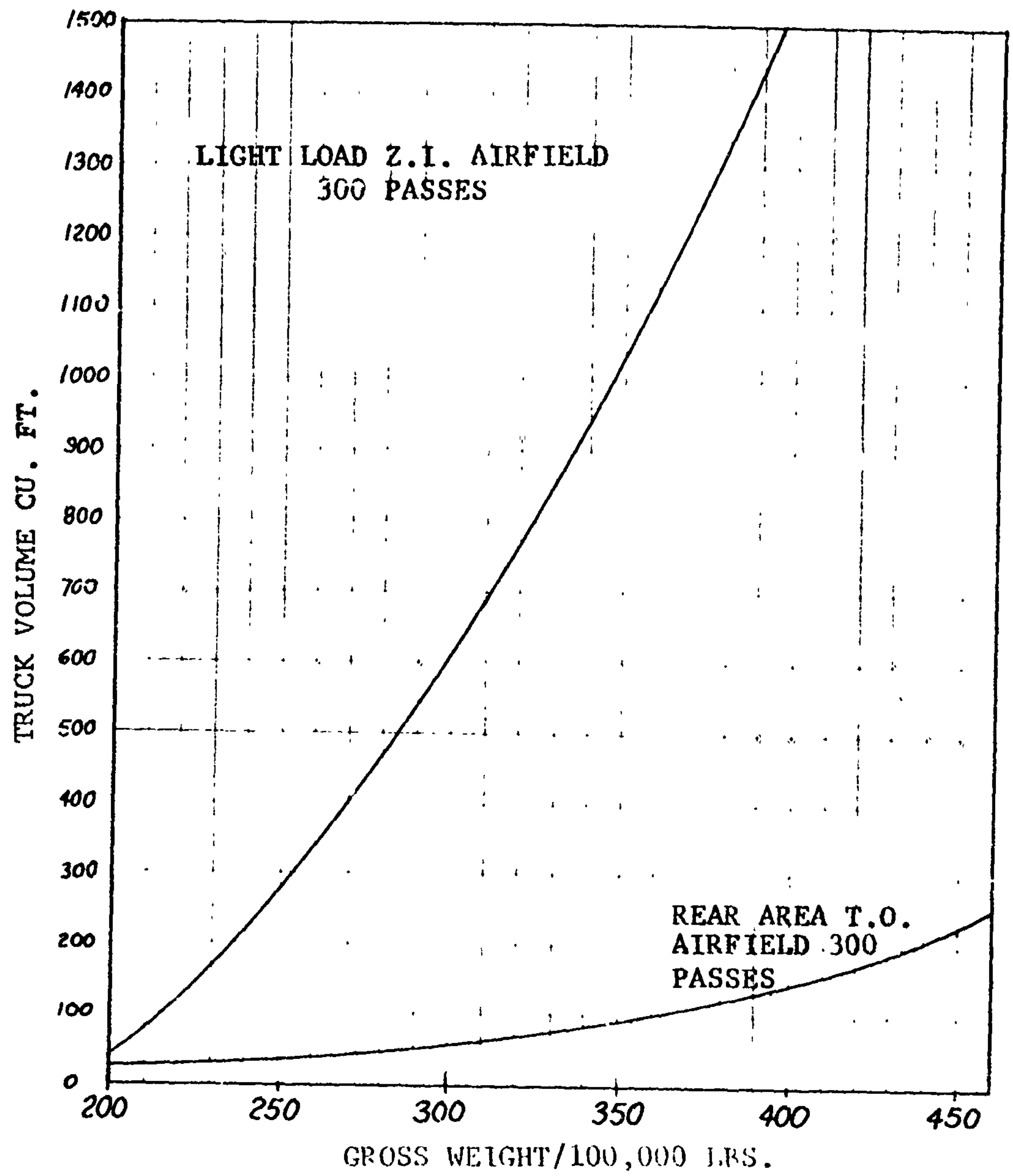


FIGURE 5.0-8

TRUCK VOLUME VS. GROSS WEIGHT FOR
LIGHT LOAD Z.I. AND REAR AREA T.O. AIRFIELDS

equivalent to a main gear loading of 25,000 lbs. on a single wheel having contact area of 100 sq.in. From this definition, Light Load Rigid pavements were defined using the Portland Cement Association (PCC) Handbook, "Design of Concrete Airport Pavement", Figure 7. Assumptions were:

f = Modulus of rupture = 700 psi

f^1 = Working stress (using a factor of safety of 1.75 for unlimited operation) = 400 psi

E = Modulus of elasticity = 4×10^6 psi

= Poisson's ratio = .15

For various subgrade reactions (K) pavement thicknesses were derived:

K = 50 lbs./in.^{2}/in.	T = 9.8 in.
10C	9.5
200	9.2
300	8.9
500	8.7

Experience indicates that a modulus or subgrade reaction of K = 100 is the minimum for airfields constructed under engineers' controls. Therefore, a rigid pavement, Light Load Airfield is defined as one of 9 1/2 inch pavement thickness on a subgrade with a K = 100 lbs./in.^{2}/in. (CBR = 3). Analysis of the gear derived for Rear Area criteria for 2120 by the PCC methods results in a pavement stress of 585 psi on this airfield. 585 psi falls between the minimum operational (approximately 3000 passes, 500 psi) and Emergency Operational (approximately 300 passes, 700 psi) categories. It is thus concluded that (1) the gear designed for 2120 to the basic criteria is also adequate for Light Load, Rigid Pavement, ZI Airfields, and (2) the applicable 4-459 evaluation curve is based on flexible pavement criteria.

Flexible Pavement Analysis

Using methods prescribed in the WES Instruction Report No. 4 (IR-4), and the 25,000 lbs. 100 sq.in. definition, the Light Load ZI Airfield characteristic curve was developed. This curve, Figure 5.0-9 shows the pavement thickness required for the base course

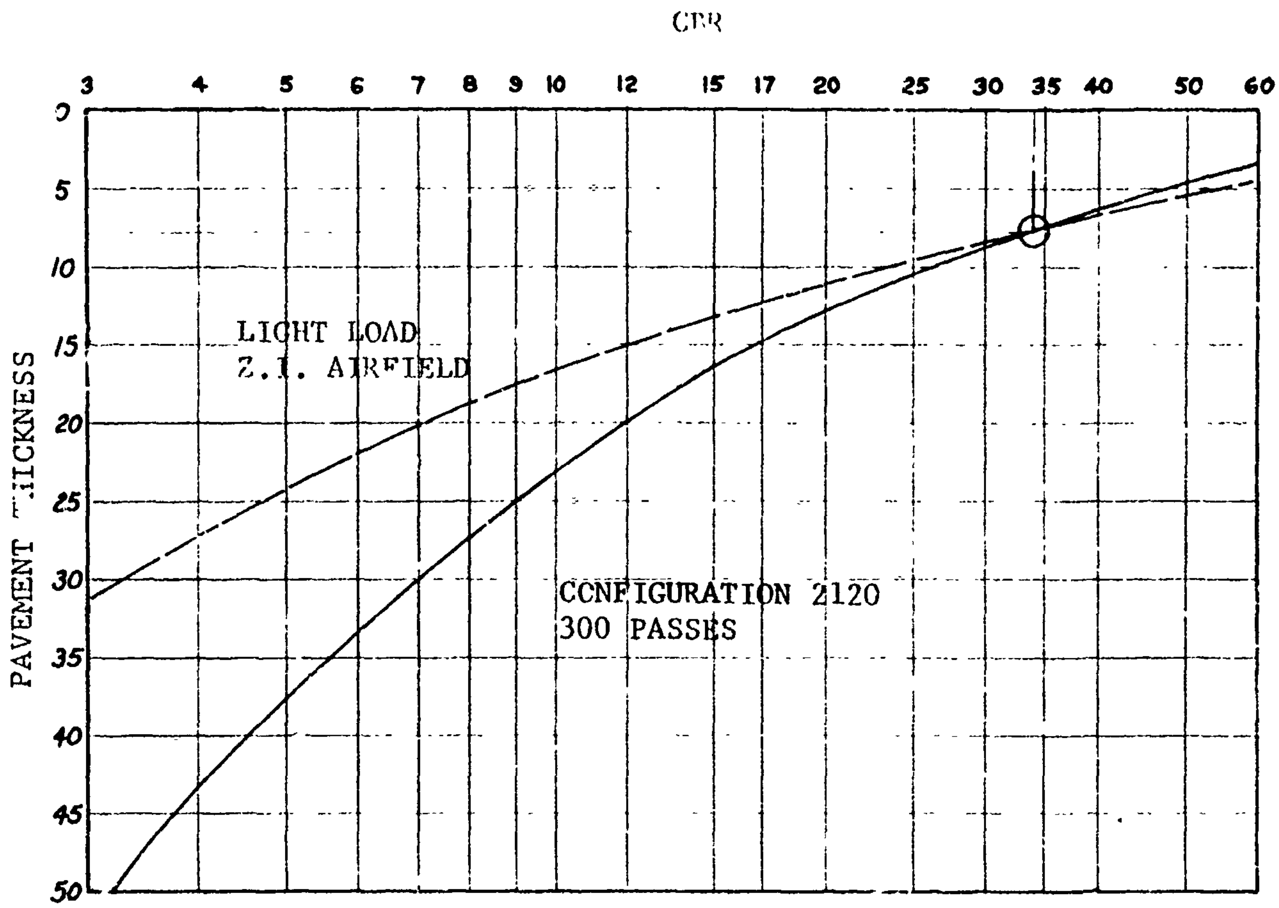


FIGURE 5.0-9

CONFIGURATION 2120 BASIC LANDING GEAR
COMPARED WITH LIGHT LOAD, FLEXIBLE
PAVEMENT, Z.I. AIRFIELD

strength measured by the California Bearing Ratio (CBR) method. Using IR-4 procedures, the characteristic curve for the 2120 basic truck was derived and plotted on Figure 5.0-9.

The intersection of the two curves indicates that the basic 2120 gear configuration meets the statement of work requirement for Light Load Airfields provided the pavement thickness is less than 8 inches or the base course strength is greater than CBR = 34.. The effect of pavement thickness and base course strength on pass capability is shown in Figure 5.0-10.

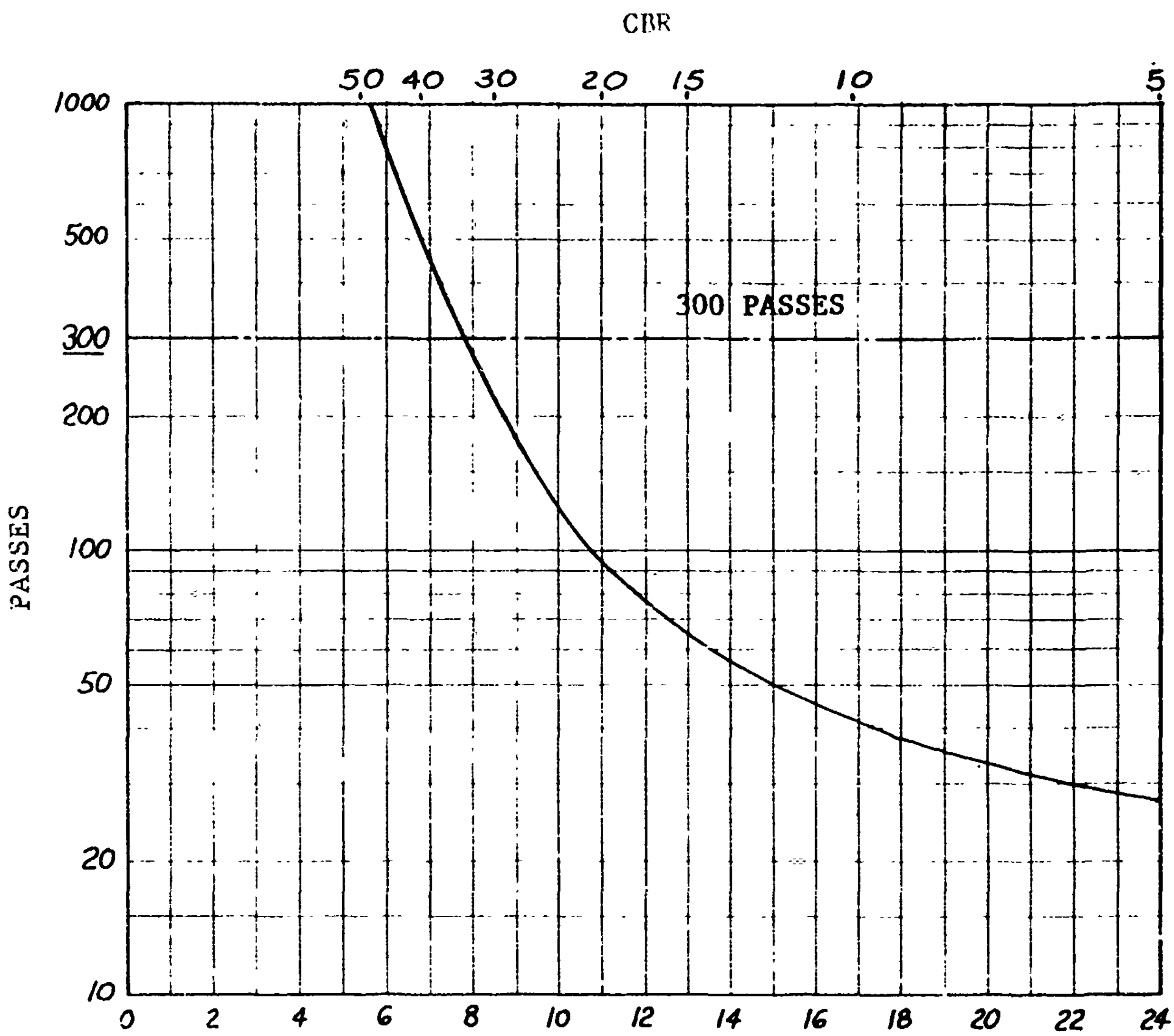


FIGURE 5.0-10

LIGHT LOAD, FLEXIBLE PAVEMENT AIRFIELD -
PASS CAPABILITY VS CBR
AND PAVEMENT THICKNESS

5.4.4.3 Conclusions

1. For Configuration 2120, a landing gear designed to the specified criteria for Light Load Airfields is so large that it is impractical for application to an airplane that will perform the required mission.

2. The basic gear derived for 2120 is adequate for all Light Load, rigid pavement airfields.

3. The basic gear derived for 2120 is adequate for some Light Load, flexible pavement airfields.

DEPARTMENT OF THE AIR FORCE
HEADQUARTERS 88TH AIR BASE WING (AFMC)
WRIGHT-PATTERSON AIR FORCE BASE OHIO

24 Aug 2006

88 CG/SCCMF
3810 Communications Blvd
Wright-Patterson AFB OH 45433-7802

Defense Technical Information Center
Attn: Ms. Kelly Akers (DTIC-R)
8725 John J. Kingman Rd, Suite 0944
Ft Belvoir VA 22060-6218

Dear Ms. Akers,

This concerns Technical Report **AD365474, Advanced Manned Precision Strike System Additional Task AFSC Planning Study No. 799086. Task 1. Baseline AMPSS Configuration. Task II. Parametric Studies (U), dated 30 June 1964.** This technical report, previously classified CONFIDENTIAL, has been officially declassified. WPAFB FOIA Control Number 06-287LK generated a mandatory classification review. The attached AFMC Form 559 and Memo for Record verifies that it was reviewed by release authorities at ASC/YD, B-1 System Program Office (currently renamed HQ 326th Aircraft Systems Group) and determined to be fully releasable to the public.

Please call me at (937) 522-3091 if you have any questions.

Sincerely

Lynn Kane

Lynn Kane
Freedom of Information Act Analyst
Management Services Branch
Base Information Management Division

Attachments
1. AFMC Form 559, RUSH – Freedom of Information Act
2. Memorandum For Record – Declassification of Technical Reports AD365474 and AD AD382213
3. Copy of Tech Report Cover