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AN ANALYTICAL STUDY FOR SUBSONIC OBLIQUE WING TRANSPORT CONCEPT

FINAL REPORT

July 1976

Prepared under Contract No. NAS 2-8686

by

THE LOCKHEED-GEORGIA COMPANY
A Division of Lockheed Aircraft Corporation
Marietta, Georgia

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16. Abstract

The Oblique Wing Concept has been investigated for subsonic transport application for a cruise Mach number of 0.95. Three different mission applications were considered and the concept analyzed against the selected mission requirements. Configuration studies determined the best area of applicability to be a Commercial Passenger Transport Mission - payload 23,768 kg (200 passenger + 10,000 lb of cargo) for a range of 5560 km (3000 n mi). The critical parameter for the Oblique Wing Concept was found to be aspect ratio which was limited to a value of 6.0 due to aeroelastic divergence. Comparison of the concept Final Configuration was made with fixed winged configurations designed to cruise at Mach 0.85 and 0.95. The crossover Mach number for the Oblique Wing Concept was found to be Mach 0.91 for takeoff gross weight and direct operating cost. Benefits include reduced takeoff distance, installed thrust and mission block fuel and improved community noise characteristics. The variable geometry feature enables the Final Configuration to increase range by 10% at Mach 0.712 and to increase endurance by as much as 44 percent. The Oblique Wing Concept Final Configuration is also shown to have alternate mission capability as an Air Force tanker and a Navy ASW airplane.

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AN ANALYTICAL STUDY FOR SUBSONIC OBLIQUE WING TRANSPORT CONCEPT

E. S. Bradley, J. Honrath, K. H. Tomlin, G. Swift,
P. Shumpert, and W. Warnock

SUMMARY

The study to assess the technical performance and economic potential of Oblique winged aircraft flying at subsonic speeds was conducted by the Lockheed-Georgia Company for the NASA under Contract NAS2-8686. A previous study, the "High Transonic Speed Transport Aircraft Study" sponsored by the NASA Ames Research Center (Reference 1), demonstrated the feasibility of the Oblique Wing Concept and showed the high potential of the concept for aircraft designed to cruise at speeds of Mach 1.2. Part of the potential can be attributed to the inherent advantage of the concept to attain low induced drag at takeoff, landing and during loiter, while maintaining good flight efficiency during cruise. These advantages applied to a subsonic design could lead to reduced takeoff gross weight, improved airport performance and community noise characteristics, improved endurance capability and better mission flexibility, and better speed matching.

The approach used in the study consisted of a survey of commercial and military missions, the selection of a number of mission possibilities, the application of the Oblique Wing Concept to these missions, the selection of the best mission-configuration combination, an analysis of the selected configuration, and a technical assessment to define key parameters and technological requirements.

Three missions, consisting of a Commercial Passenger Transport, an Executive Transport and a large Military Cargo Transport, were chosen and parametric analyses performed from which the configurations satisfying each of the mission requirements were selected.

The technology readiness for the study is consistent with a service introduction date of 1985 for the Oblique wing airplane, and background data used to predict technology availability were obtained from earlier NASA sponsored studies such as Reference 1 and the "Study of the Application of Advanced Technologies to Long Range Transport Aircraft," Reference 2.

Additional development of the Oblique Wing Concept by the NASA Ames Research Center has also been an important source of background data for the study.

The airplane structural characteristics rely upon the ability to utilize the maximum possible level of filamentary composite materials, which in this case, are mainly graphite-epoxy and Kevlar 49. The level of application for the study results in an essentially all-composite airframe structure. The level of weight reduction is compatible with that of Reference 1 and amounts to 20 percent. Critical structural design conditions arose initially from aeroelastic divergence of the leading wing.

Aerodynamic characteristics rely upon supercritical airfoil technology and since the stability and control responses of the Oblique wing are unconventional, a high degree of stability augmentation is required and a flight control system which accounts for cross-coupling effects is necessary. A system to satisfy these conditions could be developed.

Propulsion system data are based upon the Pratt and Whitney STF 433, bypass ratio 6.5 turbofan, Reference 3, which is an engine design consistent with the airplane technology time frame for noise and emissions and thrust/weight and specific fuel consumption improvements. The maximum thrust level achievable for 1985 is estimated to be 289,128 N (65,000 lb).

The Oblique wing airplanes described in this report for passenger transportation do not appear to present any insurmountable design integration problems. The Military Cargo Transport application, however, for large airplanes is precluded due to propulsion system size, wing/flap system integration problems and due to center of gravity and loadability limitations.

Evaluation of the candidate configurations indicated the mission-configuration combination best suited to the Oblique Wing Concept to be the Commercial Passenger Transport, for which the mission is that of transporting 200 passengers a distance of 5560 km (3000 n mi) cruising at Mach 0.95.

The results of the study indicate an upper limit on swept aspect ratio of 6.0 to ensure divergence-free characteristics for the wing without incurring weight penalties. The results further show that the Oblique Wing Concept has 7 percent less takeoff gross weight, 5 percent less direct operating cost, lower total installed thrust and block fuel, and requires less takeoff distance than the equivalent conventional configuration. In addition, the variable geometry feature permits a maximum increase in range at off-design conditions of 10 percent and increases endurance capability up to 44 percent.

The Oblique Wing Concept advantages also include reductions in takeoff sideline, takeoff flyover and in approach flyover noise levels of 0.5, 2.5 and 8.5 EPNdB, respectively, from the levels of the equivalent conventional configuration and a significantly smaller 90 EPNdB soundprint of 9.065 x 10^6 m^2 (3.5 mi^2) compared to 19.17 x 10^6 m^2 (7.4 mi^2) for the conventional configuration.

The airplane design for the Commercial Passenger Transport application is also shown to have military mission capability as either an Air Force tanker or Navy ASW airplane.

INTRODUCTION

The principal features of the Oblique wing aircraft have been well defined for the low supersonic, M = 1.2, speed region in a previously completed study, Reference 1, and feasibility of the concept has been established. It has been demonstrated analytically that the inherent advantages of the Oblique Wing Concept to minimize induced drag at takeoff and landing and during loiter while maximizing airplane range for high speed cruise, are independent of Mach number. Because of these and other advantages not highly sensitive to cruise Mach number the possibility of a subsonic Oblique wing transport becomes an attractive alternative to a low supersonic design. The continued uncertainty about the price and availability of jet fuel will influence the design of future subsonic transports so that the Oblique Wing Concept used in conjunction with other technology advances, Reference 2, offers an operational flexibility in a fuel market environment which may alter over the life of the aircraft but which can be adjusted to suit prevailing conditions.

This final report describes and presents the results of the "Analytical Study for the Subsonic Oblique Wing Transport Concept" which began on August 1, 1975 and was concluded on July 31, 1976.

At the start of the study a conference was held at the NASA Ames Research Center between NASA and Lockheed-Georgia representatives. At this event, which took place on August 21, 1975, determination of the methodology for assessing aeroelastic effects on wing weight and the selection of the cruise Mach number for all Oblique wing airplane studies were made.

During the initial phases of the study, missions were identified and selected, configuration studies performed, and an evaluation of the concept application was made. A review of progress was conducted at the NASA Ames Research Center on December 10, 1975, at which time agreement was reached on the selection of the Final Configuration.

The remainder of the study consisted of the development, design and performance estimation of the Final Configuration. At the conclusion of the Final Configuration analysis an evaluation of the Oblique Wing Concept was performed, a technology assessment made and conclusions presented.

The study was performed at the Lockheed-Georgia Company under the direction of Roy H. Lange, Transport Design Department Manager.

Edward S. Bradley was designated Study Manager. Responsibility for Aerodynamics, Structures, Propulsion and Design Integration was assigned to J. Honrath, W. W. Warnock, P. Shumpert and E. S. Bradley. Other contributors were C. M. Jenness - Aeroelastic Analyses, K. Tomlin - Stability and Control Analyses, and G. Swift - Acoustic Analyses.

The contribution of the NASA Ames Research Center consisting of the aeroelastic analysis of Appendix B, is acknowledged.

The data of this report are available in summary form in NASA CR-137897, Summary Report, published July 1976.

SYMBOLS

α	Angle of attack
A	Area
A/A_{MAX}	Cross sectional area ratio
$A\ /A_{MAX}$	Mass flow ratio
AR	Aspect ratio
AR_S	Swept Aspect Ratio
c	Local chord
C_{av}	Average chord
C_F	Flap chord
C_D	Drag coefficient
C_l	Two-dimensional lift coefficient
C_L	Lift coefficient or centerline
$C_{L_{APP}}$	Approach lift coefficient
C_{L_D}	Wing design lift coefficient

$C_{L_{MAX}}$	Maximum lift coefficient
$C_{L_{MAX\ TRIMMED}}$	Maximum trimmed-lift coefficient
C_m	Two-dimensional pitching moment coefficient
C_M	Wing or aircraft pitching moment coefficient
C_{M_o}	Wing or aircraft zero-lift pitching moment coefficient
C_V	Velocity coefficient
δ	Wing deflection or pressure ratio
δ_{AM}	Ambient pressure ratio
δ_F	Flap deflection
δ_{T2}	Stagnation pressure ratio at engine fan entrance plane
D	Drag
ϵ	Downwash angle
e	Wing efficiency factor
EPNdB	Effective perceived noise level
F_N	Net thrust
$F_{N_{MCR}}$	Net thrust at maximum cruise rating
FVR	Wing fuel volume ratio
g	Acceleration due to gravity
η	Non-dimensional spanwise location
θ	Wing twist or temperature ratio
θ_{T2}	Stagnation temperature ratio at engine fan entrance plane

I_{xx}	Moment of inertia in roll
I_{xy}	Product of inertia in the horizontal plane
I_{xz}	Product of inertia in the vertical plane
I_{yy}	Moment of inertia in pitch
I_{zz}	Moment of inertia in yaw
L/D_{MAX}	Maximum lift-to-drag ratio
Λ	Sweep angle
M	Mach number
M_{CR}	Cruise Mach number
M_D	Drag rise Mach number
M_∞	Free stream Mach number
P	Pressure
P_{AM}	Ambient pressure
$P_{T,EX}$	Stagnation pressure at nozzle exit
q	Dynamic pressure
SFC	Specific fuel consumption
SFC_{MCR}	Specific fuel consumption at maximum cruise rating
T	Temperature
t/c	Thickness to chord ratio
V	Speed
V_{APP}	Approach speed
V_B	Design speed for maximum gust intensity

V_C	Design cruise speed
V_D	Design dive speed
V_∞	Free stream velocity
w	Frequency of oscillation
W	Engine inlet total airflow rate
W/S	Wing loading
X/C	Non-dimensional chord location
Z/C	Non-dimensional height location

ABBREVIATIONS

ADM	Alternate Design Mission
APU	Auxiliary Power Unit
ASW	Anti-Submarine Warfare
BPR	By-Pass Ratio
DOC	Direct Operating Cost
DOT	Department of Transportation
EAS	Equivalent Airspeed
MAC	Mean Aerodynamic Chord
OW	Operating Weight
PDM	Preliminary Design Mission
SLS	Sea Level Static
TOD	Takeoff Distance
TOGW	Takeoff Gross Weight

STUDY OBJECTIVES, TECHNICAL APPROACH AND DESIGN REQUIREMENTS

Study Objectives

The objectives of this study are: a) Definition of an Oblique Wing Concept which satisfies the Statement of Work; b) The identification of key parameters and the sensitivity of the design to changes in each of these parameters; and c) An assessment of the design impact of the application of advanced technologies and definitions of critical research areas associated with the development of the concept.

Technical Approach

The methodology for the conduct of the study consisted of a program plan which divided the study into four (4) related elements. The study plan, Figure 1, consists of the following elements: 1) Mission Selection, 2) Configuration Design and Analysis, 3) Final Analysis, and 4) Technical Assessment.

The technical approach calls for a survey of suitable missions both commercial and military, the analysis of the Oblique Wing Concept performing the missions, the selection of the configuration/mission pair best suited to the Oblique Wing Concept, the analysis of the selected configuration performing the primary and alternate missions and the comparison of the airplane performance with conventional configurations. A technical assessment of the concept as a subsonic transport to establish technological development requirements concludes the study.

Design Requirements

The design requirements for the Oblique Wing Concept were established from the Statement of Work. The basic definition of airworthiness requirements in the FAR Part 25 (Reference 4), augmented by new criteria where necessary for Oblique Wing Concept definition, was used as the airworthiness guideline for this study.

Performance requirements. - The important items of airplane performance for the Oblique Wing Concept are:

- o Range - A transcontinental stage length of 5560 km (3000 n mi) for the basic mission with ranges of 2780 km (1500 n mi) to 8330 km (4500 n mi) to determine sensitivity.

- FAA takeoff and landing field length - No greater than 3048 m (10,000 ft) for 350 K (90°F) (ISA + 17.22°C) at an airfield elevation of 305 m (1000 ft).

- Cruise Mach number - The range of consideration for cruise Mach number to be not less than Mach 0.8 and no greater than Mach 0.98 at the appropriate cruise altitude.

- Fuel - Airplane performance and operation to be based upon conventional JP fuels.

- Noise - A 90 EPNdB noise contour of 12.95×10^6 m^2 (5.0 mi^2) during approach and takeoff.

- Approach speed - Not to exceed 259.3 km/hr (140 k) EAS.

Structural design requirements. - Structural design data are based upon the following parameters:

- Design air speeds - The design air speeds are those defined by Figure 2.

- Design load factor - Pitch maneuver cruise configuration + 2.5g max, -1.0g min.

- Gust load capability - (Reference 4) - Gust load capability will be based on encounter of gusts of:

  i) 22.12 m/sec (66 ft/sec) at speeds up to V_B

  ii) 15.24 m/sec (50 ft/sec) at speeds up to V_C

  iii) 7.62 m/sec (25 ft/sec) at speeds up to V_D

- Landing and ground handling:

  i) Sink speed at maximum landing weight 3.05 m/sec (10 ft/sec)

  ii) Sink speed at maximum takeoff weight 1.83 m/sec (6 ft/sec)

  Taxi load factors due to discrete bump - 2.0g.

Alternate mission load factors. - Are obtained from Military Specifications MIL-A-8861 as follows:

- Navy ASW mission - Maneuver limit load factor - 3.0g.

- Air Force tanker mission - Maneuver limit load factor - 2.0g.

Flutter deformation and fail safe criteria. - The aircraft will be designed to be free from flutter, divergence and control reversal at all speeds up to 1.2 V_D in accordance with Reference 4.

MISSIONS AND CONCEPTS

The mission selection process involved the collection of mission requirements data from as many sources as possible on those missions which appeared to have potential for the Oblique Wing Concept. These data were assembled into listings of candidate missions for consideration during the study and were further categorized into Preliminary Design Missions and Alternate Design Missions.

Data Sources

At the inception of the study a literature search was initiated to uncover possible missions applicable to the Oblique Wing Concept. Approximately 1,700,000 government and private technical abstracts of possible interest to this study were reviewed using the Lockheed DIALOG computerized data retrieval system. Key words and subject information were also given to the Defense Documentation Center and visits to the Navy Research and Development Information Center, the Advanced Systems Directorate of NASC, and to the Air Force Development Plans and Analysis Group and the Air Force Systems Command Headquarters Requirements Office augmented the data obtained through Lockheed facilities. Commercial airplane data were obtained through Lockheed-Georgia Company Operations Research and Commercial Sales organizations and from Project INTACT, Reference 5.

Candidate Missions

A summary of candidate missions for the Oblique Wing Concept is shown on Table I.

The first three missions of Table I are commercial and represent the best compromise from comments received from the airlines, DOT and other government agencies for future commercial aircraft missions. Although the mission defined in the Statement of Work has no stated requirement for cruise speed, other than it should be in the range of Mach 0.8 to 0.98, high speed will always offer some attraction for the traveling public. The highest

TABLE I CANDIDATE MISSIONS

Mission	Speed	Payload	Range	Takeoff Distance	Altitude	Remarks
Commercial Passenger	M 0 95	200 Passengers + 4,534 kg (10,000 lb)	5560 km (3000 n mi)	—	9,144 - 12,192 m (30-40,000 ft)	Baseline design mission.
Commercial Cargo	M 0 82	49,895 kg (110,000 lb) +	4815 km (2600 n mi) +	3,048 m (10,000 ft)	9,144 - 12,192 m (30-40,000 ft)	Must be compatible with military requirements.
Executive Passenger	M 0.82 +	15-18 Passengers + Baggage	7408 km (4000 n mi)	1,524 m (5,000 ft)	12,192 m (40,000 ft)	
Air Force Tanker	371 km/hr (200 k) TAS at 3,048 m (10,000 ft) M 0 88 at 11,887 m (39,000 ft)	81,648 - 113,400 kg (180-250,000 lb) and/or 27,216 - 36,288 kg (60-80,000 lb)	For 6482 km (3500 n mi) and/or For 10,186 km (5500 n mi)	3,048 m (10,000 ft)	3,048 - 10,668 m (10,35,000 ft)	
Missile Launcher	741 km/hr (400 k) TAS at 6,096 m (20,000 ft)	147,871, 178,942 or 220,672 kg (326,000, 394,500 or 486,500 lb)	6 hours at maximum TOGW and 12 hours with inflight refueling	3,048 m (10,000 ft)	9,144 m (30,000 ft) +	Could be smaller
Military Cargo	556 km/hr (300 k) TAS +	158,757 kg (350,000 lb)	6482 km (3500 n mi) or 12,964 km (7000 n mi) or 6482 km (3500 n mi) radius with payload offload and no refuel at midpoint	2,438 m (8,000 ft)	9,144 m (30,000 ft) +	80% of fleet owned by civil air carriers.
Command Post	For best endurance	Up to 45,360 kg (100,000 lb)	Max possible	1,829 m (6,000 ft)	9,144 m (30,000 ft) +	
Navy Carrier Aircraft, i.e., COD, ASW, Tanker, Early Warning, Attack Bomber	Best endurance to M 0.95 +	To 4,536 kg (10,000 lb)	To 3704 km (2000 n mi)	853 m (2,800 ft)	To 13,716 m (45,000 ft)	Several missions compatible with 1 basic airframe Wing swung to fore and aft position gives deck storage advantage.

cruise Mach number possible for the Oblique Wing Concept without compromising the airplane design based upon past studies is 0.95 and this speed was selected as the design cruise Mach number for the Commercial Passenger Transport. The range, 5560 km (3000 n mi), provides transcontinental and transoceanic capability. Commercial cargo missions result from Reference 5 program considerations for which a cruise Mach number of 0.82 appears to be the speed requirement for payloads of 49,895 kg (110,000 lb) or more. Compatibility of this aircraft with similar military missions is desirable.

The executive passenger mission of Table I is that of the Grumman Gulfstream "X." High speed for this mission could be an advantage but not at the expense of increased costs. Although the range requirement of 7408 km (4000 n mi) could not be substantiated it would provide transcontinental and transoceanic range capability. A takeoff distance not exceeding 1524 m (5000 ft) is a requirement for general aviation field usage.

Candidate missions appropriate to Air Force operations are:

- Air Force Tanker
- Missile Launcher
- Military Cargo
- Command Post

Air Force tanker requirements vary from speeds of 370.0 km/hr (200 k) TAS at 3048 m (10,000 ft) to Mach 0.88 at 11,887m (39,000 ft). A high cruise speed tanker aircraft offers the advantage of higher off-load speed which is particularly useful when refueling supersonic aircraft and for tanker recycling between base and off-load point. The missile launcher mission applies to ballistic missiles and the mission profile requires high-speed dash capability together with good cruise and loiter capability.

The military cargo mission is derived from the Air Force "ATLAS" program requirements which cover a wide band of payload and range. High speed is not a requirement but productivity and closure time would show improvement with higher cruise speed. The command post activity has always been a corollary mission for existing aircraft. Speed for this mission is not important but endurance and short runway capability are both prime criteria.

Navy missions for carrier based airplanes are shown on Table I as a group. All of these aircraft are relatively small and require short takeoff distance capability. Cruise speed requirements extend from speeds for maximum endurance to a cruise Mach number of 0.95.

Mission Selection Criteria

The major advantages of the Oblique Wing Concept over fixed wing or conventional aircraft for the same mission are:

- Lower takeoff gross weight - For missions requiring a cruise speed of M = 0.90 and above, an Oblique wing airplane will have a weight advantage over a conventional airplane designed to the same mission. The reduction in takeoff gross weight could lead to lower initial and direct operating costs for the Oblique wing airplane.

- Improved airport performance - For missions requiring a cruise speed of M = 0.90 and above, an Oblique wing airplane will require shorter runway lengths at takeoff for the same thrust-to-weight ratio as a conventional airplane.

- Improved endurance capability - An Oblique wing airplane will have improved flight endurance capability in the unswept configuration than a conventional airplane of the same gross weight and fuel load.

- Better mission flexibility - An Oblique wing airplane will exhibit greater efficiency on missions requiring both high endurance and high speed segments of a mission profile than a fixed wing aircraft performing the same mission profile.

- Better speed matching - The variable geometry feature of the Oblique Wing Concept will provide the means to more efficiently match speed with altitude and thrust at any condition below design cruise conditions than the equivalent fixed wing airplane.

The Oblique Wing Concept advantages listed above were the principal criteria applied to the candidate mission list for the selection of three Preliminary Design Missions. The versatility of each candidate mission to perform other missions was also a prime consideration in the selection of each Preliminary Design Mission.

This is exemplified by considering specialized designs such as bomber or tanker airplanes which would, due to the high density of the payload, require low-volume fuselages. Each airplane would be capable of performing the design mission efficiently. Attempting to modify such airplanes to perform missions involving large volume for fuselage payload and fuel stowage would incur severe penalties in weight and performance so that evolutionary variants of these airplanes performing alternate missions would thus be very limited.

Conversely, airplanes designed to perform commercial or military missions involving the transportation of passengers or cargo over long distances are designed for large payload and fuel volumes. The versatility of such designs to perform alternate missions is greatly enhanced so that such designs may easily be converted to perform missions such as the Air Force tanker, missile launcher, command post, or the Navy land based ASW.

Preliminary Design Mission Selection

Three Preliminary Design Missions (PDMs) were chosen from Table I for analysis. These PDMs were selected because of their potential for exploiting the unique advantages and characteristics of the Oblique Wing Concept. The selected PDMs shown on Table II are:

- o A Commercial Passenger Transport Mission.
- o An Executive Transport Mission based upon the Grumman "X" requirements.
- o A Military Cargo Transport Mission derived from the requirements of the Air Force ATLAS program.

The requirements for each mission are also shown on Table II.

The Commercial Passenger Transport Mission of 200 passengers for a range of 5560 km (3000 n mi) could benefit from the Oblique Wing Concept advantages of reduced gross weight and therefore reduced direct operating costs, and, since long-range is a feature of this mission, the Oblique Wing Concept enlarges the number of airfields available by operating the airplane at reduced gross weight.

The Executive Transport Mission was ultimately changed from that shown on Table II, i.e., 15-18 passengers, to 14 passengers and the range reduced from 7408 to 6950 km (4000 to 3750 n mi) in order to keep the resulting airplane size within the dimensional constraints imposed by carrier operation for possible Navy sea-borne corollary missions.

The Military Cargo Transport Mission consisting of a payload of 158,750 kg (350,000 lb) for ranges from 6480 to 12,960 km (3500 to 7000 n mi) results in aircraft having gross weights well in excess of 454,000 kg (1×10^6 lb). Each aircraft would therefore benefit from the reduced gross weight and lower direct operating costs obtainable with the Oblique Wing Concept.

The Commercial Passenger Transport Mission was also selected as the Baseline Design Mission for the provision of a configuration to enable the generation of design and performance data for general application throughout the study.

TABLE II PRELIMINARY DESIGN MISSIONS

1	2	3
Commercial Passenger Transport	Executive Passenger Transport	Military Cargo Transport
Payload - 200 Passengers + 4,536 kg (10,000 lb) Cargo Cruise Mach No. = 0.95 Range - 5560 km (3000 n mi) Takeoff Distance - 3,048 m (10,000 ft) Cruise Altitude - 9,144 - 12,192 m (30-40,000 ft) (gross weight, airfield performance)	Payload - 15-18 Passengers + Baggage Cruise Mach No. = 0.95 Range - 7408 km (4000 n mi) Takeoff Distance - 1,524 m (5,000 ft) Cruise Altitude - 12,192 m (40,000 ft) (gross weight, airfield performance, mission flexibility)	Payload - 158,750 kg (350,000 lb) Cruise Mach No. = 0.95 Range 1 6482 km (3500 n mi) 2 12,964 km (7000 n mi) Radius 1 6482 km (3500 n mi) Offload Payload at Midpoint No Refuel at Midpoint. Takeoff Distance - 2,438 m (8,000 ft) Cruise Altitude - 9,144 m + (30,000 ft +) (gross weight, airfield performance)
CANDIDATE ALTERNATE DESIGN MISSIONS		
Tanker (endurance, flexibility, speed matching) Command Post (endurance, gross weight) Ground Based Navy Aircraft - ASW, Rescue/Search/Surveillance (endurance, flexibility)	Navy Carrier Aircraft, e.g., COD, ASW, Tanker, Early Warning, Trainer, Attack Bomber (all characteristics in various combinations)	Tanker (endurance, flexibility, speed matching) Missile Launcher (endurance, flexibility) Commercial Cargo (gross weight, airfield performance)

Alternate Design Missions

The candidate missions remaining following the selection of the PDMs were designated Alternate Design Missions (ADMs). These ADMs were then categorized, as appropriate, for each PDM, so that at the selection of the Final Design Mission the ADMs corresponding to that mission had already been defined. The ADMs categorized according to applicability to the PDMs are shown on Table II.

The Air Force tanker and command post and the Navy land-based ASW missions result in configurations approximately the same size as the aircraft for the Commercial Passenger Transport Mission. The advantage of greater endurance and the ability to fly mission profiles involving loiter and high-speed segments, together with the unique ability to better match the speed and altitude requirements of receiver aircraft during refueling, benefit the tanker airplane. The command post application can make use of the higher endurance capability and lower gross weight and the land-based ASW airplane can benefit from the high-speed cruise and improved on-station loiter capability.

A preliminary examination of carrier-based airplane application indicates that Navy carrier airplanes would be about the same size as the Executive Transport. The ability to pivot the wing to an almost fore and aft position eliminates the need for wing folding devices for on-deck and between-deck stowage. Each of the possible Navy carrier aircraft Alternate Design Missions can utilize one or more of the Oblique Wing Concept advantages already outlined. In addition, the standardization of carrier-borne airframes is an important Navy consideration and the ability of the Oblique Wing Concept to provide this capability is therefore important.

The application of the Oblique Wing Concept to long range high-speed missions such as heavy tanker, missile launcher and military and commercial cargo transport provide improved productivity in addition to advantages such as lower gross weight and improved airport performance.

Oblique Wing Concept

The Oblique Wing Concept for the purposes of this study is defined as a high wing configuration for which the wing sweep angle can be varied from zero to some maximum angle set by flight or design conditions.

The configuration will utilize a body in which an essentially constant cross section is used over the section of fuselage serving as the passenger or cargo compartment. A single deck arrangement is desirable for passenger operation in which galley, maintenance area and baggage and cargo facilities are located beneath the deck. To ensure commonality for military and commercial cargo missions,the cargo floor width and length is determined by commercial container size requirements whereas the compartment height is dictated by the military requirement.

The wing planform is trapezoidal, has a taper ratio of 0.33 and constant thickness chord ratio from root to tip. Where possible the wing volume between the spars will provide fuel tankage for mission fuel. In cases where the wing volume is insufficient for the design mission, additional volume will be provided in the fuselage. The wing will have trailing edge flaps. If necessary, leading edge devices will be added to augment the maximum lift. Because of the variable geometry feature of the Oblique Wing Concept the wing contours must remain unencumbered by any form of protuberance whatever, so that the aerodynamic efficiency of the wing is not impaired. Leading and trailing edge high lift devices must therefore be contained within the wing airfoil contours except at those prescribed conditions of flight requiring deployment of these devices.

The empennage configuration is to be a tee-tail arrangement in which the horizontal stabilizer is of conventional configuration articulating in the pitch axis only.

The landing gear arrangement will satisfy the requirements of adequate ground clearance angle, provision of a tip-over angle of 1 rad (57.3 deg) and minimal fairing for stowage.

Conventional Concept

The conventional concept for the comparison configurations of this study is defined as a high or low wing configuration in which the body or fuselage, where required by the airplane cruise speed regime, is contoured so that the configuration cross-sectional area distribution conforms to a pre-determined area distribution curve in order to minimize drag divergence Mach number effects.

The wing configuration is a fixed swept arrangement laterally bi-symmetric. All mission fuel will be contained within the wing volume between spars. The wing high lift devices will consist of leading edge slats and trailing edge Fowler type flaps, single or double slotted.

The empennage configuration will be either a tee-tail or a conventional low tail arrangement, depending upon the location of the wing and arrangement of the propulsion system.

The propulsion system may be either wing mounted on pylons or arranged at the aft end of the fuselage externally or integrated with the rear fuselage or may be a combination of both.

The landing gear arrangement for either a high or low wing configuration will provide sufficient ground clearance on takeoff and a tip-over angle of not less than 1 rad (57.3 deg).

To ensure compatibility with the Oblique Wing Concept for passenger operation, the fuselage configuration will be a single deck arrangement with space for a galley, maintenance areas, and containerized and bulk cargo beneath the deck.

Similarly, to ensure commonality between military and commercial cargo configurations the minimum width and length of the cargo floor is determined by commercial container size requirements whereas the cargo compartment minimum height is dictated by military requirement.

CONFIGURATION STUDIES

Configuration studies were conducted for the three Preliminary Design Missions in order to develop those configurations which formed the basis of evaluation of the Oblique Wing Concept in each mission role. These studies required the synthesis of each configuration/mission pair for which airplane parametric sizing analyses were performed. The Methodology and Basic Data used for the sizing studies are contained in Appendix A. The evolution of each of the study configurations is described in the following.

Design Synthesis

The design synthesis used to generate the aircraft configurations was based upon:

- Definition of a fuselage to accommodate the mission payload.
- Number and location of engines.
- An estimated location of the wing pivot on the fuselage.
- Estimated tail arms for the horizontal and vertical stabilizers.
- Estimated location of the landing gear on the fuselage.

Commercial Transport Airplane Configuration

A 200 passenger payload, together with the 5560 km (3000 n mi) range of the Statement of Work was selected as the Baseline Mission and the configuration, developed for this mission following a number of iterations, became the Baseline Configuration for the Oblique Wing Concept. The Baseline Configuration provided the vehicle for the execution of a variety of engineering analyses, the results of which were subsequently incorporated into the Baseline Configuration to yield a Cycled Baseline Configuration for the purposes of concept evaluation.

The evolutionary process by which the Cycled Baseline Configuration was obtained consisted of several iterations as follows:

- An initial configuration was developed to establish the range of the parametric variables necessary to cover sufficient combinations of the parameters to permit optimization of and determination of configuration general characteristics.

- Determination of the configurations optimized for wing sweep angle to establish the cruise yaw angle for the Oblique Wing Concept.

- Generation of a Baseline Configuration for the execution of engineering and performance studies.

- Development of a Cycled Baseline Configuration for concept evaluation.

<u>Fuselage definition</u>. - Definition of the fuselage for Commercial Passenger Transport requirements was based upon the characteristics derived from the results of past studies which indicated that, for cruise Mach numbers up to 0.95, no area-ruling of the fuselage would be required for the Oblique Wing Concept. The fuselage configuration for the Commercial Passenger Transport is therefore able to utilize a considerable length of constant section, the cross section of which is shown on Figure 3.

In addition, the fuselage configuration is designed to provide:

- Passenger payload - 19,232 kg (42,400 lb) equivalent to 200 passengers together with 4536 kg (10,000 lb) of cargo.

- Passenger distribution - 15% first class
- 85% tourist class

- Seat sizes and arrangement to current wide-body jet standards of comfort.

- Below deck galley and passenger convenience provisions consistent with current standards.

- Ingress and egress in accordance with Reference 4.

- Containerized baggage and bulk cargo volume below passenger deck.

Initial configuration. - The parametric analysis of airplanes performing the Baseline Design Mission established that the minimum takeoff gross weight airplane occurred at a swept aspect ratio of 7.0 (aspect ratio 14.0 unswept) for an Oblique wing swept at an angle of 0.785 rad (45 deg). This analysis assumed Pratt and Whitney STF 429 engine data.

The airplane design requires all mission fuel to be located in wing tanks and a preliminary check of the volume available in the wing indicated space for 39,969 kg (88,118 lb) of fuel.

The principal characteristics of the initial configuration are shown on Tables III and IV, column 1.

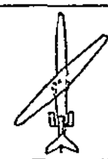
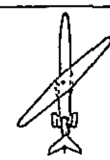
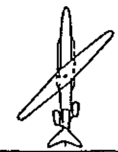
A weight and balance check of the initial configuration was performed to determine the center-of-gravity envelope and the possible existence of balance problems arising from location of the propulsion system. The balance characteristics for the Initial Configuration are shown on Figure 4. The portion of the envelope indicated by the solid line is the envelope for normal operation of the airplane with full passenger and cargo payload on the airplane. The forward portion of the envelope, shown by the broken line, represents the center-of-gravity travel diagram for full passenger payload only. The aft portion of the envelope, also indicated by a broken line, is the envelope for 4536 kg (10,000 lb) of cargo and no passengers aboard. Although this diagram is a preliminary center-of-gravity travel envelope only, it does indicate that an Oblique wing configuration, with the propulsion system mounted at the aft end of the fuselage, does not present insurmountable balance problems.

Baseline Configuration. - The selected airplane characteristics from the parametric data for the Baseline Configuration are wing loading 5772 N/m^2 (120 lb/ft^2), swept aspect ratio 6.0, takeoff gross weight 136,937 kg (301,894 lb), and a fuel volume ratio of 1.2 Fuel volume ratio is defined as the volume between the wing front and rear spars and between root and tip ribs, divided by the volume required by the mission fuel. Sufficient margin was built-in to this airplane to allow for growth without affecting mission performance capability, thus ensuring that analytical trends would not be distorted.

The complete data for this configuration are shown in column 2 of Tables III and IV.

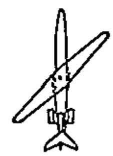
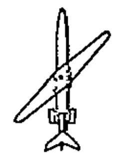
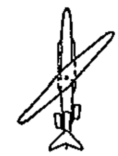
The Baseline Configuration, Figure 5, is a three-engine, high wing airplane designed to cruise at Mach 0.95. The propulsion system consists of three aft-fuselage mounted turbofans, two of which are housed in external nacelles on each side of the rear fuselage and the third, mounted on the airplane centerline at the rear of the fuselage, is supplied with air by means of an 'S'-duct arrangement. Engines are scaled Pratt and Whitney STF 433 turbofans, bypass ratio 6.5. Provision is made for a crew of two on the flight deck and 6 cabin attendants for passenger service. The fuselage passenger compartment has accommodations for 200 passengers arranged in 2, 3 and 4 abreast seating. Containerized baggage and cargo compartments and a service galley are arranged beneath the passenger deck.

TABLE III COMMERCIAL PASSENGER TRANSPORT - CONFIGURATION CHARACTERISTICS

CONFIGURATION		1 	2 	3 
		INITIAL	BASELINE	CYCLED BASELINE
Takeoff Gross Weight - kg (lb)		131,661 (290,263)	136,937 (301,894)	141,128 (311,134)
Component	Parameter			
Fuselage	Body Length - m (ft)	50.08 (166 67)	50 08 (166 67)	50.08 (166.67)
	Cabin Length - m (ft)	36.83 (120.83)	36.83 (120.83)	36.83 (120.83)
	Passenger Mix - FC/TC - %	15/85	15/85	15/85
	Seating - Min/Max Abreast - TC	5/7	5/7	5/7
	No. Aisles	2	2	2
	Fineness Ratio	10.00	10.00	10 00
Wing	Area - m^2 (ft^2)	199.5 (2,148)	224.3 (2,415)	215.7 (2,322)
	Aspect Ratio Swept	7.0	6.0	5.0
	*Pivot Normal Chord - %	38.5	38.5	38.5
	Thickness Ratio Swept Root/Tip %/%	11.02/11.02	11.66/11 66	11.09/11.09
	Taper Ratio	0 33	0.33	0.33
	Pivot Location % Body Length	58.6	58.6	58.6
Empennage	Horizontal Area - m^2 (ft^2)	17.5 (188.26)	37.0 (398)	42.9 (462)
	Aspect Ratio	4.0	4.0	4.0
	Sweep C/4 - rad (deg)	0.70 (40)	0.70 (40)	0.70 (40)
	Taper Ratio	0.4	0.4	0.4
	Volume Coef. V_H	0.442	0.60	0.67
	Thickness Ratio - %	9.5	9.5	9.5
	Vertical Area - m^2 (ft^2)	11.6 (125)	25.9 (279)	35.3 (380)
	Aspect Ratio	1.0	1.0	1 0
	Sweep C/4 - rad (deg)	0 742 (42.5)	0.742 (42.5)	0.742 (42 5)
	Taper Ratio	0.8	0 8	0.8
	Volume Coef. V_V	0.0427	.064	0.101
	Thickness Ratio - %	9.5	9.5	9.5
Propulsion	Engine Type	P&W STF 429	P&W STF 433	P&W STF 433
	No. Engines	3	3	3
	Location	Aft Fuselage	Aft Fuselage	Aft Fuselage
	Uninstalled S T SL Std Day - N (lbf)	126,543 (28,448)	130,497 (29,337)	147,507 (33,161)
	Cruise SFC - kg/hr/N (lb/hr/lb t)	0.0803 (0.788)	0.0788 (0.773)	0.0788 (0.773)

* PIVOT LOCATION % UNSWEPT CHORD AT WING CENTER LINE

TABLE IV COMMERCIAL PASSENGER TRANSPORT - CHARACTERISTICS AND PERFORMANCE

	CONFIGURATION		
Cruise Mach No. 0.95 Payload - 23,768 kg (52,400 lb) * Payload for Initial Configuration - 23,133 kg (51,000 lb) Range - 5560 km (3000 n mi)	1 	2 	3 
QUANTITY/PARAMETER	* INITIAL	BASELINE	CYCLED BASELINE
Takeoff Gross Weight, kg (lb)	131,661 (290,263)	136,937 (301,894)	141,128 (311,134)
Operating Weight, kg (lb)	68,785 (151,645)	71,272 (157,129)	71,824 (158,344)
Fuel Weight, kg (lb)	39,969 (88,118)	41,896 (92,366)	45,536 (100,391)
Wing Area, m^2 (ft^2)	199.5 (2,148)	224.3 (2,415)	215.7 (2,322)
Engine SLS Rating, N (lbf) (Uninstalled)	127,510 (28,448)	130,497 (29,337)	148,634 (33,161)
No. Engines/BPR	3/6.50	3/6.50	3/6.50
Swept Aspect Ratio	7	6	5
Sweep Angle, rad (deg)	0.785 (45)	0.785 (45)	0.785 (45)
Thrust Loading - T/W, N/kg	2.905 (0.294)	2.86 (0.291)	3.16 (0.32)
Wing Loading - W/S, N/m^2 (lb/ft^2)	6,224 (130)	5,772 (120.55)	6,200 (129.5)
Cruise Altitude, m (ft)	10,972.8 (36,000)	11,277.6 (37,000)	11,277.6 (37,000)
Cruise Lift/Drag Ratio - L/D	17.03	16.33	14.93
FAA Takeoff Field Length, m (ft) 305 K (90°F Day), 305 m (1000 ft)	2,580 (8,465)	2,700 (8,860)	2,544 (8,346)
Landing Distance, m (ft) 305 K (90°F Day), 305 m (1000 ft)	2,329 (7,643)	2,163.4 (7,098)	1,890 (6,201)
Approach Speed, km/hr (k) EAS	253.0 (136.6)	240.76 (130)	259.3 (140)

All mission fuel is contained in wing tanks and the wing sweep mechanism is arranged to vary sweep angle from 0 to 0.785 rad (0 to 45 deg).

Since an aft-fuselage mounted propulsion system is desirable to maintain an aerodynamically clean wing, the use of a tee-tail empennage configuration is necessary.

Main landing gears consist of two 4-wheel bogies located on the fuselage to provide a tip-over angle of 1 rad (57.3 deg). Retraction is lateral about a simple pivot and stowage is arranged in the fuselage beneath the passenger deck. A landing gear fairing of minimum size encloses those portions of the landing gear mechanism and support structure outside the fuselage contour.

A single-leg two-wheeled nose gear retracts longitudinally forward beneath the forward portion of the passenger deck.

A high lift system consisting of a single slotted Fowler type flap system occupying a span of 75 percent of wing trailing edge is integrated into the wing so that no portion of the flap mechanism or support structure is exposed to the airflow except during deployment of the flap system. The high lift system is arranged so that deployment of the flaps is possible only at 0 rad (0 deg) of wing sweep angle.

Flight controls comprise the lateral control system and longitudinal and directional controls. Ailerons for lateral control are placed at the outboard 25 percent of the wing trailing edge and flight and ground operable spoilers are provided over the inboard portion of the wing forward of the flaps. Longitudinal control is obtained by means of a movable horizontal stabilizer and elevators and a two piece rudder provides directional control.

Baseline Configuration development. - Development studies for the Baseline Configuration consisted of:

- o Main landing gear arrangement and determination of the cross-section required for the landing gear fairing.
- o Determination of the cross-sectional area distribution of the configuration and remedial action to correct deficiencies.
- o Analysis of the wing pivot structure in fuselage and development of a fuselage interior layout.
- o Analysis of the wing to accurately determine wing weight and the weight penalty required to avoid divergence of the leading wing in the swept configuration.

Main landing gear arrangement. - The landing gear of the Initial Configuration required a large fairing to enclose a gear long enough to provide ground clearance at rotation and a tip-over angle of 1.02 rad (58 deg). The resulting cross section of the fairing was considered unacceptable and design studies were performed to examine the problem from the standpoint of minimizing the length of the gear which in turn would produce a smaller fairing. The gear design for the Baseline Configuration consisted of an arrangement in which the gears retract laterally, using simple rotation about an axis parallel to the airplane centerline and slightly outside the fuselage contour so that, when retracted, the gears occupy space beneath the passenger deck. This is accomplished by first contracting the shock absorber strut by means of an internal system before initiating gear retraction.

Wing pivot support structure and fuselage interior layout. - The design of the wing pivot and supporting structure is similar to that of previous Oblique Wing Concept studies documented in Reference 1. A preliminary structural analysis of the support frames in the fuselage established the depths and widths of the frame structures which transfer the fuselage loads to the wing. The pivot support structure is carried by three frames 0.2 m (8.0 in) deep which encroach upon the fuselage interior and which affect the seating arrangement. The fuselage interior layout of the baseline airplane was arranged for 15 percent first and 85 percent tourist class passengers. In order to accommodate the wing pivot structure, relocation of four (4) passengers in the wing pivot area was required and was accomplished without change to the fuselage design.

Configuration cross-sectional area distribution. - The cross-sectional area distribution for the Baseline Configuration is shown on Figure 6. The area peak is caused by the maximum cross-section of the wing occurring at approximately the same fuselage station as the accumulation of the fuselage and landing fairing maximum areas. The arrangement of the aft engine nacelles placed symmetrically on the rear fuselage caused an area outcrop due to the build-up of area on the rear fuselage.

In order to smooth the area distribution curve, at the same time avoiding contouring the fuselage, filling-in of the forward "bubble" was accomplished by extending the landing fairing forward and aft as shown on Figure 6. The area outcrop on the rear fuselage was removed by relocating the external engine nacelles asymmetrically, also shown on Figure 6, so that the area build-up due to the engine nacelles was distributed over a greater length of fuselage.

Wing weight analysis. - An analysis of the cantilever behavior of the leading wing using the stiffness distribution of the strength designed aluminum swept-aspect ratio 6.0 wing of the Baseline Configuration indicated a weight of approximately 2268 kg (5000 lb) would be incurred to increase the bending stiffness to avoid wing divergence. When applied to an aspect ratio 5.0 wing, however, the analysis showed that adequate stiffness was available to prevent divergence. A swept aspect ratio of 5.0 was therefore selected for the Cycled Baseline Configuration analysis.

Cycled Baseline Configuration. - The analyses and studies of the Baseline Configuration indicated that the principal configurational changes required to produce a feasible airplane consisted of:

- o Limiting wing aspect ratio to 5.0 to avoid wing weight penalties.
- o Extending the main landing gear pod forward to FS 540 and aft to FS 1400 to fill in the area distribution "bubble" forward of the peak.
- o Relocating the external engine nacelles asymmetrically without upsetting airplane balance to eliminate the area outcrop behind the peak of the area distribution curve.

In addition, other changes affecting airplane characteristics consisted of:

- o Elimination of unnecessary conservatism from the airport performance calculations.
- o Relocation of the wing front beam from 12 to 9 percent of the wing chord to increase fuel volume.
- o Increases in tail volume coefficients to reflect configuration peculiarities.

Cycling the baseline airplane through the sizing analysis with the changes described incorporated increased the airplane takeoff gross weight from 136,937 kg (301,894 lb) to 141,128 kg (311,134 lb). Because of the increase in the level of installed thrust, takeoff distance was reduced. The complete data for the Cycled Baseline Configuration are shown on Tables III and IV, column 3, and the configurational changes on Figure 7.

The weight breakdown for the Cycled Baseline Configuration is given on Table V and the balance characteristics are shown on Figure 8.

This configuration was the Commercial Passenger Transport Mission candidate for the evaluation of the Oblique Wing Concept.

Executive Transport Airplane Configuration

The development of the Executive Transport airplane configuration was initially conducted for the selected Preliminary Design Mission, i.e., a payload of 18 passengers and baggage for a range of 7408 km (4000 n mi). As the analysis progressed, the size of the airplane increased such that the carrier-borne Navy alternate missions for this type of airplane were precluded.

Further analyses were conducted to determine the variation of airplane geometry with mission payload and range. The results of the analysis

TABLE V COMMERCIAL PASSENGER TRANSPORT - CYCLED BASELINE CONFIGURATION WEIGHT BREAKDOWN

ITEM	WEIGHT	
	kg	(lb)
WING	12,998	(28,655)
HORIZONTAL STABILIZER	1,172	(2,584)
VERTICAL STABILIZER	925	(2,040)
FUSELAGE	12,940	(28,529)
LANDING GEAR	6,207	(13,684)
NACELLE	2,740	(6,040)
PROPULSION	11,646	(25,675)
AUXILIARY POWER SYSTEM	267	(590)
SURFACE CONTROLS	1,243	(2,740)
INSTRUMENTS	388	(855)
HYDRAULICS AND PNEUMATICS	874	(1,927)
ELECTRICAL	2,144	(4,727)
AVIONICS	781	(1,723)
FURNISHINGS	8,611	(18,983)
AIR CONDITIONING AND ANTI-ICING	2,196	(4,839)
AUXILIARY GEAR SYSTEM	—	—
ARMAMENT	—	—
WEIGHT EMPTY	65,132	(143,591)
FUSELAGE FUEL SYSTEM		
OPERATING EQUIPMENT	6,692	(14,753)
OPERATING WEIGHT	71,824	(158,344)
PAYLOAD	23,768	(52,400)
ZERO FUEL WEIGHT	95,592	(210,744)
FUEL WING	45,536	(100,391)
FUEL FUSELAGE		
GROSS WEIGHT	141,128	(311,134)

indicated that an acceptable carrier-compatible design could be obtained by reducing the mission payload to 14 passengers and the range to 6950 km (3750 n mi). The characteristics of each configuration developed for the Executive Transport airplane are shown on Tables VI and VII.

Fuselage definition. - Examination of the space requirements for executive passenger transportation indicated that a fuselage internal arrangement and cross-section similar to the JetStar Model 1329-6A fuselage would be adequate for the Executive Transport Configuration.

Since fuselage shaping is not required with the Oblique Wing Concept, compromising of the seating arrangement is necessary only in the area of the wing pivot structure.

The fuselage is 2.16 m (85 in) in diameter and has seating for eighteen passengers arranged in two rows on either side of a recessed central walkway which provides the necessary head room as shown on Figure 9. The fuselage arrangement also provides for a crew of two and for comfort facilities and adequate baggage space. Nose and rear fuselage fineness ratios are consistent with cruise at Mach 0.95.

Executive Transport Initial Configuration. - The characteristics chosen for the Initial Configuration from the parametric analytical data were:

- Wing sweep angle — 0.785 rad (45 deg)
- Swept aspect ratio — 5.0
- Wing loading — 4190 N/m^2 (87.5 lb/ft^2)
- Takeoff distance — 1524 m (5000 ft)

Selection of these characteristics provided sufficient margin on second segment climb gradient to allow growth of the airplane during the configuration development studies.

The takeoff gross weight for the Initial Configuration was 32,778 kg (72,264 lb) and the configuration assumed two engines mounted in external nacelles on rear fuselage behind the wing and a tee-tail empennage. Data for this airplane are shown on Tables VI and VII, column 1. Weight and balance investigation revealed relocation of the wing and empennage would be necessary to achieve a balanced configuration.

Executive Transport Baseline Configuration. - Development of the Initial Configuration to correct the balance problem by relocating the wing and empennage and cycling the airplane through the airplane sizing procedures resulted in the Executive Transport Baseline Configuration shown on Figure 10. Resizing the airplane from the Initial Configuration increased the takeoff gross weight from 32,778 kg (72,264 lb) to 34,389 kg (75,816 lb). Data for the Baseline Executive Transport are shown on column 2 of Tables VI and VII.

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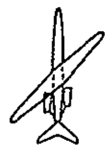
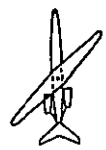
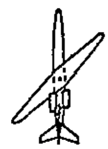
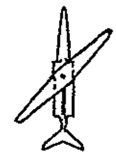
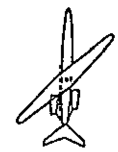
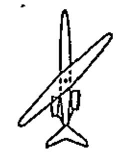
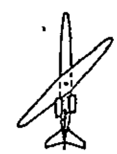
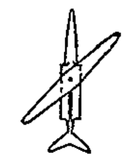
TABLE VI EXECUTIVE TRANSPORT - CONFIGURATION CHARACTERISTICS

CONFIGURATION		1  INITIAL	2  BASELINE	3  CYCLED BASELINE	4  CARRIER COMPATIBLE
Takeoff Gross Weight - kg (lb)		32,778 (72,264)	34,389 (75,816)	36,745.5 (81,010)	30,186 (66,549)
Component	Parameter				
Fuselage	Body Length - m (ft)	21.58 (70.8)	21.85 (71.70)	21.85 (71.70)	22.43 (73.59)
	Cabin Length - m (ft)	13.1 (43)	13.1 (43)	13.1 (43)	11.3 (37)
	Passenger	18	18	18	14
	Seating	2 Rows	2 Rows	2 Rows	2 Rows
	No. Aisles	1	1	1	1
	Fineness Ratio	8.85	10.13	10.13	10.4
Wing	Area - m^2 (ft^2)	73.5 (791)	75.53 (813)	80.64 (868)	65.0 (700)
	Aspect Ratio Swept	5.0	5.0	5.0	5.0
	*Pivot Normal Chord - %	38.5	38.5	38.5	38.5
	Thickness Ratio Swept Root/Tip %/%	13.32/13.32	13.22/13.22	13.22/13.22	13.13/13.13
	Taper Ratio	0.33	0.33	0.33	0.33
	Pivot Location % Body Length	48.4	54.4	56.0	68.5
Empennage	Horizontal Area - m^2 (ft^2)	14.91 (160.5)	15.52 (167.1)	25.0 (269.1)	14.29 (153.8)
	Aspect Ratio	4.0	4.0	4.0	4.0
	Sweep C/4 - rad (deg)	0.70 (40)	0.70 (40)	0.70 (40)	0.70 (40)
	Taper Ratio	0.4	0.4	0.4	0.4
	Volume Coef. V_H	0.53	0.53	0.715	0.526
	Thickness Ratio - %	9.5	9.5	9.5	9.5
	Vertical Area - m^2 (ft^2)	15.63 (168.3)	16.3 (175.2)	21.3 (229.4)	15.48 (166.6)
	Aspect Ratio	1.0	1.0	1.25	1.0
	Sweep C/4 - rad (deg)	0.742 (42.5)	0.742 (42.5)	0.742 (42.5)	0.742 (42.5)
	Taper Ratio	0.8	0.8		
	Volume Coef. V_V	0.101	0.101	0.12	0.101
	Thickness Ratio - %	9.5	9.5	9.5	9.5
Propulsion	Engine Type	P&W STF 433	P&W STF 433	P&W STF 433	P&W STF 433
	No. Engines	2	2	2	2
	Location	Aft Fuselage External Side Nacelles	Aft Fuselage External Side Nacelles	Aft Fuselage Over Wing External Nacelles	Aft Fuselage Integrated
	Uninstalled S T SL Std Day - N (lbf)	54,957 (12,355)	58,707 (13,198)	64,535 (14,508)	52,569 (11,818)
	Cruise SFC - kg/hr/N (lb/hr/lb t)	0.8117 (0.796)	0.0806 (0.791)	0.0800 (0.785)	0.0815 (0.799)

* PIVOT LOCATION % UNSWEPT CHORD AT WING CENTER LINE

TABLE VII EXECUTIVE TRANSPORT - CHARACTERISTICS AND PERFORMANCE

Cruise Mach No. 0.95 Payload - 18 Passengers Range - 7408 km (4000 n mi) * Payload - 14 Passengers Range - 6945 km (3750 n mi)	CONFIGURATION			
	1 	2 	3 	4 
QUANTITY/PARAMETER	INITIAL	BASELINE	CYCLED BASELINE	* CARRIER COMPATIBLE
Takeoff Gross Weight, kg (lb)	32,778 (72,264)	34,389 (75,816)	36,745.5 (81,010)	30,186.0 (66,549)
Operating Weight, kg (lb)	15,989 (35,251)	16,712 (36,844)	17,622.5 (38,851)	15,185.4 (33,478)
Fuel Weight, kg (lb)	15,058 (33,197)	15,946 (35,156)	17,392 (38,343)	13,654.5 (30,103)
Wing Area, m^2 (ft^2)	73.5 (791)	75.53 (813)	80.64 (868)	65.0 (700)
Engine SLS Rating, N (lbf) (Uninstalled)	54,958 (12,355)	58,716.5 (13,200)	64,535 (14,508)	52,569 (11,818)
No. Engines/BPR	2/6.5	2/6.5	2/6.5	2/6.5
Swept Aspect Ratio	5 0	5.0	5.0	5.0
Sweep Angle, rad (deg)	0.785 (45)	0.785 (45)	0.785 (45)	0.785 (45)
Thrust Loading - T/W, N/kg	3.35 (0.342)	3.42 (0.348)	3.51 (0.358)	3.48 (.355)
Wing Loading - W/S, N/m^2 (lb/ft^2)	4,190.0 (87.5)	4,280.5 (89.4)	4,280.5 (89.4)	4,362.0 (91.1)
Cruise Altitude, m (ft)	11,277 (37,000)	11,277 (37,000)	11,277 (37,000)	11,277 (37,000)
Cruise Lift/Drag Ratio - L/D	13.9	13.59	13.21	13.32
FAA Takeoff Field Length, m (ft) 305 K (90°F Day), 305 m (1000 ft)	1,524 (5,000)	1,524 (5,000)	1,524 (5,000)	1,524 (5,000)
Landing Distance, m (ft) 305 K (90°F Day), 305 m (1000 ft)	1,399 (4,590)	1,411 (4,630)	1,407 (4,618)	1,432 (4,700)
Approach Speed, km/hr (k) EAS	177.8 (96.0)	179.5 (96.9)	178.72 (96.5)	183.35 (99 0)

The weight breakdown for the configuration is shown on Table VIII and the balance characteristics on Figure 11.

Baseline Configuration development. - Configuration development of the Baseline Configuration was necessary to eliminate the problems caused by the close proximity of the wing trailing edge, particularly with flaps deployed, and the engine nacelle intakes since masking of the intakes would result in unacceptable degradation of the engine performance. Avoidance of the wing/ engine intake interference was achieved by relocating the engines on pylons on the upper portion of the fuselage slightly behind the wing. Since the engine nacelles in this location would cause masking of the tee-tail horizontal stabilizer, relocation to the fuselage in a low or conventional position was necessary. A reduction in the efficiency of the low tail coupled with a loss of tail arm caused an increase in the area of the surface.

Cycled Baseline Configuration. - Incorporating these changes into the configuration and resizing the airplane resulted in the Cycled Baseline Configuration shown on Figure 12. Due to these changes the takeoff gross weight increased from 34,389 kg (75,816 lb) to 36,745 kg (81,010 lb). Column 3 on Tables VI and VII shows the data for this airplane.

A check of this airplane as a Navy carrier-borne configuration indicated that the constraints of size and weight imposed by carrier operations were exceeded and that further study to ensure Navy carrier compatibility was necessary.

In addition the configuration continued to exhibit design integration problems due mainly to the location of the engine nacelles.

Carrier Compatible Configuration. - The constraint on carrier-borne airplanes is provided by a "foul line" dimension of 15.25 m (50 ft). A span limiting dimension of 24.40 m (80 ft) was chosen because it is a) currently the maximum span of any Navy airplane, and b) the "foul line" constraint at this span provides a "foul line" clearance of 3.0 m (10 ft). It was further determined that the span dimension could be achieved by using up to 0.26 rad (15 deg) of wing sweep for landing without degrading the maximum lift coefficient by more than 3 percent.

At a swept aspect ratio of 5.0 and with the span dimension limited to 24.40 m (80 ft), the parametric data show that the maximum wing area for a Carrier Compatible Configuration is limited to 65.00 m^2 (700 ft^2). The configuration selected from the possible candidates defined by the maximum wing area limitation is that for a range of 6950 km (3750 n mi) which provides transatlantic as well as transcontinental range capability with a payload of 14 passengers.

An airplane configuration that satisfies both the commercial and Navy requirement is shown on Figure 13. The arrangement of the passenger accommodation is similar to that of the Cycled Baseline Configuration and integration of the engines into the rear fuselage is necessary to overcome the design and aerodynamic problems of the configuration. The carrier

TABLE VIII EXECUTIVE TRANSPORT - BASELINE CONFIGURATION WEIGHT BREAKDOWN

ITEM	WEIGHT	
	kg	(lb)
WING	2,063	(4,548)
HORIZONTAL STABILIZER	377	(830)
VERTICAL STABILIZER	405	(893)
FUSELAGE	2,318	(5,110)
LANDING GEAR	1,385	(3,054)
NACELLE	846	(1,866)
PROPULSION	3,364	(7,416)
AUXILIARY POWER SYSTEM	171	(378)
SURFACE CONTROLS	521	(1,149)
INSTRUMENTS	182	(401)
HYDRAULICS AND PNEUMATICS	244	(539)
ELECTRICAL	1,229	(2,710)
AVIONICS	561	(1,235)
FURNISHINGS	1,328	(2,928)
AIR CONDITIONING AND ANTI-ICING	749	(1,651)
AUXILIARY GEAR SYSTEM	6	(14)
ARMAMENT	—	—
WEIGHT EMPTY	15,749	(34,722)
FUSELAGE FUEL SYSTEM		
OPERATING EQUIPMENT	963	(2,122)
OPERATING WEIGHT	16,712	(36,844)
PAYLOAD	1,731	(3,816)
ZERO FUEL WEIGHT	18,443	(40,660)
FUEL WING	15,946	(35,156)
FUEL FUSELAGE		
GROSS WEIGHT	34,389	(75,816)

compatible airplane features long duct engine intakes which, although increasing installation losses and therefore engine size, provide space for additional fuel tankage and landing gear stowage. The data for the configuration are shown on Tables VI and VII, column 4 and the weight breakdown is shown on Table IX. The configuration and mission changes cycled through the airplane sizing procedures produce an airplane having a takeoff gross weight of 30,186 kg (66,549 lb). This configuration was used in the evaluation of the mission/configuration suitability for the Oblique Wing Concept.

Military Cargo Transport Airplane Configuration

Preliminary estimates of airplane size indicated that for a range of 12,965 km (7000 n mi), the propulsion system requirements for the resulting large airplane would render achievement of a practical Oblique Wing configuration almost impossible. The mission range of 6480 km (3500 n mi) was therefore selected for configuration studies.

Fuselage definition. - Data for the fuselage cargo compartment definition which represents a compromise between commercial and military requirements were obtained from past mission-related studies. The fuselage cross section requires a cargo floor 6.4 m (21 ft) wide to accommodate 6.1 m (20 ft) cargo containers loaded cross-wise and a height of 4.42 m (14.5 ft) determined by military equipment requirements. The cargo compartment cross-section, Figure 14, is essentially constant, a feature made possible by the use of the Oblique Wing Concept which eliminates the need for fuselage shaping.

Military Cargo Transport Initial Configuration. - The characteristics for the Initial Configuration determined from parametric analyses and shown on Tables X and XI, column 1, are swept aspect ratio 4.75, wing sweep angle 0.700 rad (40 deg) and cruise wing loading 6225 N/m^2 (130 lb/ft^2) for a takeoff distance constrained to 2440 m (8000 ft).

The takeoff gross weight of the resulting airplane was 608,720 kg (1,342,000 lb) and preliminary checks revealed the existence of severe loadability and balance problems.

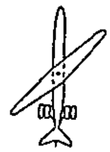
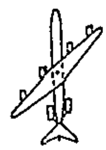
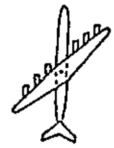
Military Cargo Transport Baseline Configuration. - The thrust requirement per engine for the four (4) engine Initial Configuration exceeded the limit set by engine technology for the airplane time-frame. It was therefore necessary to revise the propulsion system to a six (6) engine arrangement to reduce the thrust per engine required below the technology limit. Resizing the airplane to reflect a six (6) engine configuration produced the Baseline Configuration for which the takeoff gross weight increased from 608,720 kg (1,342,000 lb) to 614,081 kg (1,353,818 lb) which includes the weight increment due to the installation of six smaller engines.

A typical configuration is shown on Figure 15, and the configuration data on Tables X and XI, column 2.

TABLE IX EXECUTIVE TRANSPORT - CARRIER COMPATIBLE CONFIGURATION WEIGHT BREAKDOWN

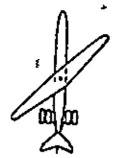
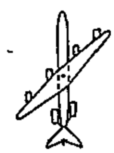
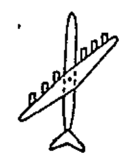
ITEM	WEIGHT	
	kg	(lb)
WING	1,745	(3,848)
HORIZONTAL STABILIZER	342	(753)
VERTICAL STABILIZER	381	(840)
FUSELAGE	2,102	(4,634)
LANDING GEAR	1,225	(2,701)
NACELLE	775	(1,709)
PROPULSION	3,017	(6,653)
AUXILIARY POWER SYSTEM	168	(370)
SURFACE CONTROLS	487	(1,073)
INSTRUMENTS	175	(386)
HYDRAULICS AND PNEUMATICS	229	(505)
ELECTRICAL	1,168	(2,575)
AVIONICS	1,134	(2,500)
FURNISHINGS	587	(1,294)
AIR CONDITIONING AND ANTI-ICING	723	(1,593)
AUXILIARY GEAR SYSTEM	5	(12)
ARMAMENT	—	—
WEIGHT EMPTY	14,263	(31,445)
FUSELAGE FUEL SYSTEM		
OPERATING EQUIPMENT	923	(2,034)
OPERATING WEIGHT	15,185	(33,478)
PAYLOAD	1,346	(2,968)
ZERO FUEL WEIGHT	16,532	(36,446)
FUEL WING	13,655	(30,103)
FUEL FUSELAGE		
GROSS WEIGHT	30,186	(66,549)

TABLE X MILITARY CARGO TRANSPORT - CONFIGURATION CHARACTERISTICS

CONFIGURATION		1 	2 	3 
		INITIAL	BASELINE	CYCLED BASELINE
Takeoff Gross Weight - kg (lb)		608,720 (1,342,000)	614,081 (1,353,818)	574,998 (1,267,653)
Component	Parameter			
Fuselage	Body Length - m (ft)	73.0 (239.5)	73.0 (239.5)	76.2 (250)
	Cargo Compt. Length - m (ft)	49.4 (162)	49.4 (162)	49.4 (162)
	Cargo Compt. Width - m (ft)	6.4 (21)	6.4 (21)	6.4 (21)
	Cargo Compt. Height - m (ft)	4.42 (14.5)	4.42 (14.5)	4.42 (14.5)
	Fineness Ratio	10.1	10.1	10.5
Wing	Area - m^2 (ft^2)	959 (10,323)	937.7 (10,093)	822.3 (8,850)
	Aspect Ratio Swept	4.75	4.75	5.0
	*Pivot Normal Chord - %	38.5	38.5	38.5
	Thickness Ratio Swept Root/Tip %/%	8.5/8.5	8.5/8.5	8.13/8.13
	Taper Ratio	0.33	0.33	0.33
	Pivot Location % Body Length	57	57	48
Empennage	Horizontal Area - m^2 (ft^2)	167.0 (1,797)	184.7 (1,988)	189.8 (2,043)
	Aspect Ratio	4.0	4.0	4.0
	Sweep C/4 - rad (deg)	0.70 (40)	0.70 (40)	0.70 (40)
	Taper Ratio	0.4	0.4	0.4
	Volume Coef. V_H	0.663	0.663	0.627
	Thickness Ratio - %	9.5	9.5	9.5
	Vertical Area - m^2 (ft^2)	116.7 (1,256)	119.3 (1,283.8)	188.7 (2,031)
	Aspect Ratio	1.0	1.0	1.0
	Sweep C/4 - rad (deg)	0.74 42.5	0.74 (42.5)	0.74 (42 5)
	Taper Ratio	0.8	0.8	0.8
	Volume Coef. V_V	0.064	0.064	0.101
	Thickness Ratio - %	9.5	9.5	9.5
Propulsion	Engine Type	P&W STF 433	P&W STF 433	P&W STF 433
	No. Engines	4	6	6
	Location	Aft Fuselage	2-Aft Fuselage 4-Wing	Wing
	Uninstalled S T SL Std Day - N (lbf)	438,150 (98,500)	300,744 (67,610)	277,618 (62,411)
	Cruise SFC - kg/hr/N (lb/hr/lb t)	0.0788 (0.773)	0.0788 (0.773)	0.0788 (0.773)

* PIVOT LOCATION % UNSWEPT CHORD AT WING CENTER LINE

TABLE XI MILITARY CARGO TRANSPORT - CHARACTERISTICS AND PERFORMANCE

Cruise Mach No. 0.95 Payload - Cargo 158,757 kg (350,000 lb) Range 6482 km (3500 n mi)	CONFIGURATION		
	1 	2 	3 
QUANTITY/PARAMETER	INITIAL	BASELINE	CYCLED BASELINE
Takeoff Gross Weight, kg (lb)	608,720 (1,342,000)	614,081 (1,353,818)	574,998 (1,267,653)
Operating Weight, kg (lb)	247,661 (546,000)	251,575 (554,628)	227,359 (501,241)
Fuel Weight, kg (lb)	202,302 (446,000)	203,749 (449,190)	188,881 (416,412)
Wing Area, m^2 (ft^2)	959.0 (10,323)	937.7 (10,093)	822.2 (8,850)
Engine SLS Rating, N (lbf)	438,150 (98,500)	300,744 (67,610)	277,618 (62,411)
No. Engines/BPR	4.0/6.5	6.0/6.5	6.0/6.5
Swept Aspect Ratio	4.75	4.75	5.0
Sweep Angle, rad (deg)	0.70 (40)	0.70 (40)	0.70 (40)
Thrust Loading - T/W, N/kg	2.88 (0.293)	2.94 (0.30)	2.896 (0.295)
Wing Loading - W/S, N/m^2 (lb/ft^2)	6,225 (130)	6,225 (130)	6,655 (139)
Cruise Altitude, m (ft)	11,277 (37,000)	11,277 (37,000)	11,277 (37,000)
Cruise Lift/Drag Ratio - L/D	16.0	15.97	16.2
* Takeoff Field Length, m (ft)	2,440 (8,000)	2,440 (8,000)	2,440 (8,000)
* Landing Distance, m (ft)	—	2,103 (6,900)	1,158 (3,800)
*305 K (90°F Day), 305 m (1000 ft)			
Approach Speed, km/hr (k) EAS.	229.6 (124)	242.6 (131)	247.4 (133.6)

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Military Cargo Transport configuration-development. - The propulsion system for the Initial Configuration was originally envisioned as an aft fuselage-mounted arrangement in order to achieve an aerodynamically clean wing. Preliminary balance analyses, however, indicated this arrangement to be unacceptable both from the airplane balance and cargo loadability standpoint, and aerodynamically because of the build-up of configuration cross-sectional area due to the concentration of engine cross-sectional area. As the analysis of the Initial Configuration showed that six engines would be required, systematic relocation of the engines in pairs from the aft fuselage to the wing was performed and the effect upon the Baseline Configuration determined as shown on Figure 15. Alternate positions for engine location include one on each wing tip arranged to swivel as wing sweep angle changes and two, one on either side of the fuselage, mounted on the forward fuselage. Although improving the balance characteristics, the forward fuselage location was clearly undesirable because of the exhaust gas ingestion by the rear engines.

The forward fuselage engines were then located at the mid semi-span of the leading and trailing wings. Finally all the engines were located on the wing and balance and loadability characteristics checked. The loadability diagram, Figure 16, shows the effect of systematically relocating the engines from the fuselage to the wing. The problems of balance and loadability are solved by mounting all the engines on the wing. The relocation to the wing, however, merely exchanges balance and loadability problems for aerodynamic and system reliability problems.

Military Cargo Transport Cycled Baseline. - Incorporating the configuration changes indicated by the configuration development studies and cycling the airplane through the sizing procedure to reflect the influence of wing leading edge devices, resized the airplane from 614,081 kg (1,353,818 lb) to 574,998 kg (1,267,653 lb) and increased the wing swept aspect ratio from 4.75 to 5.0. The resulting configuration is shown on Figure 17. The characteristics for this configuration are shown on Tables X and XI, column 3, the weight breakdown on Table XII and the balance characteristics on Figure 18.

Although no satisfactory solution for the configuration was found, the Cycled Baseline Configuration was used in the evaluation of the Oblique Wing Concept suitability.

Mission/Configuration Evaluation

The evaluation of each mission/configuration pair was performed on the basis of qualitative assessments of the Cycled Configuration for each of the three Preliminary Design Missions. In the case of the Executive Transport configuration, the original mission was changed in order to achieve a Carrier Compatible Configuration.

TABLE XII MILITARY CARGO TRANSPORT - CYCLED BASELINE CONFIGURATION WEIGHT BREAKDOWN

ITEM	WEIGHT	
	kg	(lb)
WING	78,920	(173,989)
HORIZONTAL STABILIZER	5,057	(11,150)
VERTICAL STABILIZER	3,956	(8,721)
FUSELAGE	35,168	(77,532)
LANDING GEAR	25,200	(55,557)
NACELLE	12,945	(28,538)
PROPULSION	44,159	(97,355)
AUXILIARY POWER SYSTEM	658	(1,451)
SURFACE CONTROLS	4,622	(10,190)
INSTRUMENTS	1,005	(2,216)
HYDRAULICS AND PNEUMATICS	2,154	(4,748)
ELECTRICAL	1,837	(4,049)
AVIONICS	1,088	(2,400)
FURNISHINGS	3,222	(7,101)
AIR CONDITIONING AND ANTI-ICING	2,088	(4,603)
AUXILIARY GEAR SYSTEM	107	(235)
ARMAMENT	—	—
WEIGHT EMPTY	222,186	(489,835)
FUSELAGE FUEL SYSTEM		
OPERATING EQUIPMENT	5,177	(11,415)
OPERATING WEIGHT	227,363	(501,250)
PAYLOAD	158,757	(350,000)
ZERO FUEL WEIGHT	386,120	(851,250)
FUEL WING	188,878	(416,403)
FUEL FUSELAGE		
GROSS WEIGHT	574,998	(1,267,653)

Military Cargo Transport concept evaluation. - A combination of a large payload, 158,757 kg (350,000 lb), and long range, 6480 km (3500 n mi), together with some stringent takeoff performance requirements, 2440 m (8000 ft) at altitude and elevated temperature, produces an airplane having a takeoff gross weight of 574,998 kg (1.267×10^6 lb). Due to the limitations imposed by engine technology, the maximum thrust/engine obtainable for the airplane time-frame has been assessed as 291,343 N (65,000 lbf). At this thrust level the Military Transport Configuration requires a minimum of six engines.

Locating the propulsion system mass at the rear of the fuselage, which is the best arrangement for an aerodynamically clean wing, results in an untenable balance situation as shown by Figure 16. Although the problem can be alleviated by relocating the engines on the wing such that normal balance conditions prevail, the change would merely substitute the balance problem for problems of functional reliability and aerodynamic efficiency. These new problems would arise because of the need to swivel each engine to ensure thrust line symmetry and alignment with airplane centerline during wing sweep-angle change. Thus, functional reliability is critical since the failure of any engine to maintain correct alignment during wing sweep variation could be catastrophic. Highly redundant systems would, therefore, be required to avoid such failures and would necessarily result in weight penalties.

Designing the airplane to cruise at M = 0.95 requires each nacelle/pylon/wing interface to be individually tailored in order to eliminate interference drag. Configuring the interface to be aerodynamically efficient at the cruise condition would, therefore, incur aerodynamic penalties at all other sweep angles which would adversely affect takeoff performance and off-design range capability.

Propulsion system design is critical to the development of a practical configuration and airplane maximum size would appear to be constrained by limitations on thrust per engine, the number of engines required, and the location of the engines.

Because of the problems described, this mission/configuration combination is considered to be unsuitable for the Oblique Wing Concept.

Executive Transport concept evaluation. - The parametric analysis of the Executive Transport configuration indicated that the wing area for minimum takeoff gross weight was constrained by airport and cruise performance matching, resulting in a wing having insufficient volume to contain the mission fuel. The data show that 70 to 80 percent of the required fuel can be contained in the wing so that the remainder of the mission fuel must be carried in fuselage tanks.

To achieve a balanced configuration requires the wing and engine masses and the fuselage fuel to be placed in close proximity. Assuming externally mounted engine nacelles, locating the nacelles on the aft fuselage below and behind the wing is precluded because of the masking effects of the deployed flap on the engine intakes. In addition since the size of the interacting components are of similar dimensions. i.e., the rear fuselage and nacelle diameters and lengths, integration of the components to avoid cross-sectional area build-up is difficult. Locating the engines above the rear fuselage and behind the wing impacts not only on the cross-sectional area distribution but also the empennage. Because of the masking effects of the engine nacelles in this location the horizontal stabilizer must be relocated to a low-tail configuration on the rear fuselage. The proximity of the nacelles and vertical stabilizer makes a substantial increase in vertical stabilizer area necessary. The result of relocating the engines as outlined above is to produce a substantial increase in airplane takeoff gross weight.

The most acceptable solution to these problems would entail integration of the engines into the rear fuselage and would require long ducts on the fuselage side beneath the wing as shown on Figure 13 to supply engine intake air. Integration of the airframe and propulsion system is considered possible but a number of problem areas such as the placement and housing of the landing gear, the disposition of the fuselage fuel, the proximity of the wing lower surface and the upper external face of the intake duct, the effect of long intake ducts on engine performance, the reduction of flap area and span on airport performance and wing area, are unresolved.

This mission and configuration is considered to have high potential for the application of the Oblique Wing Concept subject to the resolution of the problems outlined above.

Commercial transport concept evaluation. - In comparison to the previous configurations, the Commercial Passenger Transport Mission consisting of a 200-passenger payload and a range of 5560 km (3000 n mi) offers the best mission application for the Oblique Wing Concept.

The configuration developed for the mission is easily balanced, has a practical center-of-gravity range and good loadability. The size of the interacting components is such that sufficient space exists to permit component integration without difficulty. The encroachment of the wing pivot support frames on the fuselage interior does not unduly influence the interior seating arrangement and the need to minimize the landing gear fairing cross-sectional area does not impose constraints upon the landing gear other than to require contraction of the gear strut before retraction. Deficiencies in the configuration area distribution are easily rectified by increasing the length of the landing gear fairing forward of the gear and by staggering the external engine nacelles laterally to avoid fuselage shaping.

A proximity problem associated with the wing trailing edge and the external nacelles exists particularly when the flaps are deployed. In the cruise configuration, design of the intakes must account for the wing downwash angularity and relocation of these nacelles upward out of the wing downwash field will tend to minimize downwash effects. Deploying the flaps, however, would produce discontinuities at the fuselage which would tend to generate vortices so that the intake design must be able to accommodate vortex swirl angularities. It, therefore, becomes important to suppress vortex formation by minimizing discontinuities and limiting inboard flap angle so that airflow effects, as far as the engine intake is concerned, are downwash related rather than vortex related.

Mission/Configuration Selection

The results of the mission/configuration evaluation indicate that the best mission/configuration for the application of the Oblique Wing Concept is the 200-passenger, 5560 km (3000 n mi) range because of the following:

- Freedom from design integration problems.
- Freedom from balance and loadability problems.
- Propulsion system within the technology limitations and close to the base engine characteristics.

The Commercial Passenger Transport Mission was therefore selected as the Final Design Mission and the related configuration used to define the Final Configuration.

Final Configuration

The aeroelastic analysis, reported in Appendix B indicated that for swept-wing aspect ratios up to 6.0 no significant structural weight penalties for divergence prevention were incurred. In view of this, and with the concurrence of the NASA, the swept aspect ratio for the Final Configuration was increased from 5.0 for the Cycled Baseline Configuration to 6.0. Further cycling of the configuration through the airplane sizing procedure resulted in a reduction in the takeoff gross weight from 141,128 kg (311,134 lb) to 139,453 kg (307,441 lb) and in the following characteristics for the Final Configuration:

- Wing loading — 6057 N/m^2 (126.5 lb/ft^2)
- Swept aspect ratio — 6.0
- Wing sweep angle — 0.785 rad (45 deg)

FINAL CONFIGURATION DESIGN

The design of the Final Configuration reflects the results of the aeroelastic analysis of the wing, Appendix B, performed by the NASA Ames Research Center using structural data supplied by the Lockheed-Georgia Company.

Final Configuration Description

The Oblique wing Final Configuration is a three (3) engine, transcontinental range, high speed, pressurized commercial transport with provision for a flight crew of two, pilot and copilot, a cabin crew of 6 attendants and a maximum payload consisting of 200 passengers together with their baggage and 4536 kg (10,000 lb) of cargo.

The configuration, shown on Figure 19, features a high wing having the capability to vary wing sweep angle from 0 rad (0 deg) to 0.785 rad (45 deg), a tee-tail empennage, and is powered by three (3) Pratt and Whitney STF 433 type BPR 6.5 turbofans, each developing 135,235 N (30,402 lbf) of static thrust at sea level standard day conditions. The airplane is designed to cruise at a Mach number of 0.95 at an altitude of 11,277 m (37,000 ft). The wing area in the swept configuration is 217.78 m^2 (2344 ft^2), and the wing swept aspect ratio is 6.0.

All the mission fuel is contained in integral tanks in the wing. A trailing edge high lift device consisting of a single slotted Fowler type flap is provided on the wing and is operative only when the wing is in the unswept position. A retractable landing gear consisting of two 4-wheel bogie main gears and a two-wheel single strut nose gear provides 0.21 rad (12 deg) of ground clearance for rotation and are located to give a tip-over angle of 1.012 rad (58 deg).

The complete data for the configuration are shown in Tables XIII and XIV.

Fuselage interior arrangement. - Current wide-body standards of comfort are used in the arrangement of the accommodation on the passenger deck, Figure 20, which has a seating-split of 15 percent first and 85 percent tourist class passengers. The arrangement of the seating is slightly compromised by the presence of the wing pivot support frames but the effect is minimal. The lateral seating, arranged for two aisles, permits a maximum of seven (7) abreast at a longitudinal spacing of 0.86 m (34 in) and where possible, is staggered longitudinally to improve passenger movement.

Access to the cabin is by means of three doors on each side of the fuselage, of which two are for normal operations at passenger terminals. Comfort stations are provided at the forward and aft ends of the passenger deck and galley facilities are provided amidship below the passenger deck immediately forward of the main landing gear stowage compartment.

TABLE XIII FINAL CONFIGURATION - CONFIGURATION CHARACTERISTICS

Takeoff Gross Weight - kg (lb)		139,453 (307,441)
Component	Parameter	
Fuselage	Body Length - m (ft)	51.31 (168.83)
	Cabin Length - m (ft)	36.83 (120.83)
	Passenger Mix - FC/TC - %	15/85
	Seating - Min/Max Abreast - TC	5/7
	No. Aisles	2
	Fineness Ratio	10.03
Wing	Area - m^2 (ft^2)	217 78 (2344)
	Aspect Ratio - Swept	6.0
	*Pivot Normal Chord -%	38.5
	Thickness Ratio - Swept Root/Tip %/%	11.34/11.34
	Taper Ratio	0.33
	Pivot Location % Body Length	57.85
Empennage	Horizontal Area - m^2 (ft^2)	42.0386 (452.5)
	Aspect Ratio	4.0
	Sweep C/4 - rad (deg)	0.70 (40)
	Taper Ratio	0.4
	Volume Coef. V_H	0.705
	Thickness Ratio - %	9.5
	Vertical Area - m^2 (ft^2)	39.3 (422.7)
	Aspect Ratio	1.0
	Sweep C/4 - rad (deg)	0.742 (42.5)
	Taper Ratio	0.8
	Volume Coef. V_V	0.101
	Thickness Ratio - %	9.5
Propulsion	Engine Type	P & W STF 433
	No. Engines	3
	Location	AFT FUSELAGE
	Uninstalled S T SL Std Day - N (lbf)	135,235 (30,402)
	Cruise SFC - kg/hr/N	0.0796
	(lb/hr/lb t)	(.781)

* Pivot Location % Unswept Chord at Wing Center Line

TABLE XIV FINAL CONFIGURATION - CHARACTERISTICS AND PERFORMANCE

Cruise Mach No. 0.95
Payload - 23,768 KG (52,400 LB)
Range - 5560 km (3000 n mi)

QUANTITY/PARAMETER	
Takeoff Gross Weight, kg (lb)	139,453 (307,441)
Operating Weight, kg (lb)	72,184 (159,137)
Fuel Weight, kg (lb)	43,501 (95,904)
Wing Area, m^2 (ft^2)	217 76 (2344)
Engine SLS Rating, N (lbf)	135.235 (30,402)
(Uninstalled)	135,235 (30,402)
No. Engines/BPR	3/6.50
Swept Aspect Ratio	6.0
Sweep Angle, rad (deg)	0.785 (45)
Thrust Loading - T/W, N/kg	2,909 (0.297)
Wing Loading - W/S, N/m^2 (lb/ft^2)	6057 (126.5)
Cruise Altitude, m (ft)	11,277 (37,000)
Cruise Lift/Drag Ratio - L/D	16 05
FAA Takeoff Field Length, M (ft) 305 K (90° F Day), 305 m (1000 ft)	[illegible]3.8 (8149)
Landing Distance, m (ft) 305 K (90° F Day), 305 m (1000 ft)	1924.5 (6314)
Approach Speed, km/hr (k) EAS	259.28 (140 0)

Access to the galley from the passenger deck is by means of personnel/service cart elevators. Space is provided in below-deck compartments at the forward and aft end of the fuselage for containerized baggage and cargo. Service areas for maintenance operations are also provided below the passenger deck. The airplane is provided with air conditioning, pressurization and humidity control systems for passenger comfort.

Flap mechanisms. - Fowler-type single-slotted flaps are installed on the trailing edges of the wings. The flaps are divided into three spanwise segments for each wing semi-span as shown on Figure 21. The flaps, with support and mechanisms, when in the retracted position, are housed completely within the wing contours. When deployed, flap translation provides an increase in the wing area by extending the wing chord an average of 19 percent. Rotation of the flap provides a maximum deflection of 0.70 rad (40 deg).

Each flap segment has two box-rail tracks whose motion is translational with the flap panel. The tracks traverse fixed rollers mounted on each side of the wing rear spar by means of support fittings. The box-rails and screw actuators extend through the wing rear spar and are enclosed by a cover in the fuel tank. The box-rails are driven by screw actuators connected to the forward end of each box-rail. The leading edge of the flap panel is connected to the aft end of the box-rail track by means of a pivot fitting which allows rotation of the flap. Each flap panel is rotated by a flap link which receives power through rack and pinion devices to cause translation of a carriage inside the box-rail in such a manner that the ratio of flap movement to actuator and rail motion is 3:1. Small hinged panels attached to the fixed trailing edge and aligned with the box-rail tracks allow the rail tracks to extend beyond the wing contours during flap system deployment.

Propulsion system description. - The propulsion system for the airplane consists of three (3) 135,235 N (30,402 lbf) thrust, bypass ratio 6.5, Pratt and Whitney 433 type turbofans. The propulsion system configuration is an aft fuselage mounted arrangement in which two of the engines are mounted in external nacelles and the third on the centerline at the aft end of the fuselage. The center engine is supplied with air through an 'S' duct having the intake on top of the rear fuselage forward of and integrated with the vertical stabilizer.

The external nacelles are mounted on aerodynamically shaped pylons designed to provide constant channel area and are located to minimize the effects of the flow field from the wing downwash in the vicinity of the engine inlets. The external nacelles, Figure 22, are acoustically treated to meet the noise constraint of FAR 36 reduced by 9 and 6 EPNdB on takeoff, sideline and flyover respectively, and 4 EPNdB on approach flyover. The acoustic design consists of a splitter and wall treatment in each inlet, a splitter and wall treatment on both walls of the secondary flow duct, and wall treatment in the exhaust nozzle duct.

Fuel system. - All fuel is contained within the wing primary box structure with a dry bay in the pivot area above the fuselage. The fuel system schematic is shown on Figure 23. The total fuel volume is divided into 3 equal volume main tanks. Each of the two outboard tanks are divided into three compartments and serve as main tanks for engines No. 1 and 3. The center tank, which consists of a compartment on each side of the dry bay, is interconnected by a gravity feed system to form a single tank. Each tank contains primary and standby pumps, either of which are capable of feeding two engines at takeoff power as well as powering ejectors for fuel compartment sequencing and vent system scavenging.

The compartment sequencing for each of the outboard tanks is from inboard to outboard so that the last of the fuel used from those tanks is from the outboard compartments. The center tank has a reservoir compartment for a pump and the remainder of the tank is depleted at a common level.

The vent systems for all three tanks terminate at a common vent box approximately 4.6 m (15 ft) from the left wing tip.

A refueling system is provided consisting of two refueling adapters located in the landing gear fairing and refueling valves in each tank. The refueling rate is approximately 3785 1/min (1000 gal/min) and defuel capability is also provided. A jettison system is limited to the center tank only.

The APU is supplied with fuel from the center engine feed line.

A crossfeed system is included as shown on Figure 23 which provides the capability of feeding any engine from any tank or combination of tanks in the event of a failure of a single crossfeed valve. The feed lines for the three engines and the refuel-defuel line pass from the wing to the fuselage through swivel joints at the wing pivot.

Emergency shutoff valves are provided at both the exit point from the tank and at the engine firewall for each of the feed lines for the engines.

Structural Description

Fuselage structure. - The fuselage structure is shown in Figure 24 and is designed to maximize the use of composite filamentary materials. The maximum diameter of the fuselage is 5.13 m (16.833 ft), and the overall length is 51.31 m(168.83 ft). The fuselage is subdivided into five major segments: the crew compartment from FS 137 to 415; the forward passenger compartment from FS 415 to 1167; the barrel section from FS 1167 to 1307; the aft passenger compartment from FS 1307 to 1740; and the rear fuselage from FS 1740 to 2106.

The shell structure consists of several integrally molded panel assemblies spliced with mechanical fasteners. Each panel consists of graphite-epoxy skins and longitudinal stiffeners together with Kevlar 49 frame-to-skin attachment clips. Titanium shims are embedded in the edges of the panels and in other areas to provide sufficient bearing strength for the mechanical fasteners. Splices occur longitudinally at each segment edge and circumferentially at the upper and lower centerlines, at floor level, and along a line coincident with the upper edge of the entry doors. The rear fuselage splices are located each side of the fuselage along the maximum halfbreadth line.

Window and entry-door cutouts having molded edge members are provided in the passenger compartment side panels. The external surface of the fuselage shell is covered with aluminum wire mesh for lightning protection.

The fuselage shell is supported by molded graphite-epoxy ring frames spaced at 0.51 m (1.67 ft) intervals, which are mechanically fastened to the shell. The flight deck floor is constructed from Kevlar honeycomb panels supported by a grid of graphite-epoxy intercostals, beams, and longerons. The main passenger floor, from FS 290 to 1722 at WL 180, which consists of graphite-epoxy floor-skins, stiffeners, edge members and aluminum seat tracks is supported by graphite-epoxy transverse beams at each frame location. Additional graphite-epoxy floors are provided at WL 115 and WL 105 between FS 415 and 1237 and FS 1347 and 1602 in the baggage and galley compartment.

Cargo and baggage related equipment such as roller channels and restraint rails are aluminum. Cargo loading and passenger entry doors are of graphite-epoxy construction with metallic hinge and latching mechanisms. The aft passenger compartment terminates in a graphite-epoxy hemispherical pressure bulkhead located at FS 1722. The nose wheel-well is 1.524 m (5 ft) wide between FS 290 to FS 415, and the structure consists of side walls, an upper bulkhead, forward and aft pressure bulkheads and a nose landing gear drag-link support bulkhead, all of which are integrally molded graphite-epoxy structures. The wheel well area is protected from debris damage by Kevlar 49 shield assemblies.

The three main frames at FS 1167, 123 and 1307, which support the wing pivot, are each provided with rollers and fittings which engage a circular track mounted on the lower surface of the wing. The frame at FS 1237 also contains a bearing housing for the wing pivot pin. Intercostals located each side of the upper centerline between FS 1167 and 1237 transfer wing drag loads into the fuselage shell.

The lower ends of the frames at FS 1237 and 1307 are provided with bearings for the main landing gear trunnions.

An underfloor beam at FS 1272 provides reaction points for the landing gear side loads. All main frames and the associated under-floor beams are constructed from aluminum. Landing gear drag and braking loads are transferred into the shell structure by means of graphite-epoxy external longerons. A graphite-epoxy keel beam located on the aircraft centerline under the floor transfers fuselage bending loads through the main gear stowage area and provides support for the landing gear fairing doors.

The main landing gear area is isolated from the pressurized passenger compartment by means of pressure bulkheads located under the floor at FS 1220 and 1347, and by a horizontal pressure bulkhead beneath the floor support beams. These bulkheads are of graphite-epoxy construction and are protected from damage by Kevlar shield assemblies.

The rear fuselage structure includes provision for mounting the three engines and the empennage. The external nacelles are mounted on pylons at FS 1740, 1788, 1859 and 1907. These structures are of graphite-epoxy construction with aluminum fittings at the pylon attachment points. The center engine is suspended from a graphite-epoxy box structure extending from FS 2005 to 2106. The upper surface of this box extends forward to FS 1927 to provide a mounting plane for the vertical stabilizer. The bulkhead supporting the vertical stabilizer rear spar at FS 2005 is an aluminum structure and serves as a termination point for the engine inlet duct. The bulkhead supporting the vertical stabilizer front spar at FS 1927 is a hybrid structure in which the upper half is of aluminum construction and the lower half of graphite-epoxy construction. The upper portion of the rear fuselage between FS 1788 and 1927 is provided with a large aperture to permit penetration of the inlet duct assembly. This aperture is edged with graphite-epoxy longerons which provide a mounting face for the dorsal fairing structure. To improve the torsional capability in this portion of the fuselage, a graphite-epoxy shear web located beneath the duct between FS 1788 and 1927 is provided.

Wing structure. - The general arrangement of the wing structure is shown in Figure 25. The wing is attached to the fuselage by means of a pivot bearing and a circular track attached to the lower surface of the wing. The bearing is inserted into the fuselage bearing housing and the track is supported by a series of rollers affixed to the fuselage structure. The wing is designed to be pivotable through an angle of 0.785 rad (45 deg).

The wing consists of left and right outer structural boxes which contain the fuel tanks, a center box structure housing the pivot, leading and trailing edges, tips, flaps, spoilers, and ailerons.

Each outer box structure is designed to carry fuel and consists of upper and lower covers, front and rear spars, ribs, fuel bulkheads and access doors. Each cover is an integrally molded assembly of graphite-epoxy skins, spanwise stiffeners and Kevlar rib attachment clips. Titanium shims are embedded into the composite material at the root splices and in other areas where bearing strength is required for mechanical fasteners. Access into the deeper portion of the wing is by a series of molded Kevlar removable clamp type doors in the wing upper surface and by elongated access panels, molded from Kevlar, bolted into the lower surface. The front and rear spars, at 9 and 65 percent chord, respectively, are each molded assemblies of graphite-epoxy caps, webs and stiffeners mechanically attached to covers. Holes are provided in the rear spar web to permit penetration of the flap track mechanism. Truss type ribs and stiffened web type fuel bulkheads are used throughout the wing structure. Rib spacing is approximately 0.64 m (2.1 ft) and are of molded graphite-epoxy structures.

Construction of the center box structure which consists of covers, spars and ribs, is similar to that of the outer wing structure. This portion of the wing is dry and internal access is by means of a clamp type door in the upper surface. Rib spacing in the center box is 0.56 m (1.84 ft). A portion of each front and rear spar at the airplane centerline is fabricated from aluminum to facilitate splicing of the track internal support ring assembly. The ring assembly, which is aligned with an external circular track, consists of graphite-epoxy webs and stiffeners and aluminum caps, and is mechanically attached to the covers, to the spars, and to the root ribs through an aluminum stub fitting.

The external track is segmented and constructed from machined titanium forgings. The pivot pin fitting is an aluminum machined forging. Both the track and pivot pin fittings are mechanically attached to the box structure.

The leading edge consists of nine (9) segments in each outer wing and two (2) in the center wing. Each segment is an integrally molded assembly of graphite-epoxy skins and beaded chordwise stiffeners supported by graphite-epoxy nose ribs which are mechanically attached to the box structure.

The fixed trailing edge structure is comprised of three (3) spanwise segments in each outer wing and one (1) in the center wing. Each outer segment consists of an upper cover and removable lower panels. Upper covers are molded assemblies of graphite-epoxy skin, spanwise beaded stiffeners and shallow ribs, and are recessed to facilitate spoiler installation. Each rib is supported by a removable tubular strut. Removable lower panels are also of molded graphite-epoxy construction. The aileron shrouds which are molded graphite-epoxy structures are supported by graphite-epoxy aileron hinge brackets. Integral titanium bushings are provided at each hinge point. The center section trailing edge consists of upper and lower covers, ribs, spar and trailing-edge section. The assembly covers consist of molded skins and spanwise beaded stiffeners and are supported by molded graphite-epoxy truss type ribs and by a trailing edge spar of molded graphite-epoxy. The trailing-edge section is triangular in cross-section and is a molded assembly of graphite-epoxy skin and chordwise beaded stiffeners. All fixed trailing-edge structure is assembled to the box structure with mechanical fasteners.

Each wing tip is an integrally molded assembly of graphite-epoxy and Kevlar and is attached to the box structures with mechanical fasteners.

The spoilers are of honeycomb construction with graphite-epoxy face sheets and aluminum hinge and actuator fittings.

Each flap segment consists of an upper cover assembly, a lower cover and related hinge fittings. The upper cover is a single piece integrally molded assembly of graphite-epoxy skin, spars and ribs, and the lower cover, which is also graphite-epoxy, is attached to the upper assembly with mechanical fasteners. All hinge fittings are aluminum.

The exposed external surfaces of the entire wing box structure, leading edges, trailing edge, tip, flaps, ailerons, and spoilers are covered with aluminum wire mesh bonded in place to provide lightning protection.

Empennage structure. - The empennage, which is a tee-tail configuration, consists of a fixed vertical stabilizer with a movable horizontal stabilizer mounted at the tip as shown on Figure 26.

The structural arrangement of the horizontal stabilizer features spars located at 12 and 65 percent of the chord, ribs spaced at approximately 0.53 m (1.75 ft), and stiffeners spaced at 7.62 cm (3.0 in). The structure consists of left, right, and center box primary structures, leading and trailing edges, tips, fairing, and elevators. The outer and center boxes are spliced with straps and mechanical fasteners. Each outer box structure is an assembly of upper and lower surface panels, front and rear spars, and ribs, all fabricated from graphite-epoxy material. The center box structure is an assembly of upper and lower surface panels, front and rear spars, root ribs, and actuator and pivot fittings. The construction of the center box is similar to that of the outer structure, as are the materials used. The leading edges, fixed trailing edges, tips, fairing, and elevators consist of skins and bead type stiffeners fabricated from Kevlar -49 material. The elevator hinge brackets are integrally molded graphite-epoxy structures with titanium bushings embedded at the hinge points. Elevator skins and spars are each integrally molded from graphite-epoxy material.

The structural arrangement of the vertical stabilizer features spars located at 10 and 58 percent of the vertical stabilizer chord, a rib spacing of approximately 0.51 m (1.67 ft), and a stiffener spacing of 7.62 cm (3.0 in). The structure consists of a primary box beam, pivot and actuator fittings, leading edge and fairing, fixed trailing edge, and a two piece rudder. Materials and fabrication techniques for the structure are similar to those of the horizontal stabilizer for corresponding structural components. The vertical stabilizer is attached to the fuselage by means of splice plates and straps using mechanical fasteners. The pivot fitting materials are aluminum, titanium and steel. The exterior surfaces of the empennage box structures are covered with aluminum mesh for protection against lightning strike.

Landing gear description. - The landing gear flotation characteristics are obtained using a two-strut main landing gear configuration. Each main gear strut has four wheels and each is arranged to retract laterally by rotation around a simple pivot. The main gear arrangement, Figure 27, provides a stroke to full closure of 0.4 m (1.3 ft). Contraction of the strut is required prior to the initiation of the retraction cycle to permit stowage in the fuselage beneath the passenger deck. An internal system reduces the strut length by 25.4 cm (10 in) which minimizes the size of the landing gear fairing required. Hard surface flotation consistent with airport facilities of the 1980 - 1990 time frame is obtained with four (4) 44 x 16 Type VII tires on each wheel.

The nose gear is a two (2) wheeled, single strut arrangement having two 44 x 16 Type VII tires to ensure commonality with the main gear. The nose gear retracts forward in the vertical plane and is stowed in a compartment below the forward section of the passenger deck and is steerable.

Final Configuration Weight, Balance and Inertia

Final Configuration weight breakdown. - The weight breakdown for the Final Configuration is shown on Table XV. The structure weight reflects the extensive use of filamentary composite materials.

Center-of-gravity travel. - The center-of-gravity travel diagram is shown on Figure 28 for the weight distribution of Table XV. The balance computer program used establishes configuration loadability and calculates fuel tank volume and fuel-burn sequence.

Final Configuration inertias. - The inertia data for the configuration for various payloads and fuel combinations for wing sweep angles of 0 rad (0 deg) and 0.785 rad (45 deg) are shown on Figures 29 and 30.

Final Configuration Performance

Performance calculations are based upon the standard methods for commercial aircraft and on the installed performance data scaled from the P&W STF 433 engine.

Mission profile. - The mission profile shown on Figure 31 is typical for the operation of the airplane for the design mission and for the off-design performance estimates. The mission segments consist of takeoff and climb to 457 m (1500 ft), sweeping the wing to the desired angle and acceleration to climb speed, climb enroute to cruise altitude, cruise at the Mach number consistent with the wing sweep angle at the cruise altitude, and descend and land at destination. Range credit is taken only for the enroute climb and cruise segments. Fuel reserve allowances are those for international flight.

TABLE XV FINAL CONFIGURATION - WEIGHT BREAKDOWN

ITEM	WEIGHT kg	WEIGHT (lb)
WING	14,567	(32,114)
HORIZONTAL STABILIZER	1,151	(2,538)
VERTICAL STABILIZER	994	(2,190)
FUSELAGE	12,933	(28,513)
LANDING GEAR	6,172	(13,608)
NACELLE	2,508	(5,530)
PROPULSION	10,692	(23,572)
AUXILIARY POWER SYSTEM	266	(587)
SURFACE CONTROLS	1,207	(2,662)
INSTRUMENTS	396	(872)
HYDRAULICS AND PNEUMATICS	563	(1,241)
ELECTRICAL	2,134	(4,705)
AVIONICS	1,089	(2,400)
FURNISHINGS	8,636	(19,040)
AIR CONDITIONING AND ANTI-ICING	2,192	(4,832)
AUXILIARY GEAR SYSTEM	—	—
ARMAMENT	—	—
WEIGHT EMPTY	65,500	(144,402)
FUSELAGE FUEL SYSTEM		
OPERATING EQUIPMENT	6,684	(14,735)
OPERATING WEIGHT	72,184	(159,137)
PAYLOAD	23,768	(52,400)
ZERO FUEL WEIGHT	95,952	(211,537)
FUEL WING	43,501	(95,904)
FUEL FUSELAGE		
GROSS WEIGHT	139,453	(307,441)

Takeoff performance. - FAR takeoff field length is shown on Figure 32. The performance is computed for a 305 K (90°F) (ISA + 17.22°C) day at an airfield elevation of 305 m (1000 ft) and at a flap setting of 0.44 rad (25 deg). At a weight of 139,453 kg (307,441), i.e., the weight required to meet the X-point for the design mission, the takeoff field length is 2484 m (8149ft).

Enroute climb performance. - An investigation of the fuel saving potential of climbing to cruise altitude at a wing sweep angle other than that for the design cruise Mach number, 0.95, showed the specific range during climb worsens as sweep angle decreases as shown on Figure 33. Enroute climb is therefore performed at the maximum sweep angle of 0.785 rad (45 deg). Climb is performed at the speed for maximum rate-of-climb, and the climb time, distance and fuel to the cruise altitude of 11,277 m (37,000 ft) are 24.4 minutes, 329 km (178 n mi) and 4085 kg (9000 lb), respectively. The specific range during climb is 0.081 km/kg (0.01975 n mi/lb).

Cruise performance. - The cruise performance for the design and off-design mission capability is calculated at constant Mach number and altitude using the drag data of Figure 34. Cruise altitude for design and off-design cruise is 11,277 m (37,000 ft).

Payload-range. - The payload-range capability of the Final Configuration is summarized on Figure 35 for cruise at Mach 0.95. The X-point range of 5560 km (3000 n mi) is performed at constant cruise altitude and the wing maximum volume is sufficient to provide a Y-point capability of 20,185 kg (44,500 lb) of payload for a distance of 7149 km (3860 n mi). The corresponding block fuel and time data are also shown on Figure 35.

Descent and landing. - Descent is performed assuming the airplane to be above the destination airport so that no range credit is taken for descent.

Approach speed for landing is 259.3 km/hr (140 k) EAS and the landing distance at an airport elevation of 305 m (1000 ft) for 305 K (90°F) day at a flap deflection of 0.70 rad (40 deg) is 1925 m (6314 ft).

Endurance performance. - The endurance capability for the Final Configuration is shown on Figure 36 for sweep angles of 0 rad (0 deg) and 0.785 rad (45 deg) for loiter on three (3) engines. A typical endurance mission initiated at a gross weight of 129,274 kg (285,500 lb) and terminated at a gross weight of 85,593 kg (188,700 lb) has an endurance capability of 8.75 hours with the wing in the cruise configuration, i.e., 0.785 rad (45 deg). Unsweeping the wing to 0 rad (0 deg) increases the endurance to 12.6 hours and produces a 44 percent increase in endurance capability. These data assume 5 percent conservatism in fuel flows to conform to MIL-C-5011A. The optimum loiter altitude is approximately 10,973 m (36,000 ft) and the average loiter Mach number is 0.6.

Final Configuration off-design performance. - The estimated performance of the Final Configuration, Figure 19, is shown on Figure 37 as a function of wing sweep angle. These data show the variation of Mach number, range, cruise L/D, SFC, cruise drag coefficient and aspect ratio as wing sweep angle is decreased from the cruise setting of 0.785 rad (45 deg) to the takeoff setting of 0 rad (0 deg). These data indicate that for a swept aspect ratio of 6.0 the maximum attainable off-design range is slightly more than 10 percent greater than the design range of 5560 km (3000 n mi). Maximum range is 6130 km (3310 n mi) and occurs at a sweep angle of 0.35 rad (20 deg) and a cruise Mach number 0.715.

Final Configuration Sensitivity Data

The effects of changes in operating weight, cruise drag coefficient, specific fuel consumption, and the maximum lift coefficient, $C_{L_{MAX}}$ for landing for the Final Configuration are shown on Figures 38, 39, 40 and 41. The weight sensitivities of Figure 38 show that the growth in operating weight is 0.81 kg (1.78 lb) for an increase of 0.454 kg (1 lb) in any of the constituent weights of the operating weight; the growth in takeoff gross weight is 1.1 kg (2.42 lb), and the increase in block fuel is 0.24 kg (0.52 lb) for every 0.454 kg (1 lb) increase in operating weight.

The drag coefficient sensitivity, shown on Figure 39, indicates that an increment of one drag count changes the operating weight by 181.44 kg (400 lb) and the takeoff gross weight by 453.6 kg (1000 lb). The fuel weight change is 226.8 kg/drag count (500 lb/drag count).

The airplane sensitivity to SFC, Figure 40, shows that a one percent change in SFC changes the operating weight by 199.6 kg (440 lb), the block fuel by 453.6 kg (1000 lb), and the takeoff gross weight by 898.0 kg (1980 lb).

The effect on operating and takeoff gross weights of changes to the level of $C_{L_{MAX}}$ for landing is shown on Figure 41 for an approach speed constraint of 259.3 km/hr (140 k) EAS. A decrease in $C_{L_{MAX}} = 0.1$ will cause an increase in operating weight and takeoff gross weight of 861.8 and 2177 kg (1900 and 4800 lb), respectively.

Conventional Configuration Analysis

Conventional configuration analyses were performed to provide a basis for comparison of the Oblique Wing Concept Final Configuration. Conventional configurations for cruise at Mach 0.85 and 0.95 were developed and basic performance data obtained.

Conventional configuration for M = 0.85. - The conventional configuration for cruise at a Mach number of 0.85 is shown on Figure 42. The airplane for the 200-passenger/5560 km (3000 n mi) mission optimized at an aspect ratio of 8.25 and wing loading of 5338 N/m^2 (111.5 lb/ft^2). The propulsion system consists of four Pratt and Whitney STF 433 type engines, each developing 76,732 N (17,250 lb) of sea level static thrust. The engines are pylon mounted arrangements, two on each wing. The configuration is a tee-tail arrangement and all mission fuel is contained in the wings.

Technology levels for aerodynamics and materials and structure are the same as those for the Oblique wing Final Configuration to permit proper comparison.

Payload-range data for the configuration are summarized on Figure 43.

Conventional configuration for M = 0.95. - The parametric sizing data for the M = 0.95 conventional configuration, Figure 44, show the airplane selection to be based on the approach speed criterion of 259.3 km/hr (140 KEAS). The airplane selected for development optimized at a minimum takeoff gross weight of 146,057 kg (322,000 lb) at an aspect ratio of 6.25 for wing sweep angle of 0.785 rad (45 deg). These initial data did not account for the effects of contouring the fuselage necessary to minimize drag at a Mach number of 0.95.

A cross-sectional area distribution having the maximum cross-sectional area at 45 percent of the body length, obtained from previous studies of M = 0.95 configurations (Reference 2) was assumed and the fuselage shape developed in conjunction with the assumed curve and the wing thickness distribution of Figure 45. The resulting cross-sectional area data are shown on Figure 46.

Resizing the configuration to account for the effects of designing to the area distribution increases the airplane size to a takeoff gross weight of 149,793 kg (330,238 lb) for an aspect ratio of 6.25 and wing sweep angle of 0.785 rad (45 deg).

The airplane is a four-engine tee-tail configuration having the engines mounted on the wings. Each engine develops 112,833 N (25,366 lb) of sea level static thrust.

A payload-range summary is shown on Figure 47.

CONCEPT EVALUATION

Evaluation of Concepts Studied

Evaluation of the Oblique Wing Concept as a Commercial Passenger Transport is based upon comparison of airplane characteristics, installed thrust, maximum lift coefficients, performance and economic capability and configuration integration with similar data for conventional configurations.

The ability of the Oblique Wing Concept to perform alternate missions is also demonstrated as part of this evaluation.

Comparative data for Oblique wing and conventional configurations for cruise at Mach 0.85 and 0.95 are given in Table XVI. The data of Table XVI for the Oblique wing Mach 0.85 airplane are based upon a swept aspect ratio of 6.0 as in the case of the Mach 0.95 airplane.

Takeoff weight comparison. - The trends of takeoff gross weight shown on Figure 48 indicate that a cross-over for takeoff gross weight for the Oblique Wing Concept occurs at a cruise Mach number in the region of 0.90. Above this cruise Mach number the Oblique Wing Concept has a weight advantage over a conventional configuration.

Direct operating cost comparison. - The trends of direct operating cost, Figure 49, show the cross-over Mach number occurring at a slightly higher Mach number, M = 0.915, than that for weight.

Oblique Wing Concept - Conventional Configuration comparison. - The Oblique Wing Concept shows advantages over the conventional configuration above cruise Mach numbers of 0.90. At the design cruise Mach number of M = 0.95 these advantages are:

- o Lower takeoff gross weight - 7% less
- o Lower total installed thrust - 10% less
- o Lower block fuel for mission - 7% less
- o Less takeoff distance - 3% less
- o Lower direct operating cost - 5% less

It can also be stated that the Oblique Wing Concept produces better matching between the requirements for airport performance and cruise conditions than the corresponding conventional airplane. In addition, the ability to vary sweep angle provides further advantages over the conventional airplane as follows:

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TABLE XVI OBLIQUE WING/CONVENTIONAL AIRPLANE COMPARISON

QUANTITY \ CONFIGURATION	OBLIQUE WING				CONVENTIONAL			
	0.85		0.95		0.85		0.95	
TOGW, kg (lb)	135,695.3	(299,157)	139,453.0	(307,411)	125,031.4	(275,647)	149,793.4	(330,238)
Operating Weight, kg (lb)	69,720.8	(153,708)	72,183.3	(159,137)	66,006	(145,520)	79,173.6	(174,548)
Block Fuel Weight, kg (lb)	34,788.8	(76,696)	35,469.1	(78,196)	28,870.7	(63,649)	38,072.3	(83,935)
Wing Area, m^2 (ft^2)	230.3	(2,479)	217.76	(2,344)	222.60	(2,396)	291.05	(3,132.86)
Aspect Ratio Swept	6.0		6.0		8.25		6.25	
Wing Loading, N/m^2 (lb/ft^2)	5611	(117.2)	6057	(126.5)	5338	(111.5)	5262	(109.90)
Approach Speed, km/hr EAS(KEAS)	259.28	(140)	259.28	(140)	250.57	(135.3)	259.28	(140)
$C_{L_{MAX}}$ Takeoff/Landing	2.04/2.59		2.4/2.82		2.24/2.69		2.01/2.45	
Cruise L/D	16.25		16.05		18.79		16.32	
Total Installed Thrust, N (lb)	407,332.5	(91,572)	405,702	(91,206)	306,928	(69,000)	451,334.4	(101,464.0)
Takeoff/Landing Distance, m (ft)	2689/1947	(8824/6388)	2483.8/1924.5	(8149/6314)	2920/1897.4	(9580/6225)	2555.4/1874	(8384/6148)
Direct Operating Cost, ¢/km (¢/st mi)	1.457	(2.344)	1.409	(2.267)	1.322	(2.127)	1.483	(2.386)

- o Increased range at off-design conditions - maximum range increase, 10%
- o Increased endurance - endurance capability increased up to 44%.

The critical area affecting the performance of the Oblique Wing Concept in this application is aeroelastic stability of the wing. The parameter predominating in these effects is wing aspect ratio in the swept and unswept configurations. Based upon preliminary aeroelastic analyses, a swept aspect ratio of 5.0 was chosen. Subsequent aeroelastic analysis performed by the NASA (Appendix B) showed that the upper limit on swept aspect ratio for the Oblique Wing to be 6.0 for weight penalty avoidance. This aspect ratio is contingent upon the ability to utilize composite filamentary materials in the wing structure to the maximum possible level and by taking advantage of the improved stiffness to density ratio of these materials.

Payload-range comparison. - A comparison of the payload-range capability of the Oblique Wing Concept Final Configuration and the conventional configurations for cruise speeds of M = 0.85 and 0.95 is shown on Figure 50. These data indicate that the Oblique Wing Concept is limited by wing fuel volume at ranges in excess of 5390 km (3350 n mi) whereas the conventional configuration has sufficient wing volume to avoid this limitation.

The Oblique Wing Concept Final Configuration, however, has the ability to exceed the performance of the conventional configuration by changing sweep angle and cruise Mach number to achieve maximum off-design range which in the region of the X-point is slightly more than 10 percent on range. Applying this factor to the Final Configuration to obtain maximum off-design performance, the Oblique Wing Concept reduces the effect of fuel volume limitation to ranges below 6196 km (3850 n mi). At payloads in excess of 14,968 kg (33,000 lb), the Oblique Wing Concept applied to the Commercial Passenger Transport Mission has an advantage over the conventional configuration through the ability to vary airplane cruise configuration.

Oblique Wing Concept, weight-range sensitivity. - The effect on configuration takeoff gross weight and DOC of variations of design range is shown on Figure 51 for mission ranges from 2778 km (1500 n mi) to 8334 km (4500 n mi). These data, which are consistent with that of the Final Configuration range of 5560 km (3000 n mi), were obtained at ranges of 2778 and 8334 km (1500 and 4500 n mi). The principal effect of increased range is to increase the size of the wing at ranges above 5560 km (3000 n mi) to meet the fuel volume requirements. Thus, at short range, wing area is constrained by the 259.3 km/hr (140 k) EAS approach speed. At 5560 km (3000 n mi) the wing size is takeoff/cruise/fuel volume/approach speed matched. Above this range however wing area is governed by fuel volume requirements causing a rapid increase in takeoff gross weight. The variation in DOC between ranges of 2778 km (1500 n mi) and 5560 km (3000 n mi) is small. Above this range, however, DOC increases significantly due to the increase in takeoff gross weight.

Alternate Missions Analysis

Two Alternate Missions have been investigated for the Oblique Wing Concept Final Configuration as follows:

- o The Final Configuration as an Air Force tanker.
- o The Final Configuration as a Navy anti-submarine warfare airplane.

Air Force tanker mission. - A brief survey of tanker/receiver aircraft requirements established that the speed and altitude requirement for fuel transfer to current and projected military airplanes to be in the region of a Mach number of 0.88 and an altitude of 11,277 m (37,000 ft).

As a tanker airplane the Final Configuration can be overloaded to a 2.0g condition to maximize fuel off-load. Removing the passenger related equipment and inserting the tanker-peculiar equipment reduces the operating weight from 72,183 kg (159,137 lb) to 62,619 kg (138,051 lb). At this operating weight the overload condition is within the capability of the airplane structure as designed for the Commercial Passenger Transport Mission.

The tanker mission profile consists of:

- o Takeoff and climb to cruise altitude.
- o Cruise at optimum altitude and speed to fuel transfer point.
- o Assume transfer altitude and a Mach number of 0.88 and the appropriate wing sweep angle, rendezvous with receiving aircraft, and accomplish fuel transfer.
- o At completion of fuel transfer return to optimum cruise altitude and speed for a recovery leg of 1852 km (1000 n mi).
- o Descend and land at recovery base.

Data for fuel off-load capability are shown on Figure 52 for two cruise Mach number conditions:

- o Cruise at Mach 0.95 and a sweep angle of 0.785 rad (45 deg).
- o Cruise at the Mach number for maximum range, M = 0.715 and a sweep angle of 0.35 rad (20 deg).

The operational flexibility of the Oblique Wing Concept enables the tanker aircraft to match a wide variety of receiving aircraft characteristics which leads to a reduction in the fuel penalty currently incurred by high performance receiving aircraft on long range missions. The performance characteristics of receiving airplanes such as the A-7, B-1, F-4, F-15 and the F-16 provided the data for this assessment.

ASW mission. - The analysis of the Final Configuration executing a typical ASW mission profile was performed for payloads of 11,340 kg (25,000 lb) and 18,144 kg (40,000 lb).

The mission profile selected for the ASW mission consists of:

- Takeoff and climb to cruise altitude.
- Cruise to loiter station at either design cruise Mach number or Mach number for maximum range.
- Descend to 1524 m (5000 ft) loiter station.
- Loiter on-station using 2 or 3 engines.
- Climb from 1524 m (5000 ft) to cruise altitude.
- Cruise to base at either design cruise Mach number or Mach number for maximum range.
- Descend and land at base.

The ASW mission capability data are shown on Figure 53 for two payloads and for operation on 2 and 3 engines at design cruise Mach number and at Mach number for maximum range for the radius portion of the mission. Maximum time-on-station is obtained by operating on two engines at the power setting required for loiter with one engine windmilling.

In the ASW configuration the airplane operating weight is reduced from 72,183 kg (159,137 lb) to 63,382 kg (139,733 lb). At a takeoff gross weight of 133,511 kg (294,340 lb) the limit maneuver load factor with a payload of 18,144 kg (40,000 lb) and 4830 kg (10,647 lb) of fuel in the fuselage is 3.0g.

Alternate Missions Comparison

The weight summaries for the Final Configuration for both tanker and ASW missions are shown on Table XVII. These weights were used to derive the performance data of Figures 52 and 53.

Comparison of the airplane with current ASW capability is not possible as the pertinent ASW data are of a classified nature.

TABLE XVII ALTERNATE MISSIONS - WEIGHT BREAKDOWN FOR TANKER AND ASW MISSIONS

ITEM	MISSION			
	AF TANKER		NAVY ASW	
	WEIGHT		WEIGHT	
	kg	(lb)	kg	(lb)
WING	14,567	(32,114)	14,567	(32,114)
HORIZONTAL STABILIZER	1,151	(2,538)	1,151	(2,538)
VERTICAL STABILIZER	994	(2,190)	994	(2,190)
FUSELAGE	12,933	(28,513)	12,933	(28,513)
LANDING GEAR	6,172	(13,608)	6,172	(13,608)
NACELLE	2,508	(5,530)	2,508	(5,530)
PROPULSION	14,505	(31,977)	10,692	(23,572)
AUXILIARY POWER SYSTEM	266	(587)	266	(587)
SURFACE CONTROLS	1,207	(2,662)	1,207	(2,662)
INSTRUMENTS	418	(922)	403	(889)
HYDRAULICS AND PNEUMATICS	699	(1,541)	597	(1,316)
ELECTRICAL	1,382	(3,044)	1,772	(3,907)
AVIONICS	1,364	(3,008)	3,374	(7,438)
FURNISHINGS	1,053	(2,321)	1,883	(4,150)
AIR CONDITIONING AND ANTI-ICING	1,152	(2,540)	1,610	(3,549)
AUXILIARY GEAR SYSTEM	914	(2,015)	36	(80)
ARMAMENT	—	—	622	(1,370)
WEIGHT EMPTY	61,285	(135,110)	60,787	(134,013)
FUSELAGE FUEL SYSTEM			544	(1,198)
OPERATING EQUIPMENT	1,334	(2,941)	2,051	(4,522)
OPERATING WEIGHT	62,619	(138,051)	63,382	(139,733)
PAYLOAD			18,144	(40,000)
ZERO FUEL WEIGHT	62,619	(138,051)	81,526	(179,733)
FUEL WING	47,155	(103,960)	47,155	(103,960)
FUEL FUSELAGE	53,678	(118,340)	4,830	(10,647)
GROSS WEIGHT	163,452	(360,351)	133,511	(294,340)

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Tanker mission comparison. - A comparison of the performance of the tanker configuration with a current tanker, the Boeing KC 135, is shown on Figure 52. The overload takeoff gross weight for the KC 135 airplane is 136,078 kg (300,000 lb) as compared to 163,452 kg (360,351 lb) for the Oblique wing Final Configuration. At a range of 1852 km (1000 n mi) the Oblique Wing Concept has 25 percent more off-load capacity than the KC 135 and at a range of 11,112 km (6000 n mi) the maximum capability is 87 percent more than the KC 135.

Environmental Effects Comparison

Acoustic characteristics comparison. - The comparison of the Oblique Wing Concept with the conventional configuration which has four wing mounted Pratt and Whitney STF 433 type turbofans installed in acoustic nacelles similar to those of the Oblique Wing Concept Final Configuration, from a noise certification standpoint shows the Oblique Wing Concept to be superior at the three noise measuring locations as follows:

- o Takeoff sideline noise - lower by 0.5 EPNdB, due to lower installed thrust.
- o Takeoff flyover noise at takeoff power - lower by 2.5 EPNdB, due to lower installed thrust and greater altitude. At reduced power the noise level is lower by 6.5 EPNdB.
- o Approach flyover noise - lower by 8.5 EPNdB, due to significantly lower thrust requirements.

The proposed requirements of Reference 6 for a conventional four-engined configuration are not as stringent on takeoff as those for a three-engined Oblique wing configuration.

The acoustic benefits arising from the Oblique Wing Concept are:

- o Lower level of airframe self noise on approach - 91.5 EPNdB, with potential for reduction to 89 EPNdB, compared to 93.5 EPNdB for the conventional configuration.
- o Lower approach thrust settings than the conventional configuration leading to approach noise levels lower by 8.5 EPNdB.

Final Configuration acoustic soundprint. - The takeoff and landing 90 EPNdB footprint is estimated to be about 9.065×10^6 m^2 (3.5 mi^2), with an acoustic nacelle. The soundprint shown on Figure 54 is estimated for the test conditions of Reference 7 at maximum takeoff weight and takeoff thrust and at maximum landing weight.

Technological Requirements

The ability of the Oblique Wing Concept, as applied to the Commercial Passenger Transport Mission, to attain the stated advantages over a conventional configuration is dependent upon the technology readiness of a number of critical technology areas. These critical areas occur in aerodynamics, structures and materials, and propulsion.

Aerodynamic technology. - The use of a supercritical airfoil is necessary to achieve the maximum possible wing thickness for cruise at Mach 0.95 without incurring drag divergence Mach number penalties,and to reduce wing structure weight and increase available fuel volume by increased thickness.

Materials technology. - The ability to use composite filamentary materials for wing structure is fundamental to the achievement of the Oblique Wing Concept benefits.

Airplane characteristics and performance can be improved by an increase in wing aspect ratio. Utilization of maximum levels of composite materials application are necessary to achieve a high aspect ratio flutter-free wing structure. The properties of composite materials are such that the stiffness to density ratio for these materials is considerably greater than for aluminum so that a given wing configuration, which in aluminum would require additional stiffness material to meet the aeroelastic requirements, could be achieved without incurring such a penalty because of the improved stiffness/ density characteristics of the material.

The materials technology levels for this study were estimated from Reference 1 to ensure compatibility of the study results with those of Reference 1.

Composite materials related studies at the Lockheed-Georgia Company indicate two important points relative to the composite material application and the time frame for maximum utilization and to the weight reduction potential of the materials.

These studies show that:

- The time frame for maximum utilization of composite materials corresponds to an introduction to service date of 1990 rather than 1985. This implies a design commitment of 1985 instead of 1980.
- The weight reduction due to maximum utilization of composites from Reference 1 is found to be conservative. Lockheed-Georgia Company studies indicate greater reductions in weight are possible and a weight comparison was made using Lockheed data to determine the overall improvement in airplane weight. These data are shown on Table XVIII.

TABLE XVIII MATERIALS TECHNOLOGY COMPARISON

TECHNOLOGY	ALUMINUM	COMPOSITE (REFERENCE 1) 1985	COMPOSITE LOCKHEED PREDICTION 1990
UTILIZATION LEVEL	0	MAXIMUM	MAXIMUM
TAKEOFF GROSS WEIGHT, kg (lb)	160,811 (354,528)	139,453 (307,441)	129,461 (285,413)
STRUCTURAL WEIGHT REDUCTION, %	0	20	28

Propulsion technology. - Propulsion technology for the 1985 time frame is embodied in the Pratt & Whitney STF 433 turbofan which is a twin spool, two-stage fan, high bypass ratio engine having separate fixed primary and fan exhaust nozzles. The engine is designed to reduce noise levels so that with acoustic treatment in the nacelle, levels 15 EPNdB below FAR Part 36, may be achieved. Other technology features include high thrust/weight ratio and reduced specific fuel consumption.

Active control technology. - The application of active controls, as a means of providing active flutter suppression may permit satisfactory divergence stability to be achieved at swept aspect ratios up to 7.0 on strength designed wing structures. Employment of an active outboard aileron could result in a reduction in takeoff gross weight at the higher aspect ratio without resorting to adding material to increase stiffness.

Summary of Results

The results of the study are summarized on Table XIX. The benefits arising from the Oblique Wing Concept when compared to the conventional concept for cruise at Mach 0.95 are shown for weight, cost, thrust, airport performance and acoustic characteristics, and for off-design capability.

The domain of the Oblique Wing Concept is shown to be at speeds of Mach 0.91 and above. At Mach 0.95 the concept significantly improves weight performance and community noise characteristics.

Additional benefits are obtained for the Oblique Wing Concept by virtue of the variable geometry features which provide alternate mission capability for military use.

Recommendations

- Conduct further aeroelastic analyses to determine structural characteristics of wings at aspect ratios greater than 6.0.
- Investigate active flutter suppression systems as a means of achieving higher aspect ratios.
- Continue development of the Commercial Passenger Transport to further improve the design and performance.
- Investigate the short haul potential of the Oblique Wing Concept.
- Further develop the Executive Transport Configuration with emphasis on the Navy carrier-borne applications.

TABLE XIX SUMMARY OF RESULTS

Parameter	Oblique Wing Concept Configuration		Conventional Configuration		Change From Conventional Configuration
Takeoff Gross Weight, kg (lb)	139,453	(307,411)	149,793	(330,238)	7% Less
Direct Operating Cost, ¢/km (¢/st mi)	1.409	(2.267)	1.483	(2.386)	5% Less
Total Installed Thrust, N (lb)	405,702	(91,206)	451,334	(101,464)	10% Less
Mission Block Fuel, kg (lb)	35,469	(78,196)	38,072	(83,935)	7% Less
Takeoff Distance, m (ft)	2,484	(8,149)	2,555	(8,384)	3% Less
Acoustic Soundprint Area 90 EPNdB m^2 (mi^2)	9.065×10^6	(3.5)	19.17×10^6	(7.4)	53% Less
Airframe Self Noise Approach EPNdB	91.5 Potential to 89.0		93.5		2 EPNdB Less
Takeoff Sideline EPNdB	———		———		0.5 Less
Takeoff Flyover EPNdB	———		———		2.5 Less
Approach Flyover EPNdB	———		———		8.5 Less

OBLIQUE WING CONCEPT OFF-DESIGN CAPABILITY

Performance Item	Cruise Configuration		Off-Design Configuration		Performance Change
Range - km (n mi)	5560	(3000)	6112	(3300)	10% More
Cruise Mach No.	0.95		0.715		
Endurance - hrs	8.75		12.6		44% More

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APPENDIX A

METHODOLOGY AND BASIC DATA

Aerodynamic Design

Basic aerodynamic data. - During the early phases of the program, aerodynamic data were evaluated to provide a basis for predicting aircraft performance loads and handling characteristics.

The wing analysis was based on use of a supercritical airfoil designed at Lockheed-Georgia. The principal characteristics of the airfoil design are:

- Thickness/chord ratio = 12 percent
- $M_D = 0.76$ for $C_{L_D} = 0.6$
- Minimum drag creep

The allowable C_L - M - Λ - t/c relationships were correlated with this data base and are shown in Figure A1. It is assumed that the wing is designed to have 10 counts of compressibility drag.

The thickness distribution of the 12 percent thickness/chord ratio airfoil design was used to derive the airfoil sections for the study configurations by scaling the ordinates of the 12 percent section by the ratio of t/c %/12 where t/c is the value of the required thickness/chord ratio from Figure A1.

Drag prediction. - The cruise drag characteristics defined in this study are built up as follows:

- The zero lift drag of each component is estimated using the appropriate form factors and skin friction drag defined for the cruise Reynolds number.
- A wing profile drag increment, which is a function of variations in design Mach number and lift coefficient, is applied.
- A compressibility drag increment of 0.0010 is added as previously discussed.
- A 12 count trim drag increment is added. This high value allows for the high C_{M_0} value which results from the relatively high section lift coefficient and aft loading.
- A roughness drag of 5 percent of the parasite drag is computed and added to the polar.

- An interference drag allowance of 3 percent of the parasite drag is computed and added to the polar.
- Induced drag determination is based on an efficiency factor of 0.90 for the Mach 0.95 Oblique wing configuration.
- An increment of 5 counts of drag has been included for the Oblique wing configurations to account for the landing gear fairings.

Wing load distribution data. The load distribution on an uncambered, untwisted Oblique wing of rectangular planform is compared with that for a conventionally swept rectangular wing in Figure A2 to demonstrate the basic influence of the local flow fields. The marked reduction in the loading of the forward wing and the small increase in loading on the trailing wing is demonstrated. The reduction in pitching moment and the small rolling moment into the leading wing may also be observed.

A Discrete Element Aerodynamic Computer Program used to generate the data of Figure A2 was then used to represent the more complex planform of the Oblique wing aircraft wind tunnel model of Reference A1, as shown in Figure A3. This model includes camber and twist as well as dihedral and uses a simplified interference plane to represent the body without an empennage at this stage. The results obtained from this computer model are compared with the wind tunnel measured data on Figure A4. The predicted lift coefficient at zero angle of attack is higher than that of the wind tunnel data. This difference will be reduced by about 0.01 with the addition of the horizontal tail. The predicted lift curve slope is higher than the wind tunnel data and will be further increased by the addition of the horizontal tail. The predicted drag polar is lower than the wind tunnel data because of the body and empennage terms not represented in the analysis. The general levels of drag, however, appear to be consistent. The predicted level of pitching moment coefficient, C_M, should closely match the wind tunnel data of Figure A4 with the addition of a horizontal tail to the computer model. The inclusion of an estimated tail contribution in the airplane pitching moment coefficient resulted in the computed pitching moment coefficient at a $C_L = 0.42$ remaining the same as the tail-off level at that C_L but with a stable slope of -0.03 through that point. A small decrease in the assumed level of tail effectiveness in pitch would result in a match of the neutral stability shown by the wind tunnel data of Figure A4.

Rolling moment is expected to be entirely wing generated and good correlation of rolling moment coefficient from the wind tunnel data and that predicted by the computer analysis has been obtained.

The spanwise loading shape which causes the rolling moment is shown in Figure A5, for two angles of attack as a function of span in the 'x' direction. This shape arises from the built-in wing twist and the effect of dihedral at a sweep angle of 0.785 rad (45 deg), as well as the effective camber of the swept sections. As shown on Figure A6, the trailing wing has a considerable degree of wash-out at 0.785 rad (45 deg) of sweep.

The change in free stream camber with wing sweep angle is shown on Figure A7. These data show that the net camber on the trailing wing is less than that of the leading wing.

The effects, demonstrated above, are summarized as follows:

- o The effect of yaw angle for the basic planform is to lose lift on the leading wing and gain lift on the trailing wing relative to a conventionally swept wing.
- o The effect of yaw angle on camber is to reduce the effective camber of both wings; the leading wing loses more than the trailing wing for pivot points ahead of the 50 percent chord depending on the taper ratio.
- o The effect of yaw angle on twist is to reduce the level of washout, for linear twist, on both wings with the trailing wing losing more than the leading wing.
- o The effect of yaw angle on dihedral is to increase the washout on the trailing wing and the wash-in on the leading wing. The angular change is greater on the trailing wing than on the leading wing.

The low level of wing efficiency of the wind tunnel data, as measured by the drag due to lift, can be partially explained by this loading; however, a reduction in the dihedral should result in a better L/D ratio, a better induced drag factor, and considerably less asymmetric roll.

The correlation between the computed and wind tunnel data shown on Figure A4 was considered sufficiently accurate to permit analysis of the Baseline Configuration using the computer program, and to determine the effects of flexibility on the span loading shapes and the jig shapes required to give the wing the proper span load shape for cruise.

The work performed on the Baseline Configuration wing planform as shown on Figure A8 was synthesized using the Discrete Element Aerodynamic Program. The evaluation considered Oblique left and right wings which were compared to the same wings synthesized as conventionally swept wings; i.e., symmetric forward or aft sweep. The results of the comparison are shown on Figure A9.

In order to obtain an efficient spanwise lift distribution, the twist distribution necessary to produce an elliptical spanwise loading was approximated at Mach 0.6 which corresponds to an unswept condition. The wing configuration was then rotated to the Oblique position which resulted in an asymmetric spanloading.

It was found that the loading distribution of the Oblique wing could be controlled by the use of spanwise twist and dihedral. The desired variation of load distribution for the unswept wing, as shown on Figure A10, was obtained with the twist distribution of Figure A11. The addition of the asymmetric dihedral, also shown on Figure A11, produced the twist distribution for the optimum load distribution for cruise without changing the unswept wing loading distribution of Figure A12.

It should be noted that in the previous analyses the computer representation consisted of a wing/body combination only, as shown in Figure A8. When the empennage was added to the configuration, the asymmetric load induced on the horizontal stabilizer by the Oblique wing had no influence on the effectiveness of the horizontal stabilizer in pitch.

The design loading of the wing at cruise conditions, dynamic pressure $q = 1334\ N/m^2$ ($300\ lb/ft^2$), produces the twist and flexure shown in Figure A14. These data do not include the relief due to inertia and the deformations shown must be accounted for in the determination of the correct jigging shape for the wing in the unswept condition by changing dihedral angle to maintain equal elliptical spanwise load distribution on the wing cruising in the Oblique configuration.

High lift data development. - The basic data for available maximum lift used in the parametric analysis program for performance evaluation are based on the methods of Reference A2. The maximum lift coefficient available for the clean configuration at low speed was based on the assumption of maximum camber near the 50 percent chord point, maximum thickness near the 40 percent chord point, and a medium leading edge thickness. A flap chord of 25 percent of the wing chord is assumed for the Commercial Passenger Transport and 28 percent for the Military Cargo and Executive Transports. The flap is assumed to be 65 percent of the wing span beginning at the 10 percent span point. The pitching moment due to flaps is calculated and a $C_{L_{MAX\ TRIMMED}}$ obtained for an operational center-of-gravity. An allowance of 6 percent on the static $C_{L_{MAX}}$ is calculated as representative of the minimum speed that should be developed during a demonstration to FAR rules of an approach to stall for these aircraft. The data for takeoff and landing flaps for three configurations are summarized on Table AI.

Relaxed static stability. - Relaxed static stability is recognized to the extent that a static margin of 5 percent is reflected in horizontal stabilizer sizing for all configuration studies.

Oblique Wing Concept Airplane Sizing Studies

Airplane sizing studies were performed for the three Preliminary Design Missions using a parametric analytical method and utilizing a computer program which, when provided with basic data such as fuselage size, engine data, mission requirements, and atmospheric data, determines:

- o Drag and weight characteristics of a given configuration.
- o The capability of the configuration to meet mission requirements.

TABLE AI SUMMARY OF FLAP SYSTEM REQUIREMENTS

Configuration	Commercial Passenger Transport	Executive Transport	Military Cargo Transport
Flap Type	Single Slotted	Double Slotted	Double Slotted
Flap Chord C_F/C %	25	28	28
Takeoff $C_{L_{MAX}}$ δ_F = 0 523 rad (30 deg)			
$C_{L_{MAX}}$ - Flaps Up	1 53	1 54	1.50
$\Delta C_{L_{MAX}}$ - L/E Slats	——	——	0 31
$\Delta C_{L_{MAX}}$ - T/E Flap	0 85	1 00	0.68
$C_{L_{MAX}}$ - Flaps Down	2 38	2 54	2.49
$C_{M_{TAILOFF}}$	$\overline{0\ 37}$	$\overline{0\ 48}$	$\overline{0\ 32}$
$\Delta C_{L_{MAX\ TRIMMED}}$	-0 07	-0 10	-0 09
$C_{L_{MAX\ TRIMMED}}$	2.31	2 44	2 40
Usable $C_{L_{MAX}}$	2 45	2 59	2 54
Landing $C_{L_{MAX}}$ δ_F = 0 70 rad (40 deg)			
$C_{L_{MAX}}$ - Flaps Up	1 53	1 54	1.50
$\Delta C_{L_{MAX}}$ - L/E Slats	——	——	0 29
$\Delta C_{L_{MAX}}$ - T/E Flap	1 10	1 23	0.96
$C_{L_{MAX}}$ - Flaps Down	2 63	2.77	2 75
$C_{M_{TAILOFF}}$	$\overline{0\ 45}$	$\overline{0\ 56}$	$\overline{0.40}$
$\Delta C_{L_{MAX\ TRIMMED}}$	-0 09	-0.16	-0 11
$C_{L_{MAX\ TRIMMED}}$	2.54	2.61	2 64
Usable $C_{L_{MAX}}$	2 69	2.77	2 80
Landing $C_{L_{MAX}}$ δ_F = 0 785 rad (45 deg)			
$C_{L_{MAX}}$ - Flaps Up	1 53	1.54	1 50
$\Delta C_{L_{MAX}}$ - L/E Slats	——	——	0 29
$\Delta C_{L_{MAX}}$ - T/E Flap	1 20	1.29	1 06
$C_{L_{MAX}}$ - Flaps Down	2 73	2 83	2.85
$C_{M_{TAILOFF}}$	$\overline{0.47}$	$\overline{0\ 58}$	$\overline{0\ 42}$
$\Delta C_{L_{MAX\ TRIMMED}}$	-0 09	-0 17	-0.12
$C_{L_{MAX\ TRIMMED}}$	2 64	2 66	2 73
Usable $C_{L_{MAX}}$	2 80	2 82	2.89

The computerized process is arranged in a loop so that aircraft size, consisting principally of wing and empennage areas and engine size, can be iterated until the configuration is sized to satisfy the mission requirements. Airport performance, approach speed and direct operating costs of the aircraft are then determined within the program.

Primary variables used for parametric analysis consist of wing aspect ratio and sweep angle, cruise altitude and cruise wing loading or lift coefficient. The program arrangement is shown on Figure A15.

Commercial Passenger Transport. - The evolution of the Commercial Passenger Transport consisted of the determination of:

- An Initial Configuration
- A Baseline Configuration
- A Cycled Baseline Configuration
- A Final Configuration

The sizing studies performed for the first three configurations resulted in a configuration which was used to evaluate the Oblique Wing Concept.

The final configuration sizing determined the characteristics for the Final Configuration design.

Initial Configuration. - A parametric analysis was performed to establish an Initial Configuration and to determine the range of parametric variables for subsequent analyses. This analysis was performed for the following values of wing loading and aspect ratio at a wing sweep angle of 0.785 rad (45 deg):

Wing loading - N/m^2 (lb/ft^2) 4788/5506/6224 (100/115/130)

Aspect ratio - 7.0/7.75/8.5

The analysis assumed Pratt and Whitney STF 429 engine characteristics and resulted in the airplane sizing parametric data shown on Figure A16. The minimum takeoff gross weight airplane of this matrix occurred at a swept aspect ratio of 7.0 and a cruise wing loading of 6224 N/m^2 (130 lb/ft^2).

These data were obtained for a constant cruise altitude of 11,277 m (37,000 ft) which was obtained from the results of past related studies. Preliminary estimates of wing weight indicated that at a swept aspect ratio 7.0, considerable weight penalty, not included in the parametric results, would be incurred to achieve a divergence-free wing.

Study range of parametric variables. The range of variables selected for the execution of subsequent parametric analyses as the result of the preceding analysis were:

Wing loading -	N/m^2	3830/4788/5745/6703
	(lb/ft^2)	(80/100/120/140)
Swept aspect ratio		4/5/6/7
Sweep angle -	rad	0.610/0.70/0.785
	(deg)	(35/40/45)

Cruise power setting determination. - Matrices of Oblique wing airplanes configured to cruise at a wing sweep angle of 0.785 rad (45 deg) were obtained for a range of swept aspect ratios from 5.0 to 7.0 and for wing loadings from 3830 to 5745 N/m^2 (80 to 120 lb/ft^2). Data were obtained for cruise power settings of 0.95 and 0.85 of normal rated cruise thrust.

An upper limit on takeoff distance was set at 2743 m (9,000 ft) to allow sufficient margin on takeoff distance to permit growth in airplane size during parametric analyses.

The parametric data for cruise at the two thrust settings are shown on Figures A17 and A18 for 0.95 and 0.85 cruise rated thrust, respectively. These data establish cruise at 0.95 cruise rated thrust as producing the minimum takeoff gross weight airplane since the takeoff distance limitation places the minimum takeoff gross weight airplane beyond the boundaries of the parametric analysis for cruise at 0.85 cruise rated thrust. At 0.95 cruise rate thrust the minimum weight airplane, 131,542 kg (290,000 lb) occurs at a swept aspect ratio of 5.0 and a wing loading of 5745 N/m^2 (120 lb/ft^2). At the same aspect ratio and wing loading for 0.85 cruise rated thrust the minimum weight airplane is 136,531 kg (301,000 lb).

Baseline Configuration. - Further parametric analyses were performed to incorporate changes in data from the previous analyses relating to the aerodynamic design and the initial sizing studies. These changes included:

- A value for $e = 0.9$ for wing efficiency.
- Tail volume coefficients increased.
- Target takeoff $C_{L_{MAX}} = 2.2$.
- Cruise to be performed at 0.95 cruise rated thrust.
- International fuel reserves for mission performance analysis.

Parametric analyses were conducted for sweep angles of 0.70 and 0.785 rad (40 and 45 deg) in order to determine the effect on changing cruise sweep angle on airplane size. These data, shown on Figures A19 and A20, indicate that, although a slightly smaller airplane results at a sweep angle of 0.70 rad (40 deg), the airplane would be constrained by fuel volume and airport performance leaving little scope for refinement both at the design and off-design points. At a sweep angle of 0.785 rad (45 deg), however, the data of Figure A20 indicate that:

- o Sufficient fuel volume is available in the wing at wing loadings of 6128 N/m^2 (128 lb/ft^2) and below.
- o Minimum takeoff gross weight occurs in the region of swept aspect ratios of 6.0 and 7.0.
- o Block fuel decreases as aspect ratio increases.
- o Takeoff distance increases significantly with aspect ratio.

The choice of the Baseline Configuration from Figure A20 was made on the basis of:

- o Takeoff distance - 2743 m (9000 ft)
- o Adequate fuel volume/wing volume ratio
- o Minimum takeoff gross weight

A number of possible configurations from Figure A20 are as follows:

W/S		AR	TOD		TOGW		FVR
N/m^2	(lb/ft^2)		m	(ft)	kgs	(lb)	
5865	(122.5)	6.0	2743	(9000)	135,624	(299,000)	1.15
5745	(120)	6.3	2743	(9000)	136,531	(301,000)	1.16
5745	(120)	6.0	2686	(8811)	136,531	(301,000)	1.2

The weight difference for the three configurations is 907 kg (2000 lb), or about 0.7 percent on takeoff gross weight. The baseline selection was therefore made on the basis of minimum aspect ratio and takeoff distance and maximum fuel volume ratio.

The configuration characteristics selected for the Baseline Configuration from these data were:

- o Wing sweep angle for cruise, 0.785 rad (45 deg)
- o Wing swept aspect ratio, 6.0
- o Wing loading, 5745 N/m^2 (120 lb/ft^2)
- o Fuel volume ratio, 1.2

Cycled Baseline Configuration. - Preliminary aeroelastic analyses performed on an aspect ratio 6.0 swept wing indicated stiffness criticality which to correct would have incurred a considerable weight penalty for the wing. Since at a swept aspect ratio of 5.0 the problem of divergence did not exist, the wing aspect ratio for the Cycled Baseline Configuration was limited to 5.0. Recycling the airplane through the sizing procedures and incorporating a number of configuration changes described in "Configuration Studies" increased the takeoff gross weight of the airplane to 141,128 kg (311,134 lb), and increased the wing loading to 6200 N/m^2 (129.5 lb/ft^2).

Cruise altitude selection substantiation. - The effect of design cruise altitude on airplane size and performance was determined for the Cycled Baseline Configuration. These data, given on Figure A21, show that the minimum takeoff gross weight configuration, 141,128 kg (311,134 lb), which has a takeoff distance of 2544 m (8346 ft) and is constrained by the approach speed limitation of 259.3 km/hr (140 k) EAS, is consistent with a cruise altitude of 11,277 m (37,000 ft).

Executive Transport sizing and selection. - The parametric data for this configuration are shown on Figure A22. These data show that a lower limit on aspect ratio is imposed by second segment climb gradient, takeoff field length and wing fuel volume. The characteristics for the Initial Configuration selected from these data were:

- o Wing sweep angle, 0.785 rad (45 deg)
- o Wing swept aspect ratio, 5.0
- o Wing loading, 4190 N/m^2 (87.5 lb/ft^2)
- o Takeoff distance, 1524 m (5000 ft)

Selection of these characteristics provided sufficient margin on second segment climb gradient to allow for airplane growth and a wing with sufficient volume to contain approximately 80 percent of the mission fuel. The takeoff gross weight for this airplane is 32,778 kg (72,264 lb). These data were used to develop the Initial Configuration.

Executive Transport Baseline Configuration. - The characteristics of this configuration are identical to the Initial Configuration. Resizing of the configuration was required to account for changes in wing location and empennage size. The takeoff gross weight increased from 32,778 kg (72,264 lb) to 34,389 kg (75,816 lb).

Executive Transport Cycled Baseline Configuration. - Recognition of the configurational problems of the Baseline Configuration required further resizing of the configuration due to configuration changes. Cycling the revised configuration through the sizing procedure increased the configuration size to a takeoff gross weight of 36,745 kg (81,010 lb).

Effect of payload and range on airplane size. - As a result of the increase in size of the Executive Transport Cycled Baseline Configuration, an analysis of the effects of changing payload and mission range on the size of the airplane was performed. Payloads of 12, 14 and 16 passengers as well as the basic 18 passenger configuration were considered. Mission ranges of 5560 km (3000 n mi) and 6480 km (3500 n mi) in addition to the base 7408 km (4000 n mi) were included. The data were generated for a swept aspect ratio of 5.0 and a takeoff distance of 1524 m (5000 ft) and are shown on Figure A23. These data were used to establish the airplane mission characteristics resulting from the imposition of geometric constraints - in this case wing span - to ensure Navy carrier compatibility for possible alternate missions.

Military Cargo Transport sizing and selection. - The parametric analysis for this mission consisted of matrices of four engined aircraft for various wing loadings and swept aspect ratios at wing sweep angles of 0.70 and 0.785 rad (40 and 45 deg),for cruise at 0.95 cruise rated thrust, and at angles of 0.61 and 0.700 rad (35 and 40 deg) for cruise at 0.85 cruise rated thrust. Data for cruise at 0.95 cruise rated thrust and at the related sweep angles are shown on Figures A24 and A25. Neither of the matrices is limited by wing fuel volume. For a takeoff distance at 2440 m (8000 ft) a lower takeoff gross weight airplane is obtained at a sweep angle of 0.700 rad (40 deg).

Takeoff distances were computed using the high lift characteristics previously developed with and without a leading edge device. The data of Figure A24 indicate that a decrease in aircraft size could be achieved if an increase in the value of $C_{L_{MAX\ TRIMMED}}$ could be obtained. The data of Figure A24 show that a takeoff gross weight reduction of 4.5 percent could result if an increment of $C_{L_{MAX}}$ of 0.3, as might occur with the addition of an efficient leading edge device, could be obtained. The data of Figure A24 ignore weight differences due to wing leading edge configuration changes and show that the optimum airplane for the mission will occur at swept aspect ratios below 5.0 for cruise at 0.95 cruise rated thrust using a leading edge device to obtain a takeoff distance of 2440 m (8000 ft).

The second approach to airplane optimization involved the use of over-sized engines as a means of minimizing takeoff distance without resorting to the use of leading edge devices. The parametric data of this analysis are shown on Figures A26 and A27 for wing sweep angles of 0.61 and 0.70 rad (35 and 40 deg), respectively for cruise at 0.85 cruise rated thrust. The data show that wing fuel volume is not a constraining feature and that for a takeoff distance constrained to 2440 m (8000 ft), a slightly lower weight airplane is obtained at a sweep angle of 0.70 rad (40 deg) than at 0.61 rad (35 deg).

Initial Configuration. - Data for each selected point from Figures A24, A25, A26 and A27 are summarized on Table AII, from which the selection of the principal characteristics for the Military Cargo Transport Initial Configuration were obtained. These characteristics are:

- Wing sweep angle, 0.700 rad (40 deg)
- Cruise power setting, 0.95 cruise rated thrust
- Leading edge device to augment high lift
- Wing loading, 6225 N/m^2 (130 lb/ft^2)
- Swept aspect ratio, 4.75

Military Cargo Transport Baseline Configuration. - The parametric data for the Initial Configuration assumed four engines located at the aft end of the fuselage. Since the data from Table AII show thrust levels in excess of the technology limit for the time frame and the configuration studies

TABLE AII MILITARY CARGO TRANSPORT - OPTIMUM AIRPLANE CHARACTERISTICS

Cruise Power Setting - % Cruise Rated Thrust	TOGW kg (lb)	Operating Weight kg (lb)	Mission Fuel kg (lb)	Wing Area m^2 (ft^2)	Aspect Ratio	Sweep Angle rad (deg)	Loading N/m^2 (lb/ft^2)	$C_{L_{MAX}}$* Takeoff	Wing Fuel	V_{APP} km/hr (k) EAS	No. Engines	Static Thrust SLS N (lb)
95%	639,565 (1,410,000)	257,187 (567,000)	223,621 (493,000)	1,091 (11,744)	4 0	0.700 (40)	5554 (116)	2.0	100%	240 8 (130)	4	511,545 (115,000)
95%	608,720 (1,342,000)	247,661 (546,000)	202,302 (446,000)	959 (10,323)	4.75	0.700 (40)	6225 (130)	** 2 3	100%	*** 229 6 (124)	4	438,150 (98,500)
95%	644,101 (1,420,000)	263,573 (581,000)	221,806 (489,000)	980.4 (10,553)	4 2	0 785. (45)	6225 (130)	2.13	100%	259 3 (140)	4	500,425 (112,500)
85%	662,698 (1,461,000)	272,155 (600,000)	231,785 (511,000)	1,079 (11,618)	4.0	0 61 (35)	5817 (121 5)	1.85	100%	259.3 (140)	4	573,820 (129,000)
85%	644,554 (1,421,000)	268,073 (591,000)	217,724 (480,000)	981 (10,561)	4.5	0.700 (40)	6225 (130)	2 0	100%	251.9 (136)	4	529,338 (119,000)

Mission - Payload - 158,757 kg (350,000 lb) Range - 6482 km (3500 n mi) M_{CR} = 0 95

* Indicates $C_{L_{MAX\ TRIMMED}}$

** Indicates use of leading edge devices to achieve $C_{L_{MAX}}$

*** Approach speed without leading edge devices indicated the existence of severe balance and loadability problems, the propulsion system configuration was changed to six engines which were progressively relocated from the aft fuselage to the wing.

Resizing was therefore conducted to establish the characteristics of the Baseline Configuration which were:

- o Takeoff gross weight, 614,081 kg (1,353,818 lb)
- o Swept aspect ratio, 4.75
- o Wing loading, 6225 N/m^2 (130 lb/ft^2)
- o Takeoff distance, 2440 m (8000 ft)

These characteristics were used to develop the Military Cargo Transport Baseline Configuration.

Cycled Baseline Military Cargo Transport. - The parametric analyses for the Initial and Baseline Configurations were performed assuming no leading edge device. To establish the effect of a leading edge device on the configuration sizing matrix, takeoff distance data were calculated for an increment of $C_{L_{MAX}}$ for 0.30 and the takeoff distance data for 2440 m (8000 ft) superimposed upon the takeoff gross weight matrix. These data, shown on Figure A28, indicate the improvement to be derived by using a leading edge device.

Due to this, it was necessary to perform additional parametric analyses with the effects of the leading devices included in the sizing procedures. The resulting parametric data shown on Figure A28 indicated the Cycled Baseline Configuration characteristics to be:

- o Wing loading, 6655 N/m^2 (139 lb/ft^2)
- o Swept aspect ratio, 5.0
- o Wing sweep angle, 0.700 rad (40 deg)

These characteristics were used to develop the Cycled Baseline Configuration.

Conventional Passenger Configuration Sizing Studies

Parametric sizing studies were performed to determine the characteristics required for conventional passenger airplanes designed to perform the Baseline Mission and to cruise at Mach 0.85 and 0.95.

Mach 0.85 conventional configuration sizing. - The parametric analysis for the Mach 0.85 conventional configuration was conducted over a range of wing loadings from 4309 N/m^2 (90 lb/ft^2) to 5745 N/m^2 (120 lb/ft^2) and aspect ratios of 6.0, 7.0 and 8.0. Fuel volume in the wing was obtained by locating the front and rear spars at 12.5 and 58 percent of the wing chord. The parametric data shown on Figure A29 show the minimum weight configuration occurring at an aspect ratio 8.0 constrained by the approach speed limitation of 259.3 km/hr (140 k) EAS. The data, however, show the wing to be fuel volume limited. To overcome this limitation the rear spar was relocated at 65 percent of the wing chord which provided a slight excess of fuel volume. By iterating on aspect ratio and wing loading in the region of 8.0 and 5267 N/m^2 (110 lb/ft^2) the following characteristics were obtained:

- Aspect ratio, 8.25
- Wing loading, 5338 N/m^2 (111.5 lb/ft^2)
- Wing sweep angle, 0.523 rad (30 deg)

Mach 0.95 conventional configuration sizing. - Preliminary analyses to determine the characteristics of the Mach 0.95 conventional airplane are shown on Figure A30. The minimum weight airplane is constrained by the approach speed limitation of 259.3 km/hr (140 k) EAS. The characteristics for the configuration from these data are:

- Aspect ratio, 6.25
- Wing loading, 5291 N/m^2 (110.5 lb/ft^2)
- Wing sweep angle, 0.785 rad (45 deg)

These data reflect configuration geometry effects which include a highly contoured fuselage and a wing thickness distribution which assumed a constant thickness chord ratio across the span. The assumption tended to distort the wing weight since the actual thickness of the wing at the root was based upon the full chord at the root with the result that the wing thickness was greater than the depth of fuselage below the passenger deck. It would therefore be necessary to allow a considerable portion of the wing lower surface to project beyond the fuselage lower contour, and would result in a poor cross-sectional area distribution. The wing thickness at the root and across the fuselage was adjusted to remain within the fuselage contours. Iterations of the configuration sizing were performed to reflect the change in wing root thickness and no change in aspect ratio or sweep angle were indicated. Takeoff gross weight increased slightly and wing loading decreased. The characteristics for the iterated configuration are:

- Aspect ratio, 6.25
- Wing loading, 5262 N/m^2 (110 lb/ft^2)
- Wing sweep angle, 0.785 rad (45 deg)

High lift system definition. - The high lift system requirements resulting from the parametric sizing studies for each of the Preliminary Design Missions are summarized on Table AIII.

It is a requirement that each of the trailing edge high lift devices be capable of stowage within the wing trailing contours when the system is in the retracted position.

Oblique Wing Concept climb technique analysis. - An analysis of the climb technique for the Oblique Wing Concept showed the maximum specific range during climb to cruise altitude to be obtained with the wing swept to the cruise configuration. The drag polars used for the analysis assume simple sweep theory correction to Mach number and design lift coefficient and each polar includes ten counts of compressibility drag. The climb performance computations recognized reduced levels of compressibility drag increment as cruise Mach number was reduced for a given sweep angle. The basic relationship for airplane climb performance is:

$$\text{Rate-of-climb } R/C = \frac{\Delta TV}{W}$$

$$\Delta T = (T - D) = \text{excess thrust}$$

$$V = \text{Airplane speed} = \text{KTAS}$$

$$W = \text{Airplane weight} = \text{lb}$$

In the case of Oblique Wing Concept, the speed for maximum rate-of-climb is reduced as cruise speed is reduced and wing sweep angle decreases. The increase in excess thrust, ΔT, which occurs as a result of the higher L/D_{MAX} capability of the unswept polars is overpowered by the reduction in climb speed, thereby decreasing the value of ΔTV and reducing the rate-of-climb. A summary of these effects is given in Table AIV.

In addition to the specific range benefit associated with climb at cruise sweep angle, there are benefits to be derived by operating at a constant sweep angle. During transition and acceleration from climb to cruise conditions, a decrease in transition time occurs due to the lower differential between climb and cruise speed.

TABLE AIII HIGH LIFT SYSTEM DEFINITION

Preliminary Design Mission	Takeoff Distance m (ft)	Flap System Type	Flap % Chord Extension	Flap Geometry		Flap Deflection rad (deg)		Wing t/c, % Unswept
				% b	% c	T.O.	LDG.	
200 Passenger Commercial Transport	3050 (10,000)	Single Slotted Fowler	20	65	25	0.523 (30)	0.70 (40)	16.5
158,757 kg (350,000 lb) P/L Military Cargo Transport	2440 (8,000)	Double Slotted Fowler with 18% c L/E Device	20	65	28*	0.523 (30)	0.70 (40)	11.1
18 Passenger Executive Transport	1524 (5,000)	Double Slotted Flap	20	65	28*	0.523 (30)	0.70 (40)	19.56

* Includes retractable vane

b - wing span

c - wing chord

TABLE AIV

CLIMB PERFORMANCE SUMMARY

Sweep Angle	Time	Distance	Fuel	Specific Range
rad (deg)	min	km (n mi)	kg (lb)	km/kg (n mi/lb)
0.523 (30)	19.9	229.6 (124)	3315.3 (7309)	0.069 (0.0170)
0.700 (40)	22.1	281.5 (152)	3700 (8158)	0.076 (0.0186)
0.7854 (45)	24.4	329.6 (178)	4085 (9006)	0.081 (0.01975)

Structures and Materials

Weight and balance estimation. - Weight estimation of all configurations was accomplished by the use of a series of computerized parametric equations. Structure weight in composite materials was obtained by the application of weight technology factors to the equations for all structural components. The weight technology factors for the study were derived from the data of Reference A3 to ensure compatibility of the present study results with those of Reference A3. Since the factors used reflect considerable conservatism with respect to the weight reduction potential for the level of composite utilization, estimates of the weight reduction of the Final Configuration using more recent data, were also made.

Configuration balance was accomplished by using a computer program which positions the wing on the fuselage for a desired balance envelope and provides check data for the location of the horizontal stabilizer. This program also calculates configuration loadability, fuel volume and burn sequence and, using the derived mass distribution and the burn sequence, computes the moment of inertia of the configuration for various combinations of payload and fuel.

Wing weight analyses. - Due to the unique geometry of the Oblique wing, corrections applied to the parametric equations were obtained by the use of an analytical wing weight estimation program called Wing - ANSWER (ANalytical Structural Weight Estimation Routine). This program has the capability of either deriving external loads or accepting input data for distributed external loads. Internal load distribution is then determined for the external load distribution for specified types of wing structural configurations. Wing flexibility effects are considered during the external and internal load generation by determining available bending and torsional stiffnesses either through direct input information or by program derived data in an iterative mode. The data for required stiffnesses are used to identify wing rigidity constraints. Material properties and wing surface panel construction can be specified and assessed. Optimization logic included in the program can derive the primary box structure weight based on the constraints of strength,stiffness, geometry and producibility. The secondary structure component weights are derived from geometry related considerations and are based on empirical weight relations. The output of the program includes wing weight, tabulated by the major items of AN-9102 Detail Weight Statement, the bending and torsional stiffness distributions, weight increments, and geometric data.

Fuselage weight.- The fuselage weight is estimated by use of a Lockheed-derived statistical equation using a data base composed entirely of transport airplanes. This equation is based upon geometry, airplane function, design dive speed, design landing weight and flight load factors. Coefficients are computed, based upon structural increments to account for unique features. An independent estimate was performed on the pivot attachment and back-up structure in the fuselage to account for this feature.

Aeroelastic analysis. - The primary aeroelastic consideration associated with the Oblique Wing Concept is the divergence of the leading wing. When the fuselage is restrained, the leading wing bending-deflection due to lift increases the streamwise angle of attack and causes increases of the lift and bending deflection in opposition to the structural stiffness. At sufficiently high airspeeds, the rate of change of aerodynamic force due to bending exceeds that of the restoring elastic forces and "static" divergence occurs in a manner similar to that on a bilaterally symmetrical wing.

An unrestrained fuselage, however, causes flutter of the unsymmetrical Oblique wing to manifest itself as an instability involving principally wing bending and fuselage rolling motions. This phenomenon has been shown by several studies to occur at higher airspeeds than the static divergence of the leading wing. The prediction of this phenomenon, however, can only be obtained by an unsymmetrical flutter analysis program. Since this was not available at the beginning of the study, the initial approach to the determination of wing weight increments to prevent divergence was based on static divergence speed calculations performed with an existing computer program (DIVROL). This program utilizes a modified strip theory aerodynamic representation in which the local lift-curve-slope and aerodynamic center are based on compressible lifting-surface theory calculations.

Wings were initially sized for maneuver, gust and ground loads using a structural synthesis program (ANSWER). The bending and torsional stiffness distributions of the strength-sized structure were then used to determine the static divergence speed of the leading wing using program DIVROL. Where deficiencies were found to occur, the divergence speed was raised to 1.2 times the limit dive speed by increasing the bending stiffness. The required bending stiffness distribution to avoid divergence was then input into program ANSWER for resizing of the wing structure.

The available and required bending stiffness distributions for the swept aspect ratio 6.0 Baseline Configuration wing are shown in Figure A31. The divergence velocity predicted for the available stiffness distribution was approximately 796 km/hr (430 k) EAS. The stiffness distribution required to raise the divergence speed to 1.2 V_D, 910 km/hr (491 k) EAS resulted in a substantial structural weight penalty. This analysis procedure was repeated for an aspect ratio 5 wing configuration, the result of which indicated no weight increase required for divergence prevention. This led to the development of the Cycled Baseline Configuration for the Passenger Transport.

During the course of the study, the NASA developed a computer program capable of properly analyzing the unsymmetric Oblique Wing Concept for divergence and flutter stability. A description of the program and the analysis results for an aspect ratio 6.0 wing are contained in Appendix B. The results indicated that the stiffnesses associated with strength-sized structure were essentially adequate for prevention of both oscillatory divergence (low frequency flutter) and classical wing bending-torsion flutter at speeds up to 1.2 V_D as shown in Figure A32. Thus, it was concluded that the initial study approach, based on static divergence speed calculations, was overly conservative. The study was therefore redirected toward the development of a Final Configuration at an aspect ratio of 6.0.

Propulsion System Design

Engine. - The Pratt and Whitney STF 433 engine, Reference A4, was selected for the study as representative of 1985 technologies. This engine is designed to produce noise level 15 EPNdB lower than the Reference A5 requirements at sideline, takeoff, and approach conditions with acoustic treatment in the nacelle, including wall treatment and inlet and fan duct splitters. The STF 433 is a twin spool, two-stage turbofan designed to operate with separate fixed primary and fan exhaust nozzles. It is sized to produce 177,929 N (40,000 lbf) of uninstalled takeoff rated thrust at sea level static standard day conditions, and the data have a scaling range of 88,964 to 231,308 N (20,000 to 52,000 lbf) thrust.

The engine aerodynamic design point is at 11,582 m (38,000 ft) at a flight Mach number of 0.95 for maximum cruise standard day operating conditions. The engine has a bypass ratio of 6.5, a fan pressure ratio of 1.92, an overall pressure ratio of 25 and maximum combustor exit temperature of 1343°C (2450°F) for the cruise design point conditions. At the sized rated thrust of 177,929 N (40,000 lbf), the engine has a thrust to weight ratio of 5.26.

Nacelle. - The nacelle, shown in Figure A33 is for the reference P&W STF 433 engine having a thrust rating of 177,929 N (40,000 lbf). The nacelle is acoustically treated to comply with the noise certification limits of Reference A5 modified as proposed in Reference A6 by the application of a splitter and wall treatment to both the inlet and fan ducts and wall/plug treatment to the primary duct.

The nacelle inlet length is that required to provide a nacelle fineness ratio compatible with the flight design Mach number of 0.95. NACA 1-series sections are employed for the nacelle forebody contours. The inlet leading edge is raked aft 0.28 rad (16 deg) to reduce inlet normal shock far-field effects at high-speed cruise conditions, and to reduce the effect of inlet pressure distortion resulting from high flow field downwash angles at low-speed flight conditions. C-141 type lip slot and blow-in doors are installed over the upper half of the inlet periphery and are required to control lip flow separation in static and crosswind conditions, and at high-flow downwash angles.

Separate exhaust nozzles are used for the fan and primary flows as required by the STF 433 engine. Due to the fan duct acoustic treatment requirements,a coplanar exit is employed. The primary nozzle uses an extended plug to minimize the external afterbody boattail angle and to provide an expansion surface for the supercritical nozzle discharge flow. A circular arc contour is used on the afterbody having a radius to nacelle maximum diameter ratio of 9 which results in a boattail angle of 0.16 rad (9 deg).

Installation effects. -

Inlet total pressure recovery. - Total pressure recovery characteristics of the one-splitter nacelle inlet configuration are presented in Figure A34. The cruise data of this curve are based on conventional inlet loss calculation methods assuming pipe flow with friction. Friction factors on these surfaces are increased by 25 percent to account for the increased roughness of the acoustic material, and are based on actual test data of the equivalent roughness of acoustic materials similar to the perforated plate used in the inlet. Lip loss factors for low-speed operation, based on the data of Reference A7, are applied. The lip geometry is elliptical with a 12 percent area contraction. Diffusion losses are minimal as a result of the relatively long inlet dictated by acoustic treatment requirements.

Additive drag. - Forebody pressure drag coefficients for the NACA 1-series forebody design are presented in Figure A35. These drag coefficients are based on wind tunnel tests of a series of forebody shapes run at design and off-design conditions of mass-flow ratio and freestream Mach number. The design mass-flow ratio for Mach 0.95 is 0.57.

After body pressure drag. - Nacelle afterbody pressure drag including power effects was assessed using the method of Reference A8. Afterbody drag coefficient as a function of fan nozzle pressure ratio is given on Figure A36. These data show a favorable drag trend as nozzle pressure ratio increases. Afterbody pressure drag is minimized by designing the aft fan cowl with a boattail angle of 0.16 rad (9 deg).

P&W 433 Nacelle skin friction drag. - The skin friction coefficient is determined by the Prandtl-Schlichting equation, and the nacelle outer surface friction drag is calculated based on a wetted area of 50.1 m^2 (539 ft^2) and a length of 7.14 m (23.43 ft). The nacelle drag is shown in Figure A37 as a function of altitude and flight Mach number.

Nozzle velocity coefficients. - Nozzle performance calculations are based on velocity coefficients utilized in the Pratt & Whitney Customer Computer Deck Program, Reference A9. The fan and primary nozzle velocity coefficients are shown in Figure A38 as a function of nozzle pressure ratio.

Cooling drag. - Corrected nacelle cooling drag is shown in Figure A39 as a function of fan nozzle pressure ratio. These drag levels are obtained from similar data calculated for the C-5 nacelle installation.

Bleed flow schedule. - The schedule of high pressure compressor interstage bleed flow, as used in the performance calculations, is shown in Figure A40. Engine bleed is assumed to be shut-off to maximize power for takeoff and climb to the cruise altitude. Auxiliary power requirements are supplied by an inflight-operable APU. Bleed is turned-on at cruise altitude and a constant flow of 1.11 kg/sec (2.45 lb/sec) per engine is maintained up to 7010 m (23,000 ft) which provides sea-level atmosphere for the cabin up to that altitude. Above 7010 m (23,000 ft), approximately 60,500 N/m^2 (8.8 psi) cabin differential is maintained by a linear reduction in bleed flow.

Power extraction. - Engine power extraction is based on an estimated average power requirement of 82,027 W (110 HP) per engine to power the airplane electrical and hydraulic systems.

Exhaust duct pressure losses. - Fan duct and primary tailpipe total pressure losses are accounted for in performance calculations as constant percentages of the fan and turbine discharge total pressures respectively. The fan duct has one acoustic splitter in approximately two-thirds the fan duct length with acoustic treatment on the outer and inner duct walls opposite the splitter. The fan duct total pressure loss is 3.4 percent. The primary exhaust duct has acoustic treatment on the outer wall of the tailpipe and upstream portion of the plug inside the tailpipe. The primary duct total pressure loss is one percent.

Installed performance. - Installed performance data for the P&W STF 433 engine in the aircraft are calculated using Reference A9 and the installation losses developed by the Lockheed-Georgia Company. All installed engine performance data are for a reference rated thrust of 177,929 N (40,000 lb) and for a nacelle installation of the type used for the study configurations. The Final Configuration, however, has an additional one percent increase in SFC to compensate for inlet flow downwash on all inlets and for inlet bend losses on the center aft fuselage mounted engine.

Engine performance data were generated for all flight conditions necessary for the aerodynamic and acoustic performance analyses. Thrust and SFC were computed for a range of altitudes from sea level to 15,250 m (50,000 ft), for standard and non-standard day operation over a range of engine power settings, and for a range of flight Mach numbers. A portion of the more pertinent performance data only are presented herein for ICAO standard atmosphere except as noted.

C-3

Takeoff. - Installed engine pylon net thrust and SFC, at takeoff rating, are shown in Figures A41 and A42 for altitudes of sea level and 305 m (1000 ft), respectively. These data are shown as a function of flight Mach number and standard day and standard day plus 19.44°C (35°F).

Maximum cruise. - Engine maximum cruise rated net pylon thrust and SFC are shown in Figures A43 and A44, respectively. These data are presented as carpet plots for a range of altitudes from sea level to 15,250 m (50,000 ft) and a range of flight Mach numbers from 0.3 to 1.0.

Partpower cruise. - Carpet plots of engine partpower cruise SFC are presented in terms of maximum cruise SFC at the specific flight Mach number condition as shown in Figures A45, A46 and A47 for altitudes of 1524 m (5000 ft), 6096 m (20,000 ft) and 11,000 m (36,089 ft), respectively. These data are shown as a function of fractional pylon net thrust referenced to maximum cruise for a range of values from 0.2 to 1.0, and flight Mach numbers over the range of 0.3 to 0.7 on Figure A45, and 0.4 to 1.0 on Figures A46 and A47.

Parametric scaling data. -

Engine scaling. - The P&W STF 433 engine scaling range of 88,964 to 231,308 N (20,000 to 52,000 lb) has been expanded to a range of 44,482 to 289,134 N (10,000 to 65,000 lb) for the configuration sizing studies. The engine scaling data are shown in Figure A48, and are based on a reference rated thrust of 177,929 N (40,000 lb). The P&W STF 433 scaling data, from Reference A4, for engine weight, length and diameter have been curve-fitted to the mathematical expressions shown for the respective parameters. The specific SFC scaling data are based on other P&W parametric engine scaling data, and show an SFC penalty for engines whose rated thrust is below 88,964 N (20,000 lb).

Nacelle scaling. - The nacelle scaling data used in the configuration sizing studies are shown in Figure A49. These data are based on material from Reference A10. The nacelle dimensional data are referenced to the engine maximum diameter, and the weight data are referenced to the engine weight.

Nacelle external location. - A study of the flow field in the vicinity of the external engine intakes was made for the Cycled Baseline Configuration, Figure A50, in the landing and cruise configurations. The flow downwash angles, Figure A50, relative to the engine inlets are greater for the right-hand nacelle installation as compared to the left-hand nacelle. This potential flow study indicated that the flow approaches the right-hand inlet at an approximate mean angle of 0.17 rad (10 deg) during landing and at 0.04 rad (2 deg) during cruise. Cruise flight does not, therefore, pose an inlet flow distortion problem but the inlet configuration and location was found to be undesirable for the landing configuration.

The external nacelles for the Final Configuration were therefore moved upward 0.25 of a nacelle diameter and the inlet planes raked aft 0.28 rad (16 deg). The upper half of each inlet incorporates blow-in doors for low speed flight. These changes were considered sufficient to provide the engines with suitable airflow characteristics.

Environmental Effects

Community noise requirements. - Aircraft whose application for type certificate is dated after November 5, 1975 and whose type certification and introduction into service will take place in the mid 1980's will be required to comply with the noise certification limits of Reference A5 as modified by Reference A6. A reference airplane weighing 136,077 kg (300,000 lb) and powered by three engines, under the new requirements must be capable of meeting existing FAR 36 levels reduced by 9 EPNdB on takeoff sideline, 6 EPNdB on takeoff flyover, and 4 EPNdB on approach flyover. These new limits recognize the presence of the aircraft self noise floor (airframe noise) and engine noise floors (core, combustion, and jet noise) of conventional designs. It further recognizes the need for some degree of noise design tolerance, and the need for designing aircraft below the noise standards so that growth derivatives will also comply with the noise standards.

The noise design goal for the Final Configuration is for a 90 EPNdB soundprint area of $12.94 \times 10^6 m^2$ ($5.0\ mi^2$), considering both approach and takeoff. This soundprint area for the Final Configuration corresponds approximately to the new noise certification limits previously outlined.

Takeoff profiles. - The takeoff profiles for the Oblique Wing Final Configuration and the conventional configuration are shown on Figure A51. The noise at the Reference A5 measurement locations depends upon airplane height, speed and power requirements. The Final Configuration takeoff flyover at takeoff power attains an altitude of 432.8 m (1420 ft) at the 6.48 km (3.5 n mi) point. On approach, over the 1.852 km (1.0 n mi) point, engines are operating at about 26 percent of the available net power. Since approach noise often controls the extent of the acoustic treatment in the nacelles, the low thrust setting resulting from the achievement of low drag by the use of the high unswept aspect ratio of the wing is an attractive feature of the Oblique Wing Concept.

Oblique wing acoustic configuration. - The noise floor of the airplane is that due to the airframe which is particularly limiting on approach. For the Oblique Wing Final Configuration at the 1.852 km (1.0 n mi) measuring point, with a landing approach lift coefficient, $C_{L_{APP}} = 1.45$, in combination with the high unswept aspect ratio of 12.0, a "clean" airframe noise level 3.7 EPNdB lower than that of the equivalent conventional configuration is generated. During approach, these noise levels are increased due to the deployment of the wing trailing edge flaps and to the open wheel wells created by extension of the landing gears. The noise increment due to the deployment of the Final Configuration single slotted Fowler type flap system is less than that of the conventional configuration which requires a double slotted Fowler type flap system having external tracks, actuators, and fairings and which experiences engine exhaust impingement from the wing mounted engines. The noise level associated with the Final Configuration is 91.5 EPNdB with potential for reduction to 89.9 EPNdB with improvements in landing gear design and elimination of wheel fairing cavities, as compared to 93.5 EPNdB for the conventional configuration.

The airframe noise floors for the Final Configuration are 10.5 and 12.5 EPNdB below the 102 EPNdB level proposed in Reference A6.. The airframe noise floor corresponding to the 102 EPNdB limit is the result of measurement on fixed wing aircraft such as the Boeing 727, 747,and the Lockheed C-5A. The lower airframe noise level of the Final Configuration indicates the possibility of compliance with lower noise certification limits on approach since the Oblique Wing Concept provides a partial solution to the problem of reducing the airframe noise floor. This proposed limit could be reduced by a further 4 EPNdB from 102 EPNdB to 98 EPNdB.

Additional noise reduction benefits accrue to the Oblique Wing Concept due to the high unswept aspect ratio and low thrust settings on approach leading to low noise levels on approach.

Engine and nacelle acoustic characteristics. - The Pratt & Whitney STF 433 engine is designed for low noise characteristics. Fan source noise control is obtained by using a low noise two-stage fan. At the cruise design point the fan pressure ratio is 1.9 and 1.7 at takeoff. Fan and primary nozzles are co-planar.

Jet noise control is exercised through the engine cycle selection such that the primary jet does not dominate the jet noise resulting in a relatively low jet noise floor.

Nacelle acoustic characteristics are obtained using the data of Reference A10. Compliance with the noise certification regulations of Reference A6, requires a noise reduction of 9 EPNdB at the critical location. This includes a 3 EPNdB prediction/design/test tolerance. The nacelle design necessary to achieve aircraft compliance with the noise certification limits is shown on Figure A33. An acoustic liner of advanced design fully integrated into the nacelle and load carrying has the absorption characteristics approaching those of a bulk absorber, which include high peak attenuation, increased band width, and improved directivity over current current technology liners.

The impact of the engine nacelle acoustical design on engine performance is included in all configuration performance analyses and includes weight, thrust, and specific fuel consumption changes.

APPENDIX B

AEROELASTIC ANALYSIS

By Erwin Johnson *
NASA-Ames Research Center
Moffett Field, California

The Subsonic Oblique Wing Transport Concept described in this report was analyzed for aeroelastic stability by NASA-Ames Research Center. Due to time limitations, only the design with an unswept aspect ratio of twelve was analyzed. This appendix summarizes the results of this analysis, pointing out the unique features of the aeroelastic response of an Oblique Wing Concept. After a brief description of the method used in the analysis, results are presented and the critical flutter conditions are identified. Finally, some conclusions and observations are made based on this preliminary analysis.

Oblique Wing Aeroelastic Behavior

The asymmetric nature of an Oblique wing causes unique behavior in almost all disciplines of aircraft analysis that cannot be accounted for by methods developed for conventional aircraft. Aeroelastic behavior, which is no exception to this, is in fact, the area of greatest controversy as to the correct analytical methodology. In particular, the data of References B1 and B2 show that analyses that model the swept forward portion of the wing as a beam clamped at the root can be unnecessarily conservative. When clamped at the root, the swept forward wing is susceptible to static aeroelastic divergence. However, if the wing is allowed a rigid body roll degree of freedom, the divergence no longer occurs and the critical aeroelastic condition becomes typically a low frequency flutter mode that is due to the coupling of wing bending and rigid body roll.

Thus, to adequately analyze the Oblique wing, it is necessary to consider the entire asymmetric wing and to include rigid body degrees of freedom in the analysis. The results reported here were obtained at NASA-Ames using computer algorithms and included the above features.

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* National Research Council Research Associate.

Methods of Analysis

The basis for the flutter analysis performed for this study is contained in Equation (1).

$$\left(\frac{(1+ig)}{\omega^2}\,[K] - [M] - [C]\right)\{w\} = 0 \tag{1}$$

where $[K]$, $[M]$, and $[C]$ are the stiffness, mass and aerodynamic matrices respectively, obtained from a finite element analysis of the aircraft, ω is the frequency of oscillation, g is the structural damping, and $i = \sqrt{-1}$.

The methods used for the construction of each of the matrices in Equation (1) are:

The stiffness matrix, $[K]$, was generated by means of a finite element beam representation using the stiffness properties, i.e., EI and GJ distributions, supplied by Lockheed. These data are listed in Table B1 for reference purposes.

A consistent mass matrix, $[M]$, listed in Table B1, was also generated using Lockheed supplied data. In addition to the inertial properties of the wing, the fuselage mass and inertia were represented by the appropriate values assuming these quantities to be concentrated at the wing pivot.

The aerodynamic matrix, $[C]$, was obtained using the computer program of Reference B3 which is based on doublet lattice methods. The aerodynamic considerations are functions of parameters such as Mach number, reduced frequency of oscillation, wing sweep angle, and atmospheric density and the computer program allows the input of asymmetric wing planforms.

Instead of solving Equation (1) directly, the system is reduced by the use of generalized modes corresponding to the normal modes of vibration. Typically, ten normal modes were retained in this procedure. Where rigid body modes were present in the analysis, such modes were treated in the manner described in Reference B4.

The well known V-g method of analysis, (Reference B5), was used to evaluate the aeroelastic stability of the system expressed in terms of generalized coordinates.

Aeroelastic Study Results

Velocity, dynamic pressure, and frequency of the aeroelastic instabilities for a series of sweep angles are listed in Table BII. Three different constraint conditions at the wing pivot were investigated in order to determine the effects of rigid body degrees of freedom. These constraints were:

TABLE B1 MASS AND INERTIA DATA

2y/b: Fraction of Span	Stiffness				Lumped Mass Data			
	EI		GJ		Mass		Torsional Inertia	
	$N\text{-}m^2$	($lb\text{-}in^2$)	$N\text{-}m^2$	($lb\text{-}in^2$)	kg	(slugs)	$kg\text{-}m^2$	($slug\text{-}in^2$)
0.05	44.6×10^6	(49.9×10^{10})	20.6×10^6	(23.1×10^{10})	1514.8	(103.8)	3840.	(40.8×10^4)
0.15	33.4	(37.5×10^{10})	18.8	(21.0×10^{10})	1312.0	(89.9)	2880.	(30.6×10^4)
0.25	24.3	(27.2×10^{10})	13.4	(15.0×10^{10})	1123.7	(77.0)	2112.	(22.4×10^4)
0.35	16.8	(18.8×10^{10})	9.01	(10.1×10^{10})	950.4	(65.12)	1510.	(16.0×10^4)
0.45	11.0	(12.3×10^{10})	6.15	(6.90×10^{10})	791.1	(54.21)	1046.	(11.1×10^4)
0.55	6.57	(7.37×10^{10})	3.75	(4.21×10^{10})	646.8	(44.32)	700.	(7.43×10^4)
0.65	3.57	(4.00×10^{10})	2.14	(2.40×10^{10})	516.6	(35.40)	446.	(4.74×10^4)
0.75	1.52	(1.70×10^{10})	0.919	(1.03×10^{10})	401.3	(27.5)	269.	(2.86×10^4)
0.85	0.606	(0.680×10^{10})	0.499	(0.560×10^{10})	300.5	(20.6)	151.	(1.60×10^4)
0.95	0.316	(0.354×10^{10})	0.294	(0.330×10^{10})	214.5	(14.7)	77.	(0.814×10^4)

Fuselage Inertia Data: m_f = 53660 kgs (3677 slugs)

I_{xx_f} = 73218 kg - m^2 (7.78×10^7 slug-in^2)

I_{yy_f} = 1.14×10^7 kg - m^2 (1.21×10^9 slug-in^2)

I_{zz_f} = 1.11×10^7 kg - m^2 (1.18×10^9 slug-in^2)

The elastic axis of the wing is at the 38.5% chord

The center of mass of the wing is at the 45% chord

TABLE BII AERODELASTIC STUDY RESULTS

DIVERGENCE

	Λ = 0 rad (0 deg)			Λ = 0.523 rad (30 deg)			Λ = 0.785 rad (45 deg)		
	Velocity m/sec	(knots)	Frequency hz	Velocity m/sec	(knots)	Frequency hz	Velocity m/sec	(knots)	Frequency hz
Clamped	454	(883)	0	207	(403)	0	219	(427)	0
Free to Roll	454	(883)	0	Does not occur		0	Does not occur		
Free to Roll Pitch and Plunge	Not calculated		-	Does not occur		-	Does not occur		

LOW FREQUENCY FLUTTER

	Λ = 0 rad (0 deg)			Λ = 0.523 rad (30 deg)			Λ = 0.785 rad (45 deg)		
	Velocity m/sec	(knots)	Frequency hz	Velocity m/sec	(knots)	Frequency hz	Velocity m/sec	(knots)	Frequency hz
Clamped	Does not occur			Does not occur			Does not occur		
Free to Roll	Does not occur			305	(593)	1.03	250	(486)	1.11
Free to Roll Pitch and Plunge	Not calculated			293	(569)	1.24	258	(502)	1.15

HIGH FREQUENCY FLUTTER

	Λ = 0 rad (0 deg)			Λ = 0.523 rad (30 deg)			Λ = 0.785 rad (45 deg)		
	Velocity m/sec	(knots)	Frequency hz	Velocity m/sec	(knots)	Frequency hz	Velocity m/sec	(knots)	Frequency hz
Clamped	280	(545)	10.3	343	(667)	9.5	427	(830)	9.9
Free to Roll	280	(545)	10.3	343	(667)	9.5	427	(830)	9.9
Free to Roll Pitch and Plunge	Not calculated			Not calculated			469	(912)	10.3

- Clamped - which refers to a pivot constrained in all six degrees of freedom.
- Free to roll - which means that the aircraft can roll about an axis which is parallel to the fuselage and passes through the wing pivot, assuming the remaining pivot degrees of freedom are fixed.
- Free to roll, pitch and plunge - in which additional freedom in plunge and in pitch about an axis perpendicular to the fuselage, is allowed.

The results in Table BII were obtained at the following schedule of Mach numbers.

Wing Sweep and Mach Number Schedule	
Wing Sweep Angle - rad (deg)	Mach Number
0.0 (0.0)	0.40
0.523 (30.0)	0.5
0.7854 (45.0)	0.612

Table BII lists three types of flutter instability. The first of these is divergence, which is a static aeroelastic instability. Whenever rigid body roll is included in the analysis of wing with non-zero sweep, this divergence mode changes to the second type of instability which can be characterized as interaction between rigid body roll and wing bending. Since the frequency of this dynamic instability is less than the first vibration frequency of the wing, this is referred to as "low frequency flutter." The third instability is an interaction between wing bending and torsion deformations, referred to as "high frequency flutter."

V-g diagrams showing the response of the most important branches of the flutter roots are presented in Figures B1, B2 and B3.

The unswept case depicted in Figure B1 is a symmetric case that could have been analyzed without resort to special techniques to allow for asymmetry. Two types of instability occur for this case. The first is a symmetric bending torsion flutter which occurs at a velocity of 280 m/sec (545 knots). The second instability has the characteristics of a divergence and occurs at 454 m/sec (883 knots), which is beyond the range of concern. This second mode, as shown on Figure B1, loops to high values of structural damping and then approaches, but does not cross, g = 0 perpendicularly. For the unswept wing, the divergence mode is driven by the torsional, as opposed to transverse, deflection of the wing.

At 0.523 rad (30 deg) of sweep, there are three distinct types of instability, as shown in Figure B2. Assuming the wing clamped, the curve annotated 'A' is seen to diverge at a speed of 207 m/sec (403 knots). When the aircraft is allowed to roll, the divergence transforms into a low frequency flutter instability at a much higher velocity of 305 m/sec (593 knots) as shown by the curve at B. This is the behavior that previous investigations have predicted and indicates the importance of including a roll degree of freedom in the flutter analysis of an Oblique wing. When pitch and plunge degrees of freedom are included, a slight reduction in the flutter speed to 293 m/sec (569 knots) is observed as shown by the curve at C.

The high frequency flutter condition, curve D, is seen to occur at a velocity of over 330 m/sec (641 knots) and can thus be dismissed as being noncritical.

At 0.785 rad (45 deg) of sweep, the behavior is seen to be analytically similar to the 0.523 rad (30 deg) case as shown on Figure B3. The clamped divergence speed shows an increase to 219 m/sec (427 knots) while the low frequency flutter speed decreases to 250 m/sec (486 knots) when degree of freedom in roll is allowed and to 259 m/sec (503 knots) when three rigid-body degrees of freedom are included. The high frequency instability is again noncritical.

Flutter point matching. - The preceding results were obtained using aerodynamics valid only at the specified Mach number and sea level density. Although this gives an adequate indication of the critical aeroelastic condition in many cases, it is possible to perform a more accurate analysis by computing results for a series of Mach numbers and altitudes. In this study, the additional computations were performed for what was considered to be the most important condition, i.e., 0.785 rad (45 deg) of sweep with freedom to roll, pitch, and plunge.

The results obtained for the analysis are summarized in Figures B4 and B5. The nearly horizontal lines of Figure B4 were obtained by curve fitting through points obtained at a constant altitude and for a series of Mach numbers. The velocity and Mach numbers are consistent at one point only along these lines. This point is determined by finding the intersection of the constant altitude lines and lines of $V = a_\infty M$ (were a_∞ is the speed of sound at the given altitude).

Aeroelastic study conclusions. - A definite conclusion on the acceptability of this aircraft from a flutter standpoint would require a more detailed analysis that would necessarily take other flight conditions into account. In addition, it would be informative to determine the sensitivity of the flutter speed to such parameters as the fuselage flexibility and inertias, wing static unbalance, and stiffness. Some preliminary conclusions drawn from the results obtained from this study are:

- o The inclusion of the rigid-body roll degree of freedom significantly increases the speed of instability. At 0.785 rad (45 deg) of sweep, this increase amounts to 14 percent over the clamped divergence speed.

- o The further addition of rigid-body pitch and plunge degrees of freedom does not significantly alter the flutter speed.

- o The aircraft appears to flutter near to the prescribed flutter boundary as shown in Figure A32.

It should be observed that the aerodynamic methodology used for the flutter model is valid only for subsonic oscillations and that extrapolation of the results into the transonic range is therefore considered to be highly qualitative. Because the lift coefficient values, and therefore the aerodynamic forces obtained from subsonic theory, are higher than the actual transonic values of these coefficients, it can be argued on physical grounds that the results obtained here are conservative in the transonic region.

It is also pointed out that the data used in this analysis does not include the effects of fuel in the wing. The studies of References B1 and B2 indicate that if fuel were included, the critical flutter speed increases as these studies show that increasing the value for the ratio of wing to fuselage inertia in roll tends to increase the speed of the low frequency flutter.

APPENDIX REFERENCES

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A2. USAF. DATCOM.

A3. Kulfan, R. M., et al, "High Transonic Speed Transport Aircraft Study," Boeing Commercial Airplane Company, NASA CR-114658, September 1973.

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B2. Weishaar, T. A. and Crittenden, J. B., "Flutter of Asymmetrically Swept Wings." Submitted for publication to the AIAA Journal.

B3. Giesing, J. P., Kalman, T. P., and Rodden, W. P., "Subsonic Unsteady Aerodynamic for General Configurations," AFFDL-TR 71-5, Part 1, November 1971.

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B5. Bisplinghoff, R. L. and Ashley, H., "Principle of Aeroelasticity," Dover Publications, Inc., New York, 1975.

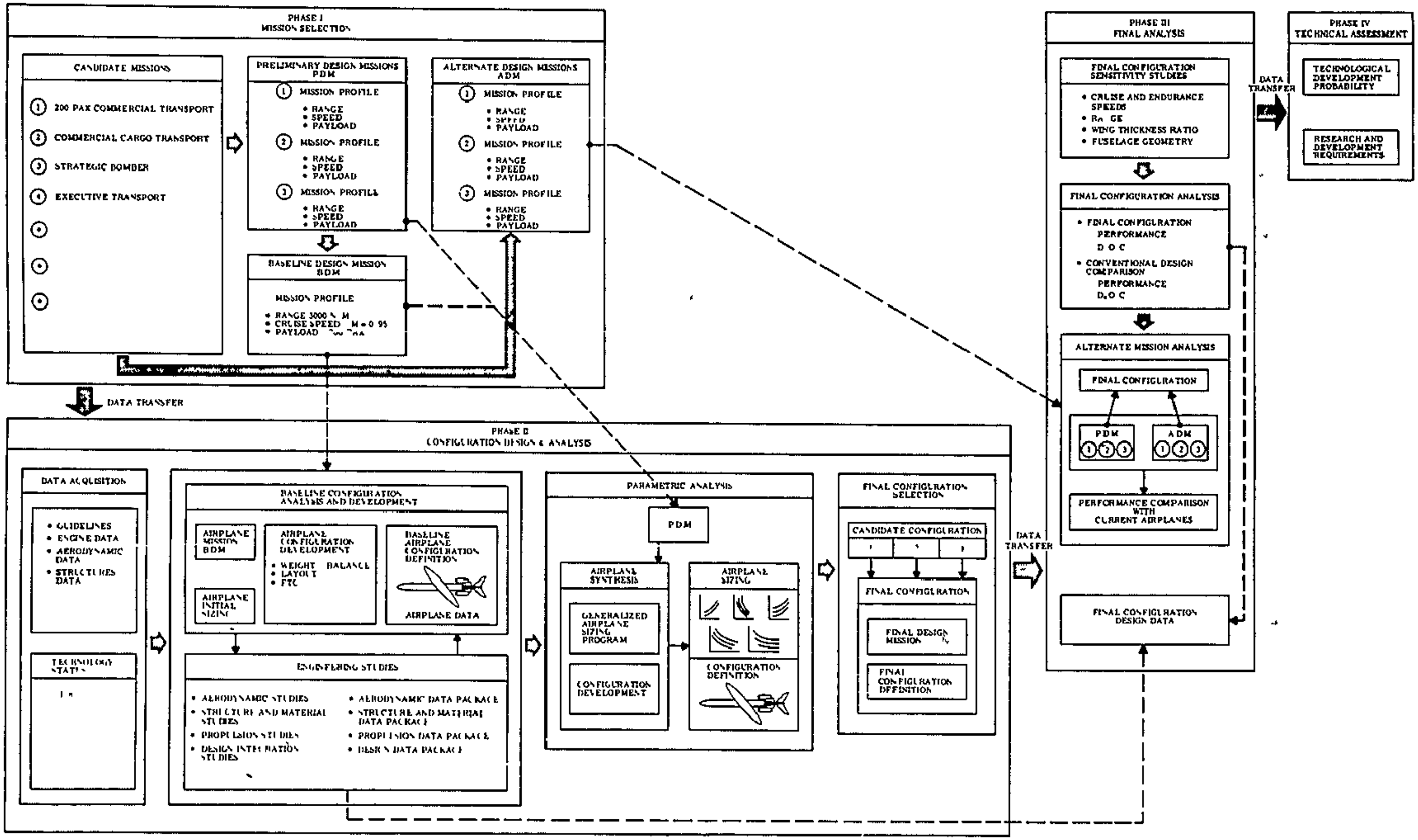

FIGURE 1 STUDY PLAN

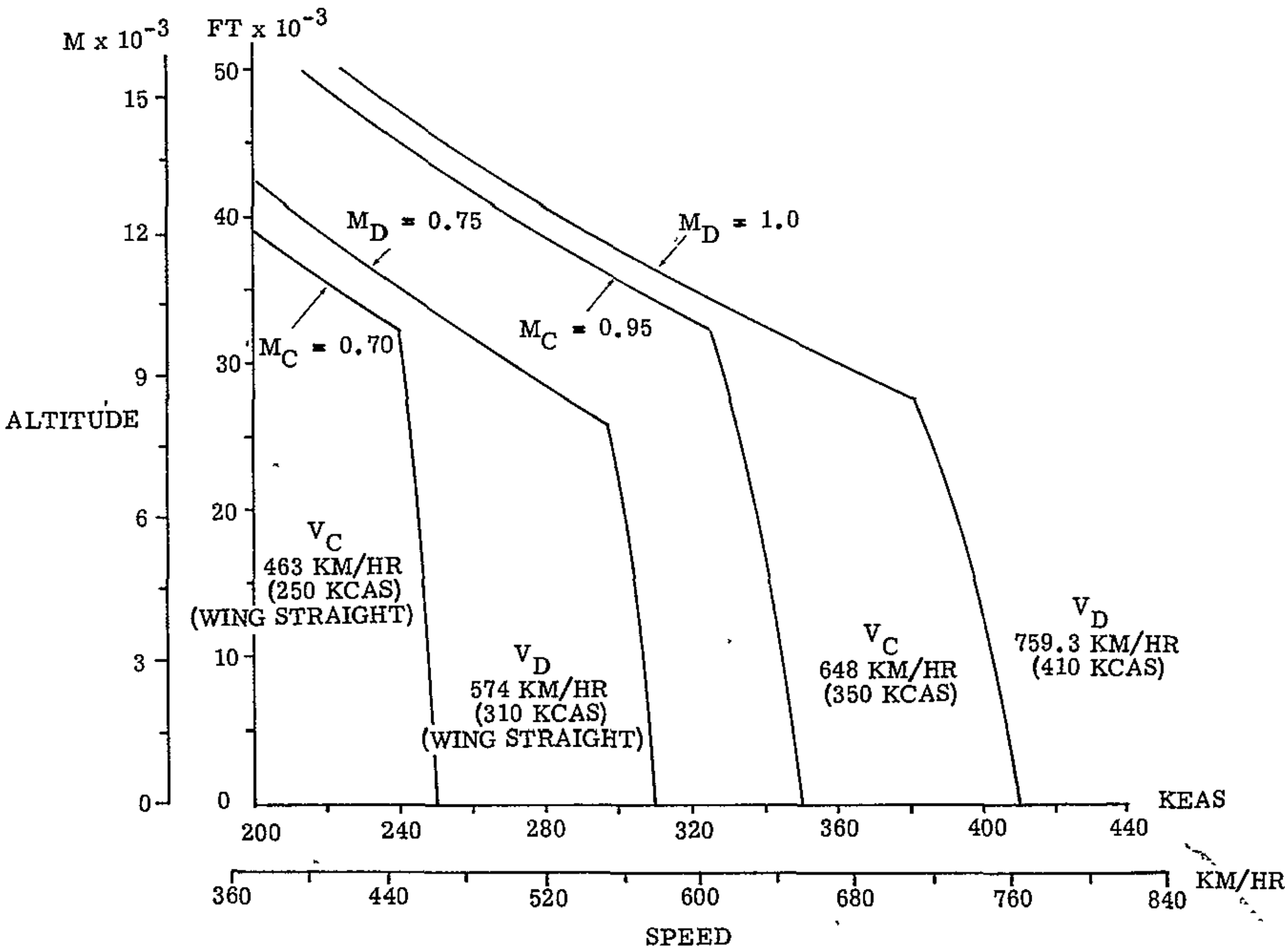

FIGURE 2 OBLIQUE WING CONCEPT SPEED-ALTITUDE SCHEDULE

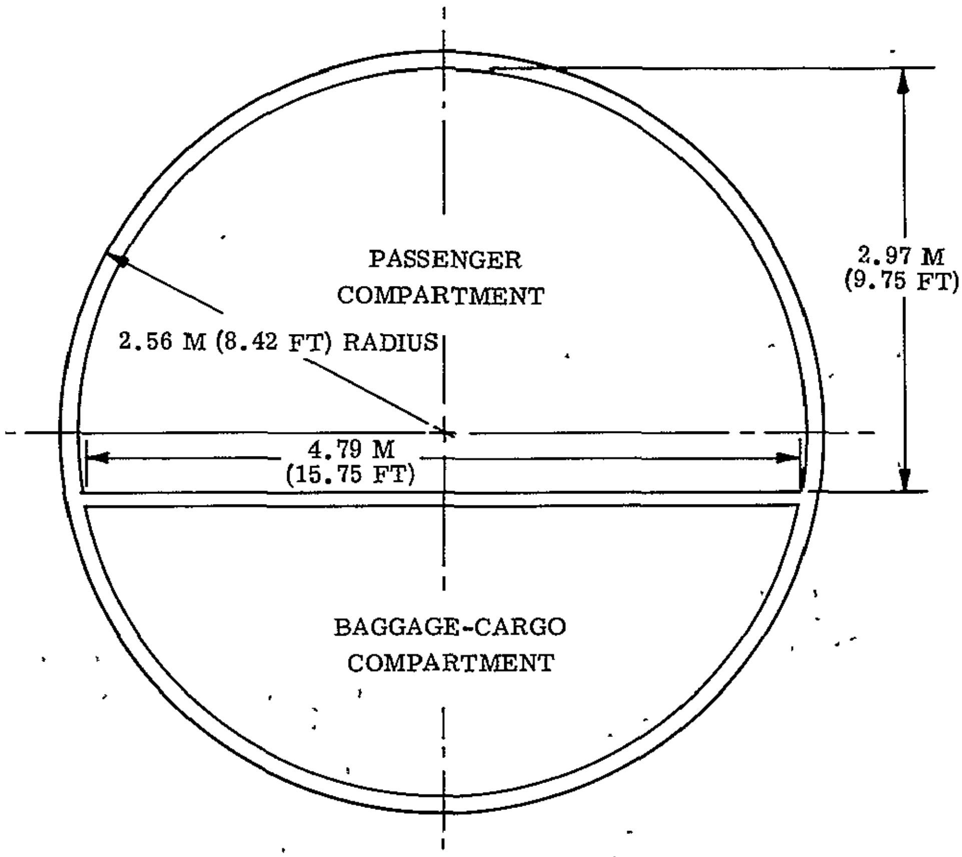

FIGURE 3 COMMERCIAL PASSENGER TRANSPORT - FUSELAGE CROSS SECTION

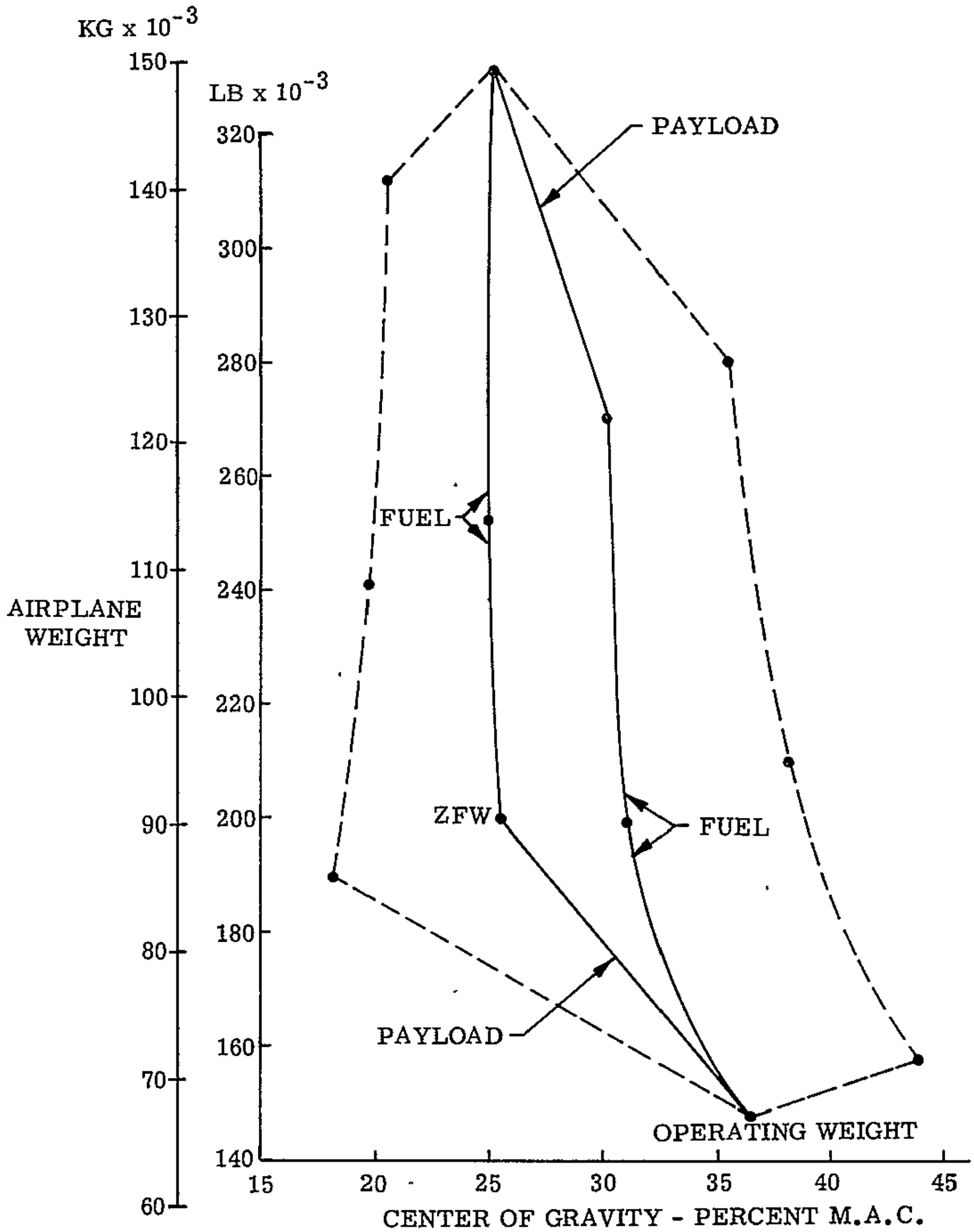

FIGURE 4 COMMERCIAL PASSENGER TRANSPORT - INITIAL CONFIGURATION C.G. ENVELOPE

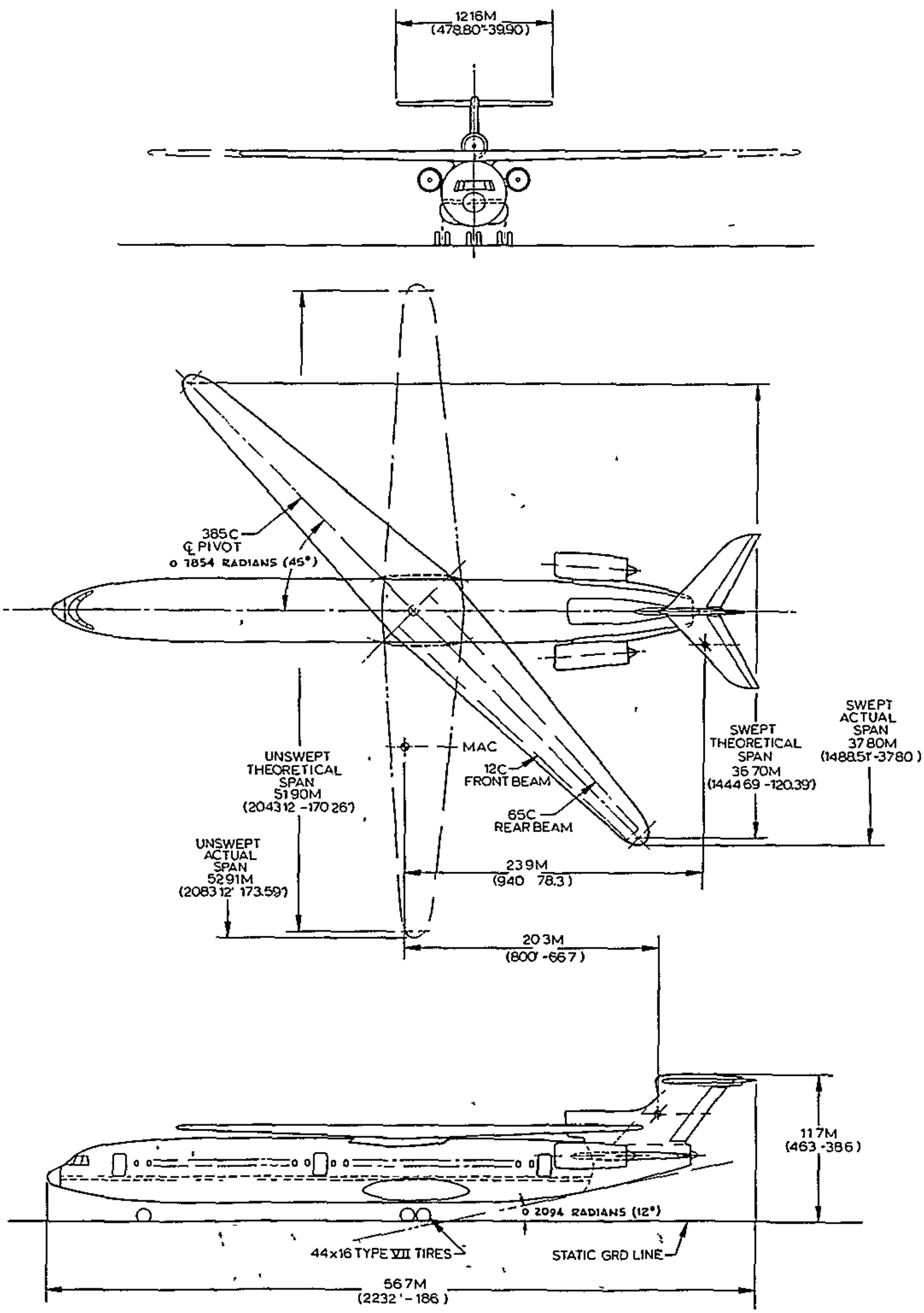

FIGURE 5 COMMERCIAL PASSENGER TRANSPORT BASELINE CONFIGURATION

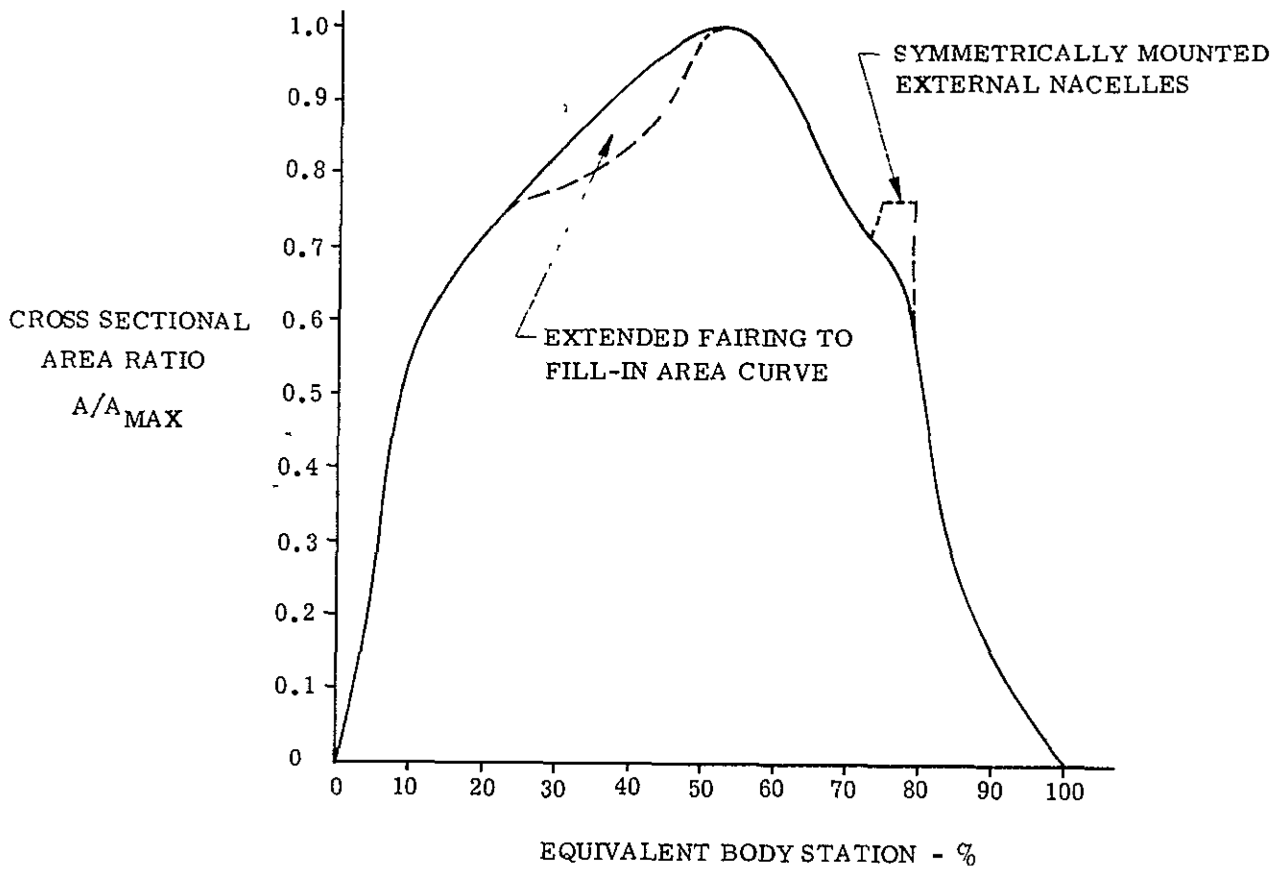

FIGURE 6 COMMERCIAL PASSENGER TRANSPORT BASELINE CONFIGURATION AREA DISTRIBUTION

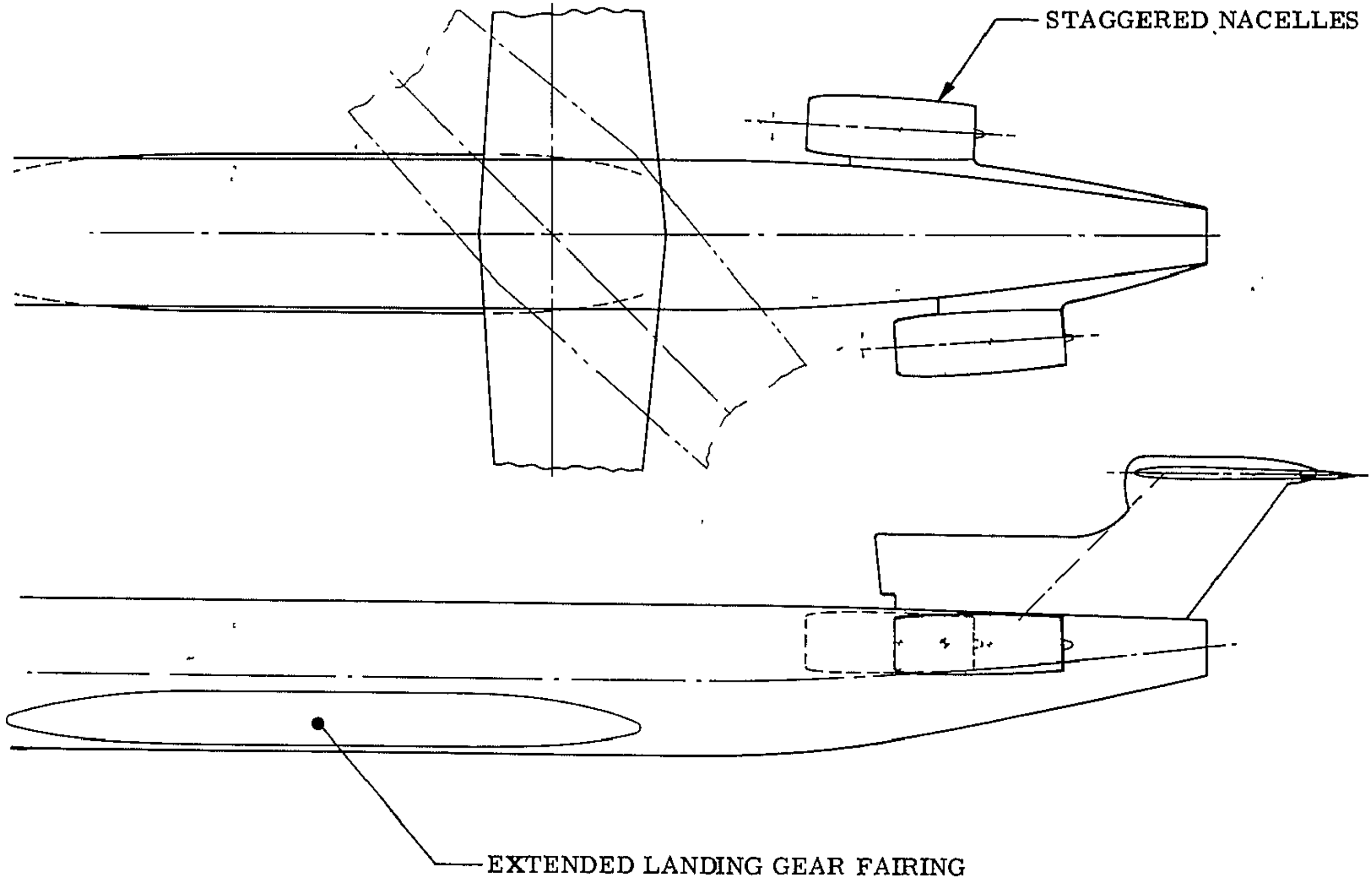

FIGURE 7 COMMERCIAL PASSENGER TRANSPORT
CYCLED BASELINE CONFIGURATION - CONFIGURATIONAL CHANGES

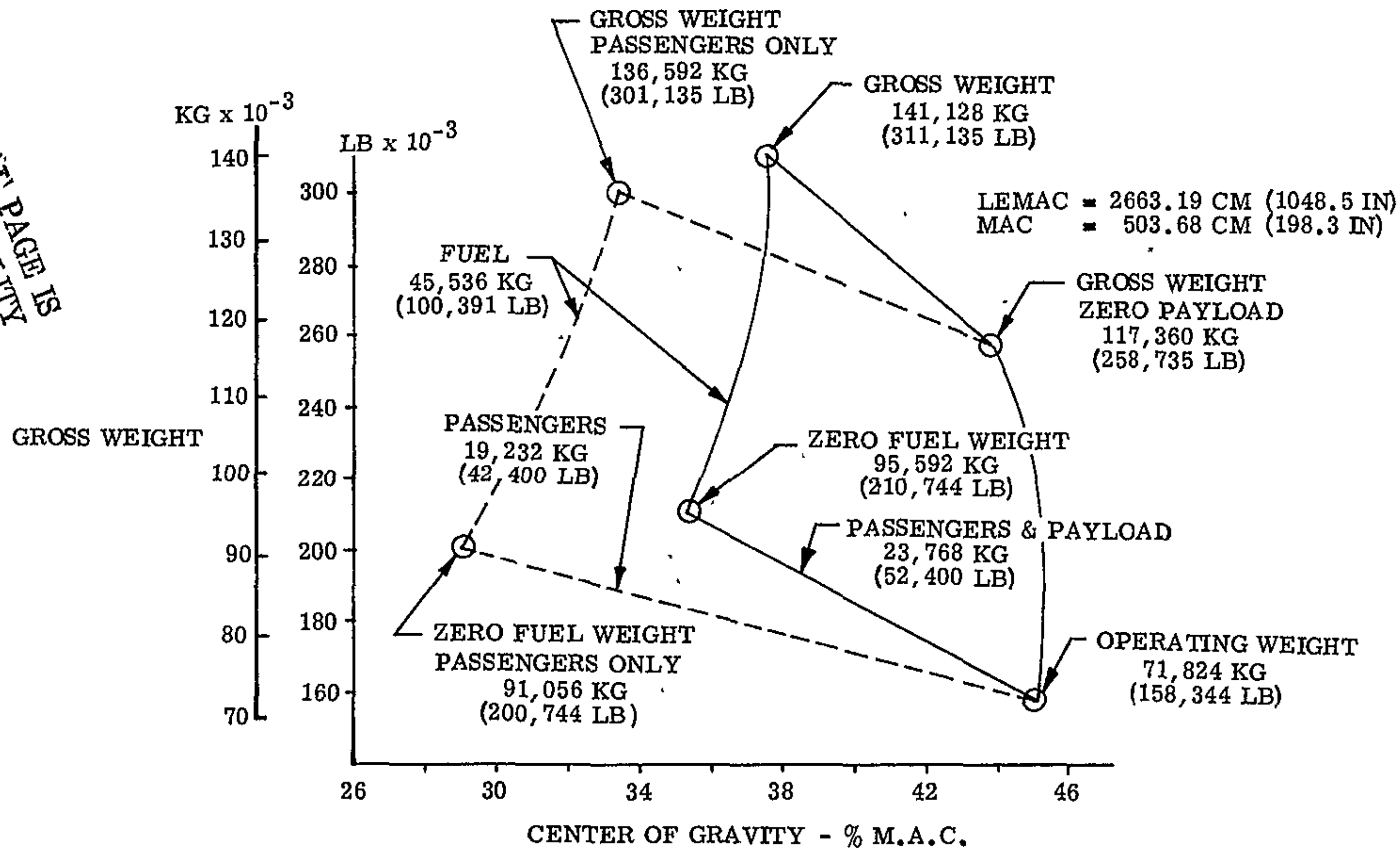

FIGURE 8 COMMERCIAL PASSENGER TRANSPORT
CYCLED BASELINE CONFIGURATION C.G. ENVELOPE

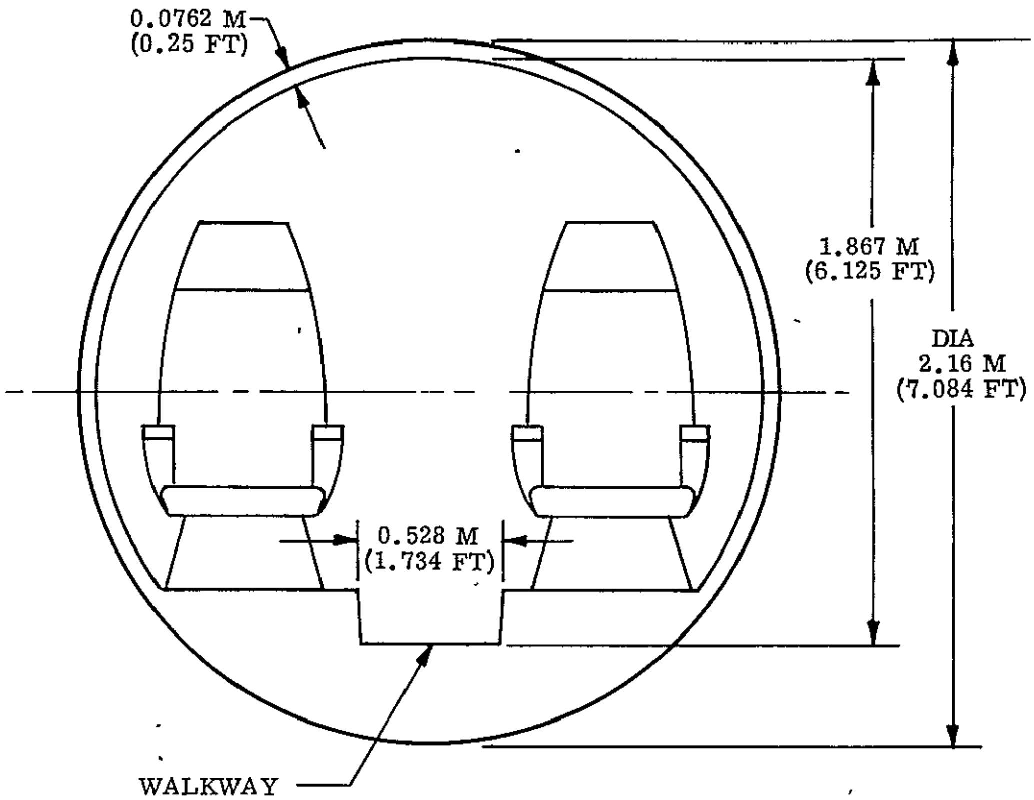

FIGURE 9 EXECUTIVE TRANSPORT - FUSELAGE CROSS-SECTION

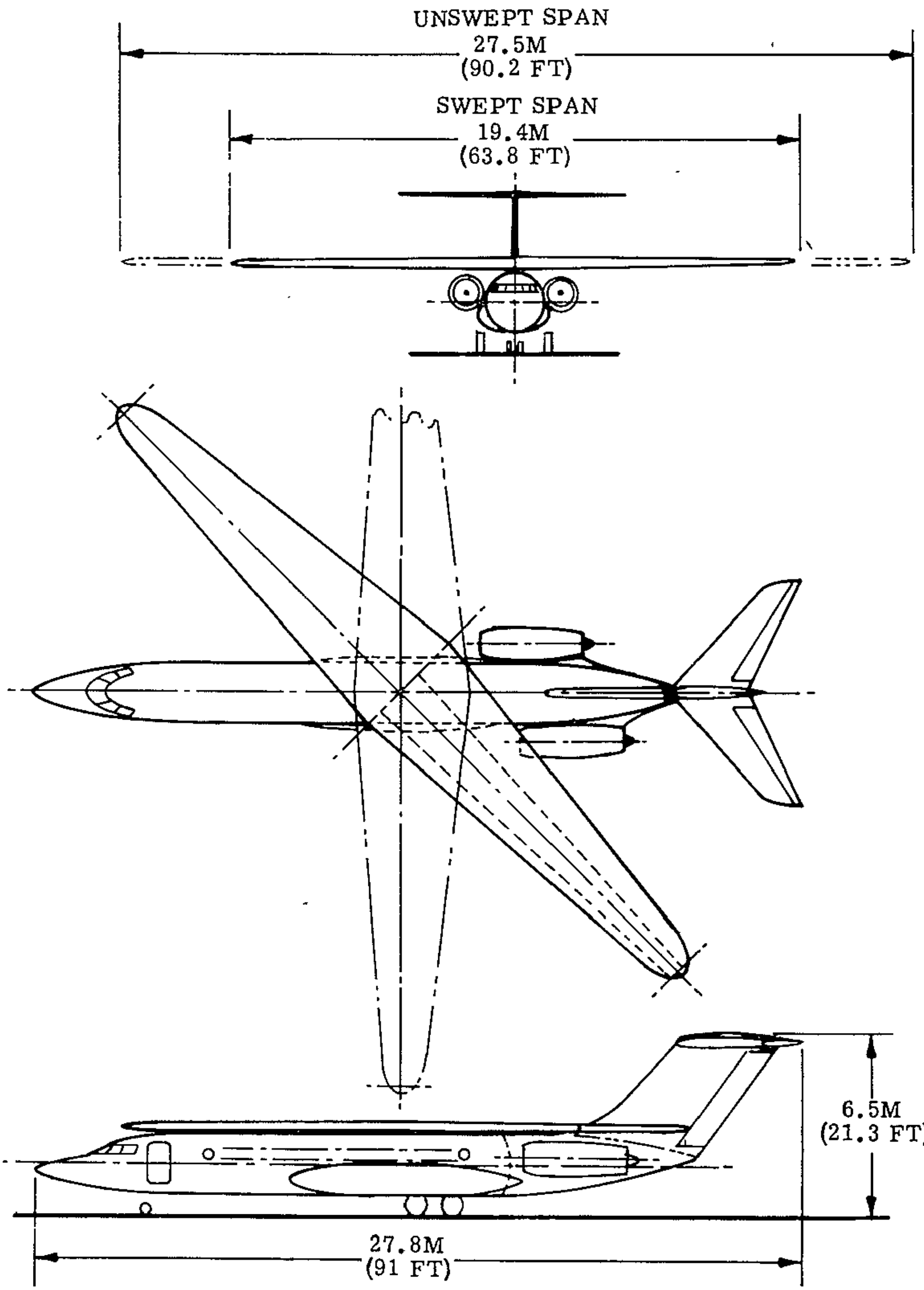

FIGURE 10 EXECUTIVE TRANSPORT - BASELINE CONFIGURATION

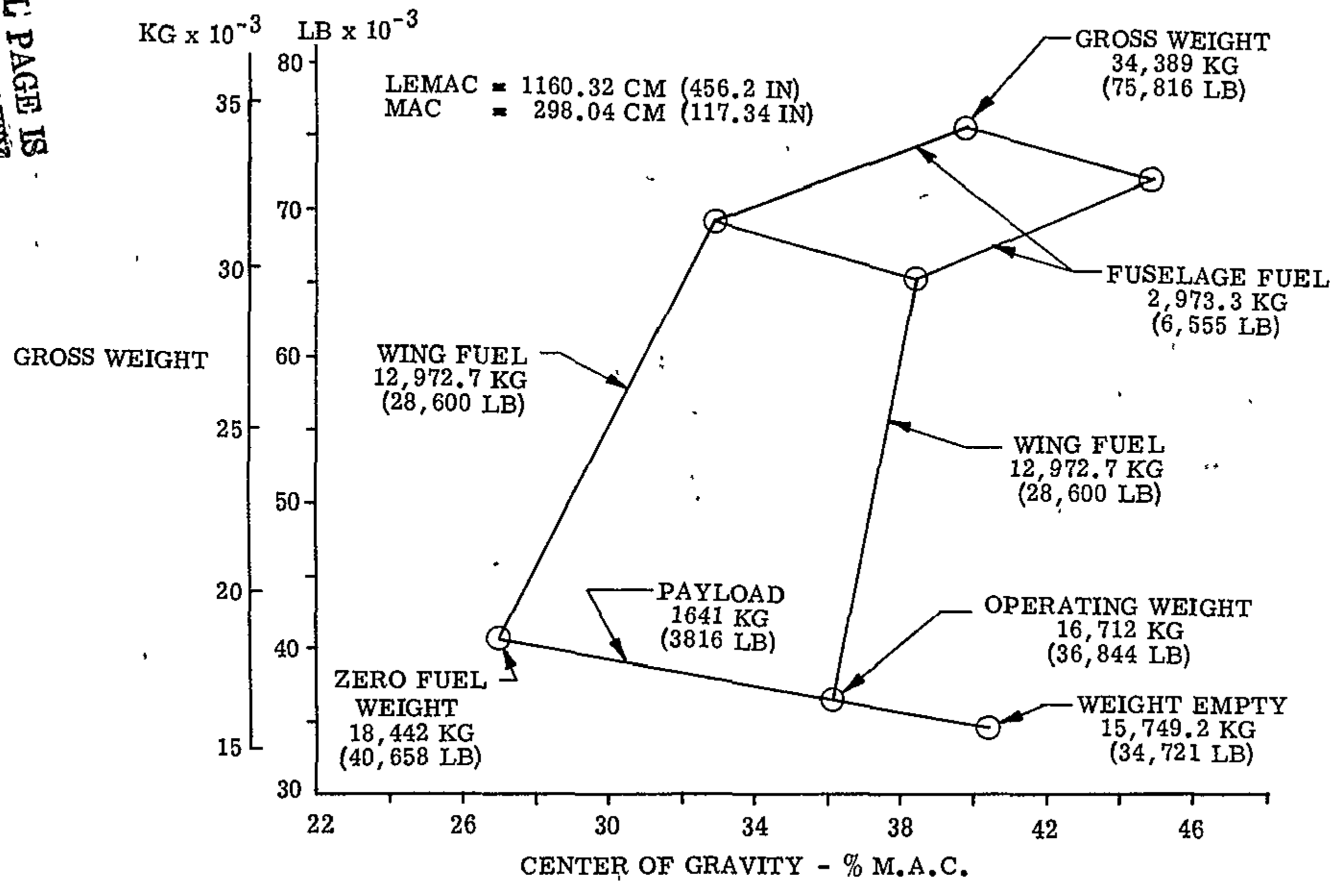

FIGURE 11 EXECUTIVE TRANSPORT - BASELINE CONFIGURATION C.G. ENVELOPE

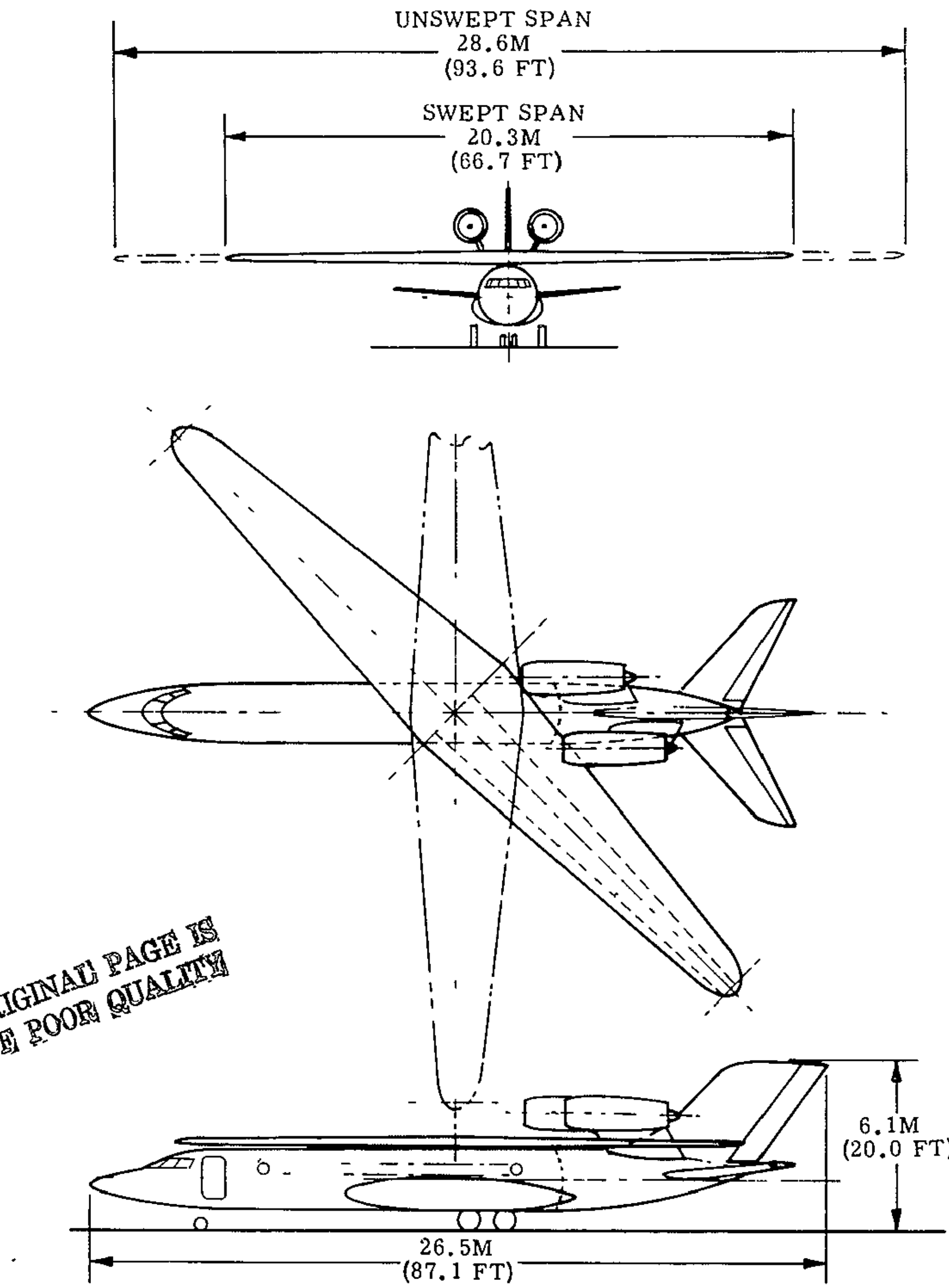

FIGURE 12 EXECUTIVE TRANSPORT -
CYCLED BASELINE CONFIGURATION

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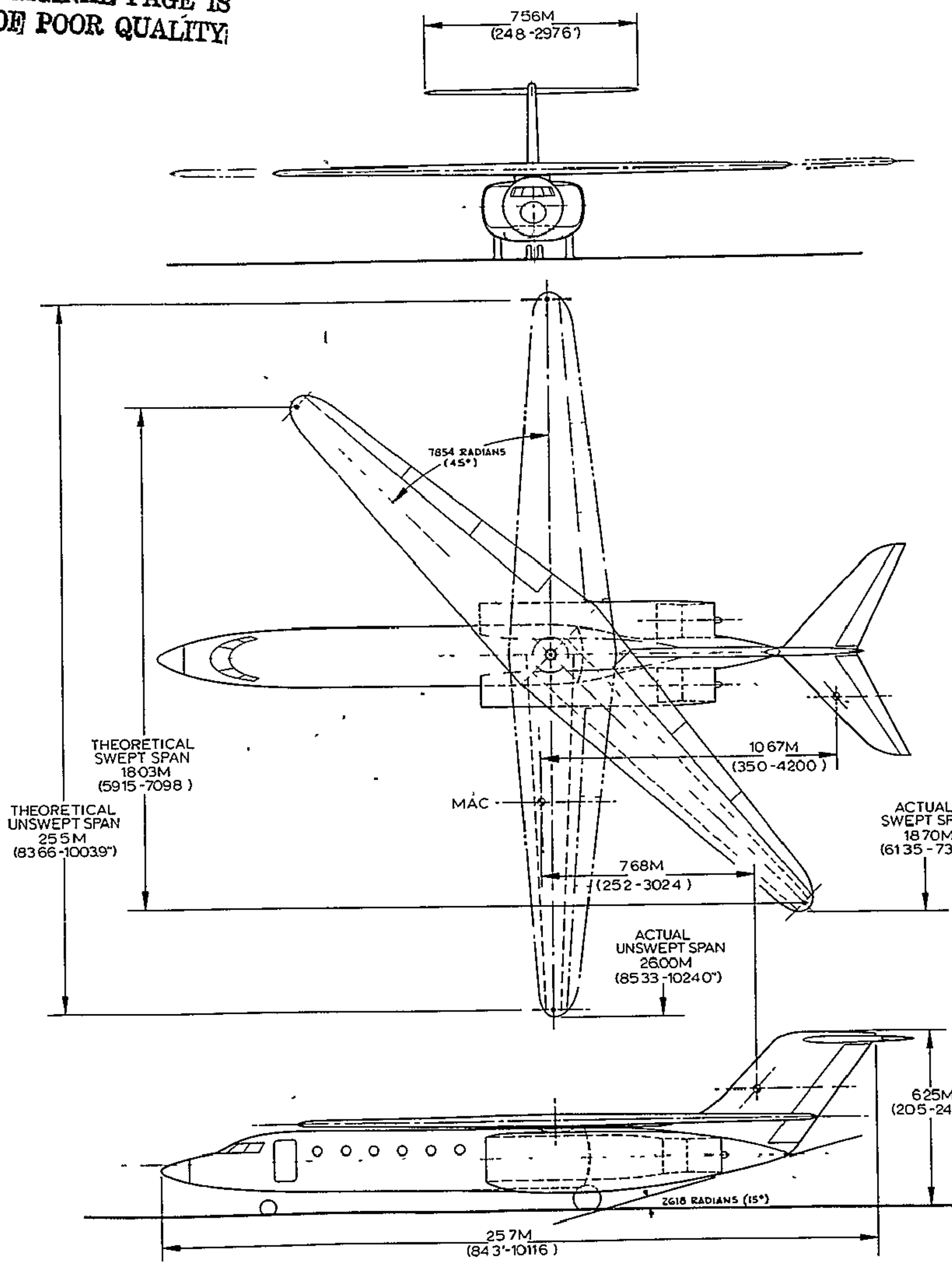

FIGURE 13 EXECUTIVE TRANSPORT -
CARRIER COMPATIBLE CONFIGURATION

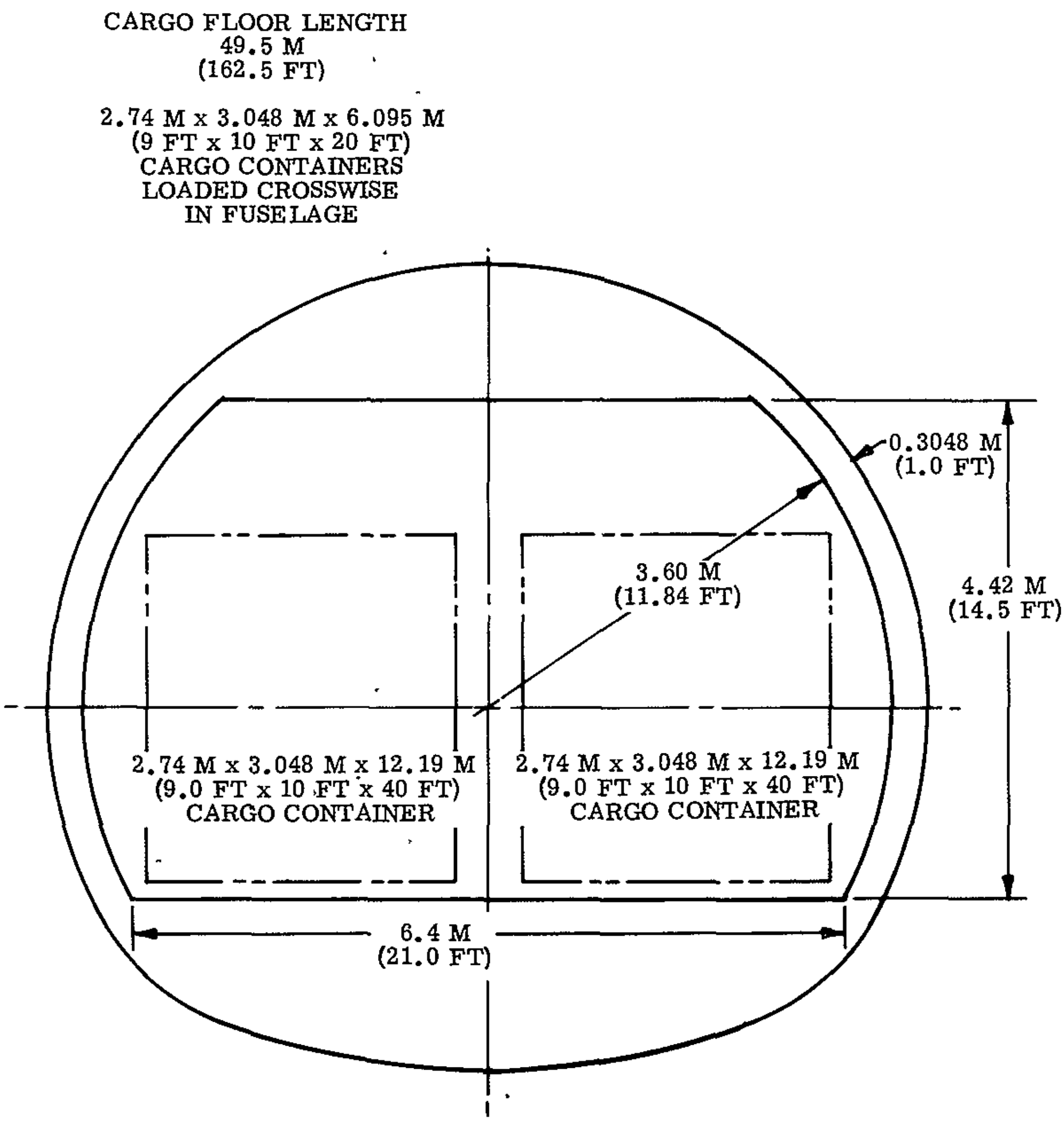

FIGURE 14 MILITARY CARGO TRANSPORT - FUSELAGE CROSS-SECTION

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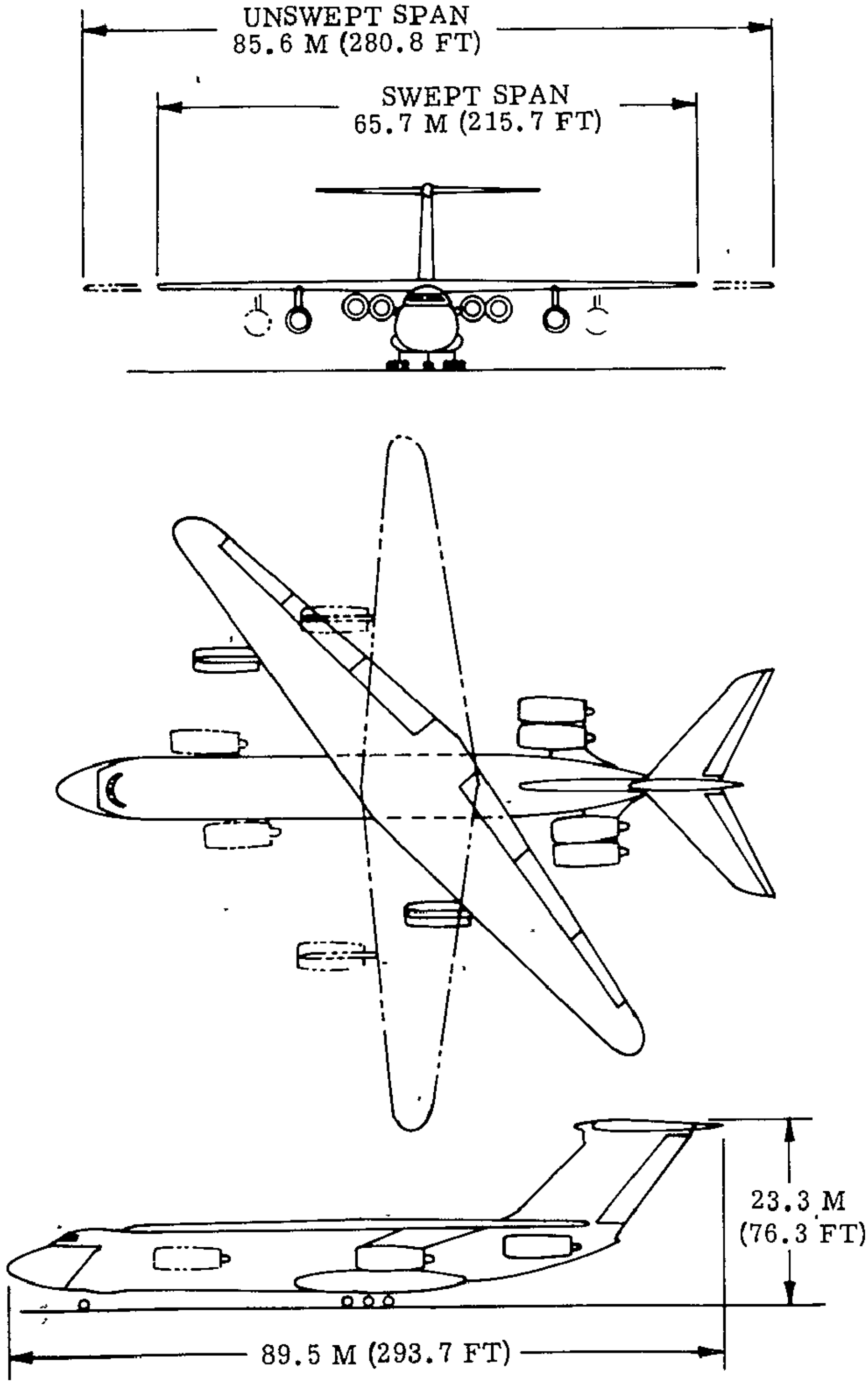

FIGURE 15 MILITARY CARGO TRANSPORT - TYPICAL BASELINE CONFIGURATION

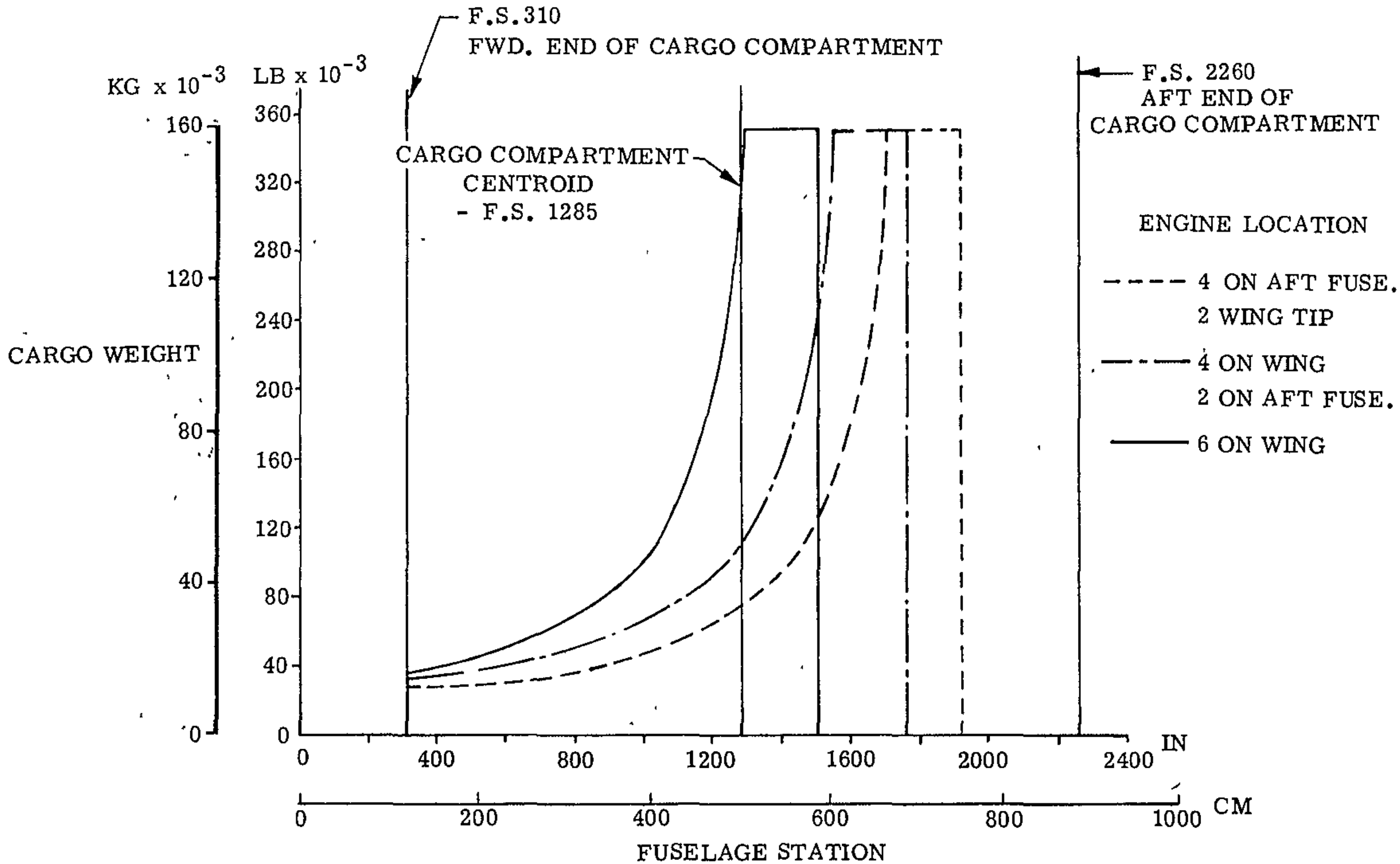

FIGURE 16 MILITARY CARGO TRANSPORT - LOADING DIAGRAM

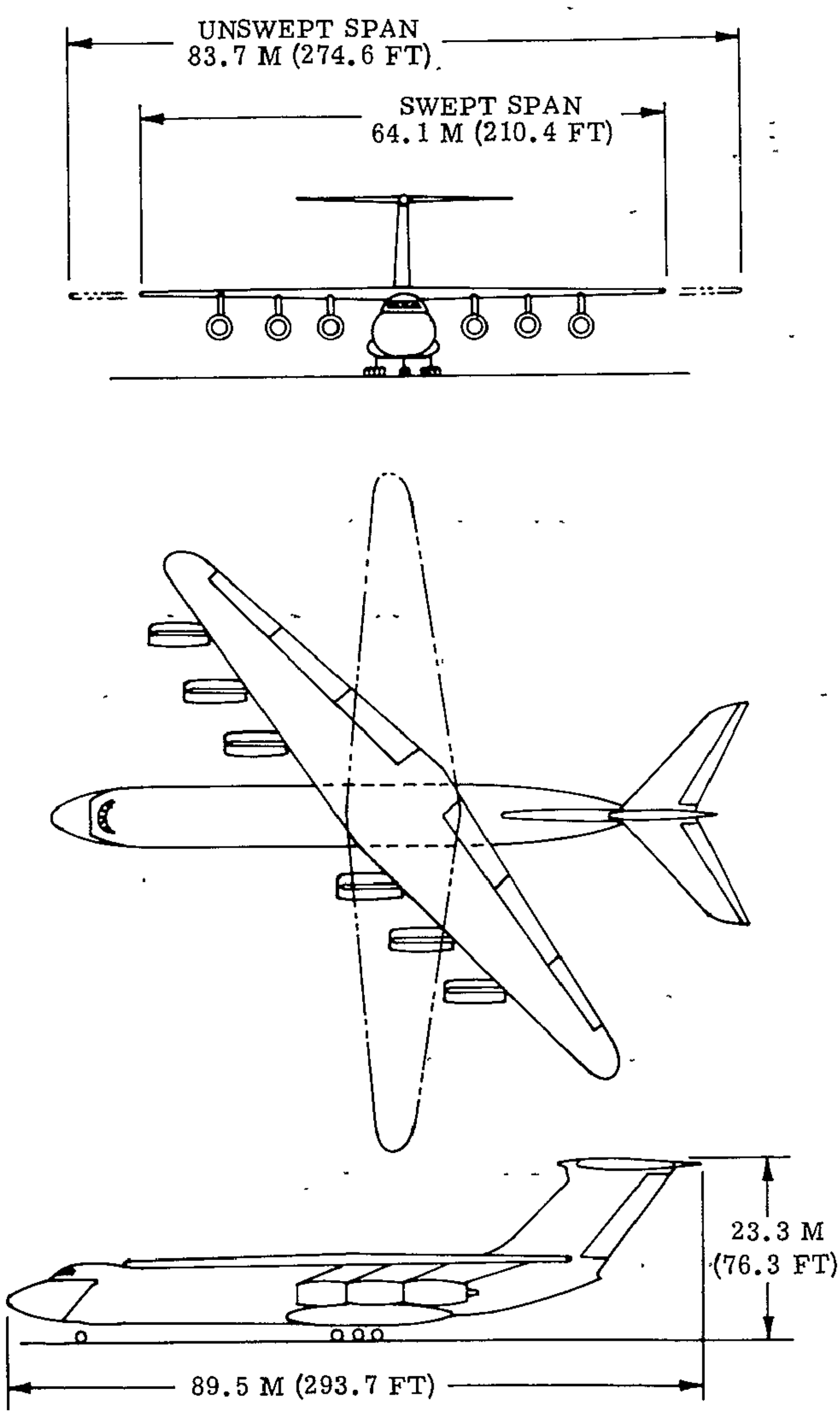

FIGURE 17 MILITARY CARGO TRANSPORT
CYCLED BASELINE CONFIGURATION

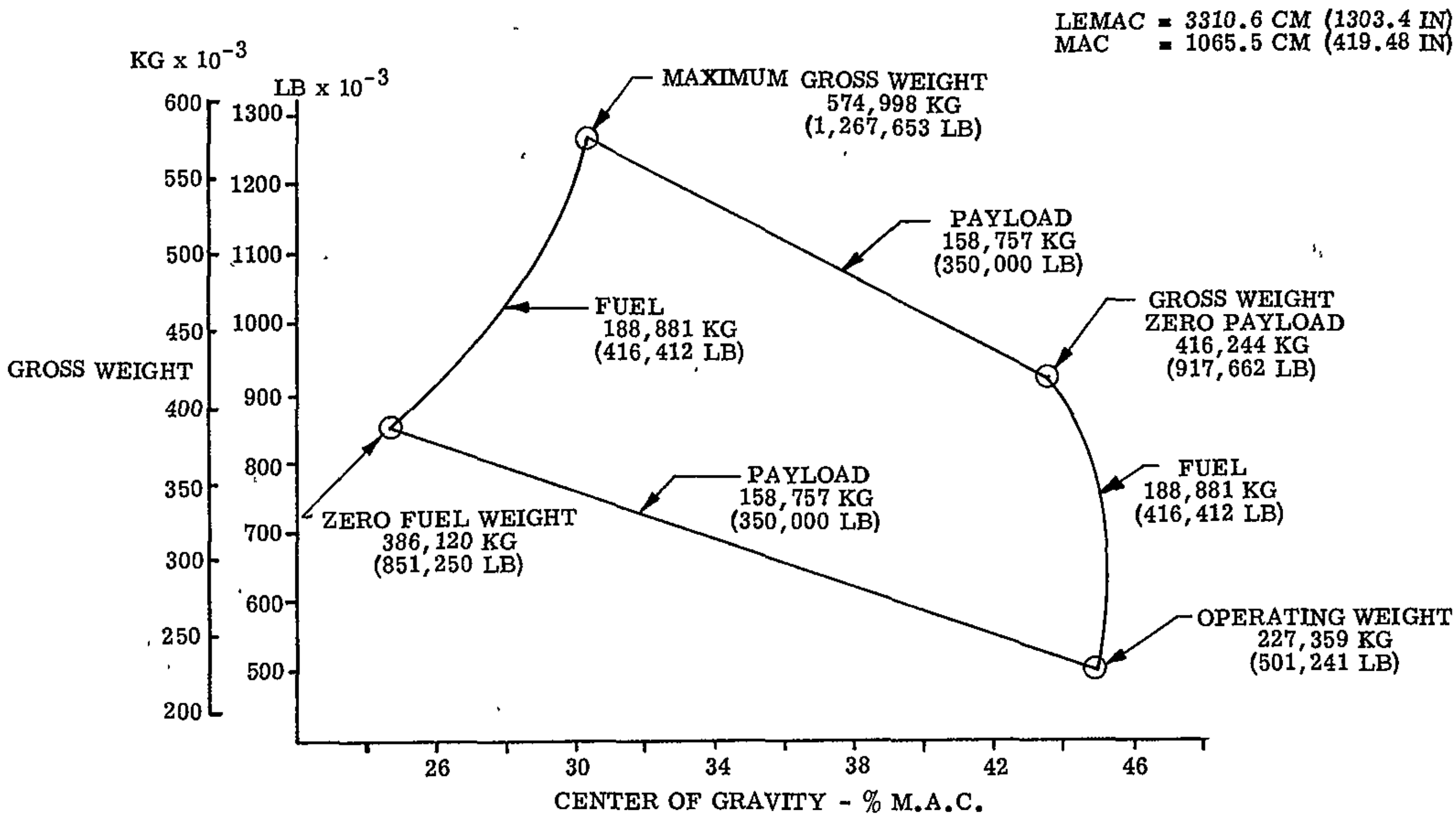

FIGURE 18 MILITARY CARGO TRANSPORT C.G. ENVELOPE

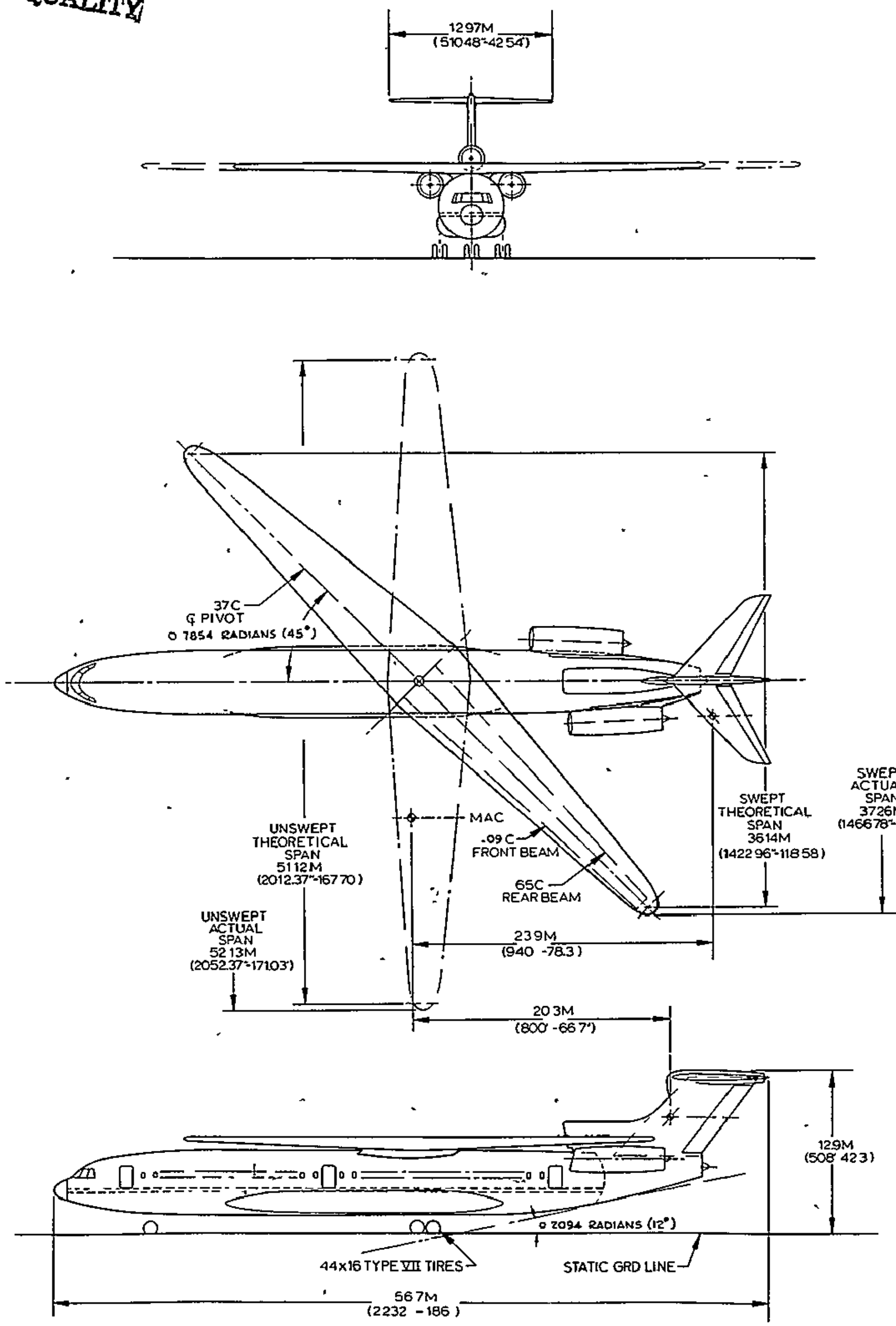

FIGURE 19 FINAL CONFIGURATION

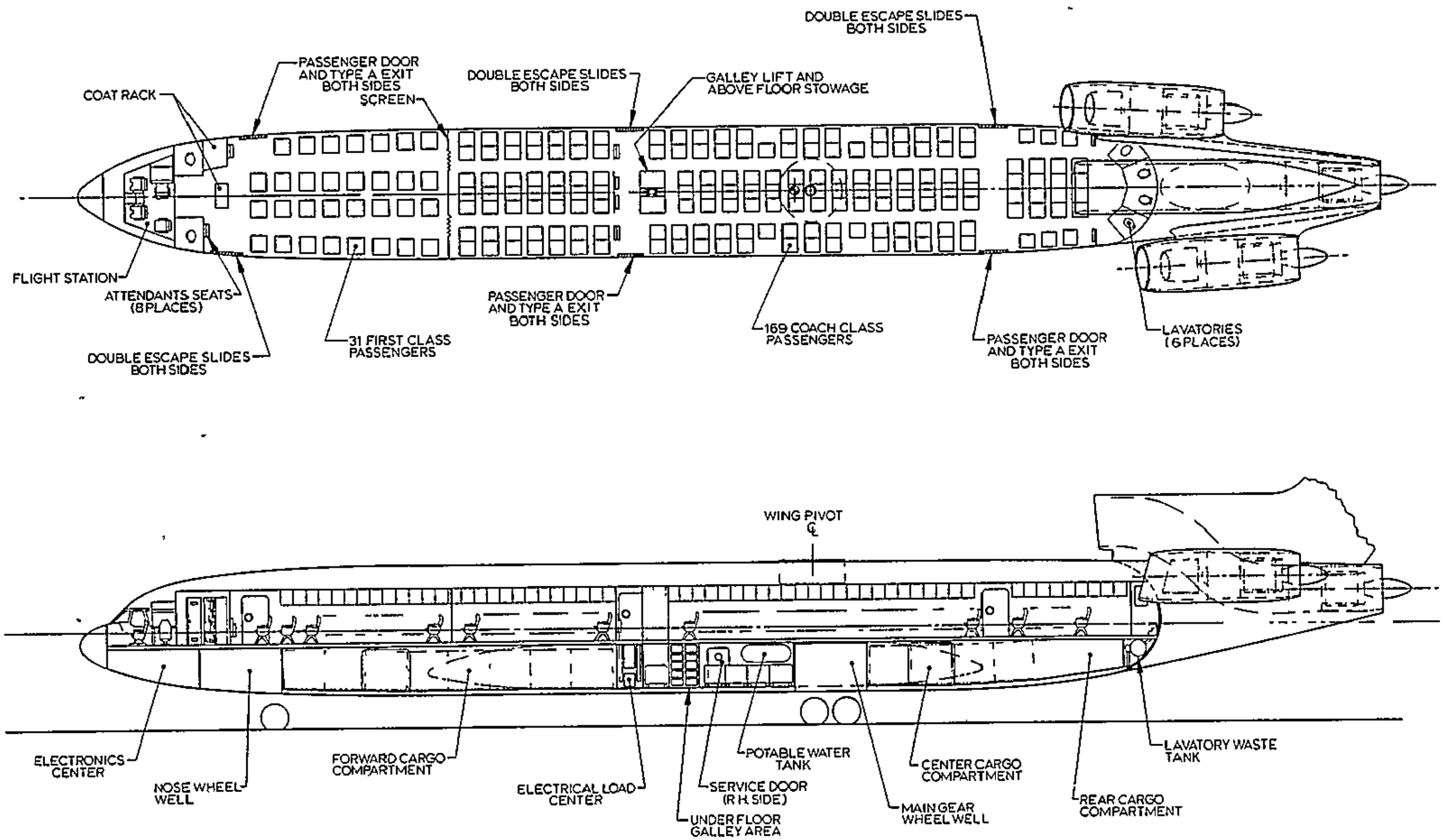

FIGURE 20 FINAL CONFIGURATION FUSELAGE INTERIOR ARRANGEMENT

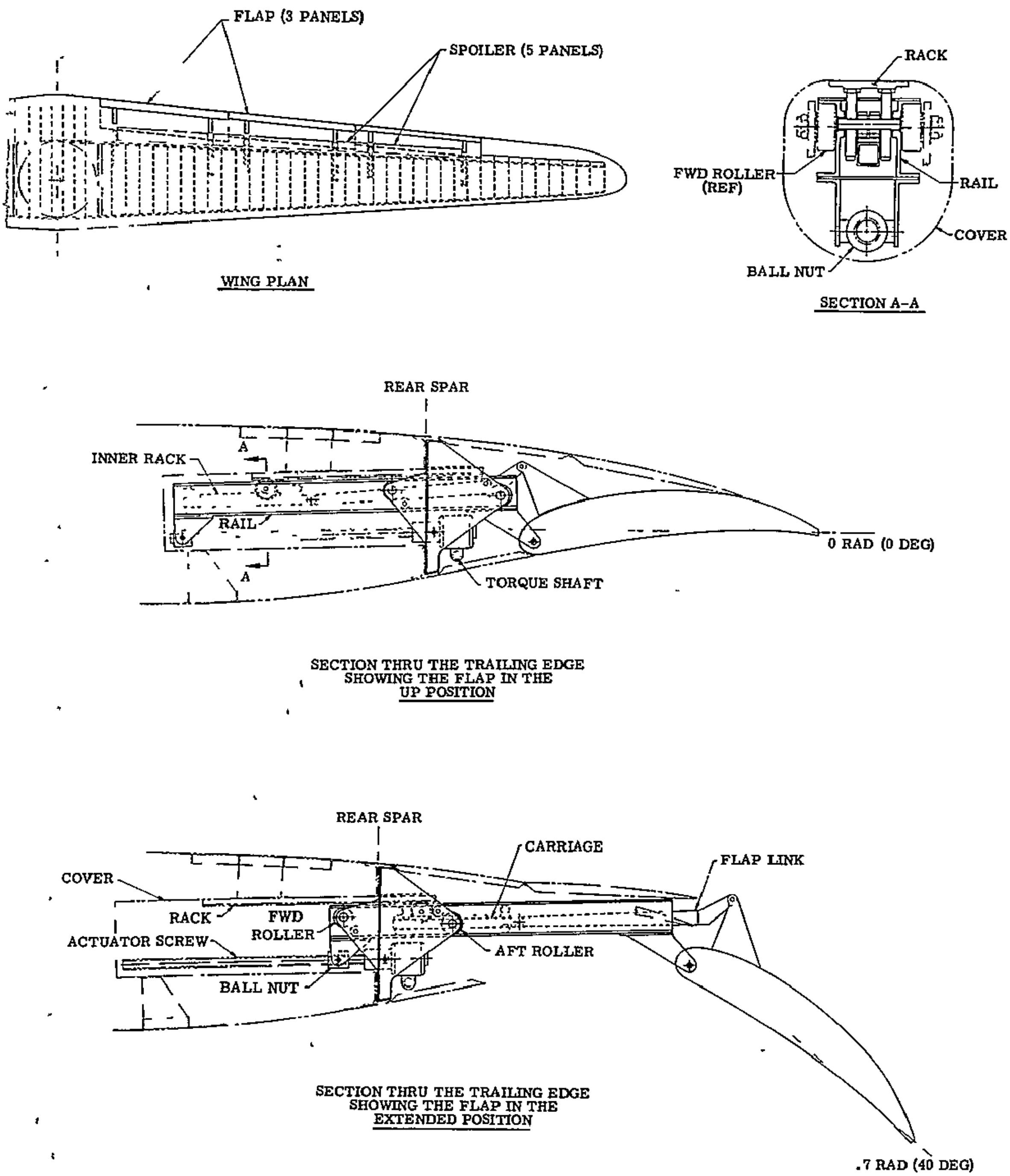

FIGURE 21 FINAL CONFIGURATION FLAP MECHANISM

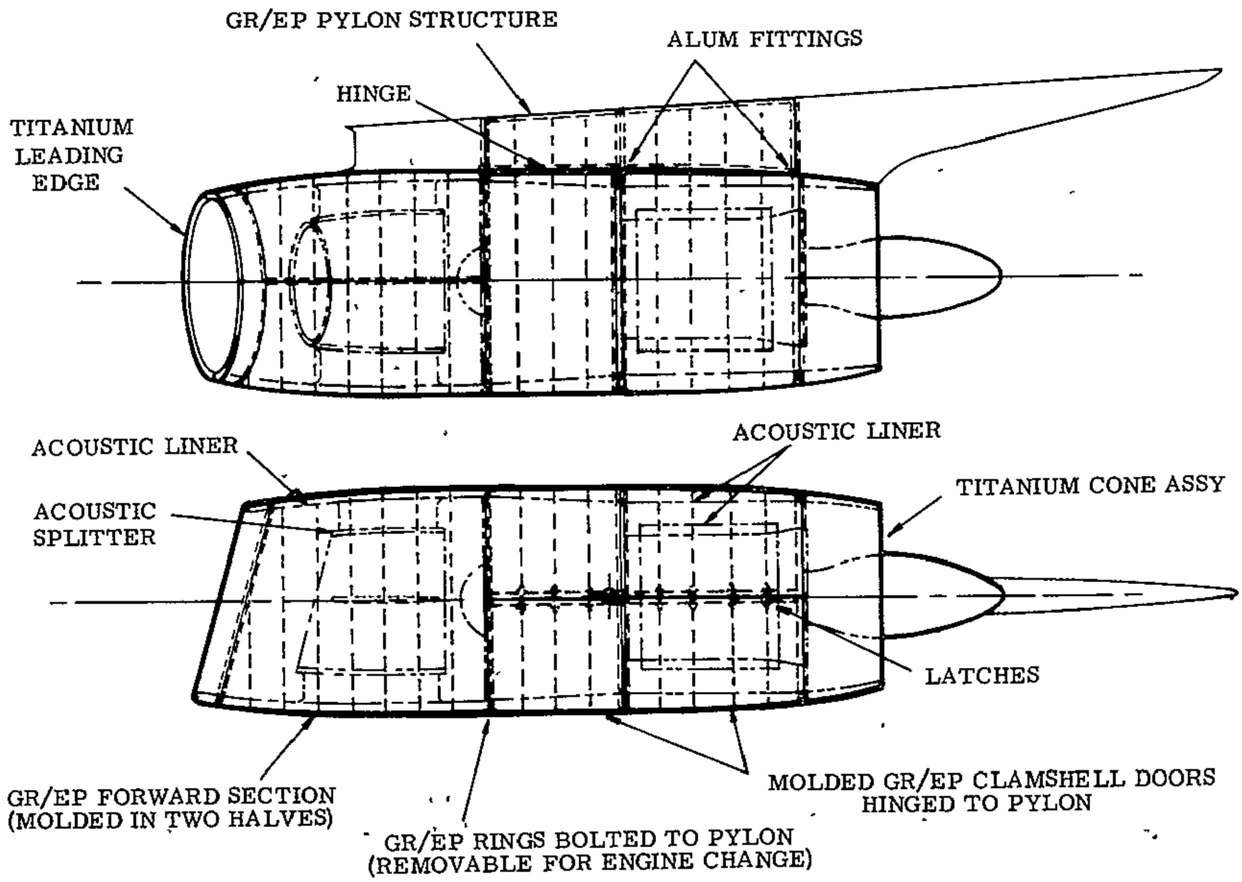

FIGURE 22 FINAL CONFIGURATION EXTERNAL NACELLE

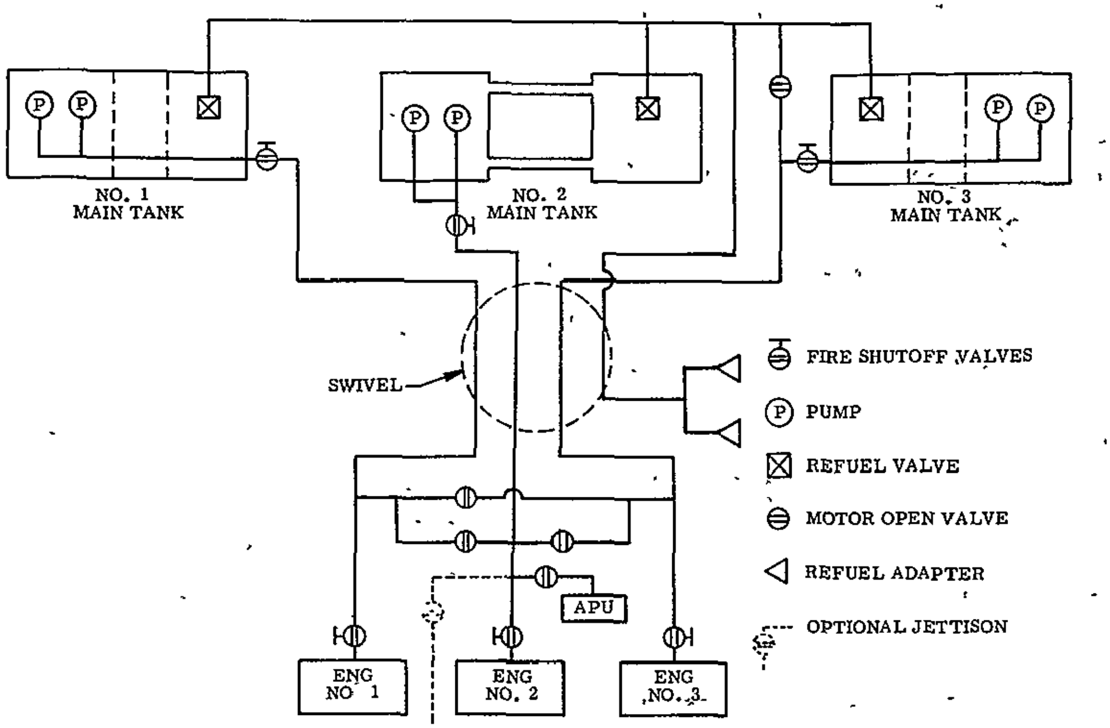

FIGURE 23 FINAL CONFIGURATION FUEL SYSTEM ARRANGEMENT

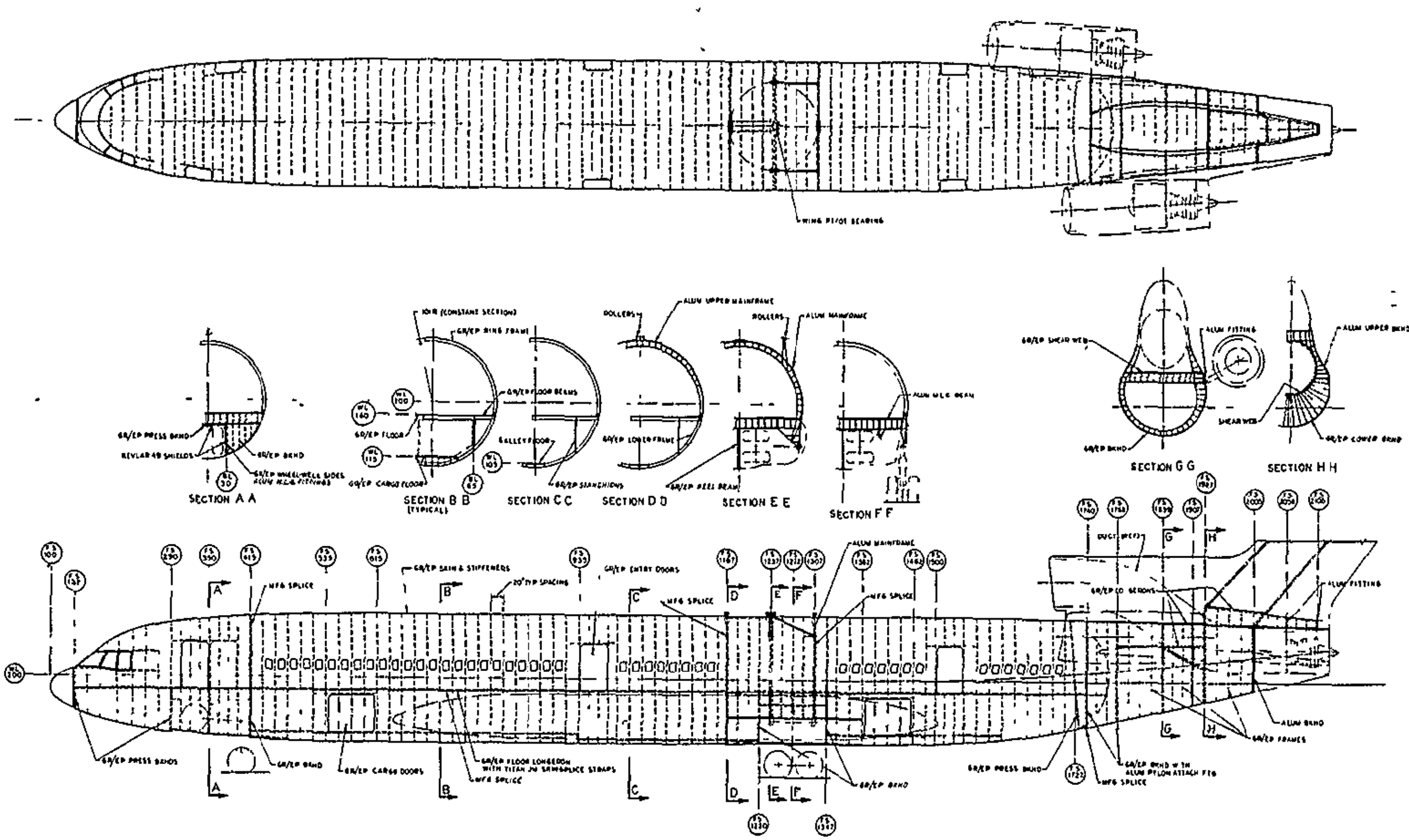

FIGURE 24 FINAL CONFIGURATION - FUSELAGE STRUCTURE

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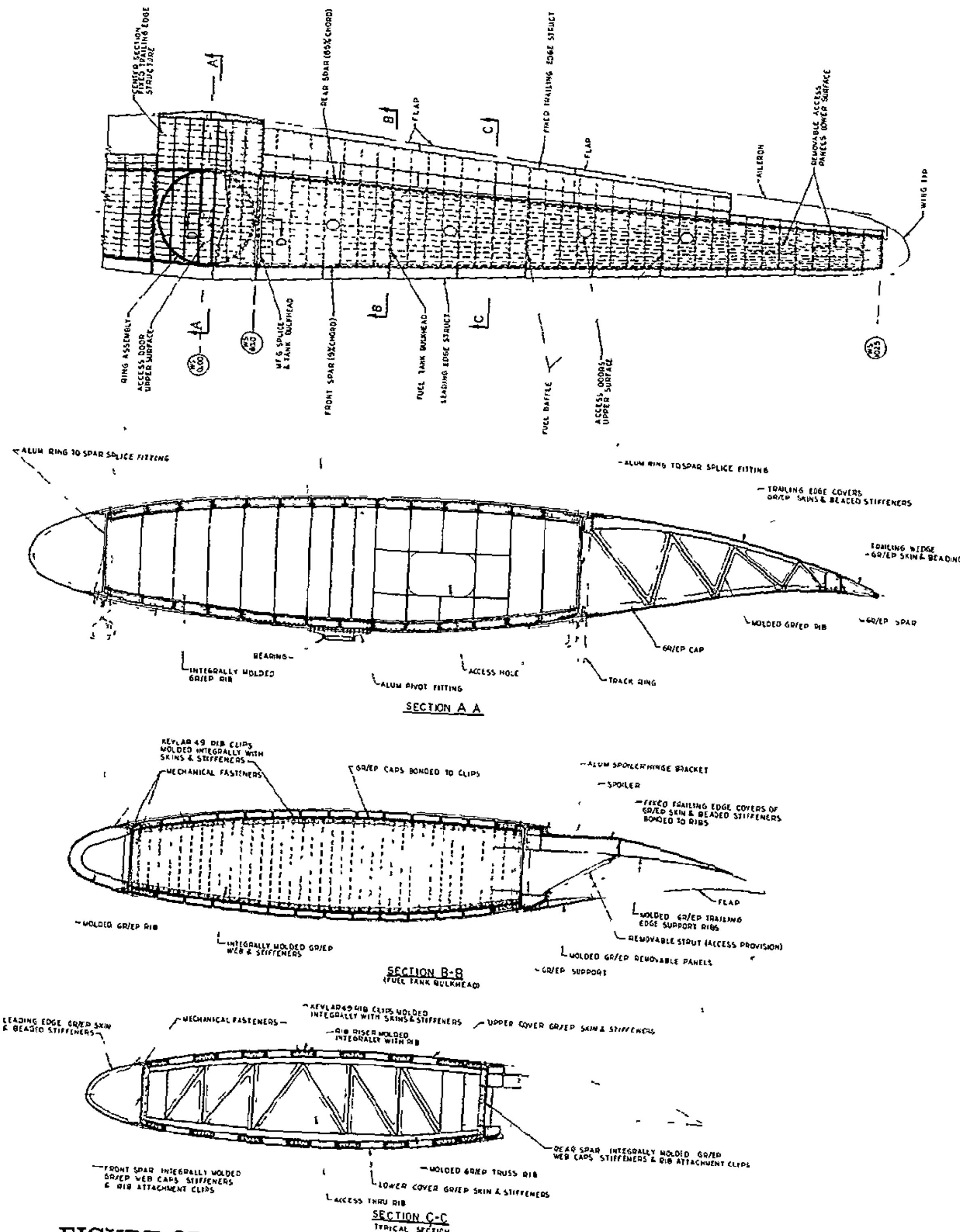


FIGURE 25 FINAL CONFIGURATION - WING STRUCTURE

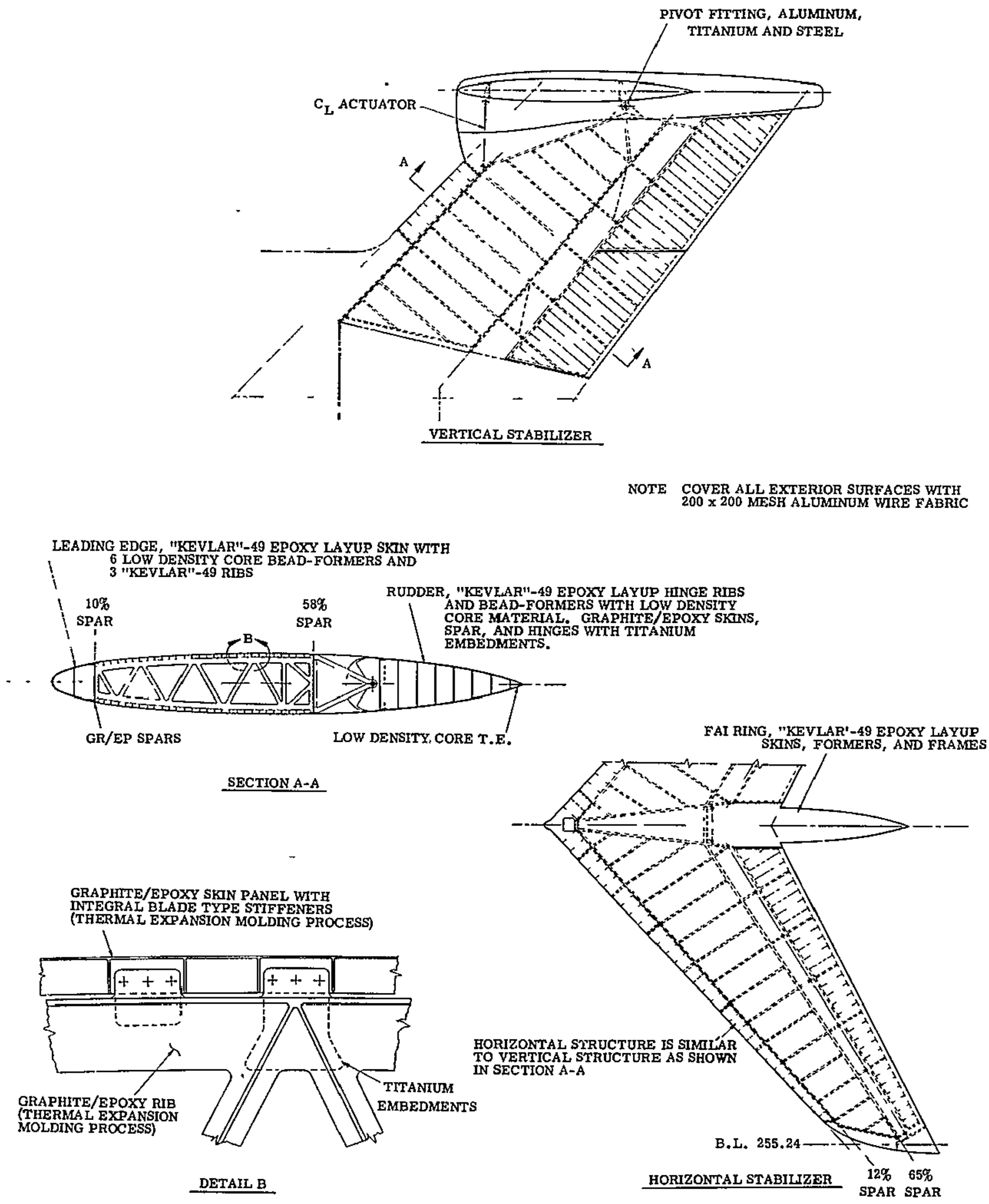


FIGURE 26 FINAL CONFIGURATION - EMPENNAGE STRUCTURE

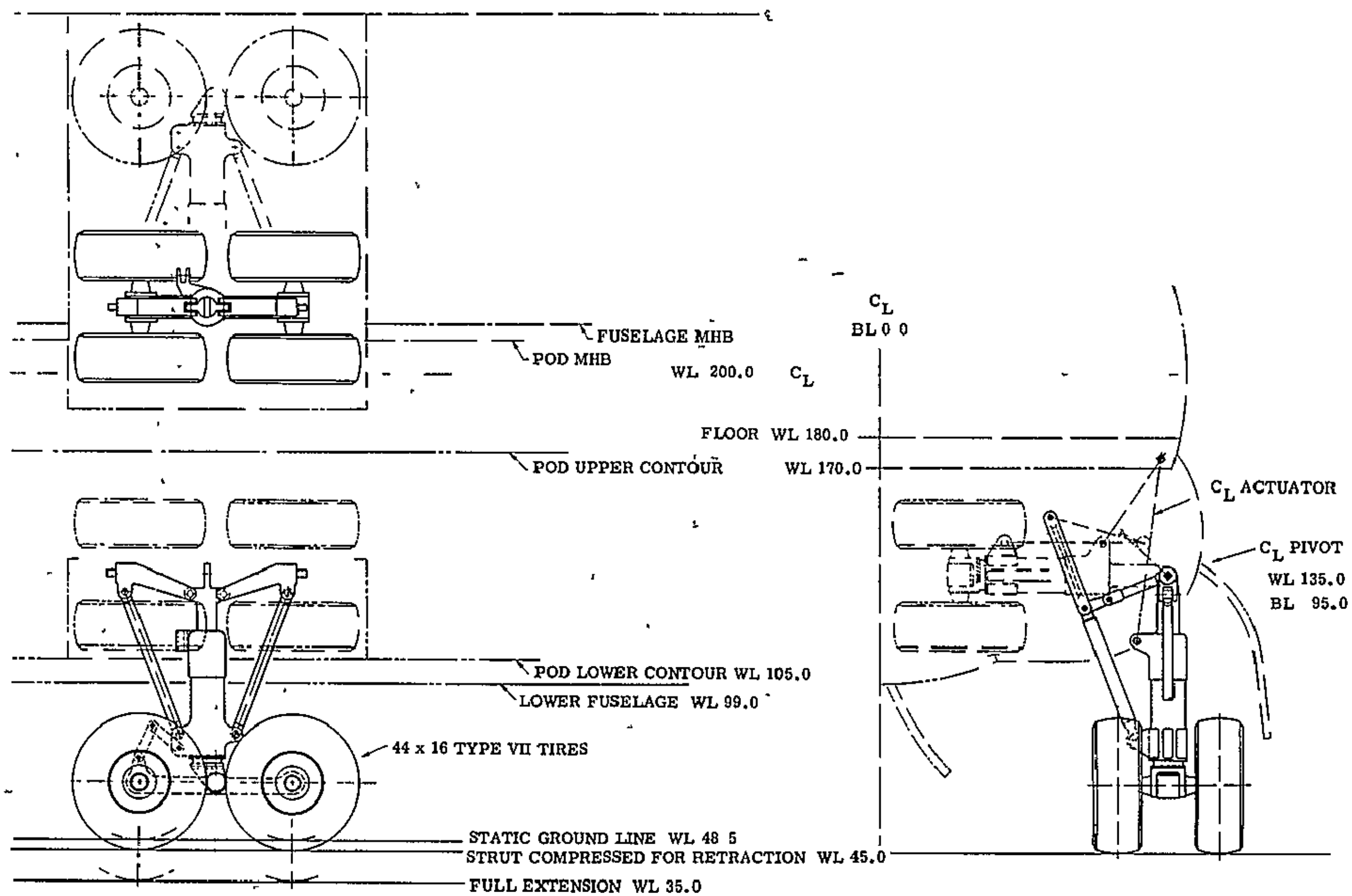


FIGURE 27 FINAL CONFIGURATION - MAIN LANDING GEAR ARRANGEMENT

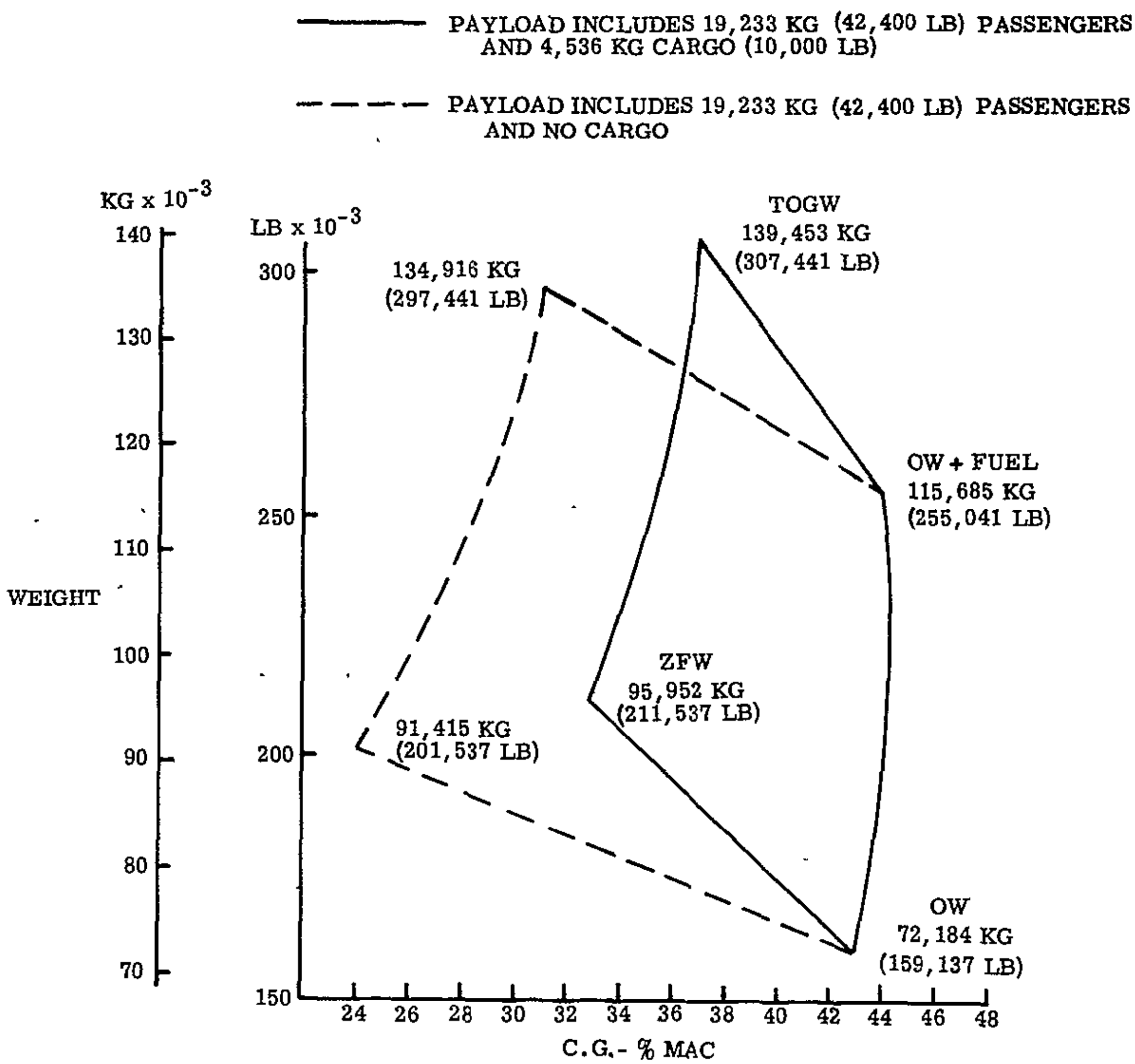


FIGURE 28 FINAL CONFIGURATION C.G. ENVELOPE

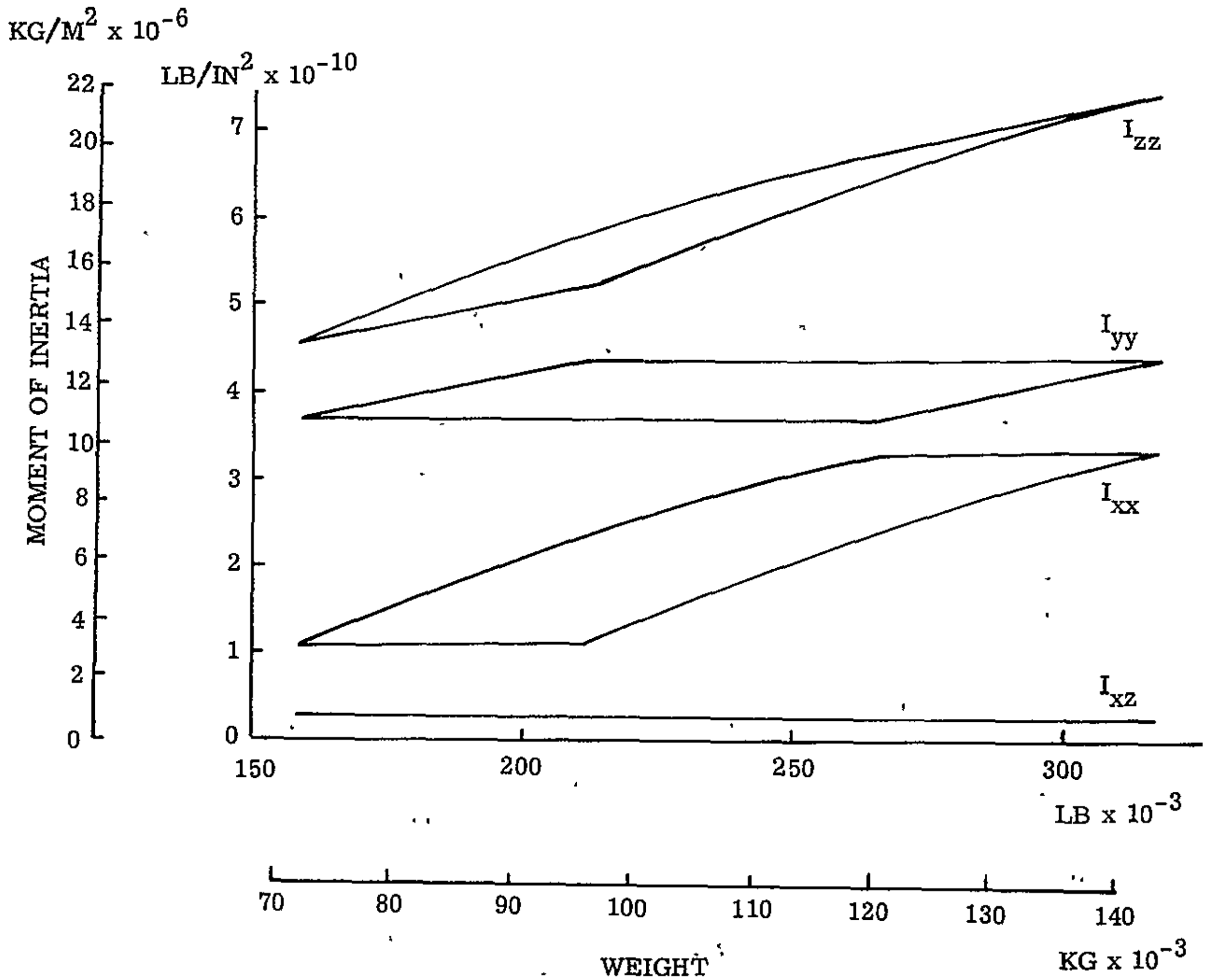


FIGURE 29 FINAL CONFIGURATION - INERTIA Λ = 0.0 RAD (0 DEG)

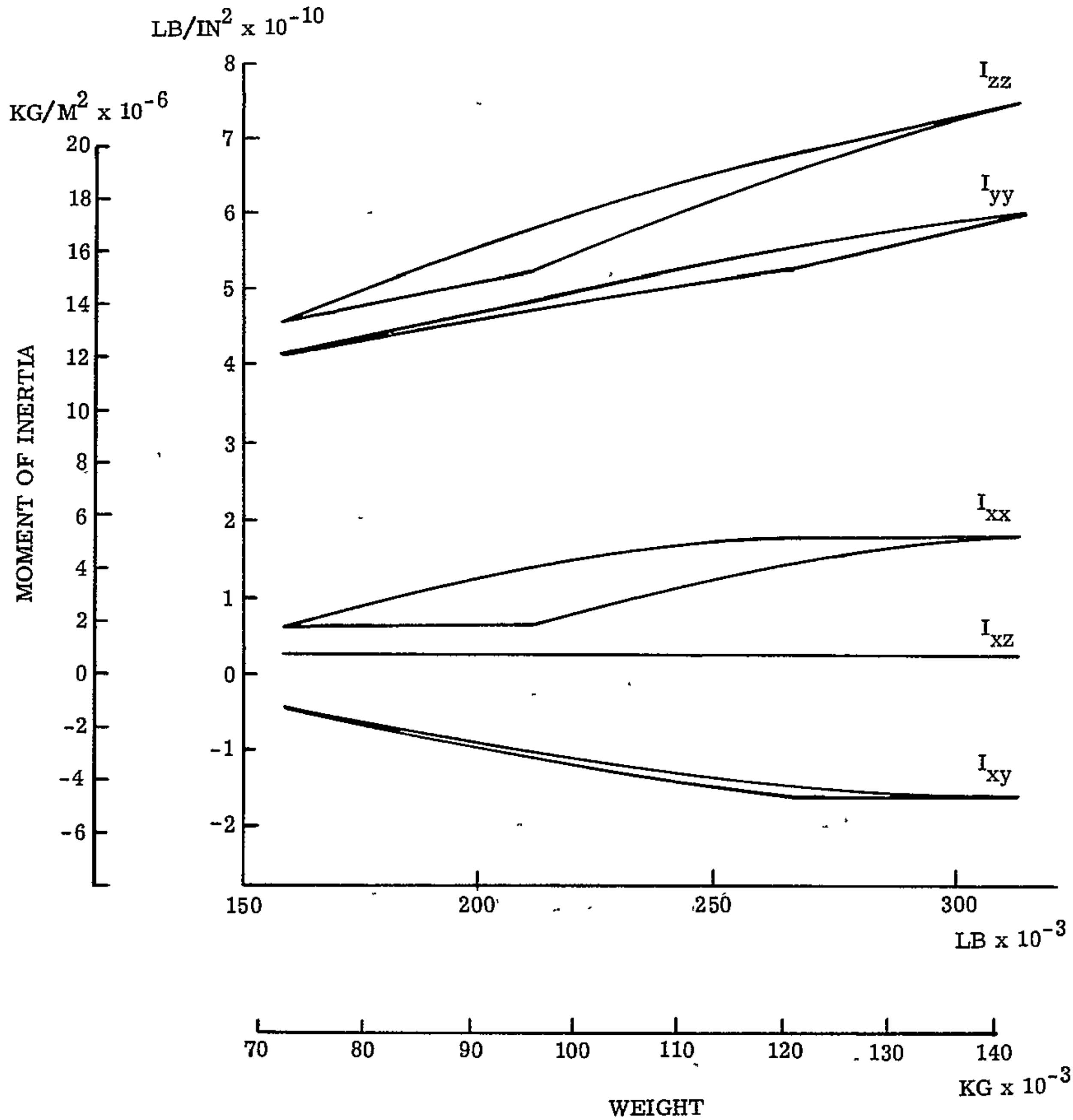
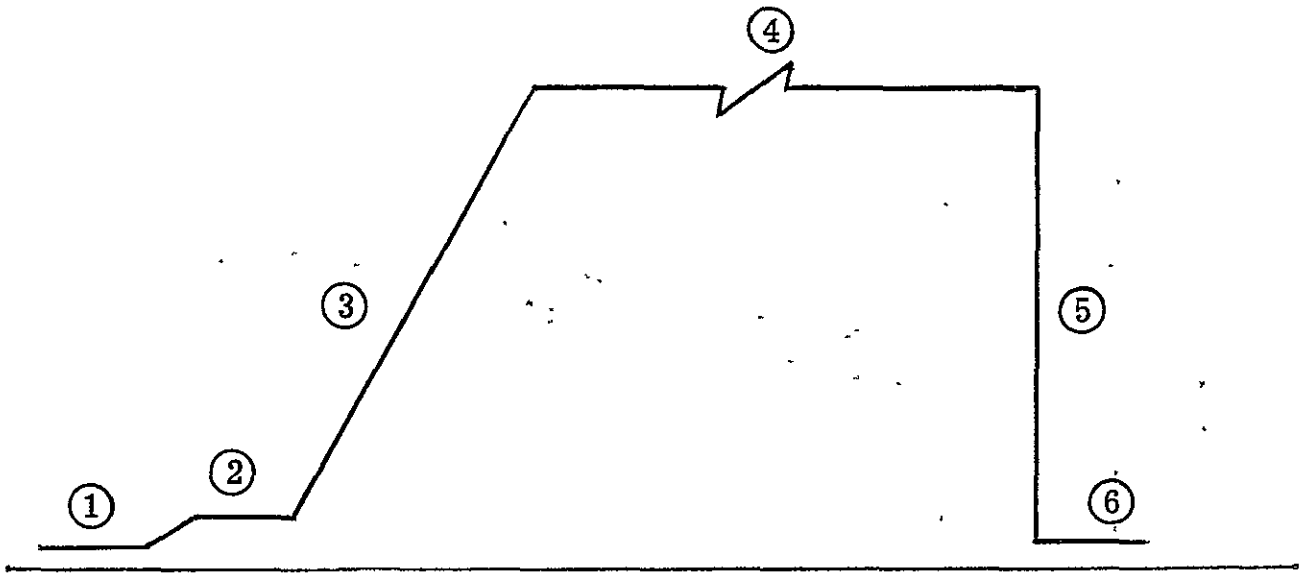


FIGURE 30 FINAL CONFIGURATION - INERTIA Λ = 0.785 RAD (45 DEG)



FINAL CONFIGURATION MISSION PROFILE

1. TAKEOFF AND CLIMB TO 1500 FEET
2. SWEEP WING AND ACCELERATE TO CLIMB SPEED
3. CLIMB TO CRUISE ALTITUDE
4. CRUISE SEGMENT
5. DESCENT
6. LAND $\leq$ 9000 FEET

FIGURE 31 FINAL CONFIGURATION - MISSION PROFILE

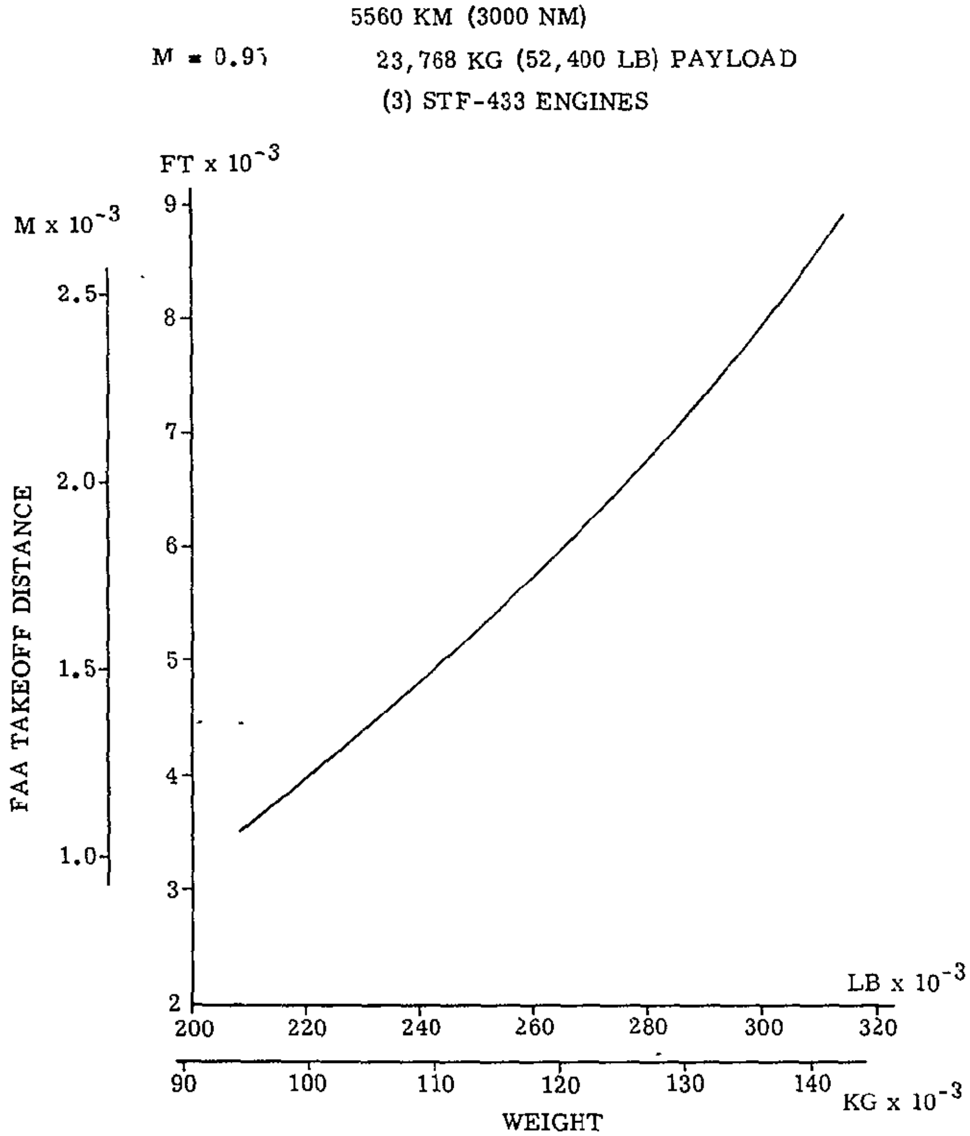


FIGURE 32 FINAL CONFIGURATION - TAKEOFF PERFORMANCE

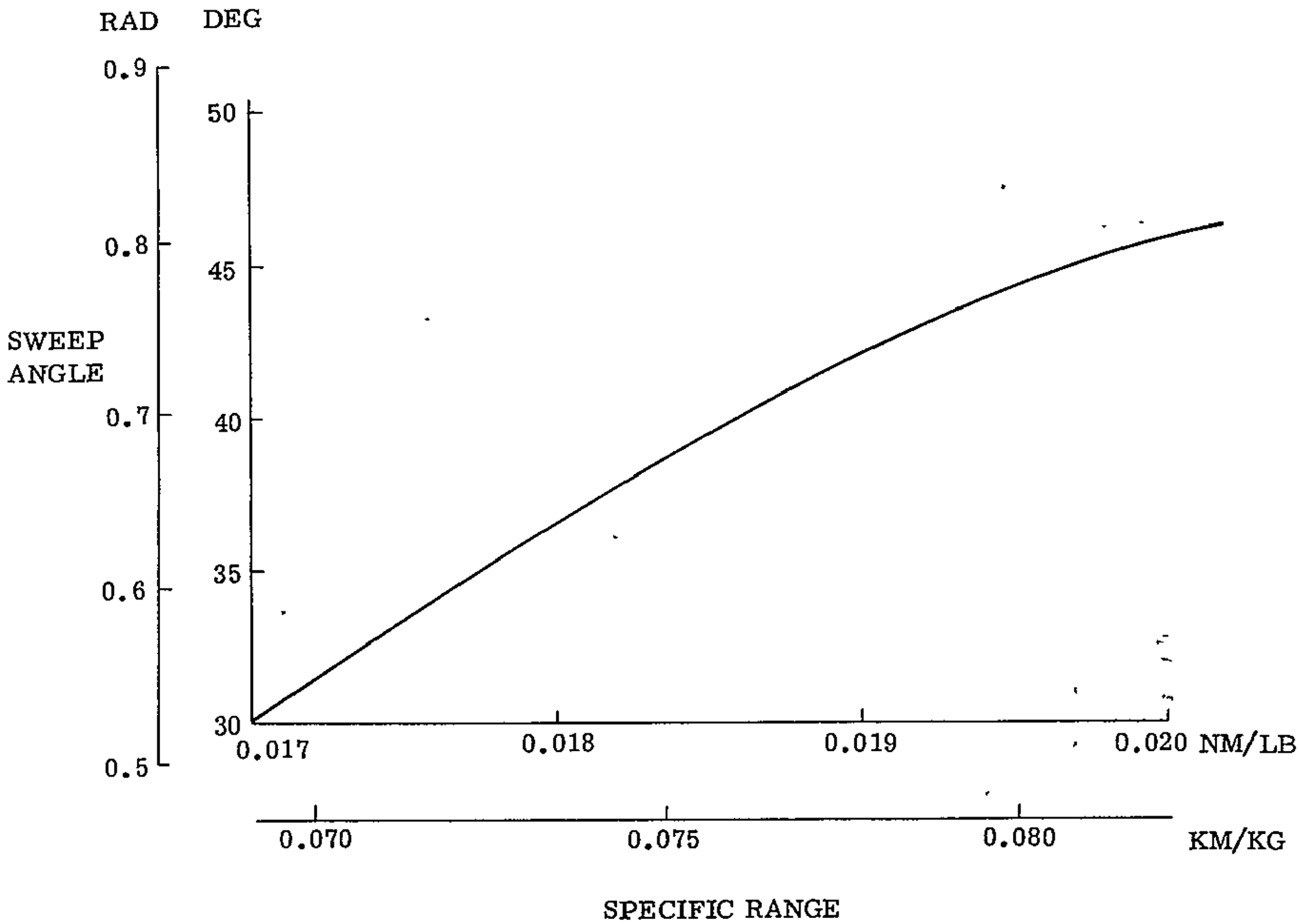


FIGURE 33 FINAL CONFIGURATION - SPECIFIC RANGE DURING CLIMB

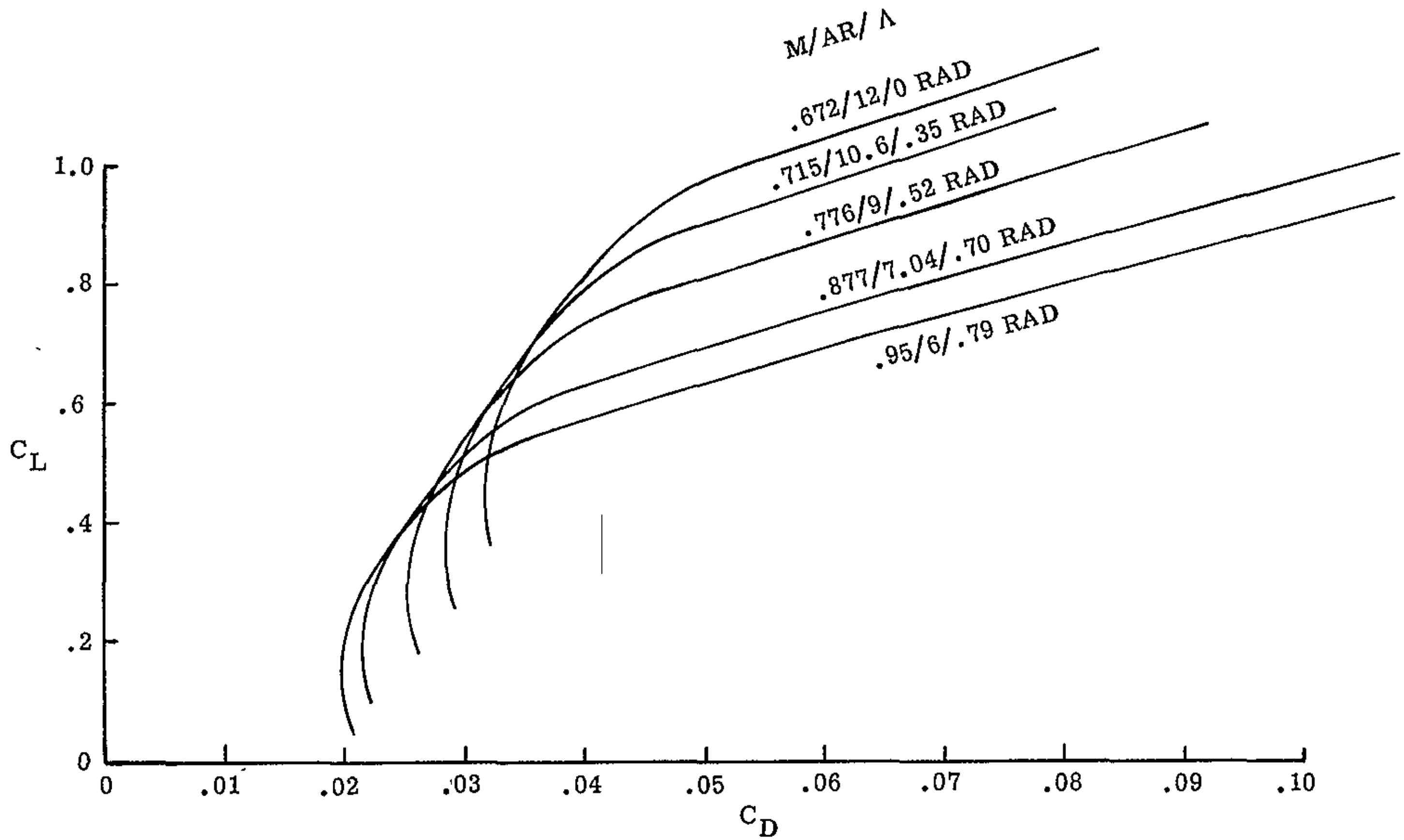


FIGURE 34 FINAL CONFIGURATION - DRAG POLARS

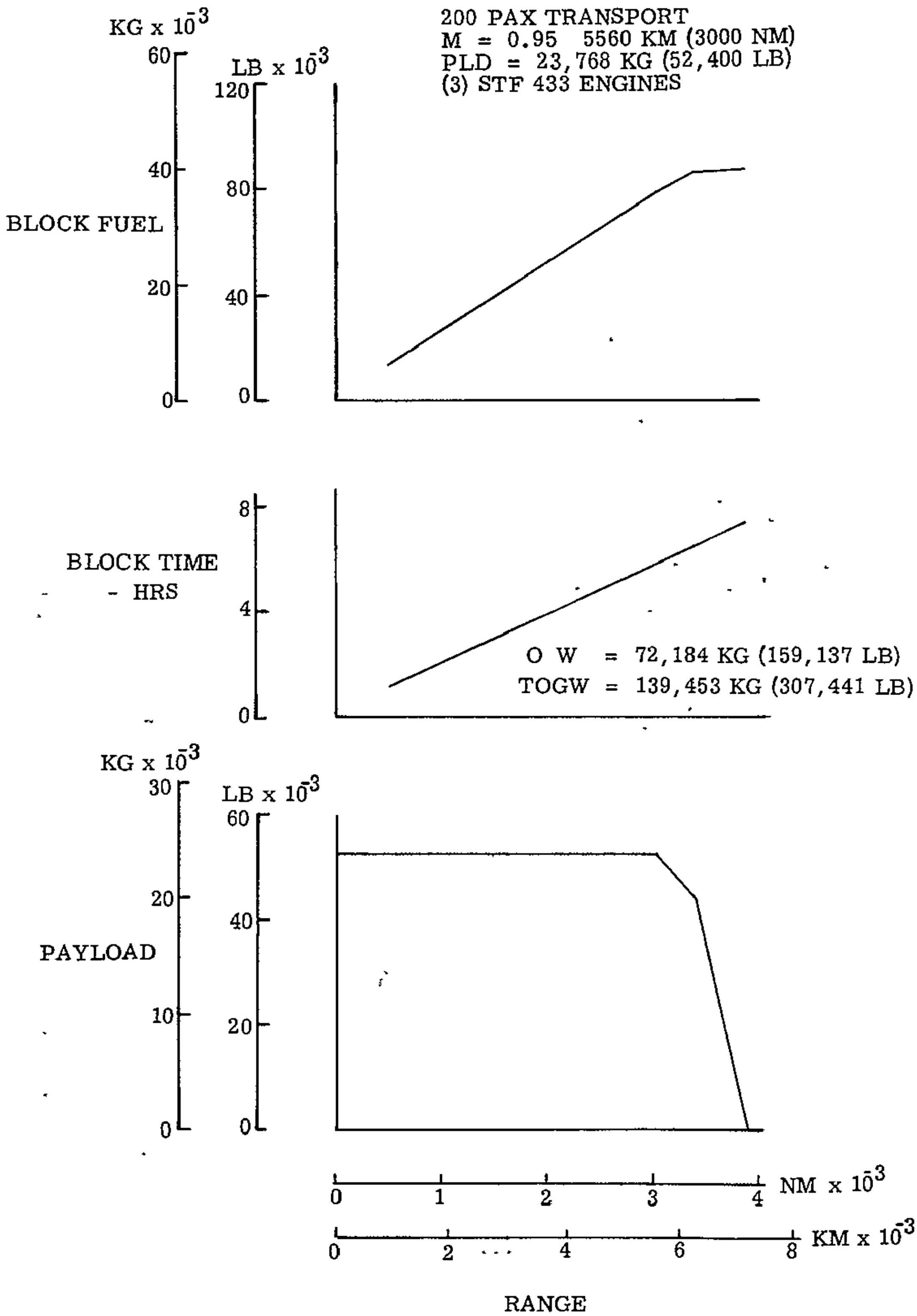


FIGURE 35 FINAL CONFIGURATION - PAYLOAD-RANGE

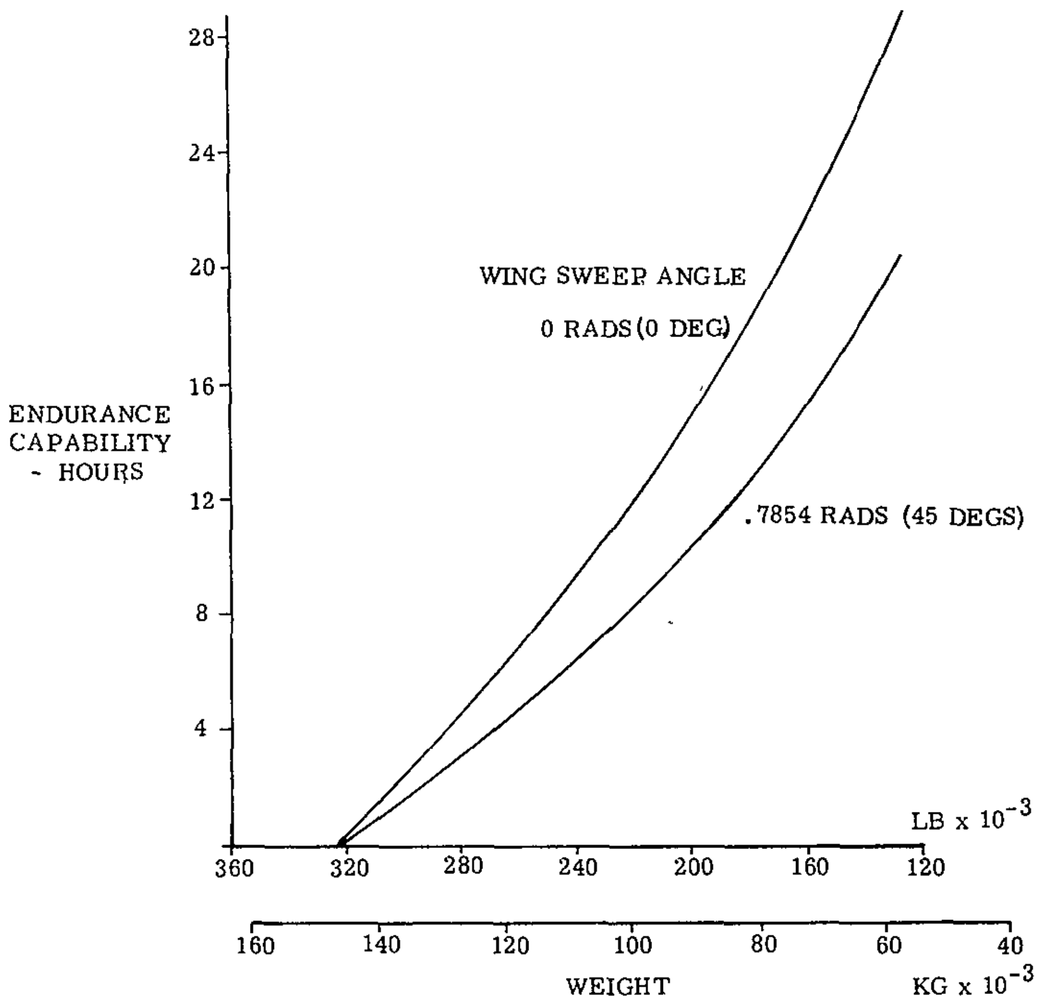


FIGURE 36 FINAL CONFIGURATION - ENDURANCE CAPABILITY

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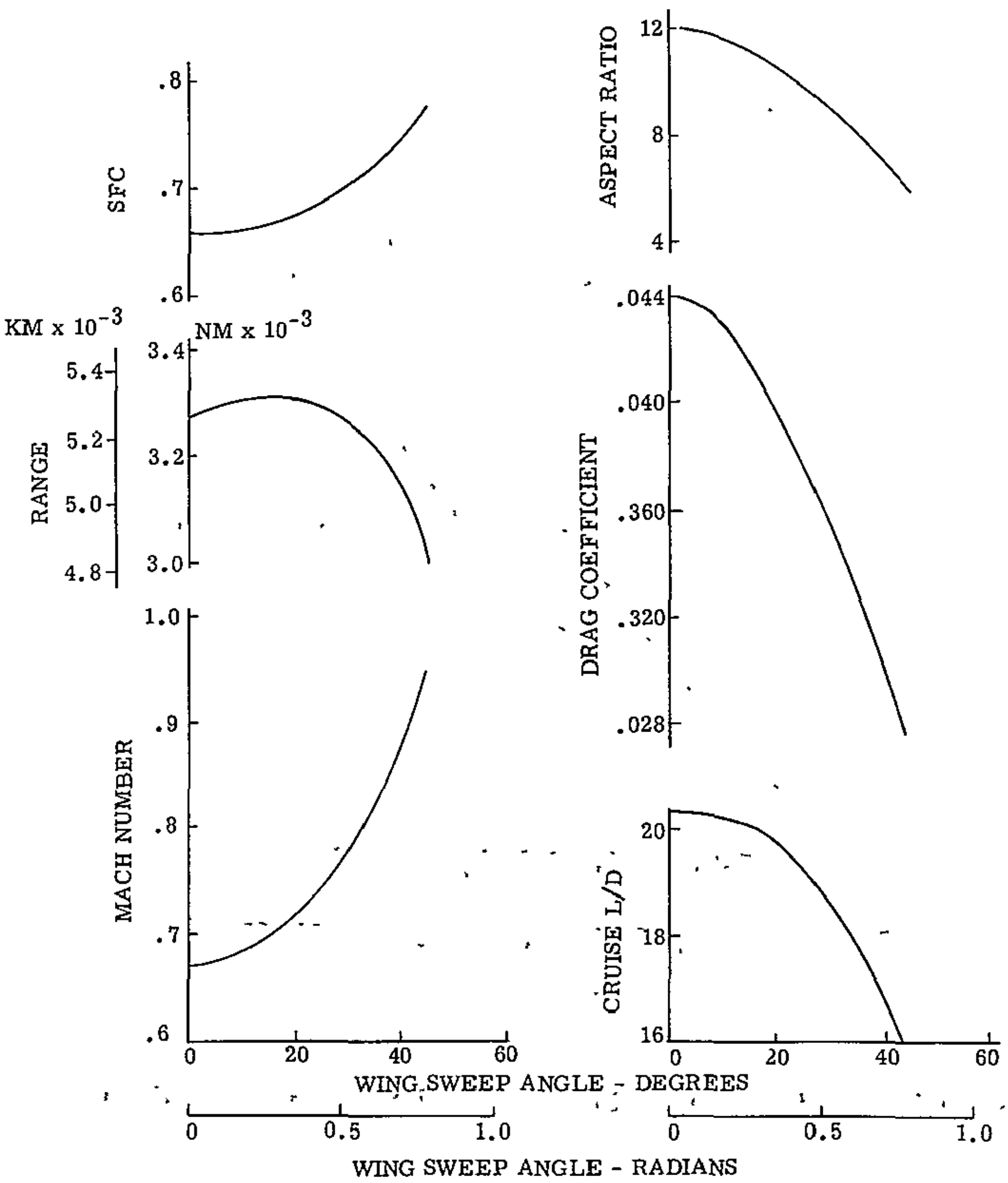

FIGURE 37 FINAL CONFIGURATION - OFF-DESIGN PERFORMANCE

M = 0.95 5560 KM (3000 NM) 23,768 KG (52,400 LB) PAYLOAD

X_W = INCREMENT OF WEIGHT OF ANY CONSTITUENT WEIGHT OF THE OPERATING WEIGHT

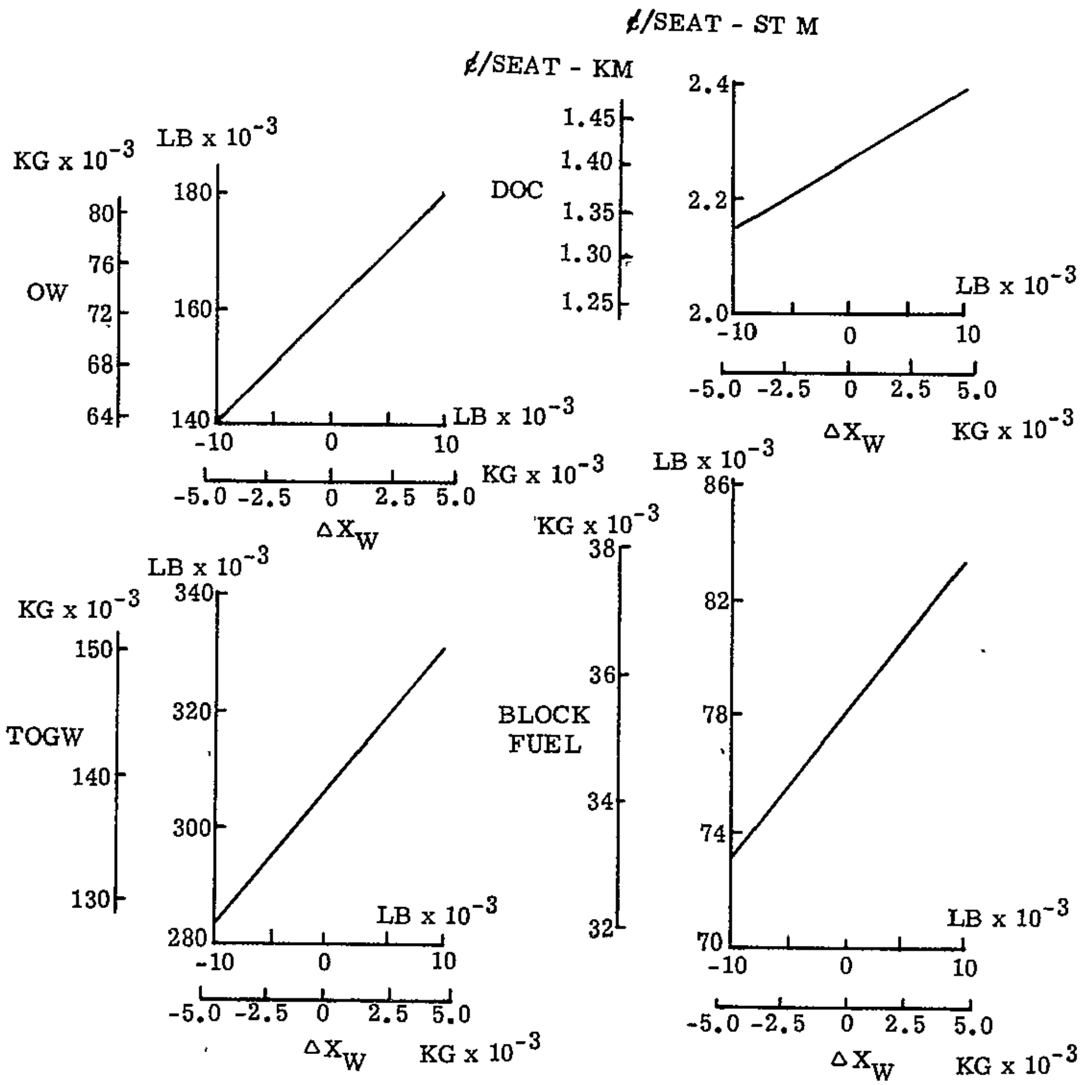

FIGURE 38 FINAL CONFIGURATION - WEIGHT SENSITIVITY

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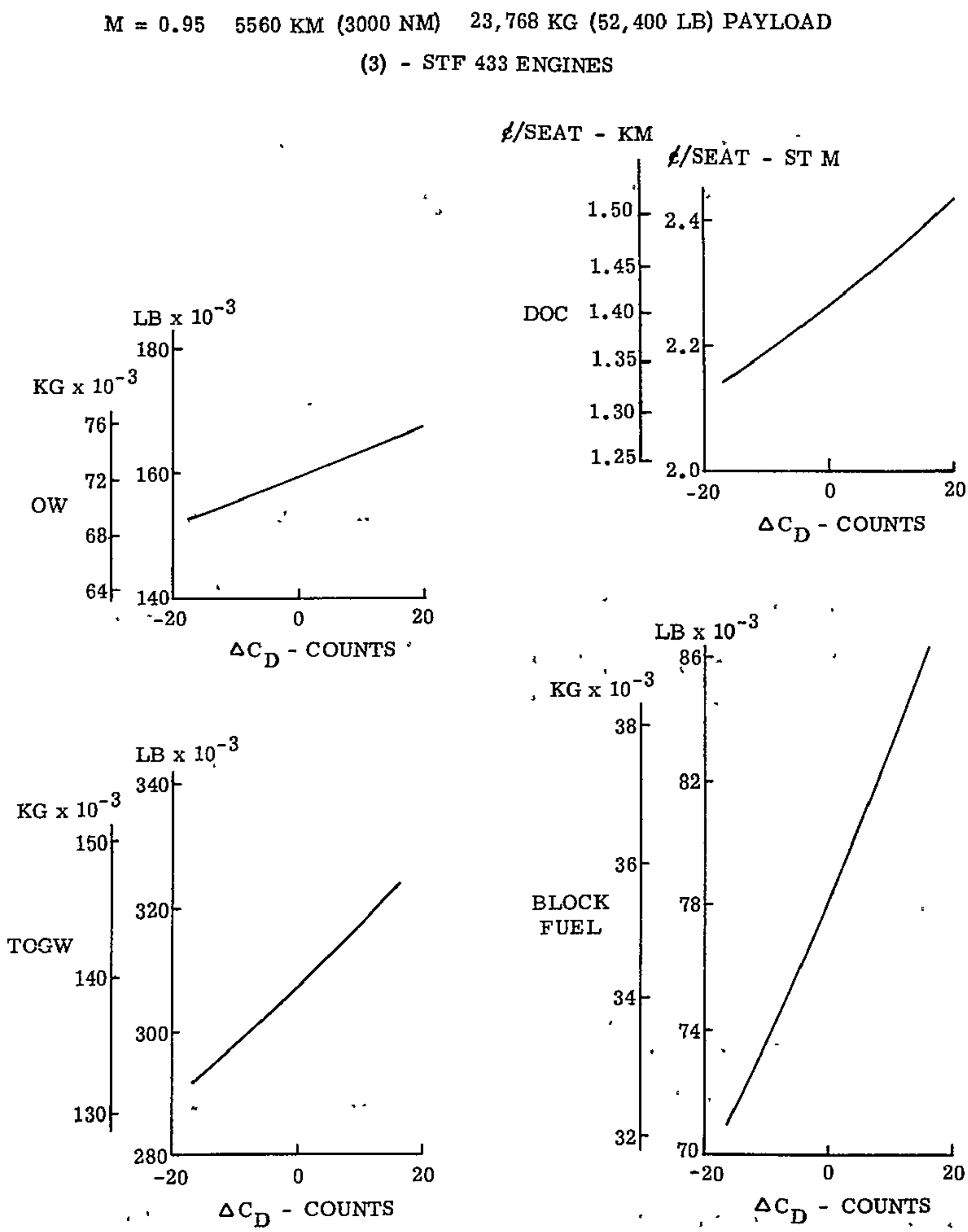

FIGURE 39 FINAL CONFIGURATION - DRAG SENSITIVITY

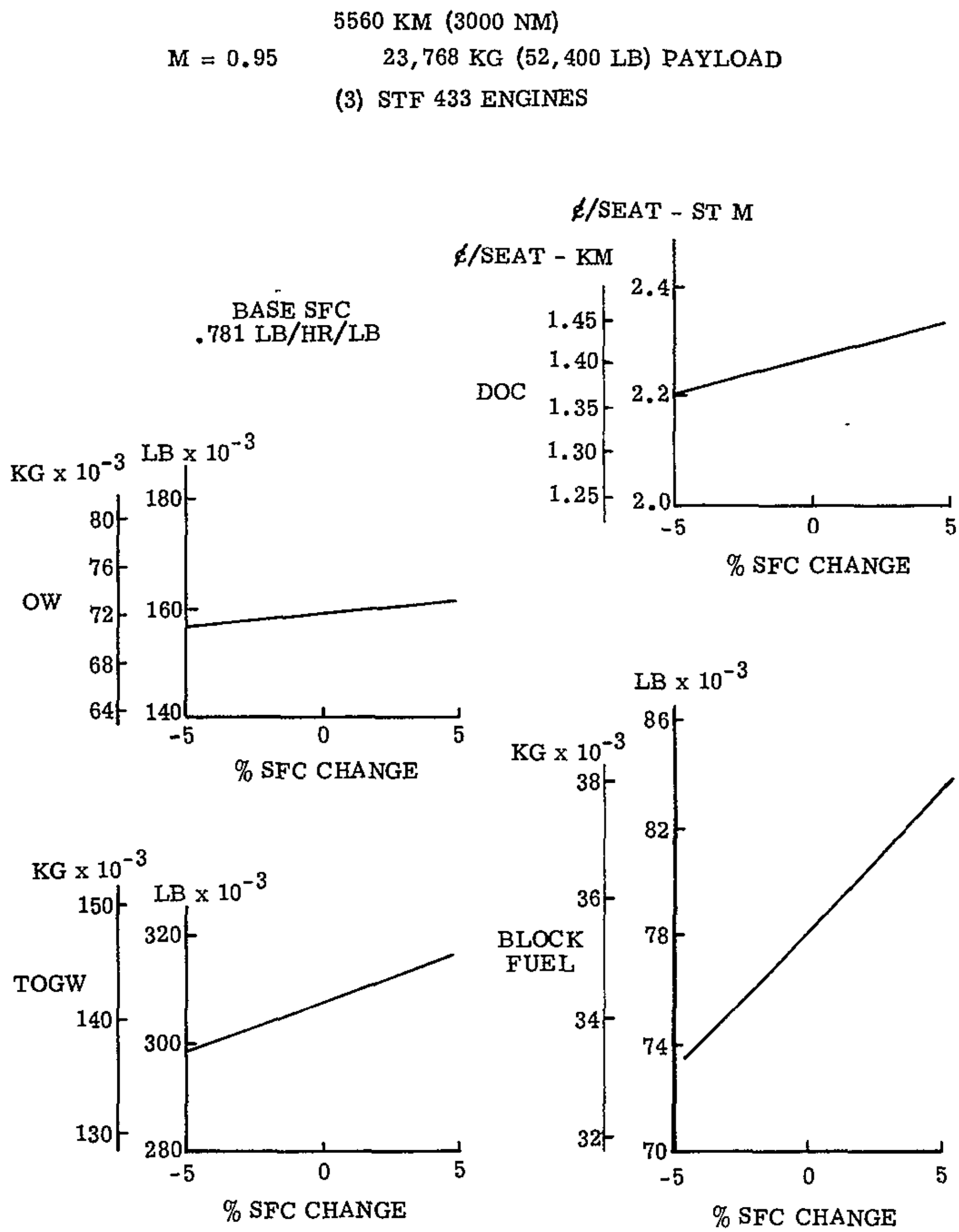

FIGURE 40 FINAL CONFIGURATION - SFC SENSITIVITY

M = 0.95 5560 KM (3000 NM) 23,768 KG (52,400 LB) PAYLOAD

(3) STF 433 ENGINES

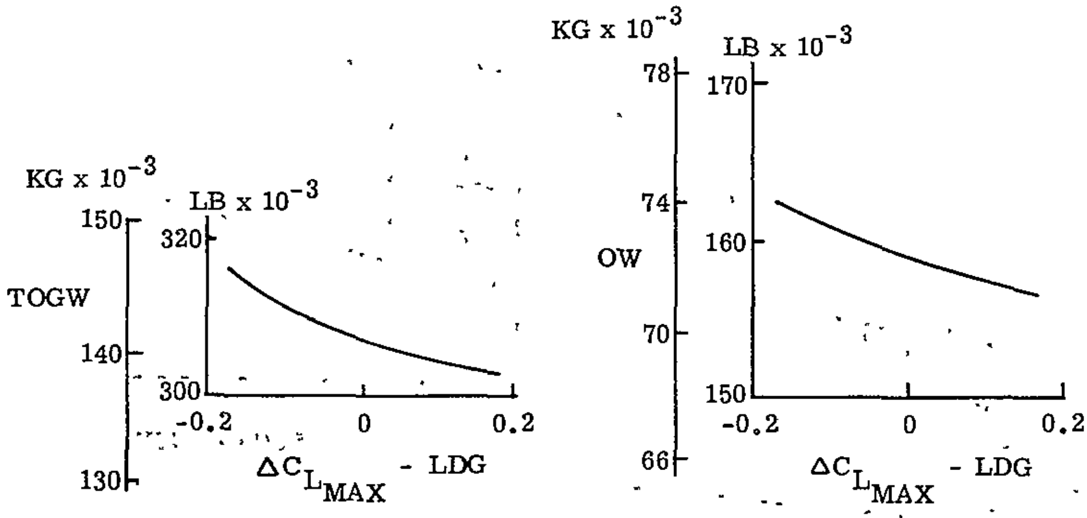

FIGURE 41 FINAL CONFIGURATION -
MAXIMUM LIFT COEFFICIENT SENSITIVITY

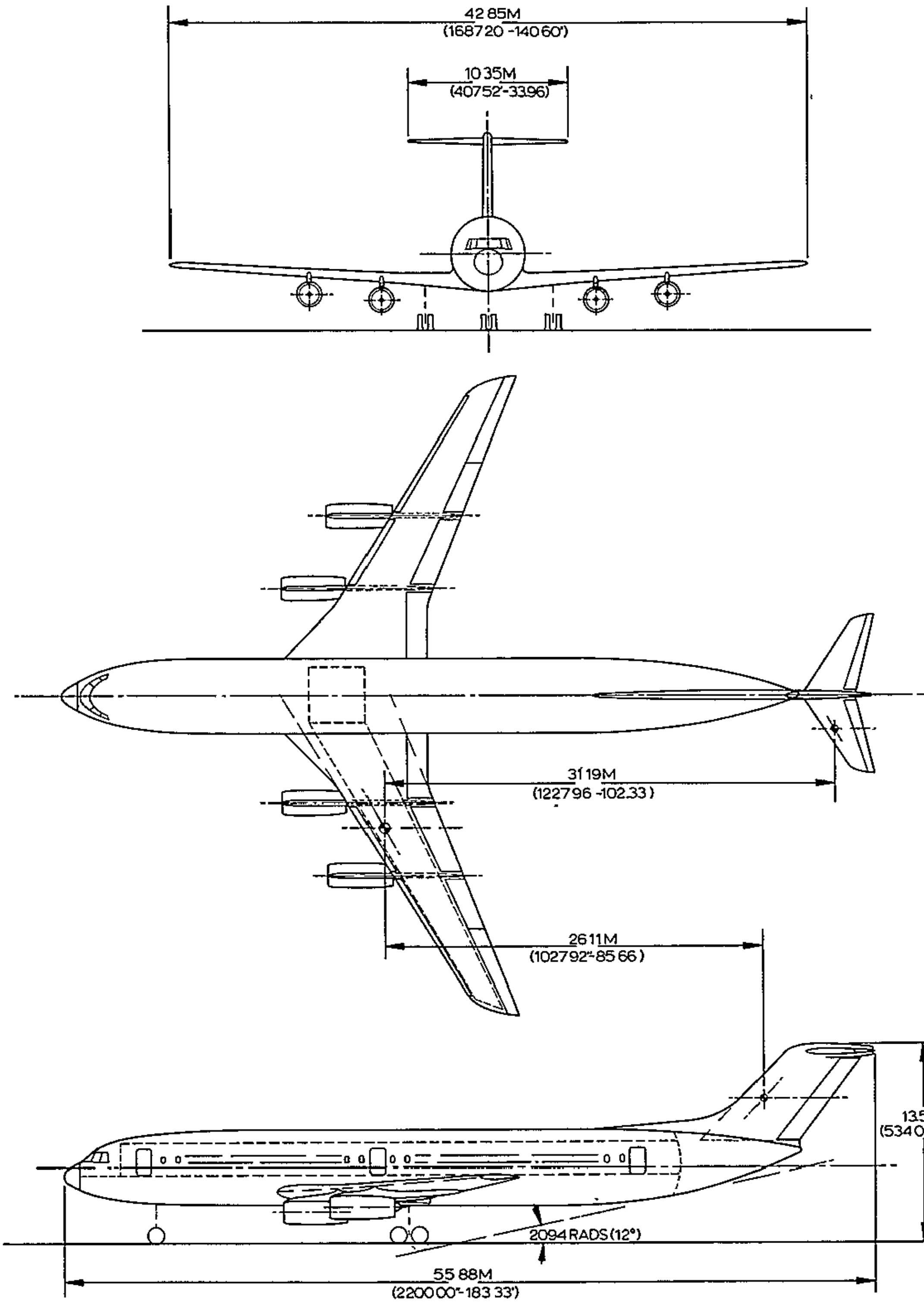

FIGURE 42 CONVENTIONAL CONFIGURATION - MACH 0.85 CRUISE

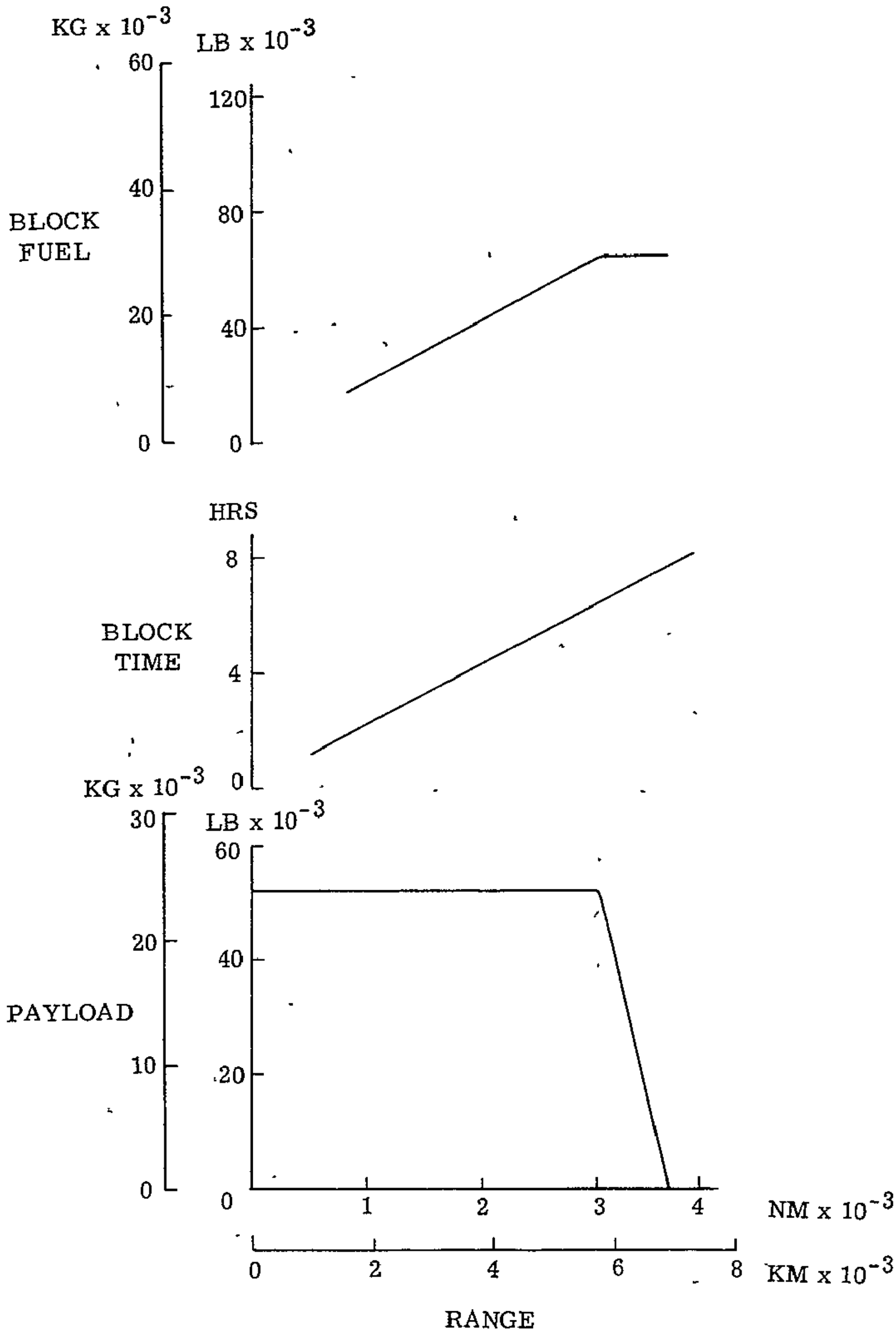

FIGURE 43 CONVENTIONAL CONFIGURATION - MACH 0.85 CRUISE - PAYLOAD-RANGE

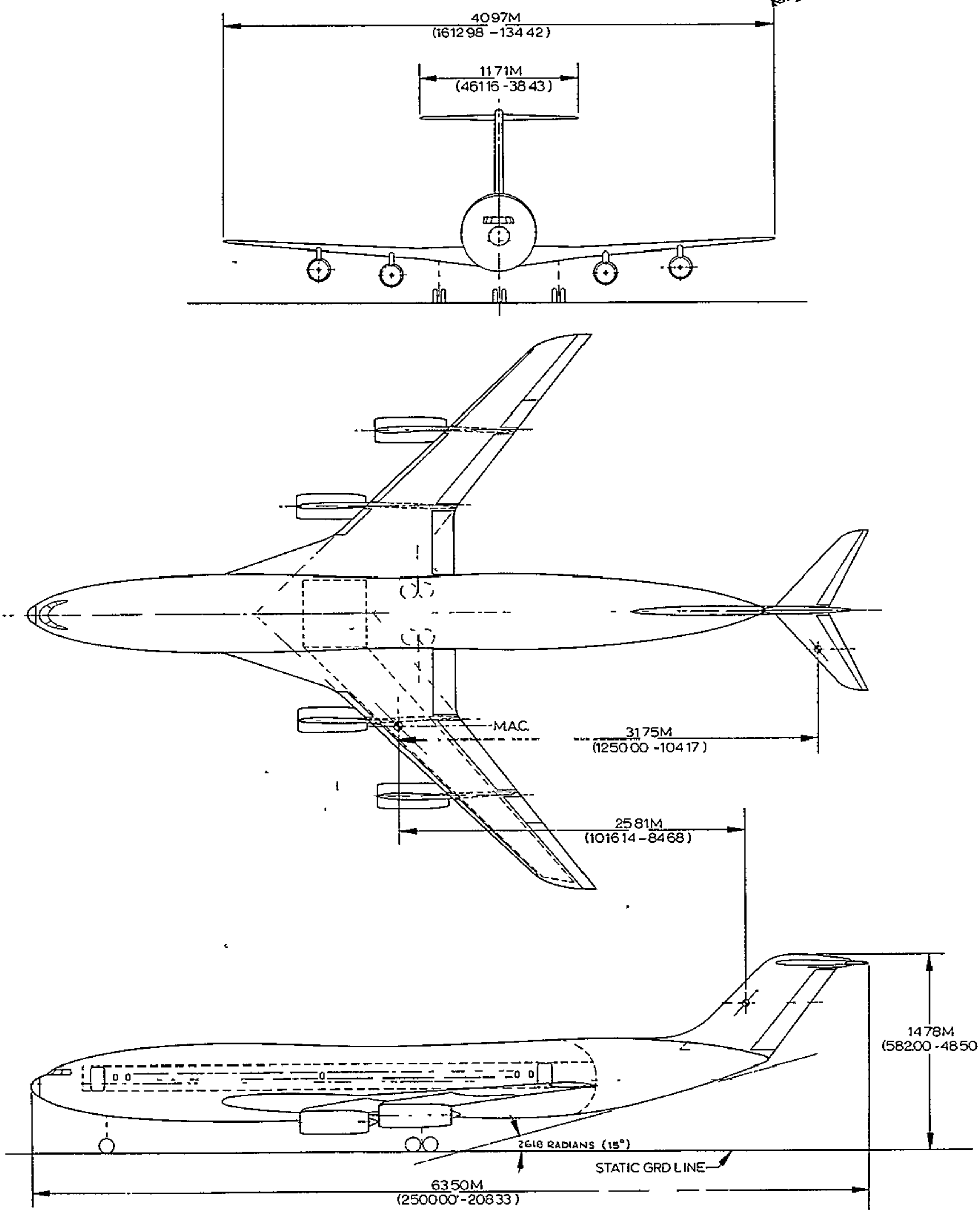

FIGURE 44 CONVENTIONAL CONFIGURATION - CRUISE AT M = 0.95

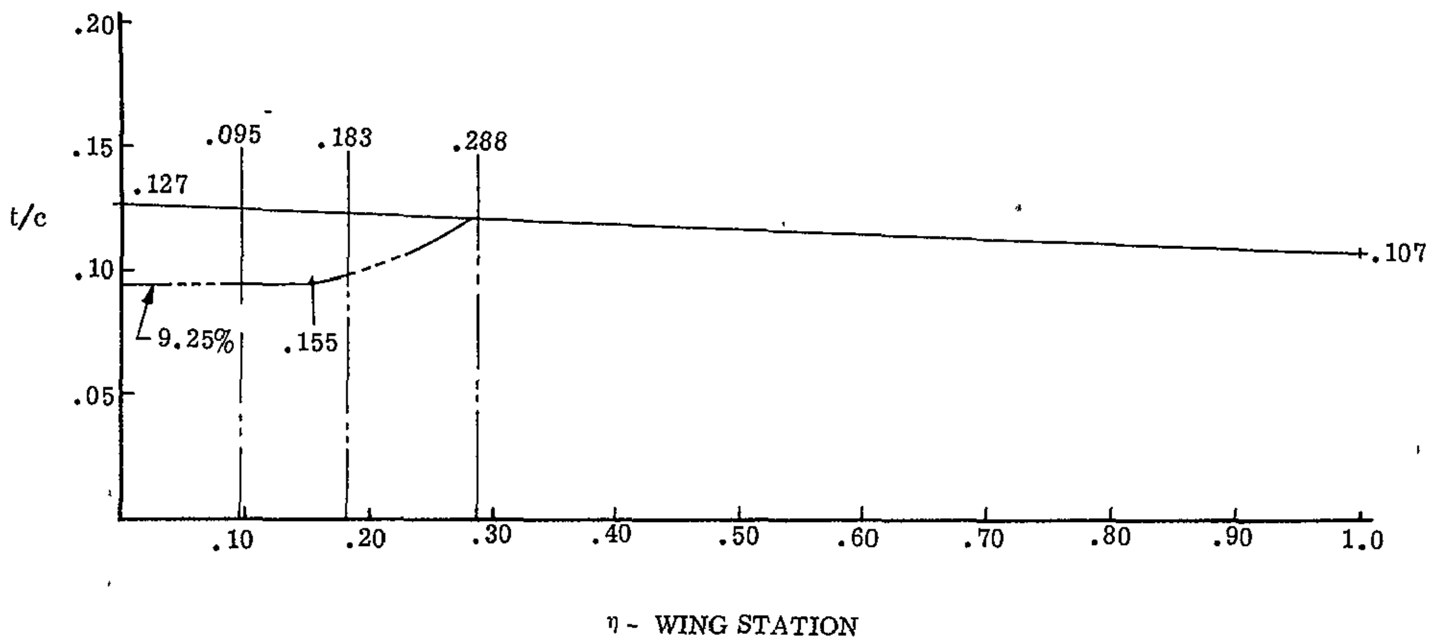

FIGURE 45 CONVENTIONAL CONFIGURATION -
MACH 0.95 CRUISE - WING THICKNESS DISTRIBUTION

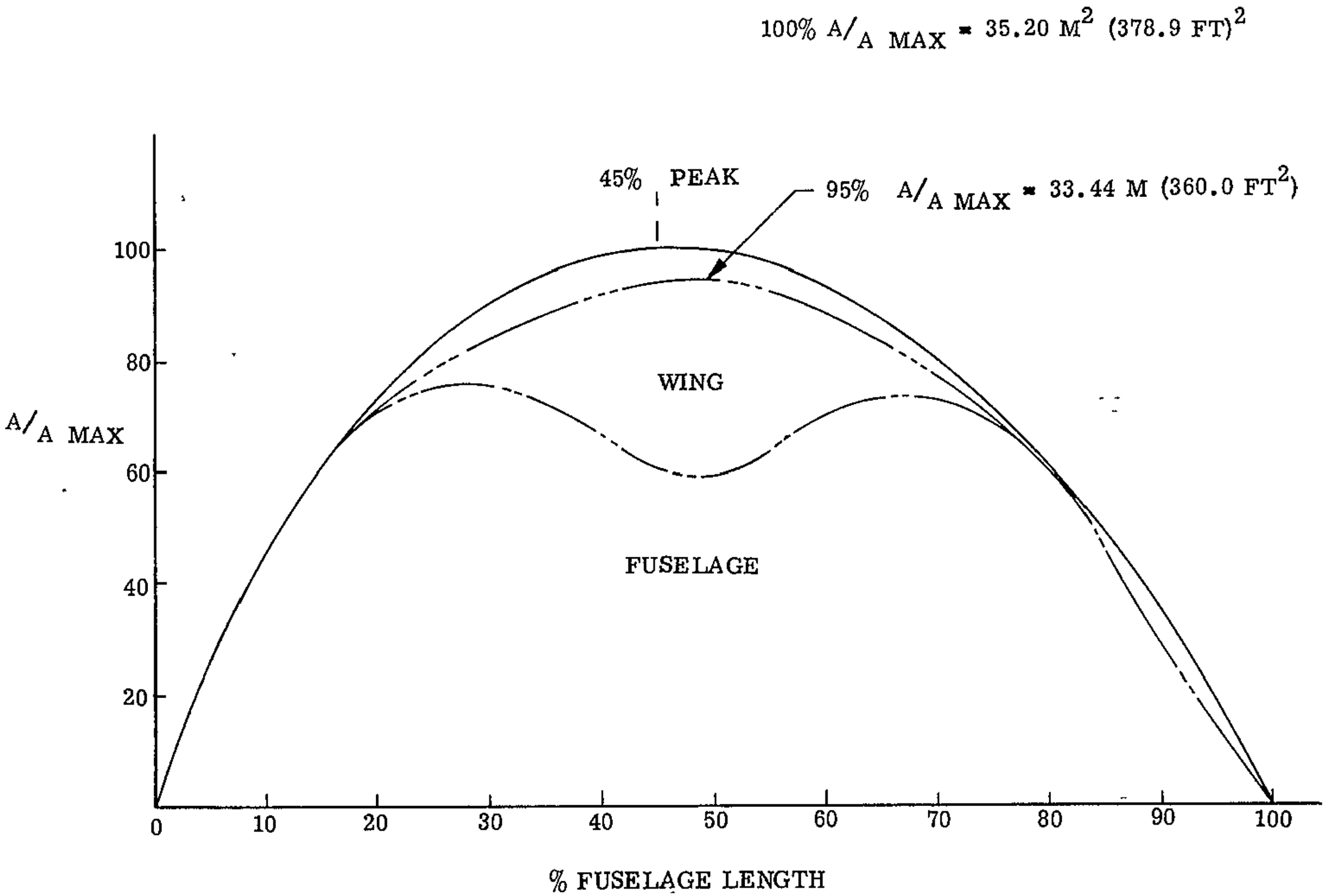

FIGURE 46 CONVENTIONAL CONFIGURATION - MACH 0.95 CRUISE - CROSS SECTIONAL AREA DISTRIBUTION

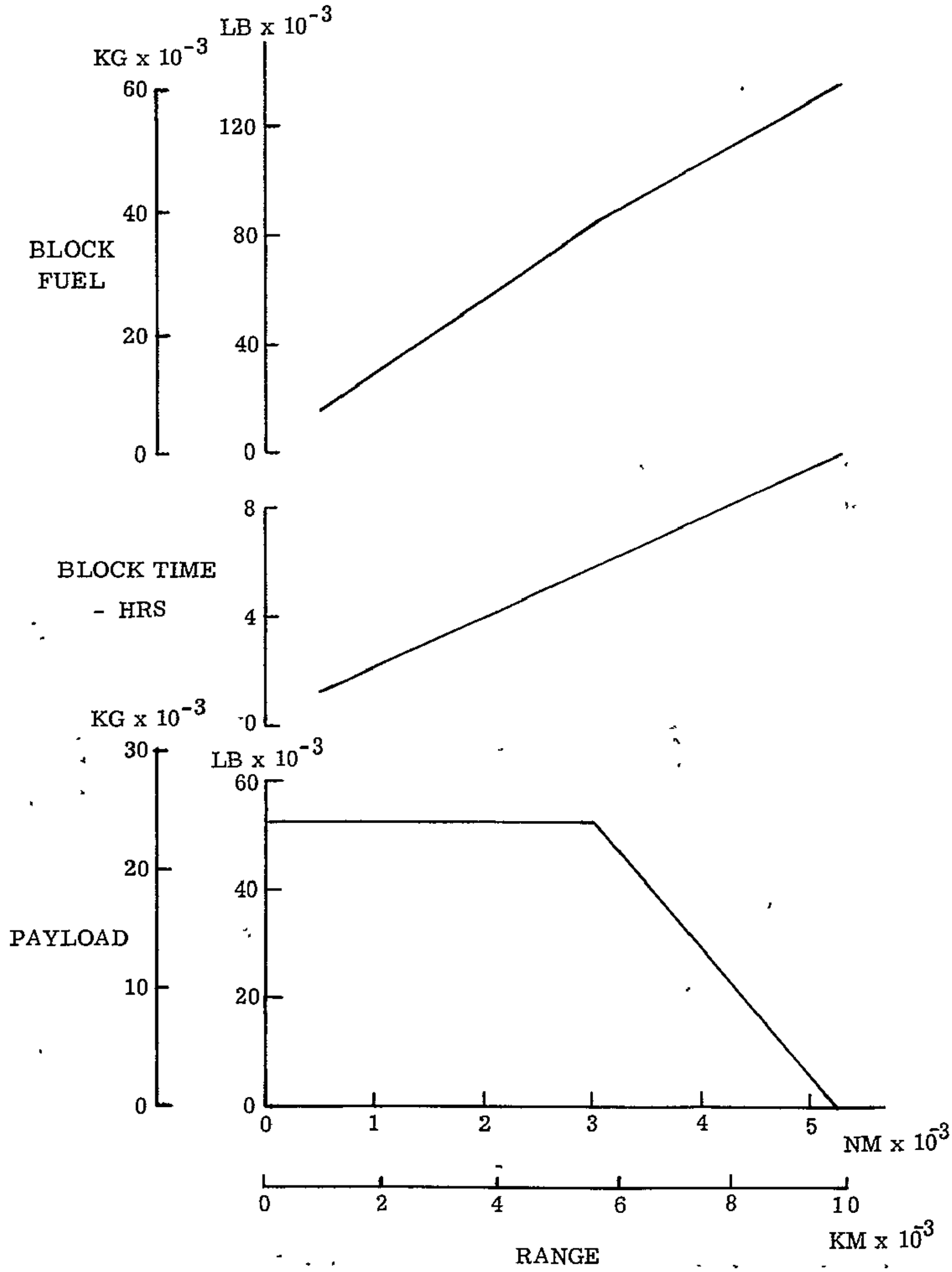

FIGURE 47 CONVENTIONAL CONFIGURATION - MACH 0.95 CRUISE - PAYLOAD-RANGE

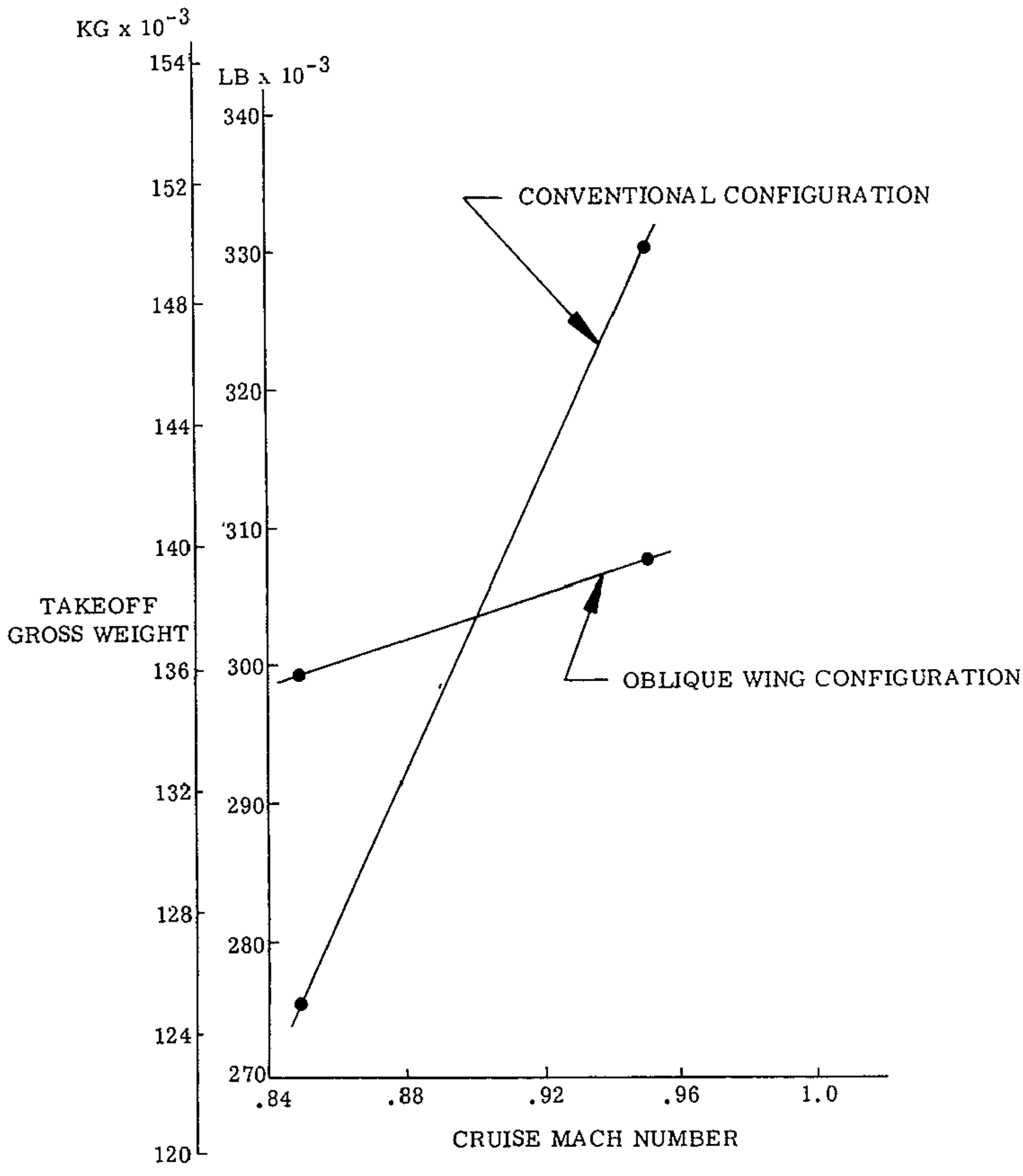

FIGURE 48 CONCEPT EVALUATION -
TAKEOFF GROSS WEIGHT COMPARISON

CRITERION - DIRECT OPERATING COST

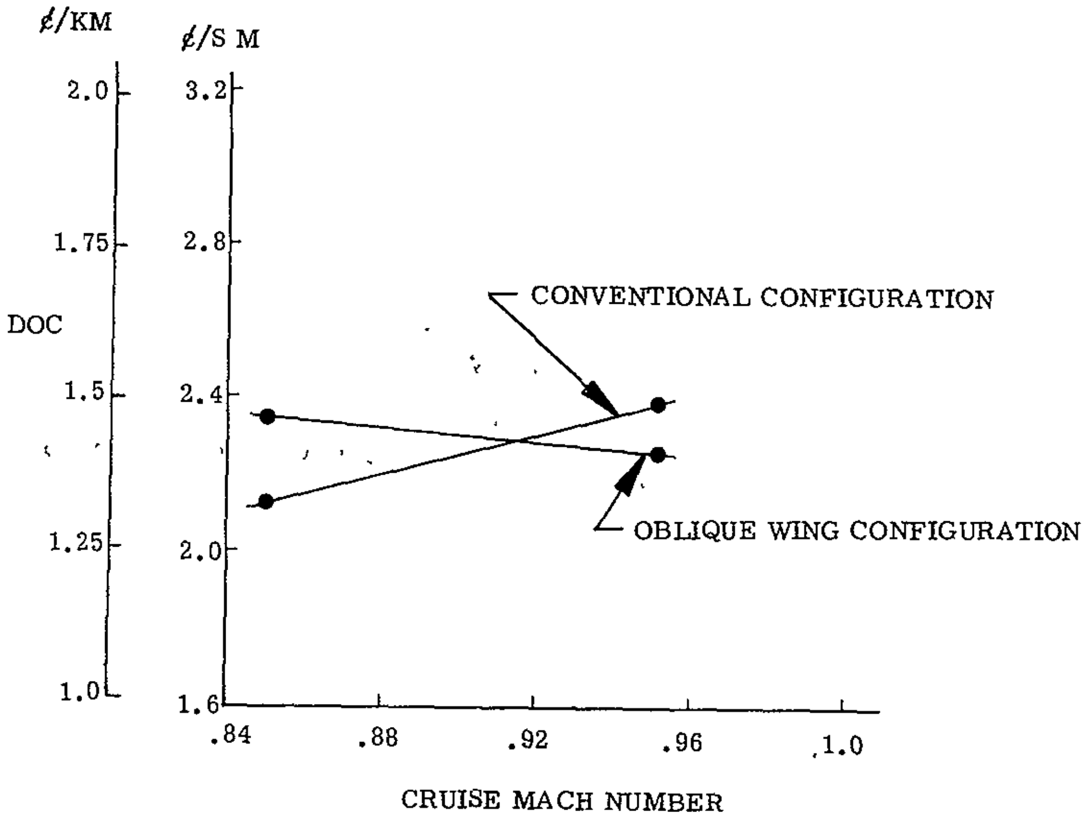

FIGURE 49 CONCEPT EVALUATION - DOC COMPARISON

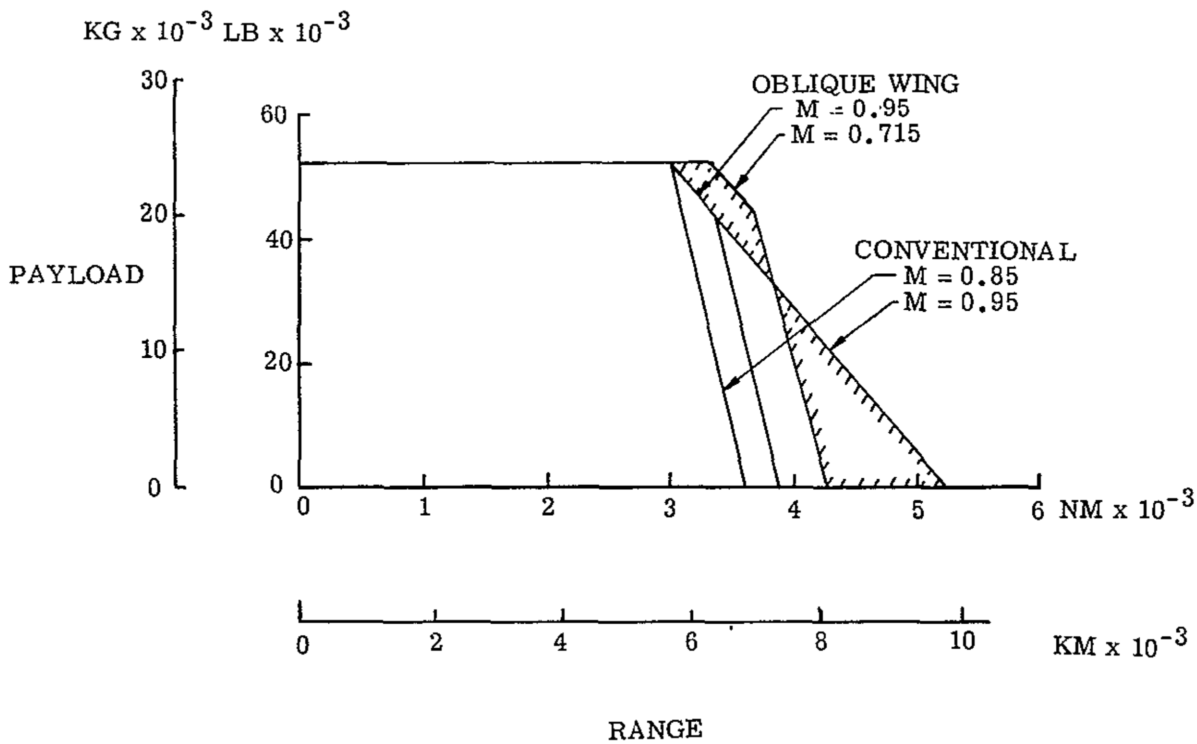

FIGURE 50 CONCEPT EVALUATION - PAYLOAD-RANGE COMPARISON

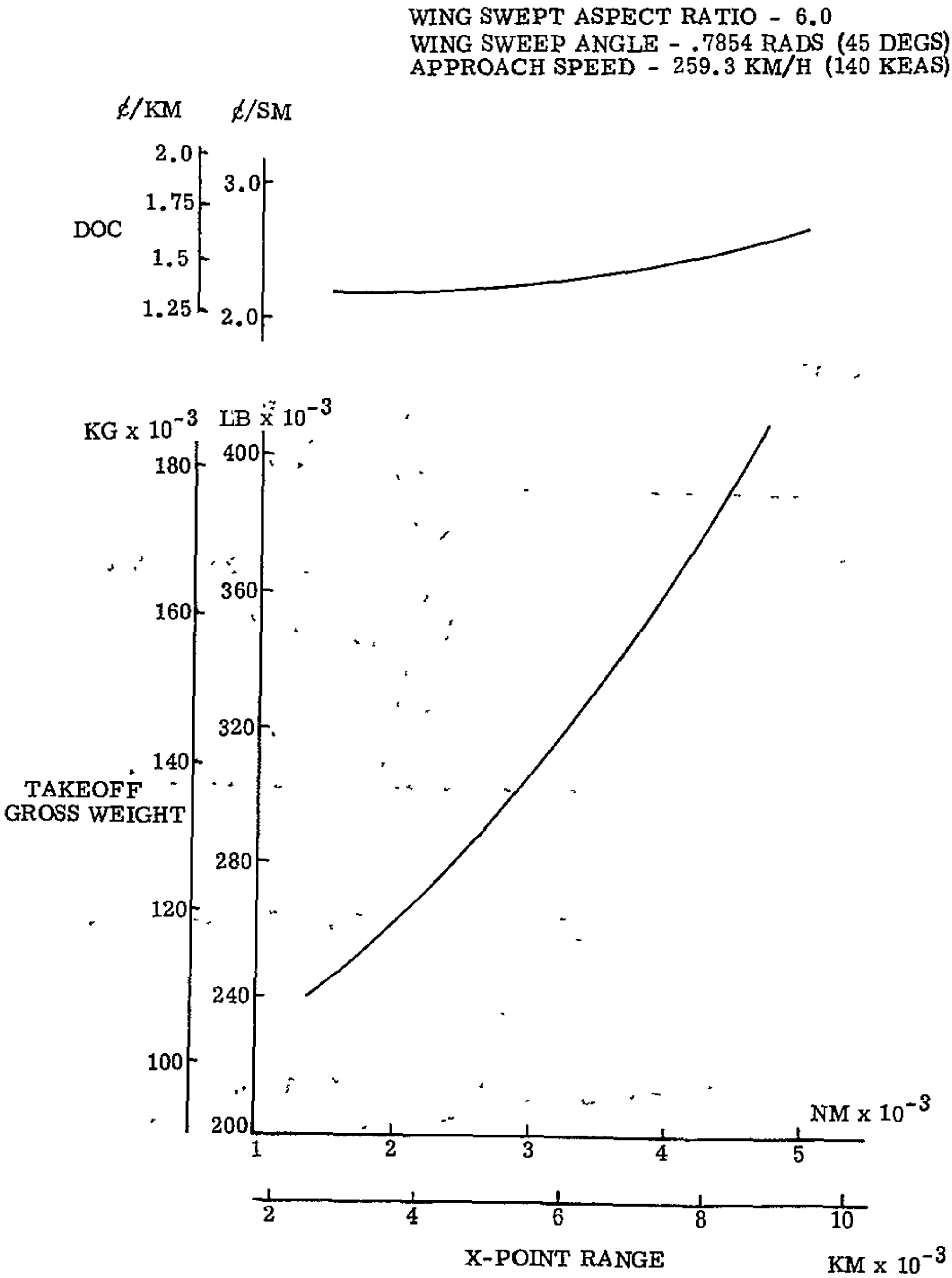

FIGURE 51 CONCEPT EVALUATION - WEIGHT-RANGE SENSITIVITY

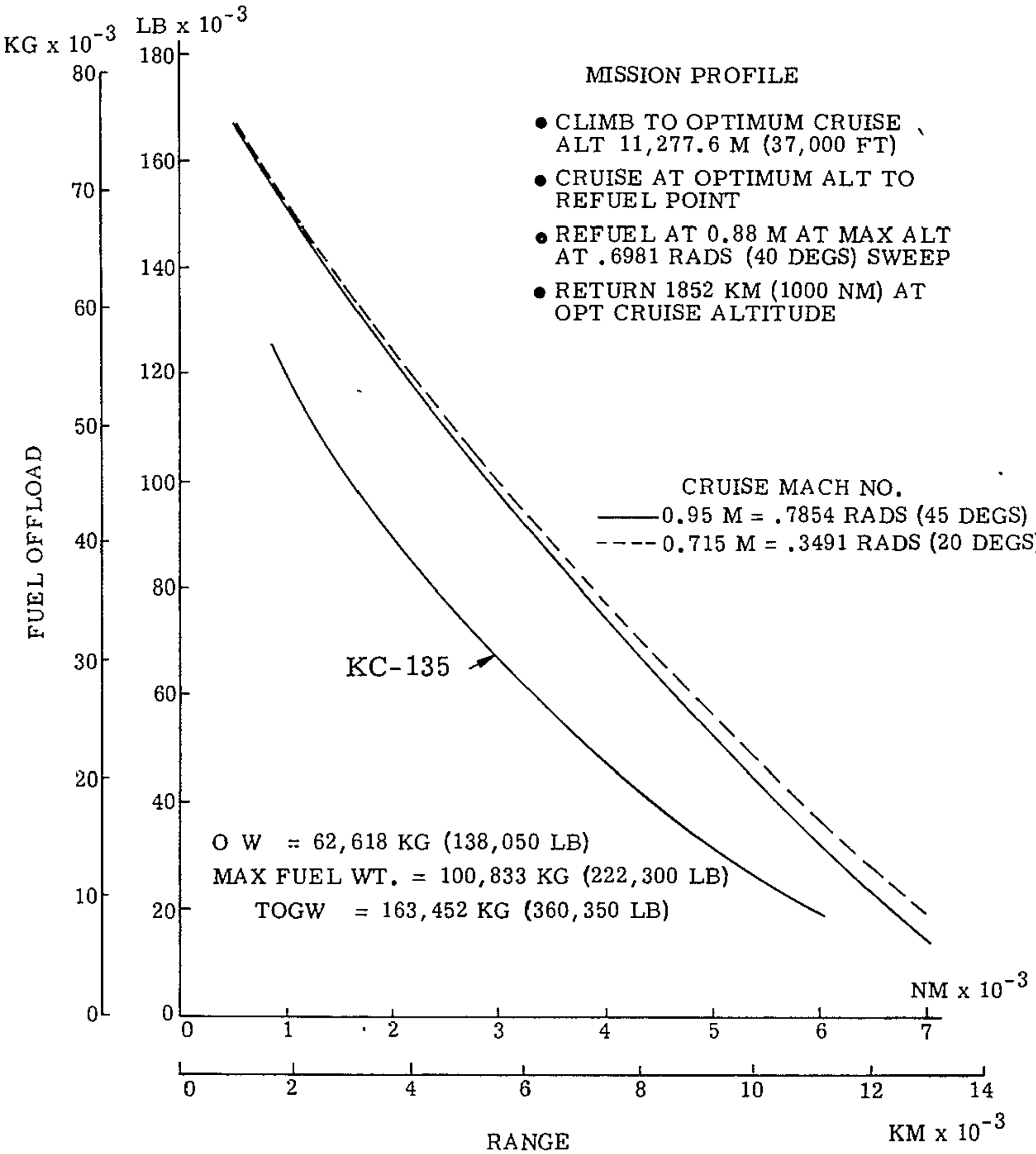

FIGURE 52 ALTERNATE MISSION - AIR FORCE TANKER FUEL OFF-LOAD CAPABILITY

PROFILE

- CLIMB TO OPTIMUM CRUISE ALT
- CRUISE AT OPTIMUM ALT AT SPECIFIED SPEED
- LOITER AT 5000'
- CLIMB TO OPTIMUM ALT
- RETURN TO BASE

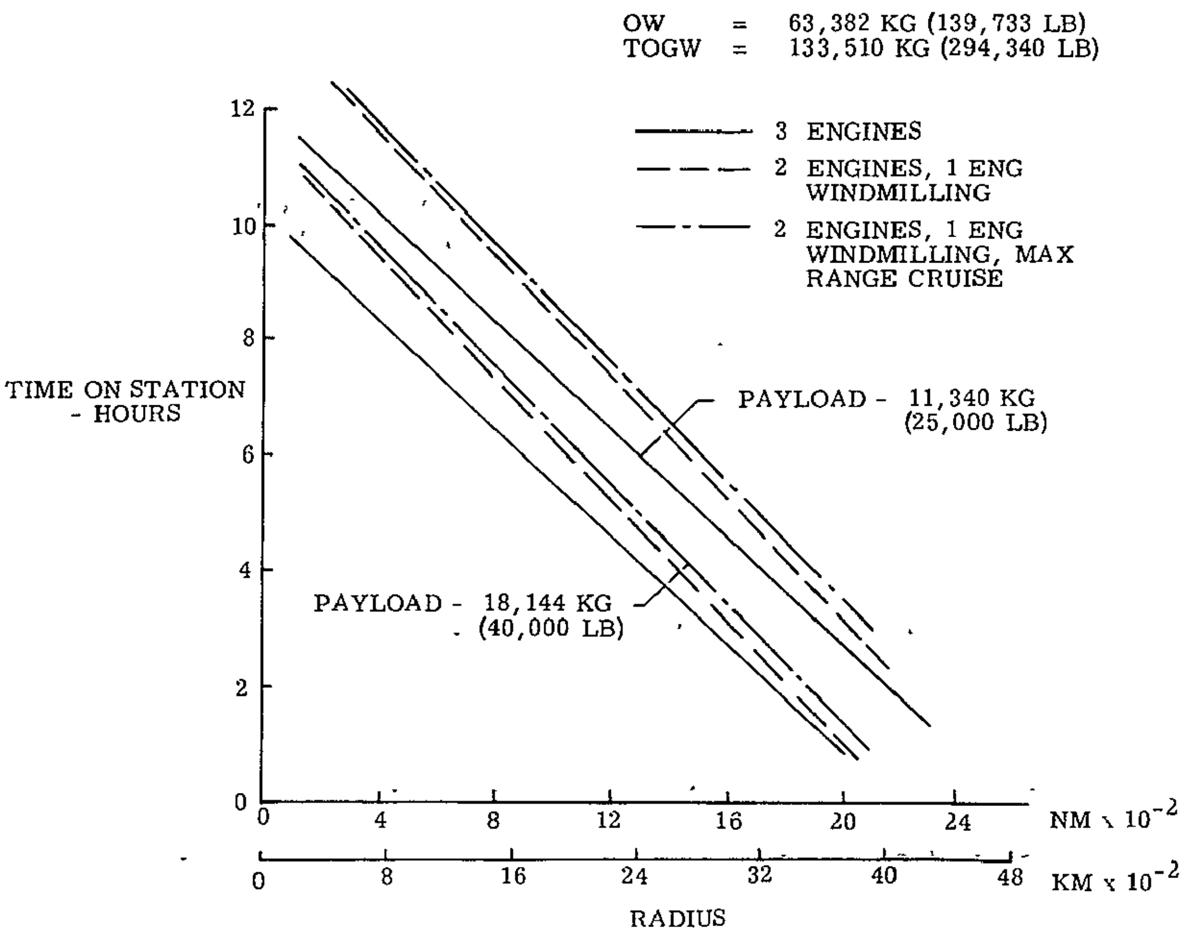

FIGURE 53 ALTERNATE MISSION - NAVY ASW ENDURANCE CAPABILITY

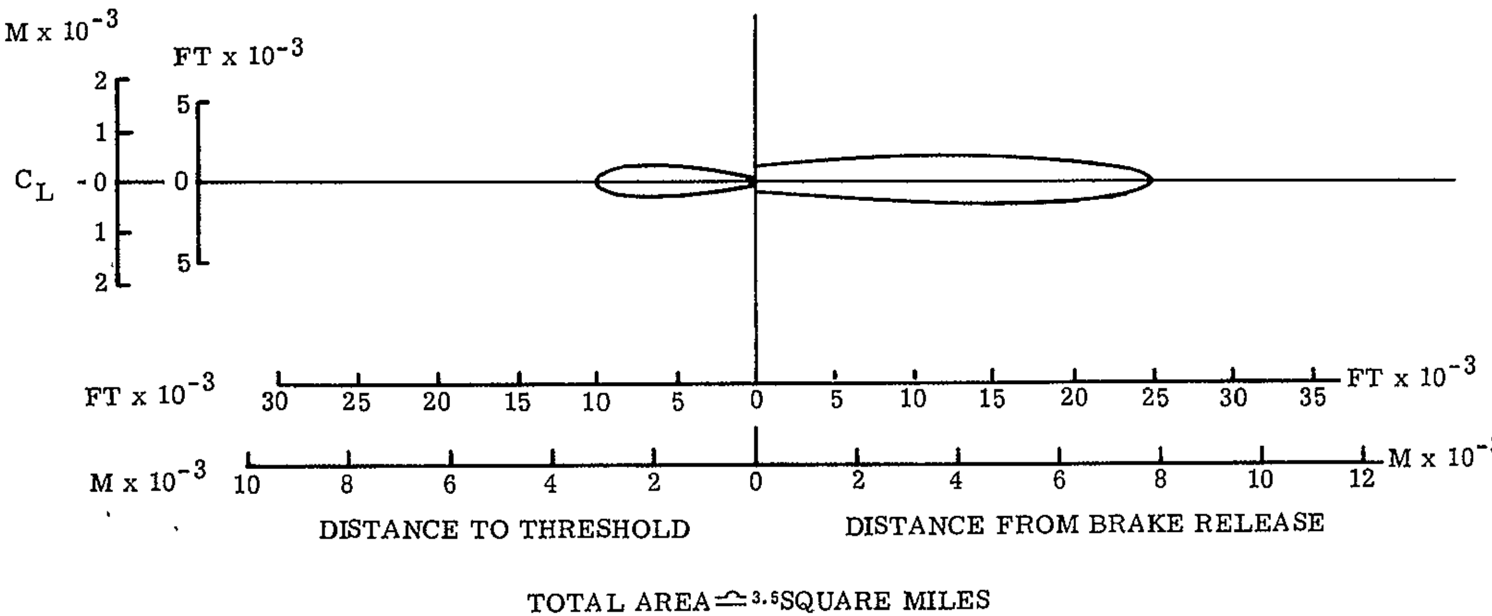

FIGURE 54 FINAL CONFIGURATION - ACOUSTIC SOUNDPRINT

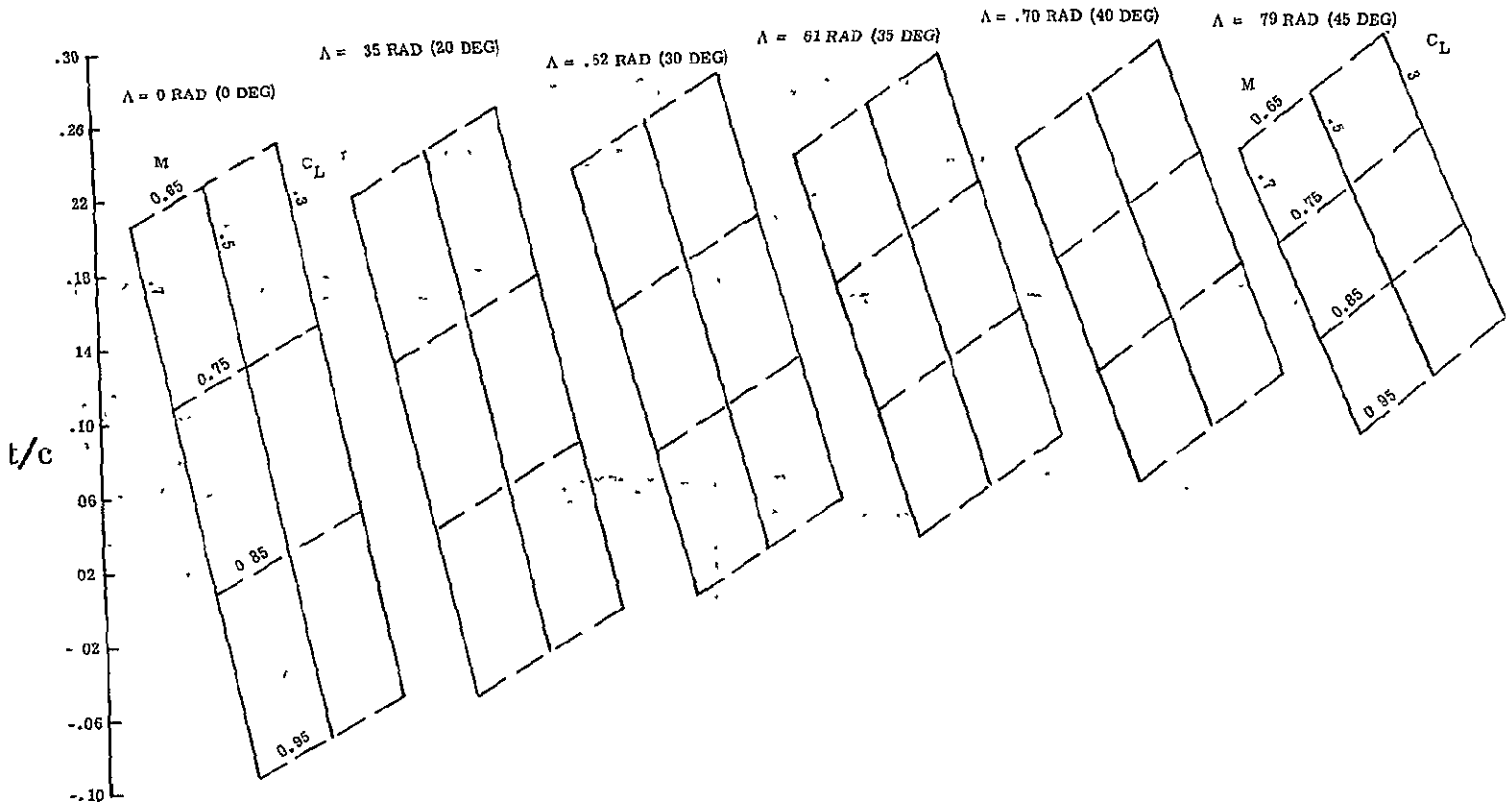

FIGURE A1 C_L - M - Λ - t/c RELATIONSHIP

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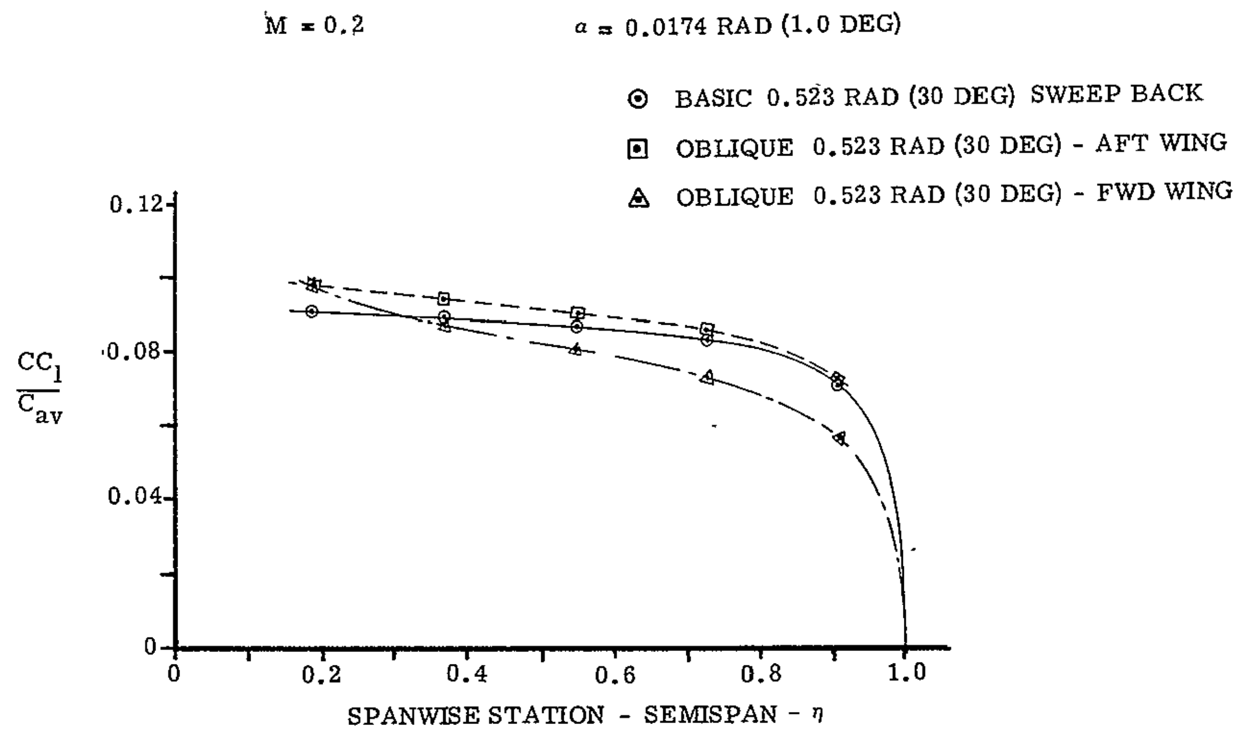


FIGURE A2 LOAD DISTRIBUTION COMPARISON

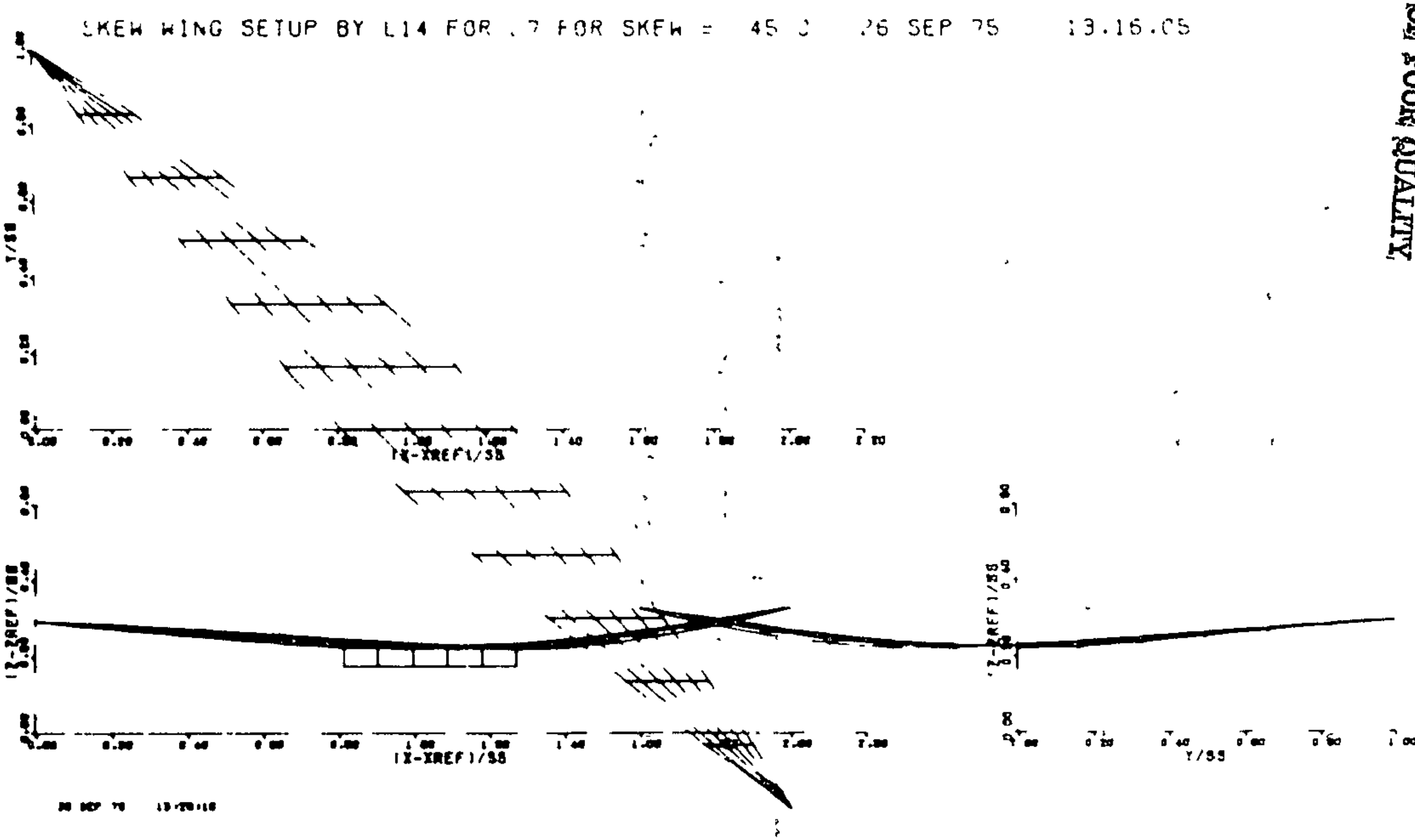
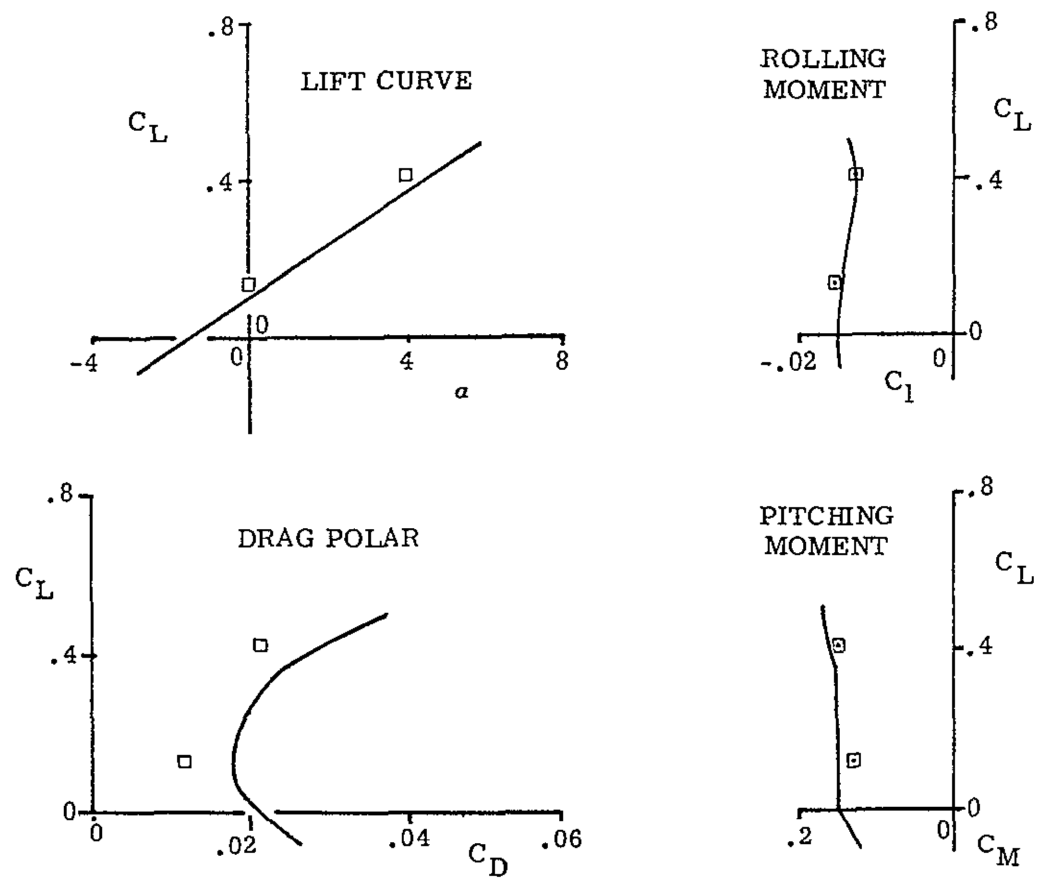


FIGURE A3 BOEING WIND TUNNEL MODEL COMPUTER REPRESENTATION

M = 0.8 WING SWEEP ANGLE Λ= -0.785 RAD (-45 DEG)



⊡ DISCRETE ELEMENT AERODYNAMIC PROGRAM COMPUTED DATA - TAIL-OFF
— WIND TUNNEL DATA FROM REFERENCE A1, TAIL-ON

FIGURE A4 COMPARISON OF WIND TUNNEL AND THEORETICAL DATA

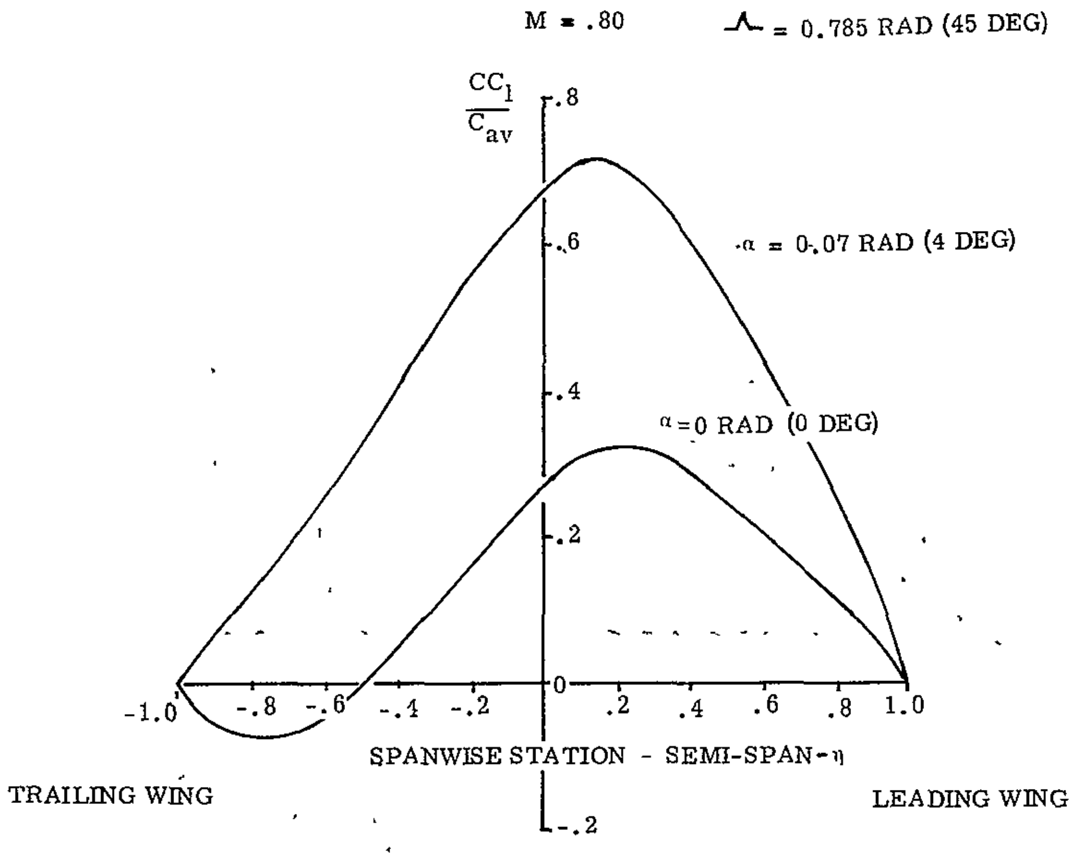


FIGURE A5 BOEING MODEL TEST WING LIFT DISTRIBUTION

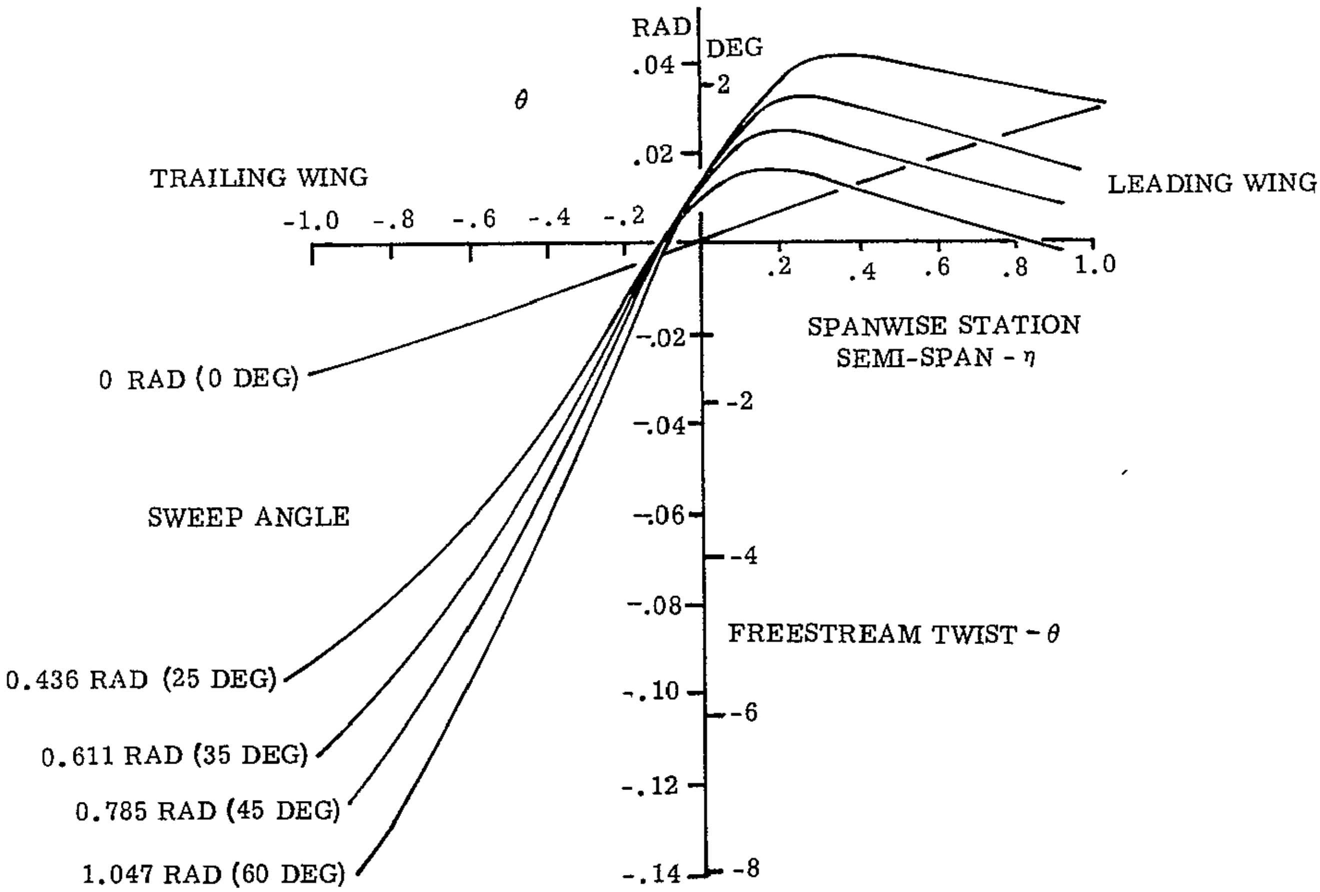


FIGURE A6 GENERAL DYNAMICS TEST OF BOEING MODEL FREESTREAM TWIST

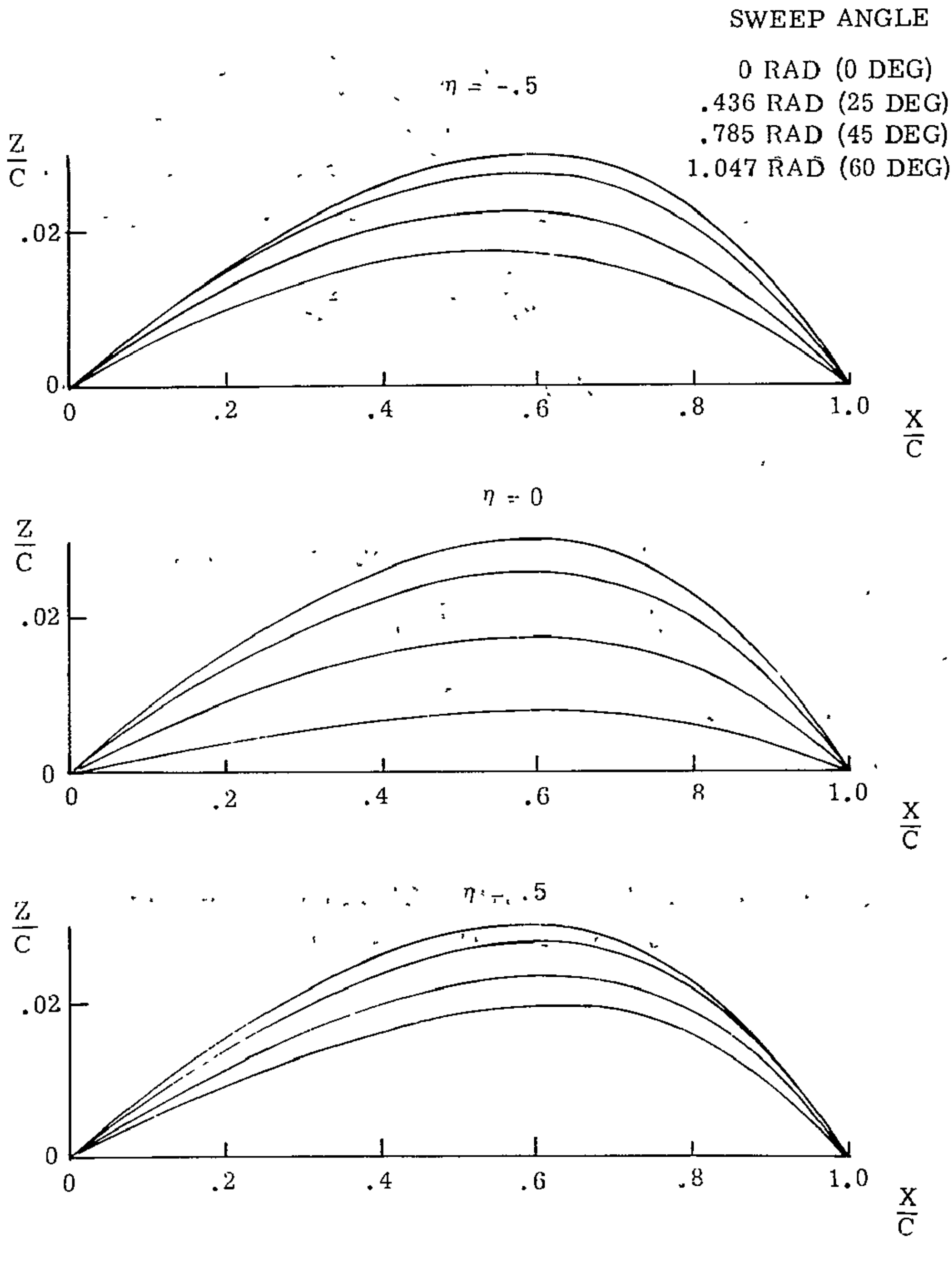


FIGURE A7 GENERAL DYNAMICS TEST OF BOEING MODEL FREESTREAM CAMBER

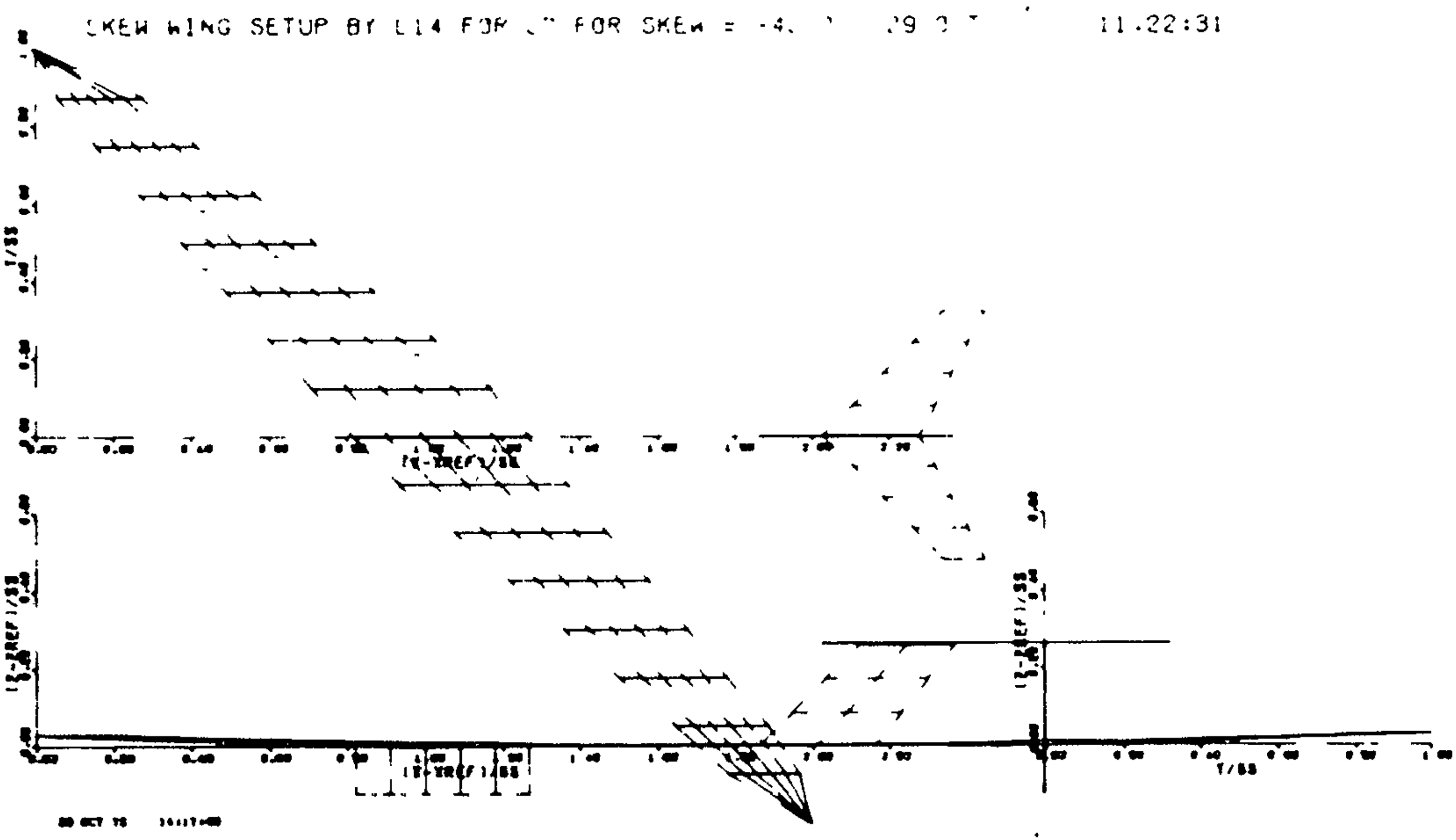
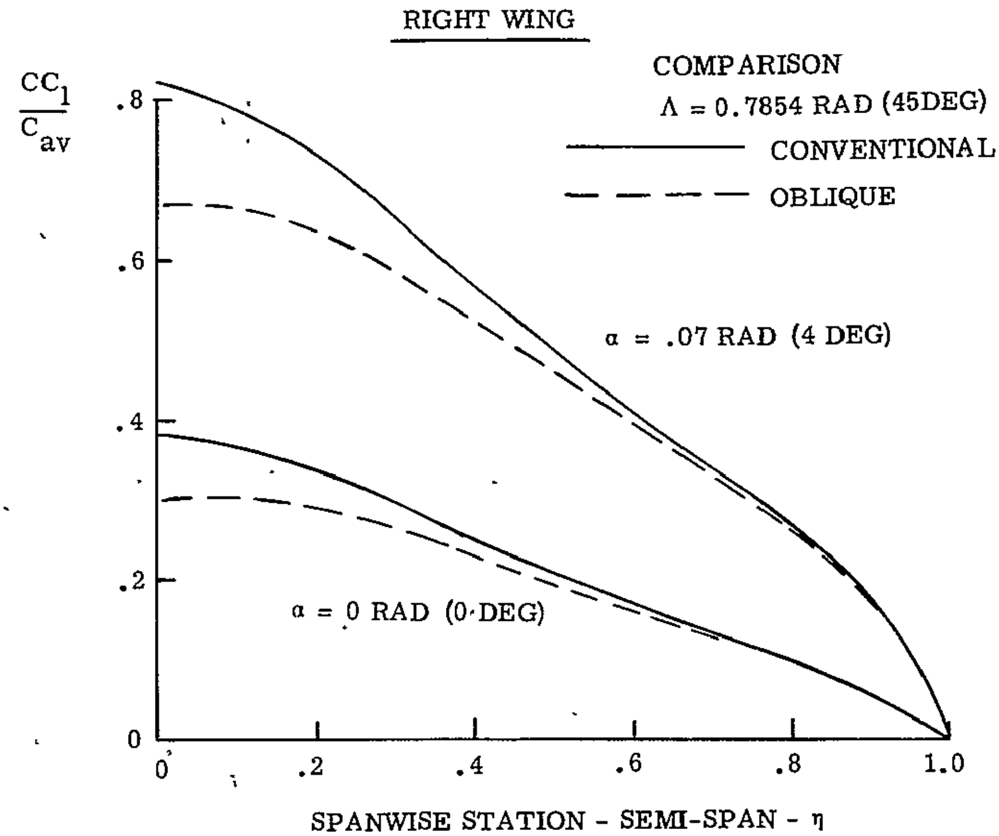


FIGURE A8 COMMERCIAL PASSENGER TRANSPORT -
BASELINE CONFIGURATION COMPUTER REPRESENTATION



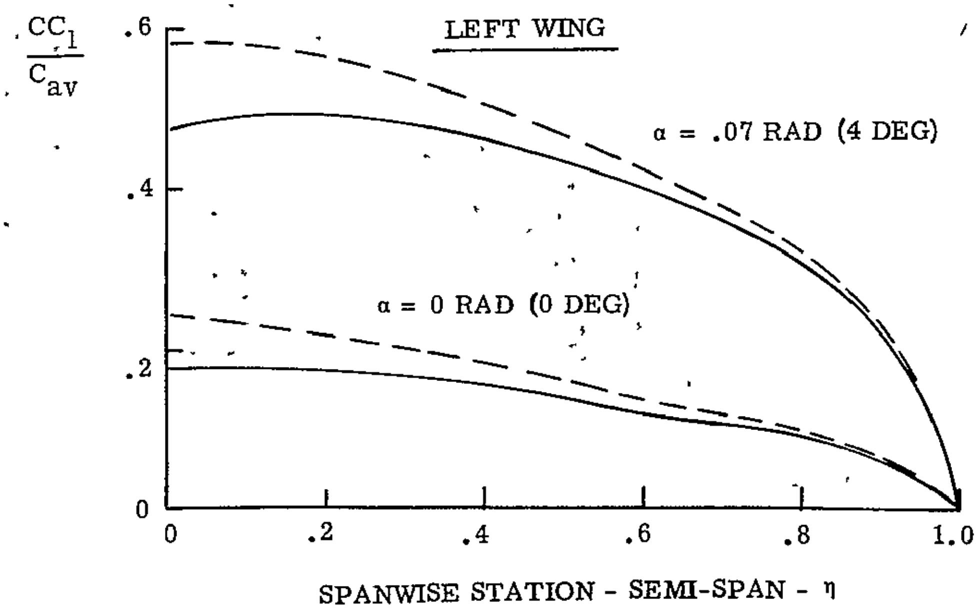


FIGURE A9 COMMERCIAL PASSENGER TRANSPORT - WING LIFT DISTRIBUTION COMPARISON

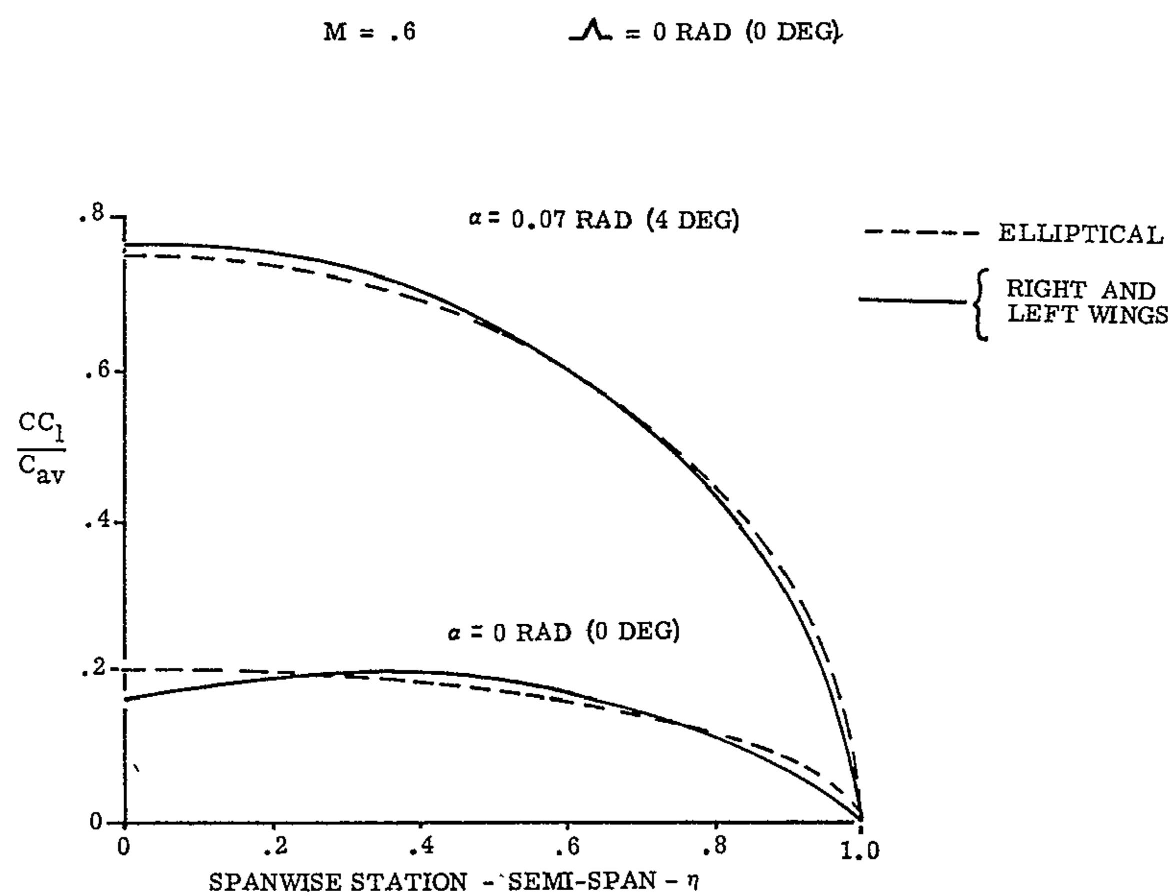


FIGURE A 10 COMMERCIAL PASSENGER TRANSPORT - SPANWISE WING LIFT DISTRIBUTION COMPARISON - UNSWEPT

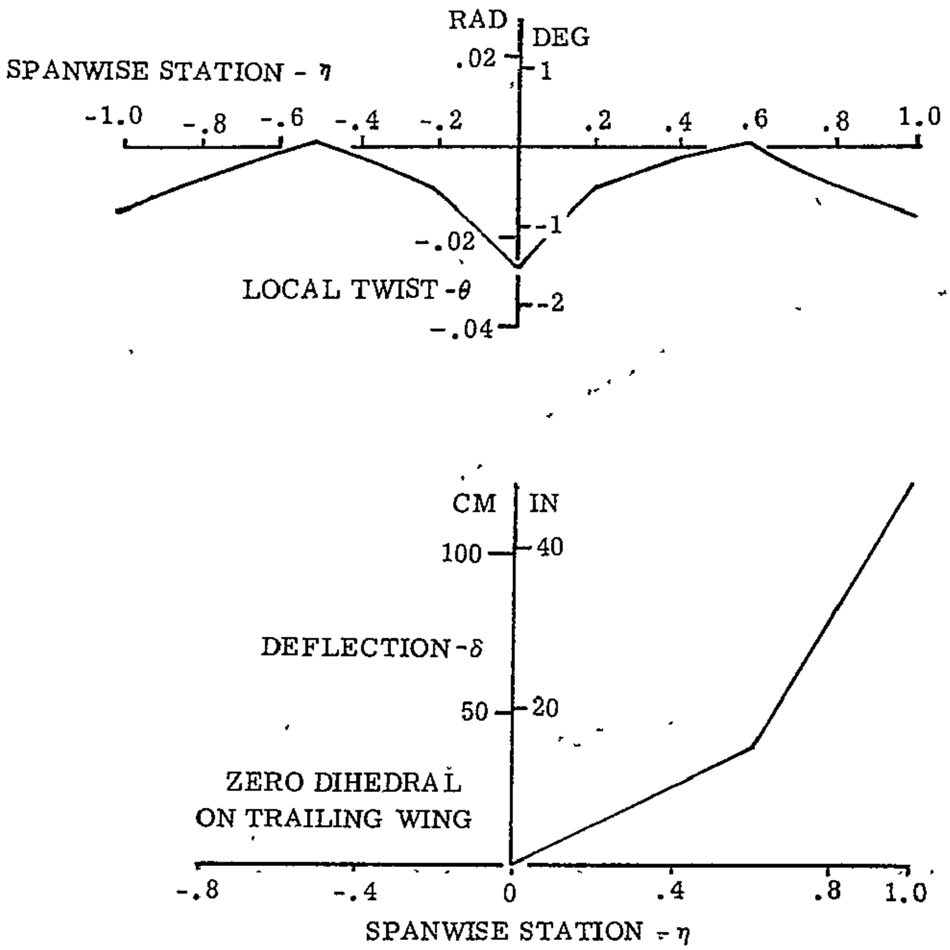


FIGURE A11 WING TWIST AND DEFLECTION

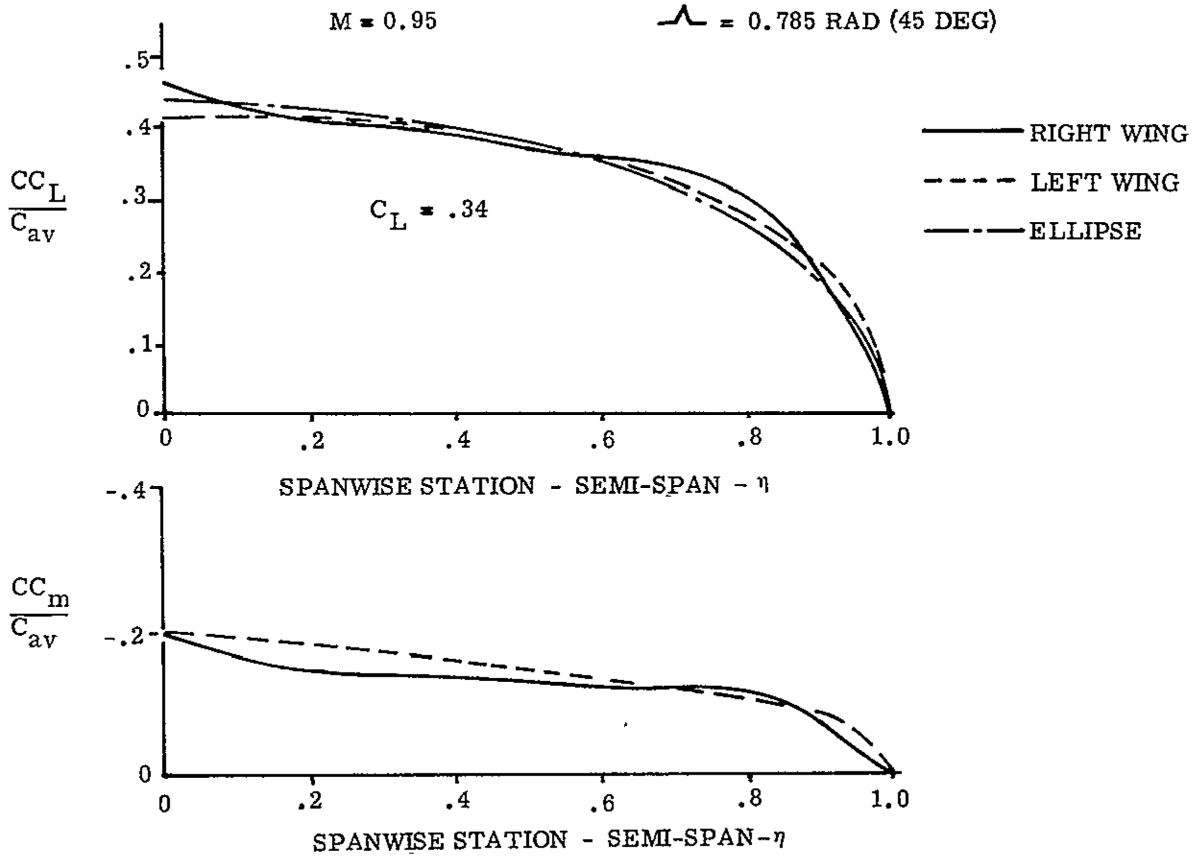


FIGURE A12 WING LIFT DISTRIBUTION Λ = 0.785 RAD (45 DEG)

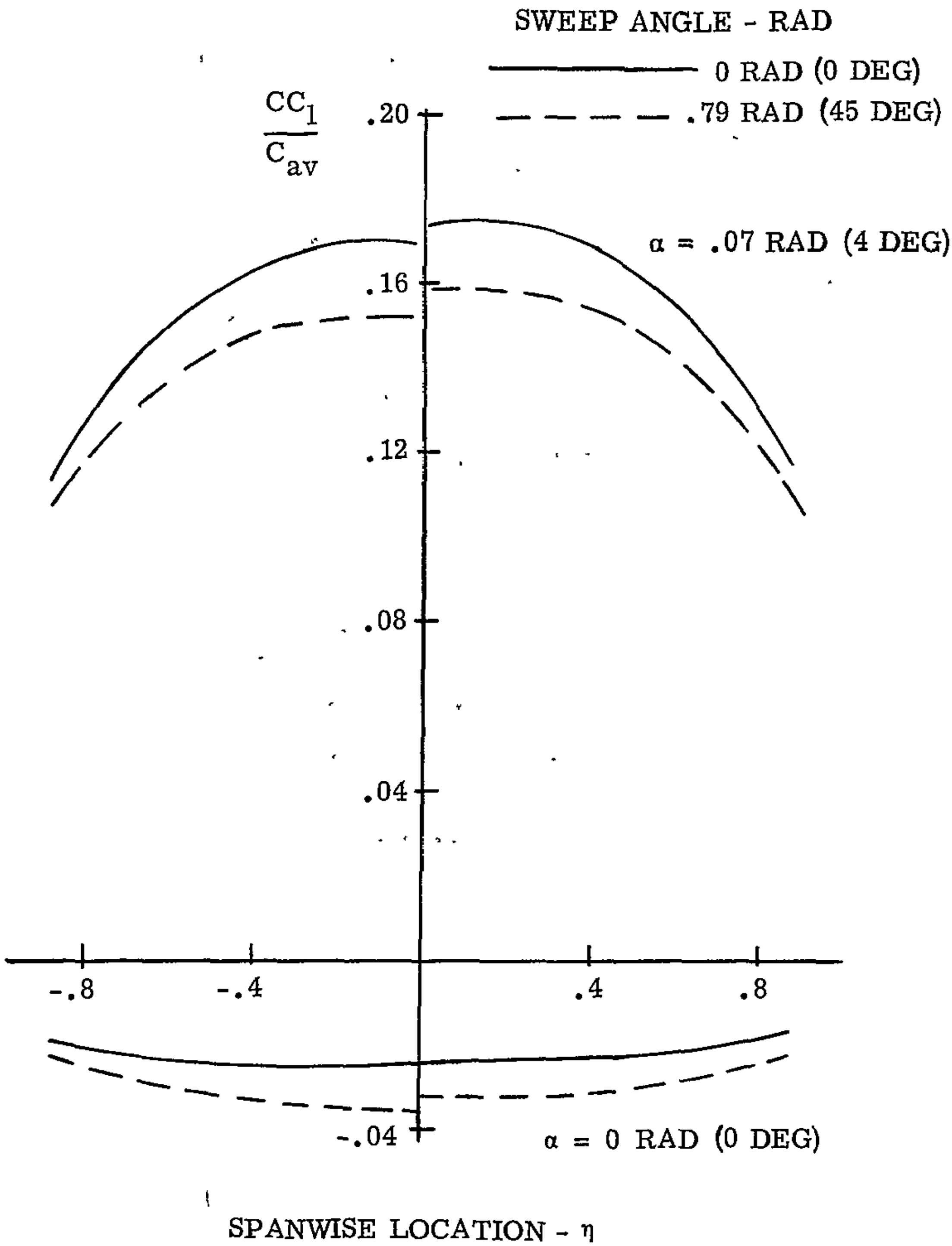


FIGURE A13 HORIZONTAL STABILIZER LIFT DISTRIBUTION

$M = 0.95$ $\Lambda = 0.785$ RAD (45 DEG) ALTITUDE = 11,000 M (36,000 FT)

$\alpha_{FRL} = 0.0436$ RAD (2.5 DEG) DYNAMIC PRESSURE = 14,344 N/M^2 (300 LB/FT^2)

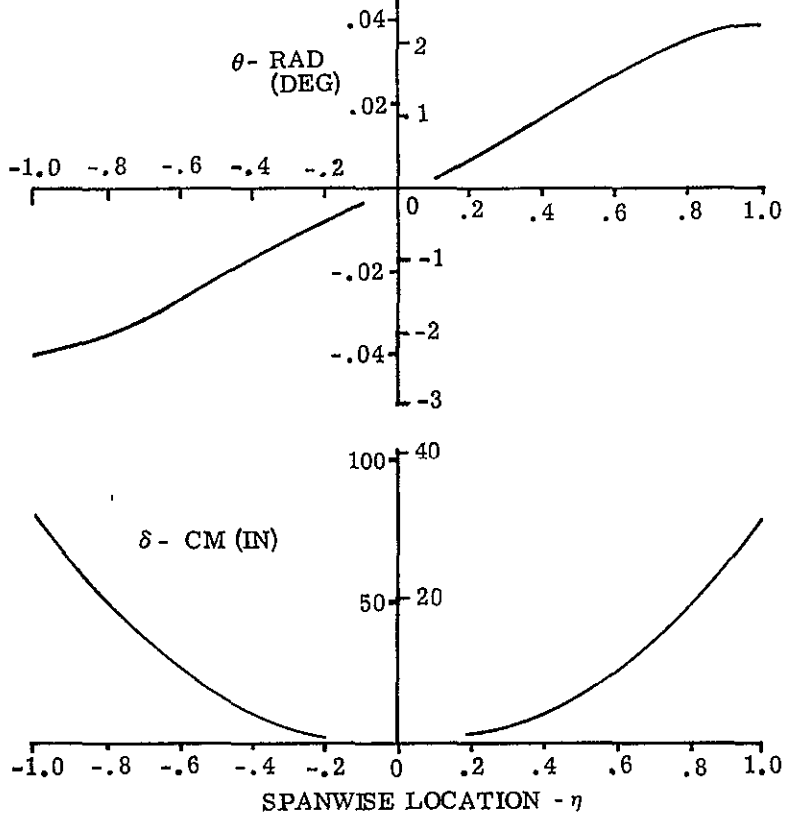


FIGURE A14 INCREMENTAL TWIST AND DEFLECTION

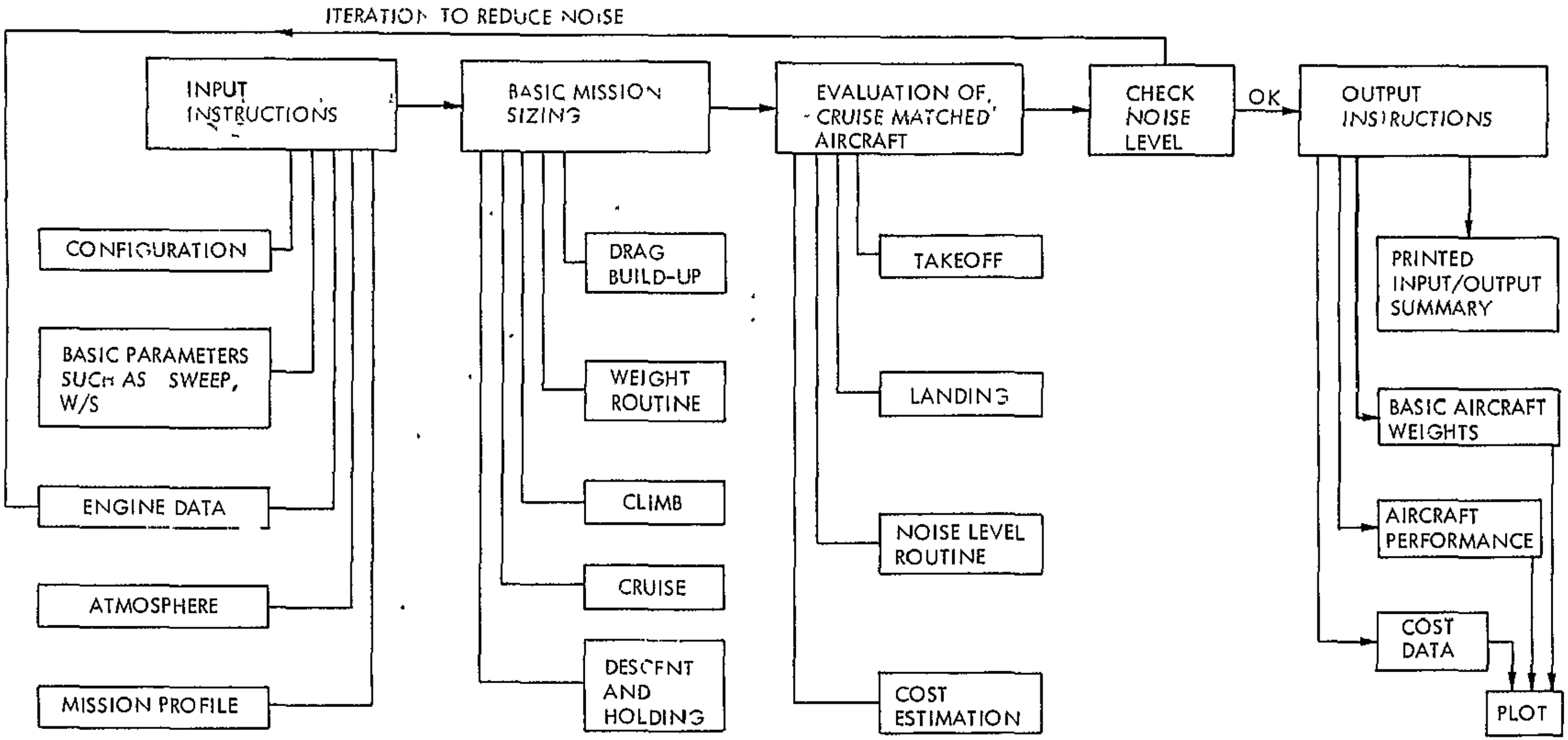


FIGURE A15 GENERALIZED AIRCRAFT SIZING PROGRAM

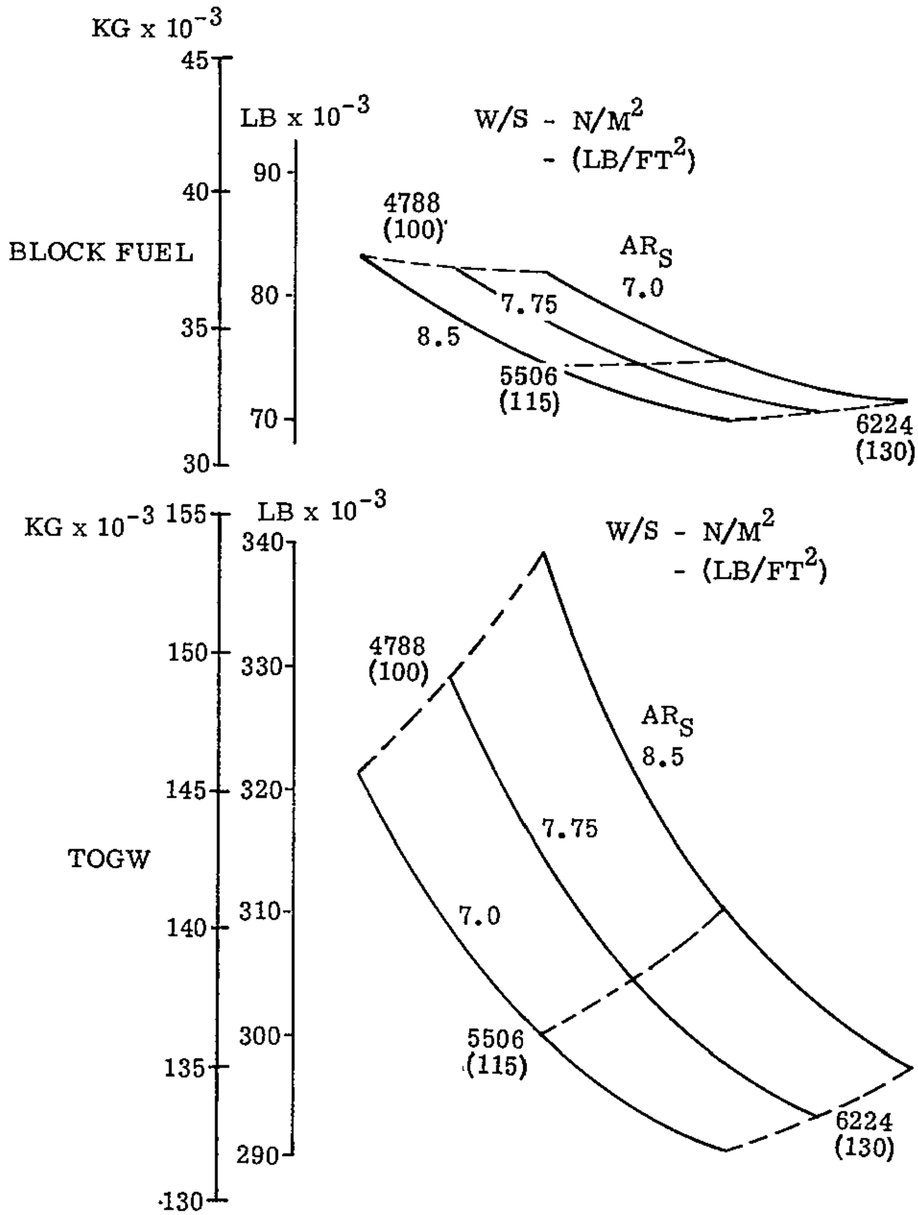


FIGURE A16 COMMERCIAL PASSENGER TRANSPORT INITIAL CONFIGURATION SIZING CHART

23,768 KG (52,400 LB) PAYLOAD 5560 KM (3000 NM)

Λ = 7854 RADS (45 DEGS) (3) STF 433 ENGINES η_{CR} = 0.95

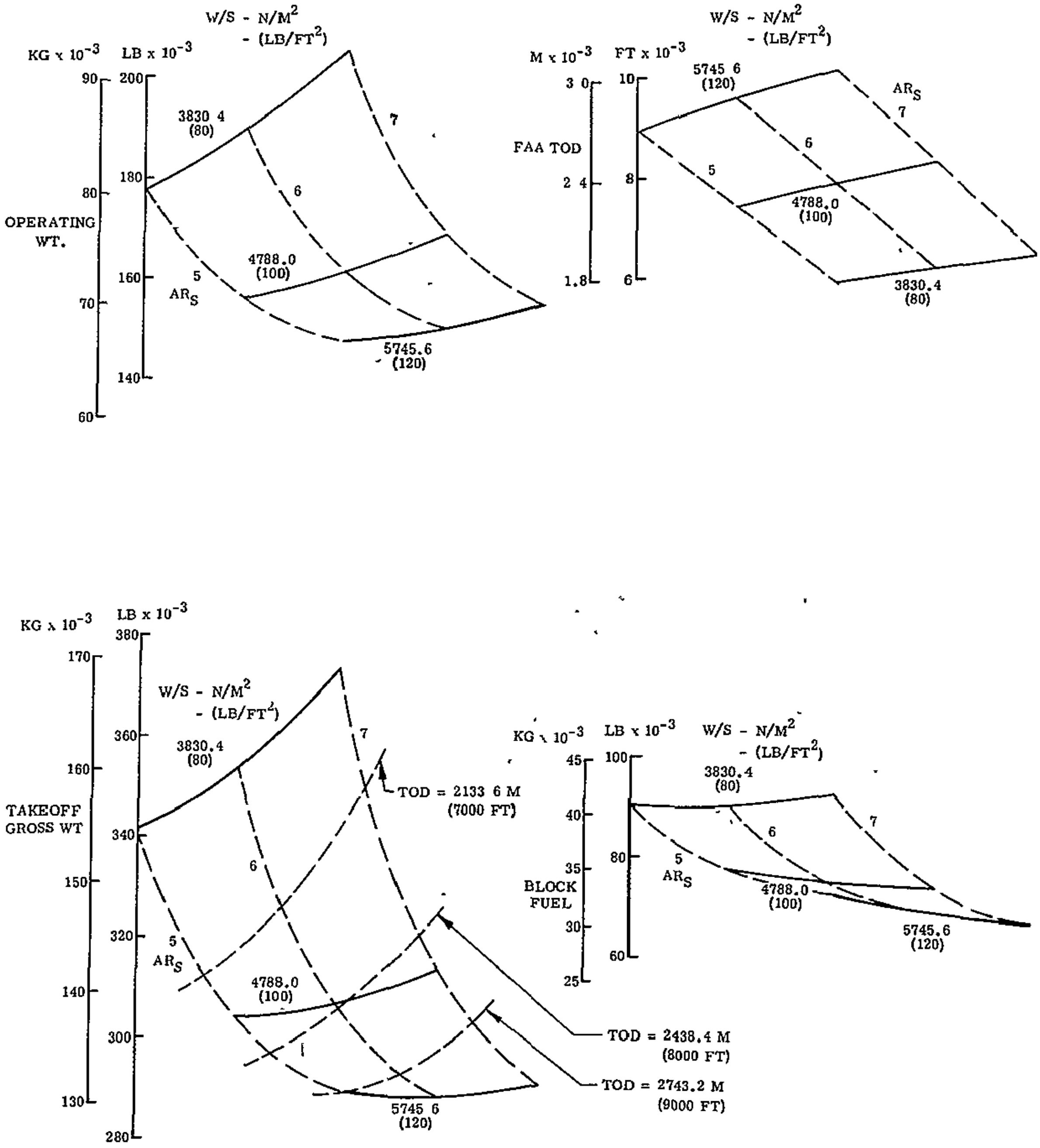


FIGURE A17 COMMERCIAL PASSENGER TRANSPORT CONFIGURATION SIZING CHART - 0.95 CRUISE RATED THRUST

23,768 KG (52,400 LB) PAYLOAD 5560 KM (3000 NM)
(3) STF 433 ENGINES, η_{CR} = 0.85

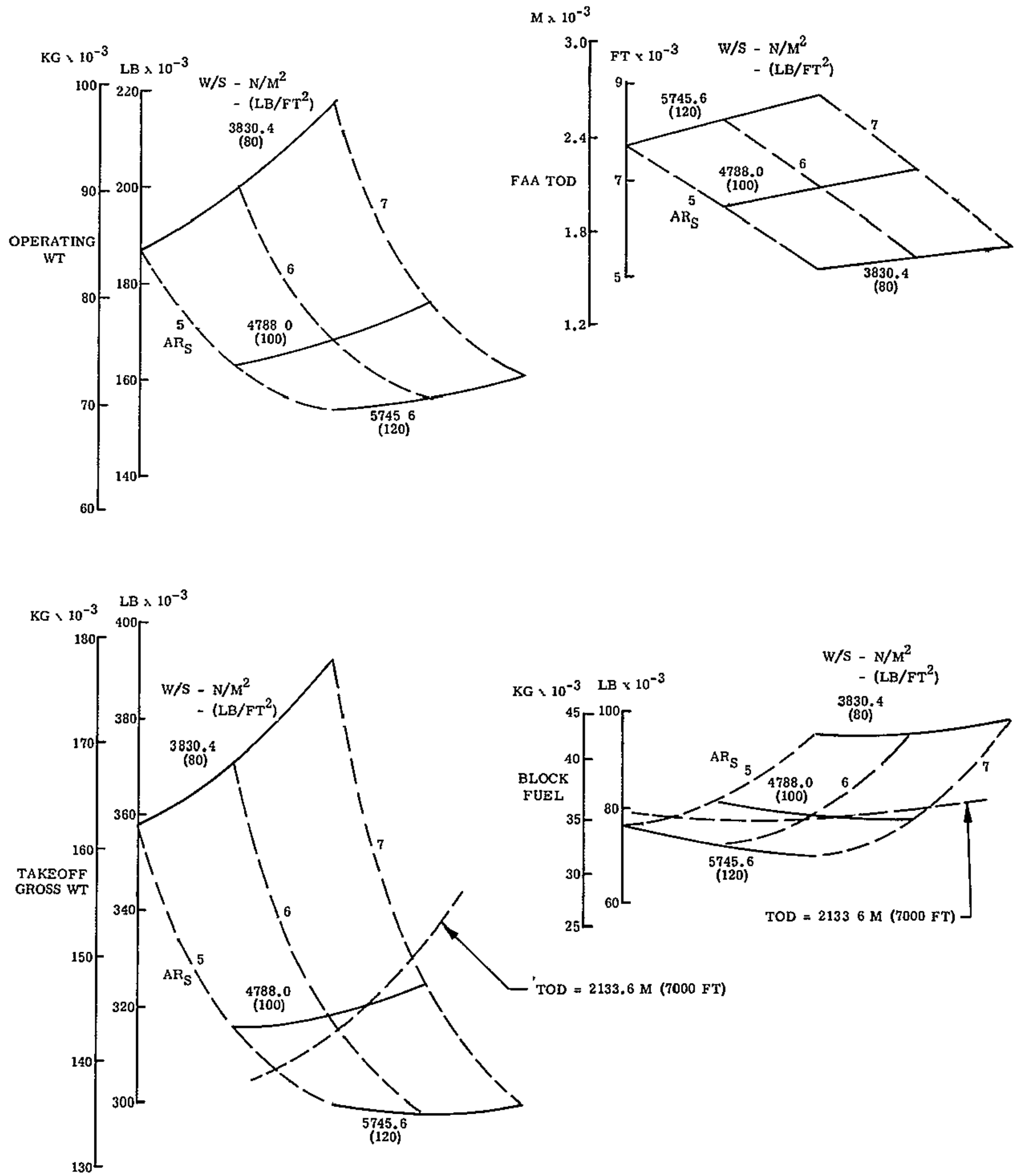


FIGURE A18 COMMERCIAL PASSENGER TRANSPORT CONFIGURATION SIZING CHART - 0.85 CRUISE RATED THRUST

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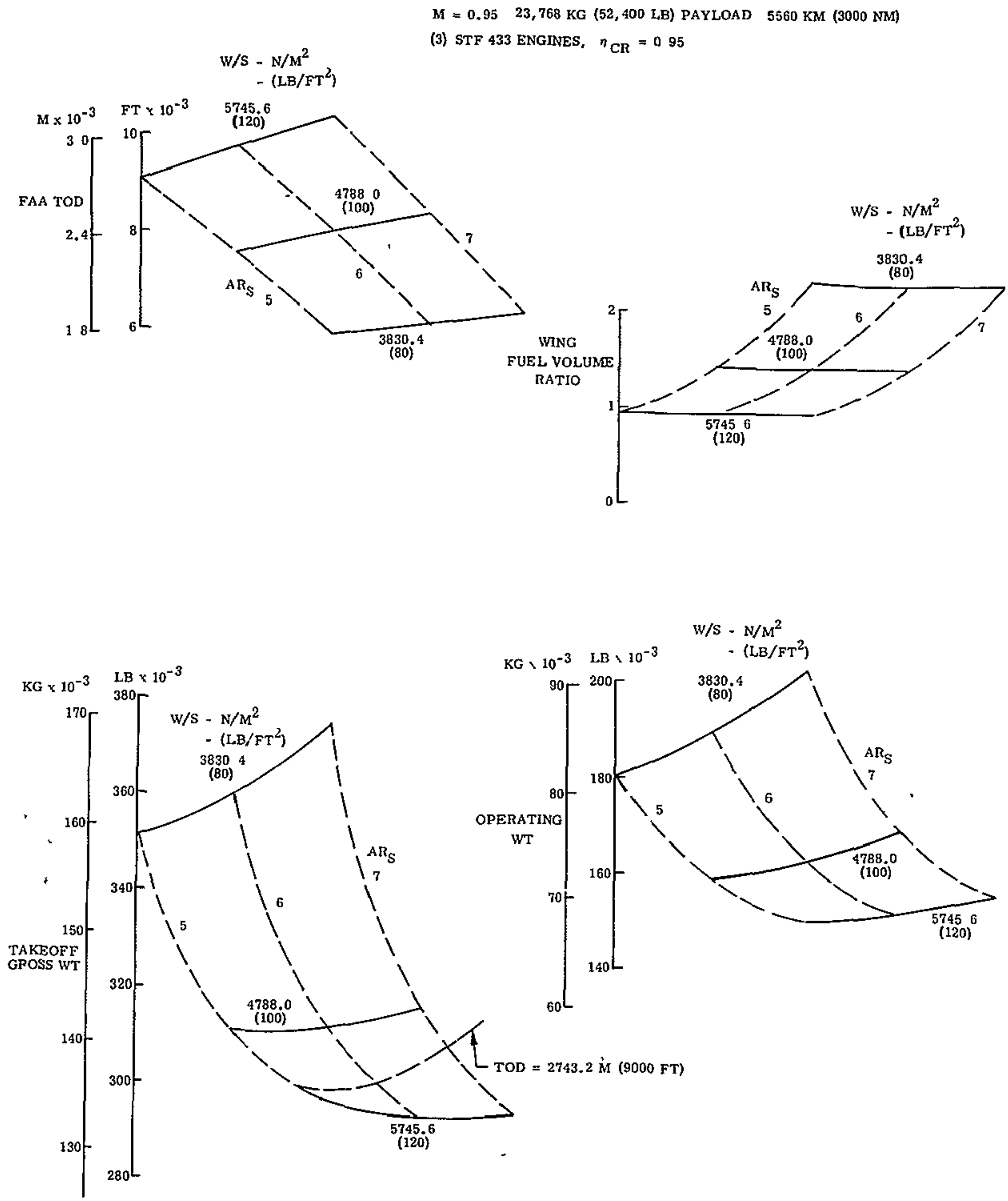

FIGURE A19 COMMERCIAL PASSENGER TRANSPORT CONFIGURATION SIZING CHART
Λ = 0.70 RAD (40 DEG)

M = 0.95 23,768 KG (52,400 LB) PAYLOAD 5560 KM (3000 NM)

(3) STF 433 ENGINES, η_{CR} = 0.95

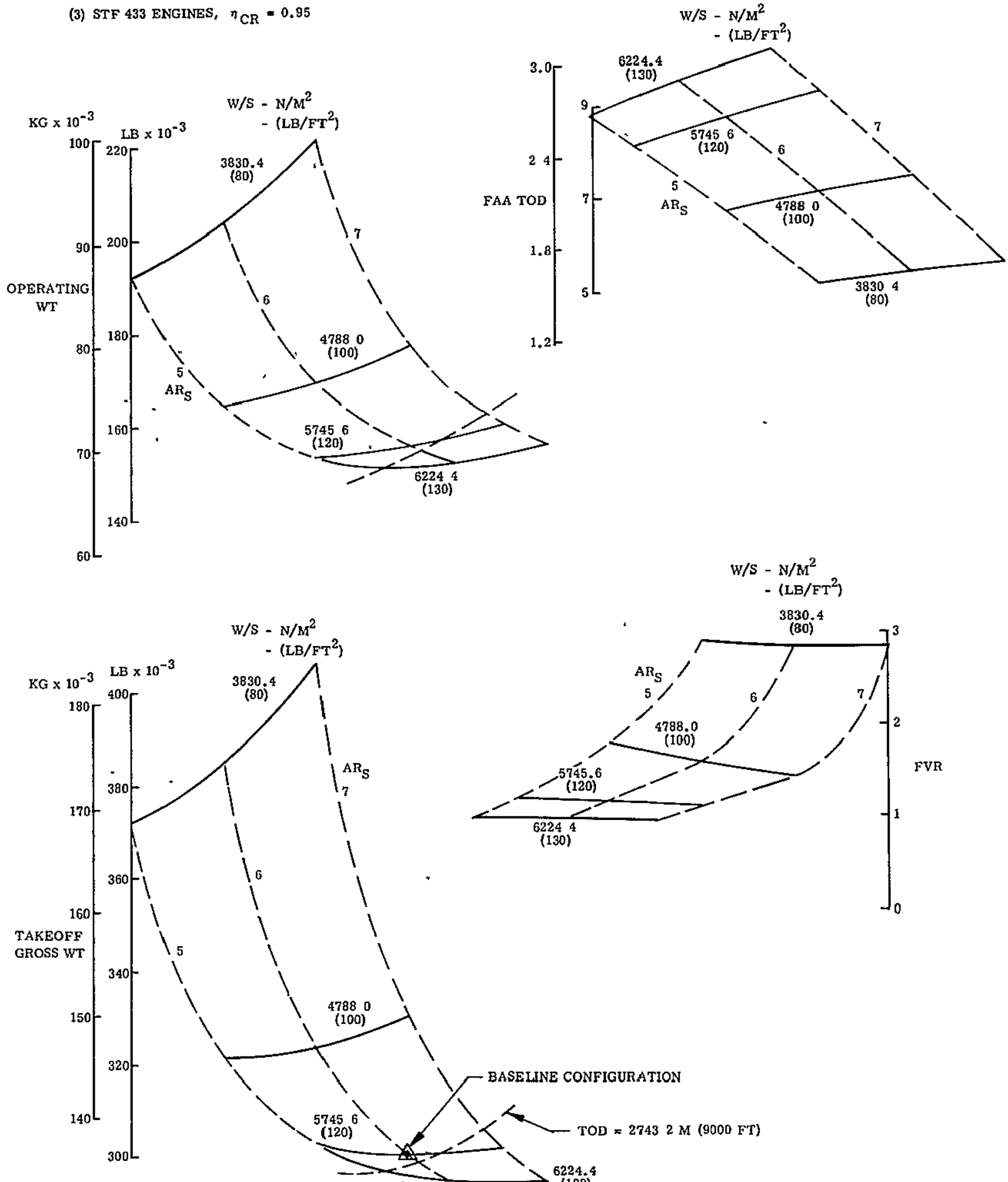

FIGURE A20 COMMERCIAL PASSENGER TRANSPORT CONFIGURATION, SIZING CHART
Λ = 0.785 RAD (45 DEG)

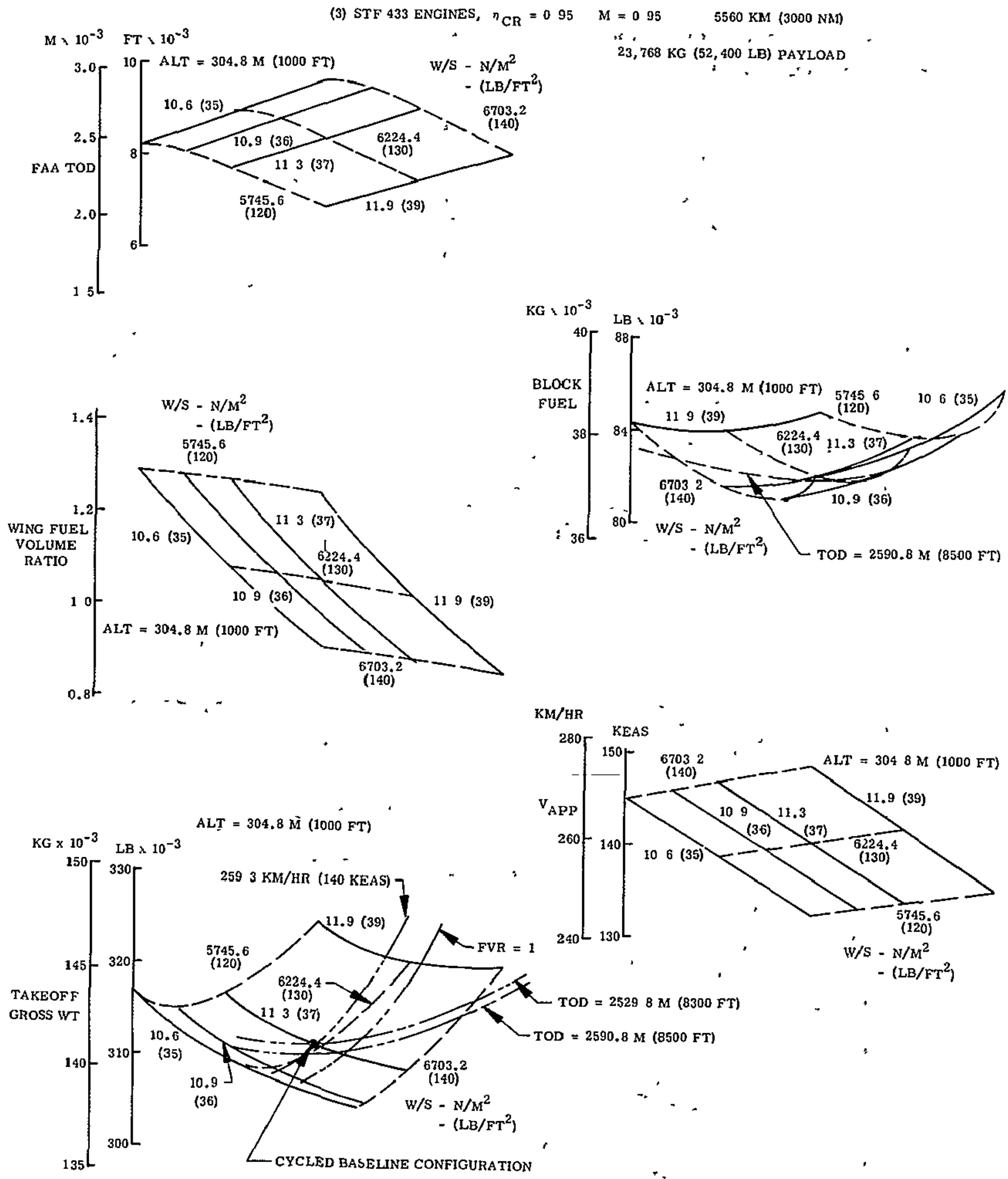

FIGURE A21 COMMERCIAL PASSENGER TRANSPORT - EFFECT OF DESIGN CRUISE ALTITUDE ON AIRPLANE SIZE

M = 0 95 18 PAX 7408 KM (4000 NM)

(2) STF 433 ENGINES, η_{CR} = 9 5 Λ = 7854 RADS (45 DEGS)

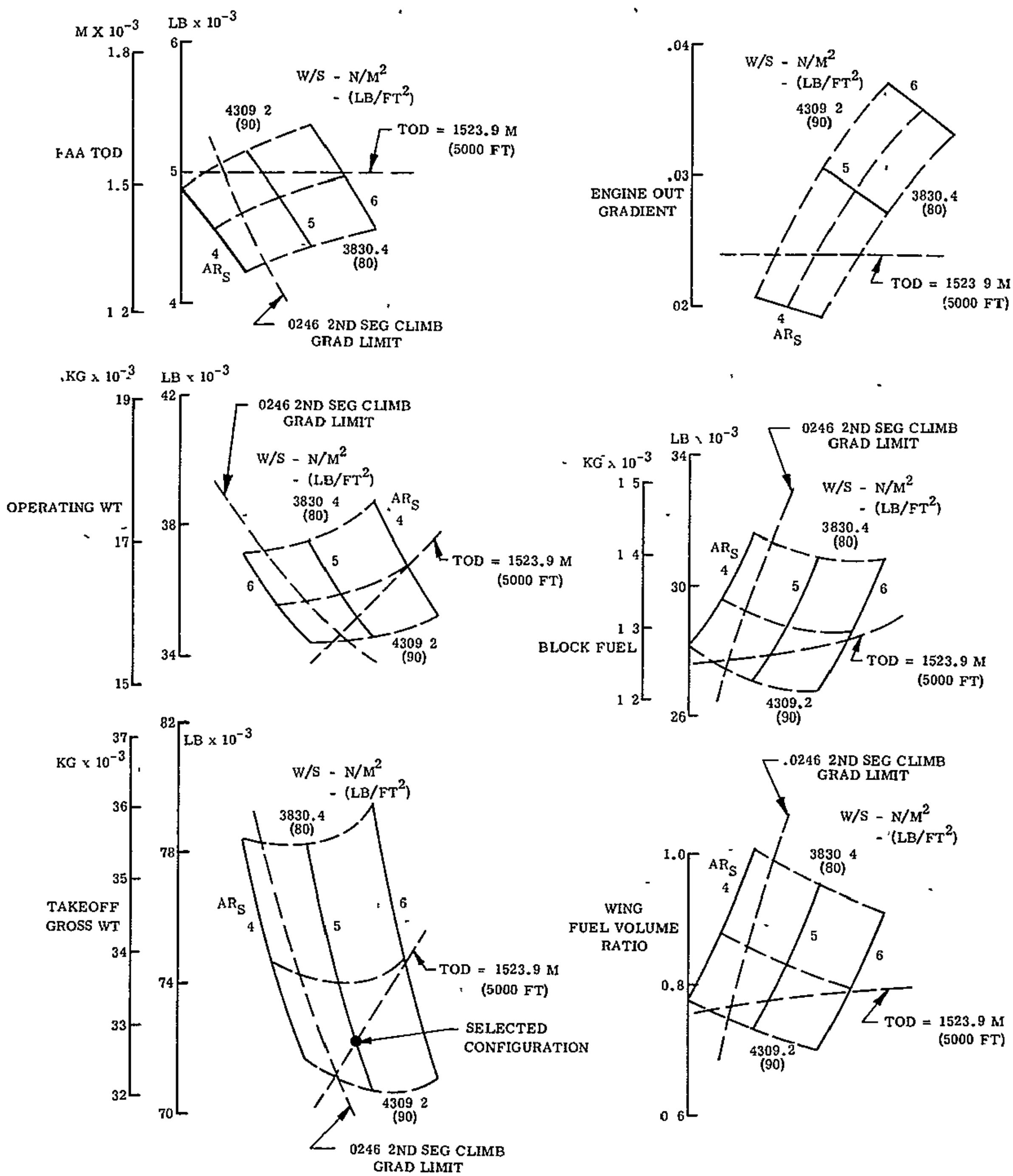

FIGURE A22 EXECUTIVE TRANSPORT CONFIGURATION SIZING CHART

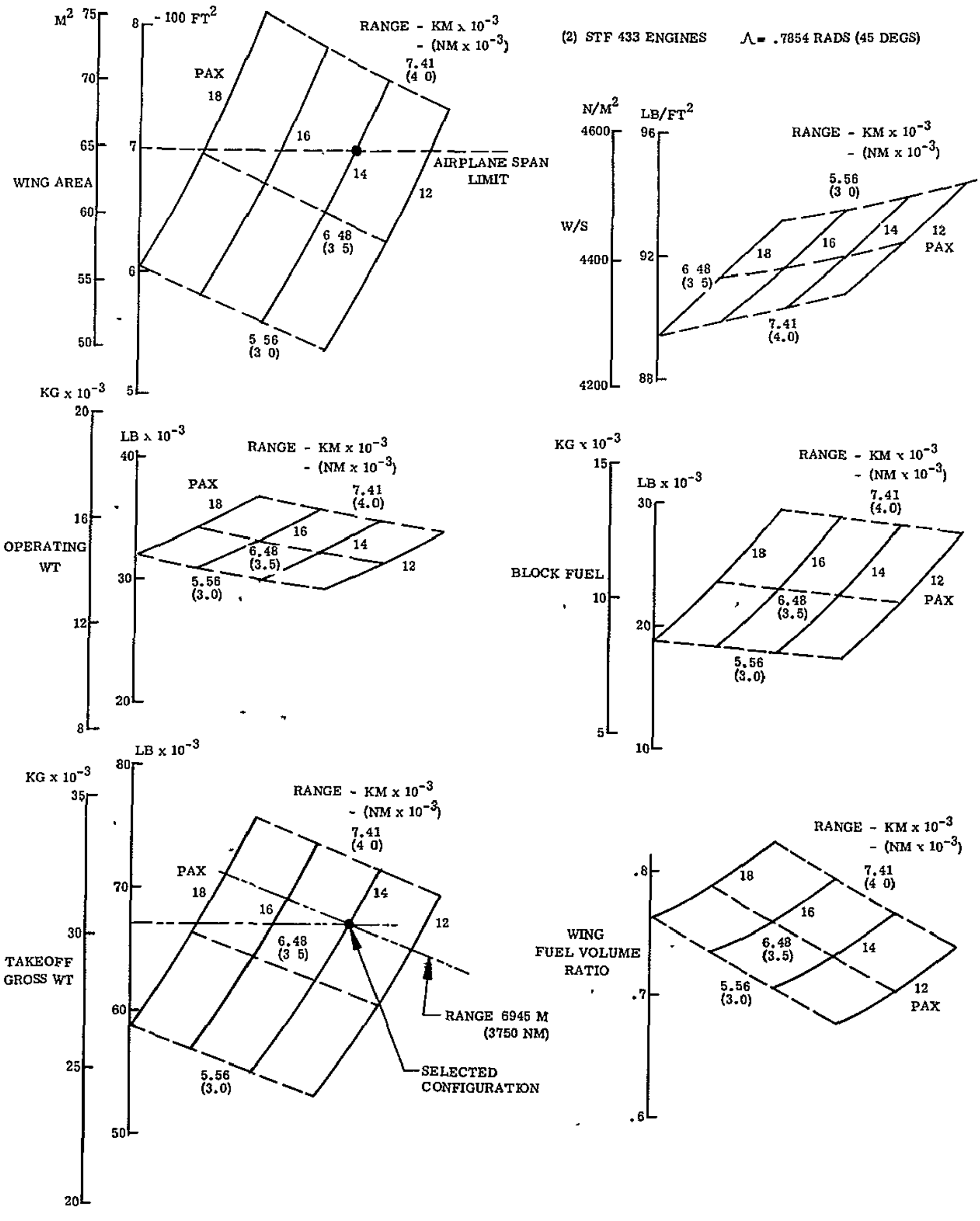

FIGURE A23 EXECUTIVE TRANSPORT - EFFECT OF PAYLOAD-RANGE ON AIRPLANE SIZE

M = 0.95 6482 KM (3500 NM) Λ = .6981 RADS (40 DEGS)
(4) STF 433 ENGINES, η_{CR} = 0.95 158,757 KG (350,000 LB) PAYLOAD

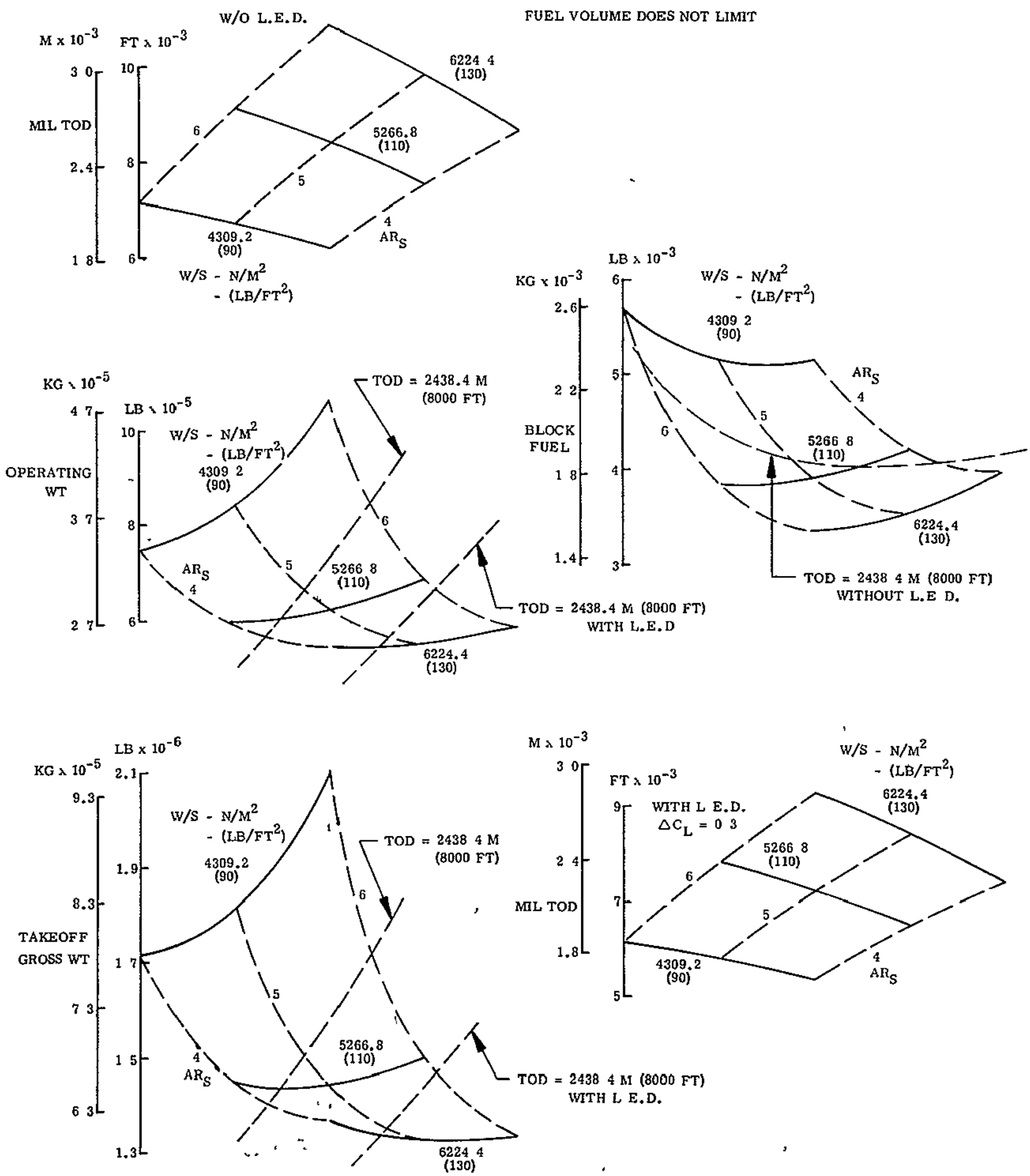

FIGURE A24 MILITARY CARGO TRANSPORT CONFIGURATION SIZING CHART
Λ = 0.70 RAD (40 DEG) - 4 ENGINES

6482 KM (3500 NM)

158,757 KG (350,000 LB) PAYLOAD

(4) STF 433 ENGINES, η_{CR} = 0 95

FUEL VOLUME DOES NOT SIZE THE WING

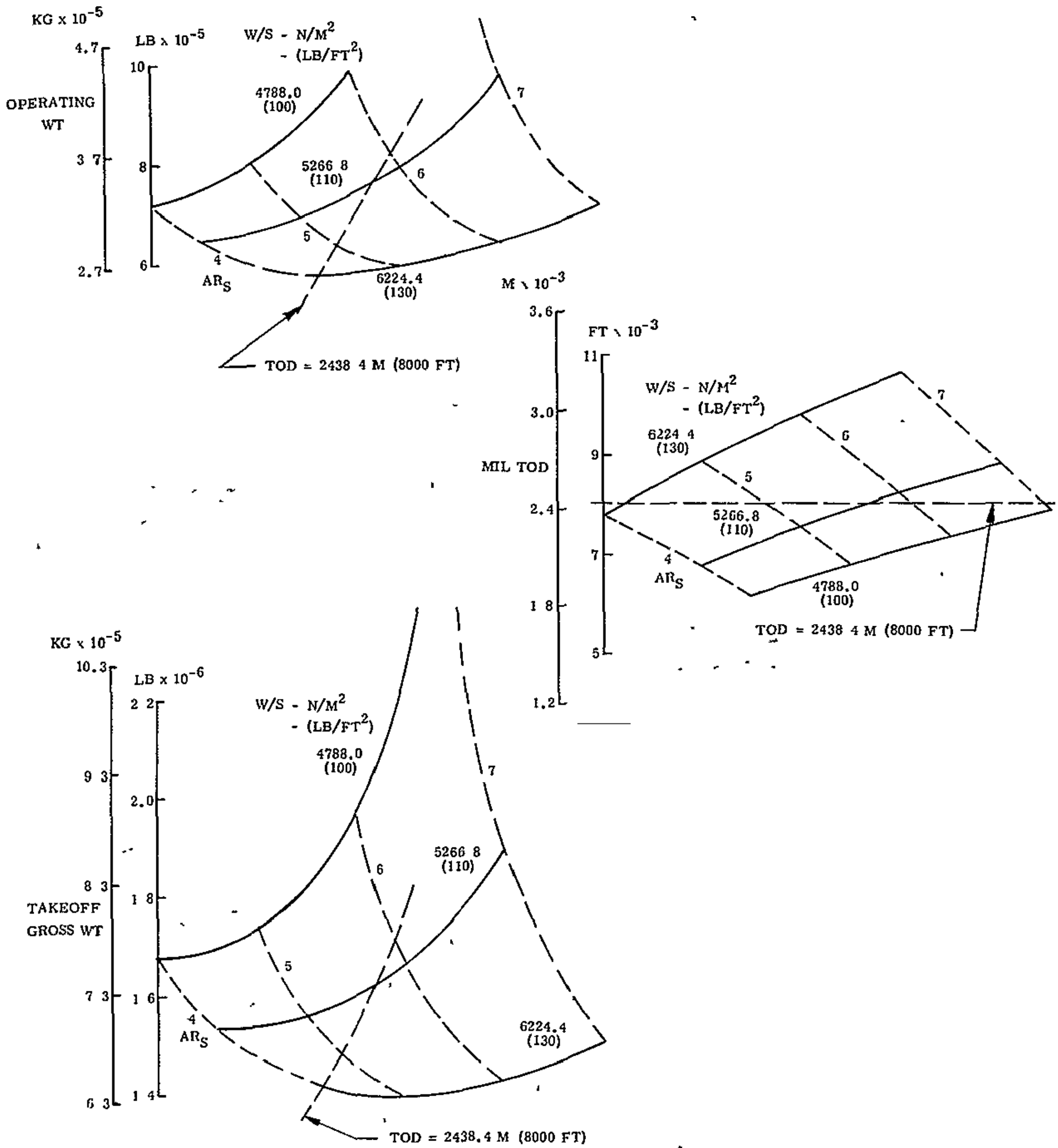

FIGURE A25 MILITARY CARGO TRANSPORT CONFIGURATION SIZING CHART
Λ = 0.785 RAD (45 DEG) - 4 ENGINES

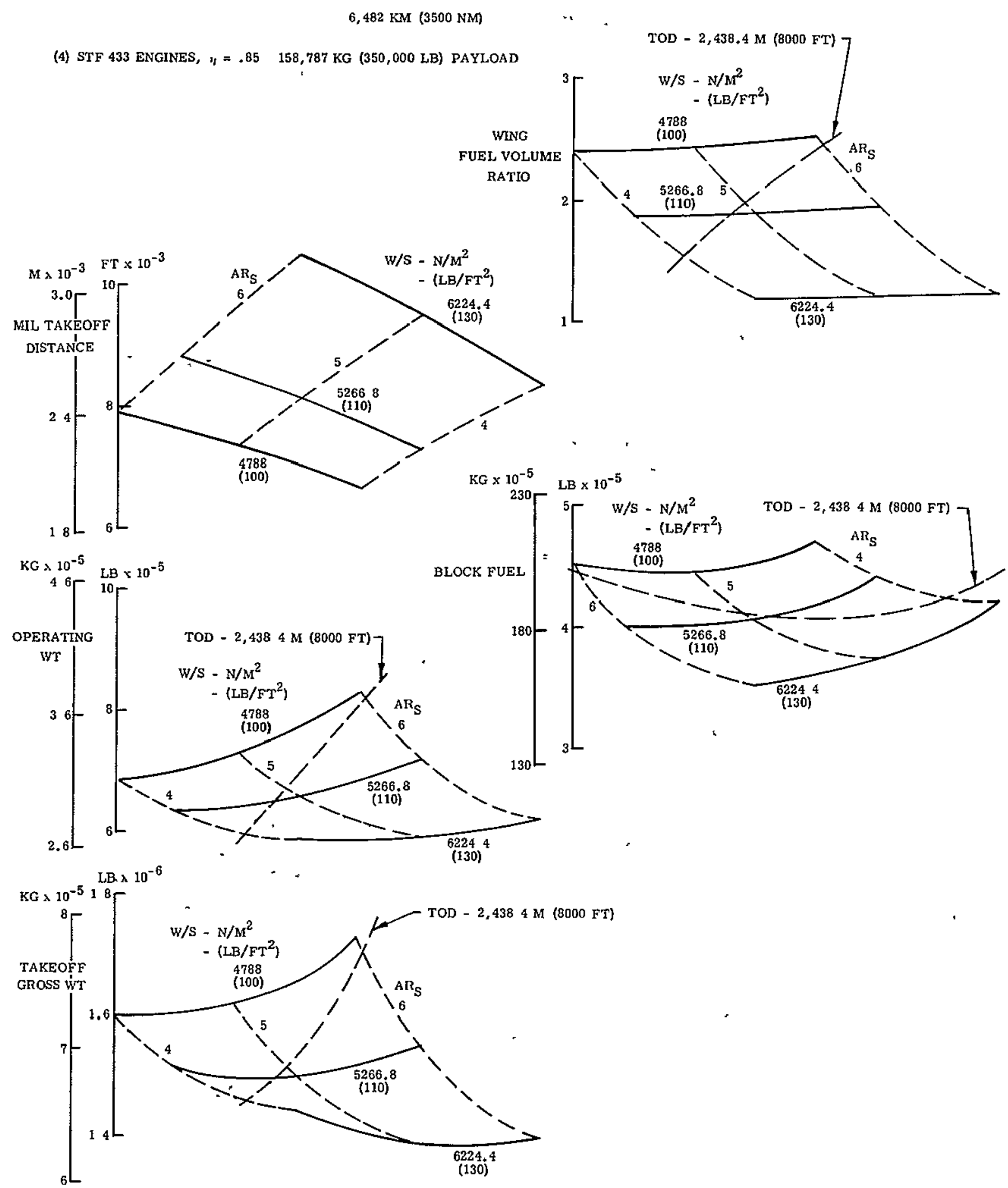

FIGURE A26 MILITARY CARGO TRANSPORT
CONFIGURATION SIZING CHART
Λ = 0.61 RAD (35 DEG) - 4 ENGINES -
0.85-CRUISE RATED THRUST

6482 KM (3500 NM)

158,757 KG (350,000 LB) PAYLOAD (4) STF 433 ENGINES, η_{CR} = 0 85

NO LEADING EDGE DEVICE FUEL VOLUME DOES NOT LIMIT

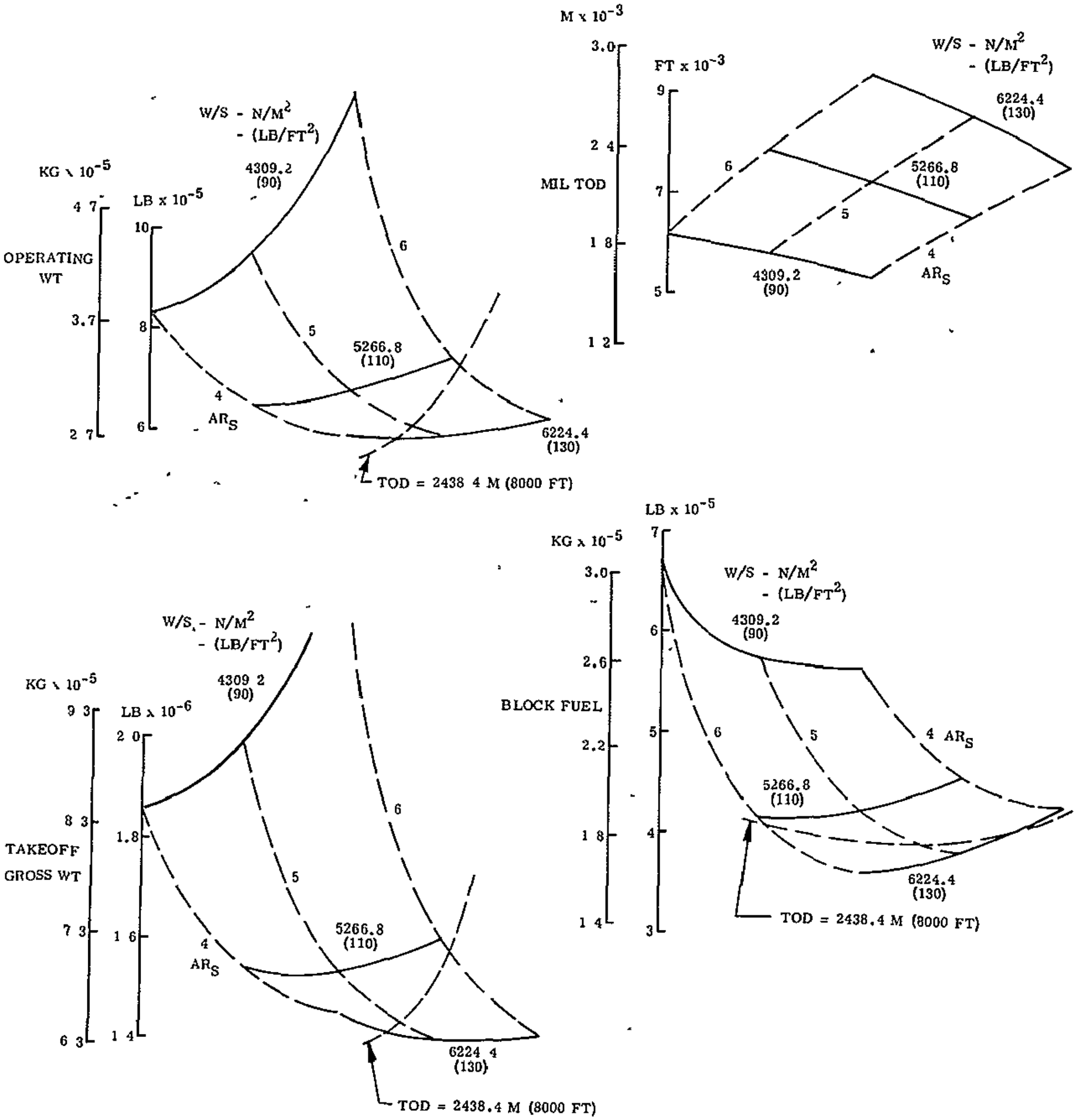

FIGURE A27 MILITARY CARGO TRANSPORT CONFIGURATION SIZING CHART
Λ = 0.70 RAD (40 DEG) - 4 ENGINES
0.85 CRUISE RATED THRUST

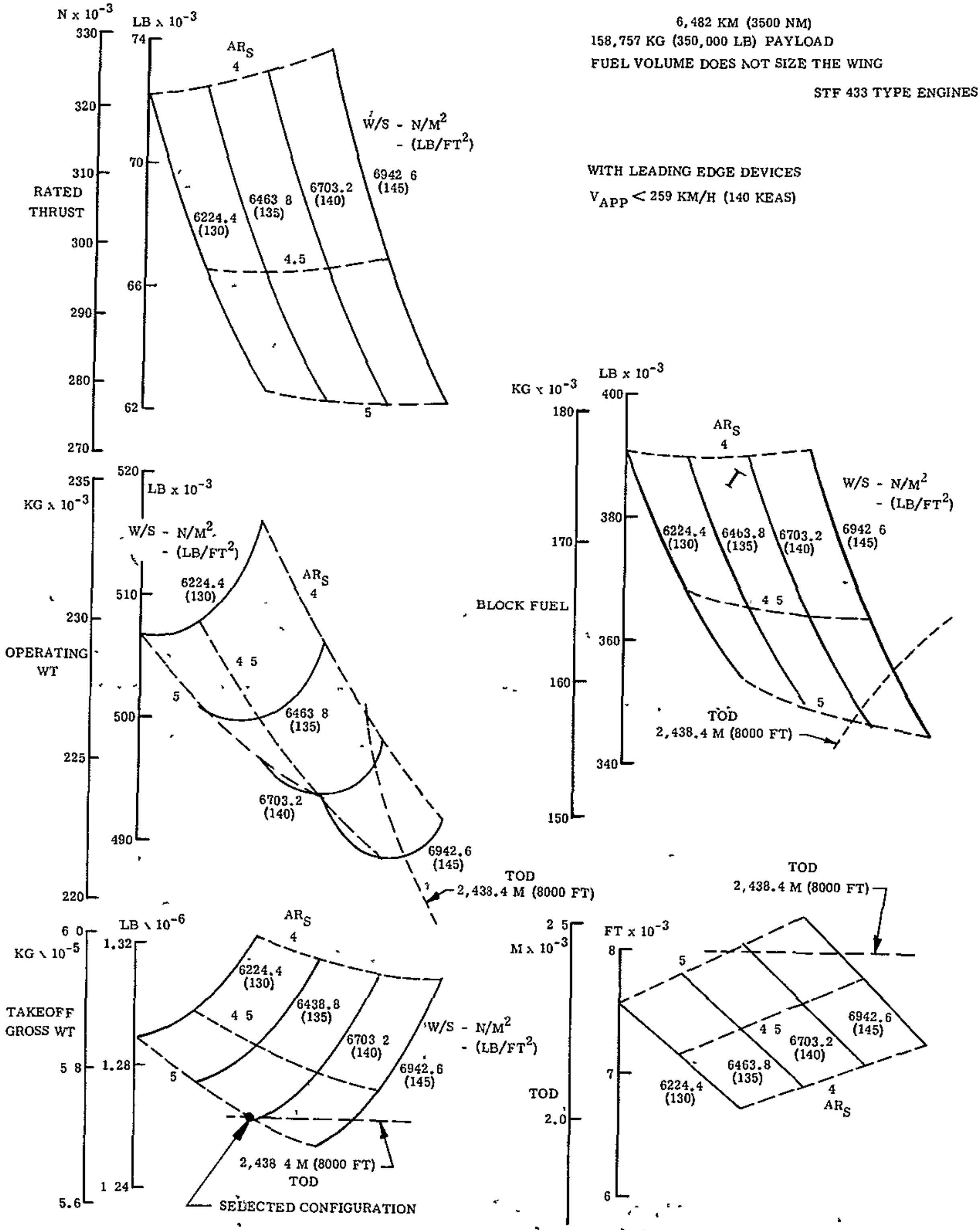

FIGURE A28 MILITARY CARGO TRANSPORT
CYCLED BASELINE SIZING CHART
Λ = 0.70 RAD (40 DEG) - 6 ENGINES
0.95 CRUISE RATED THRUST

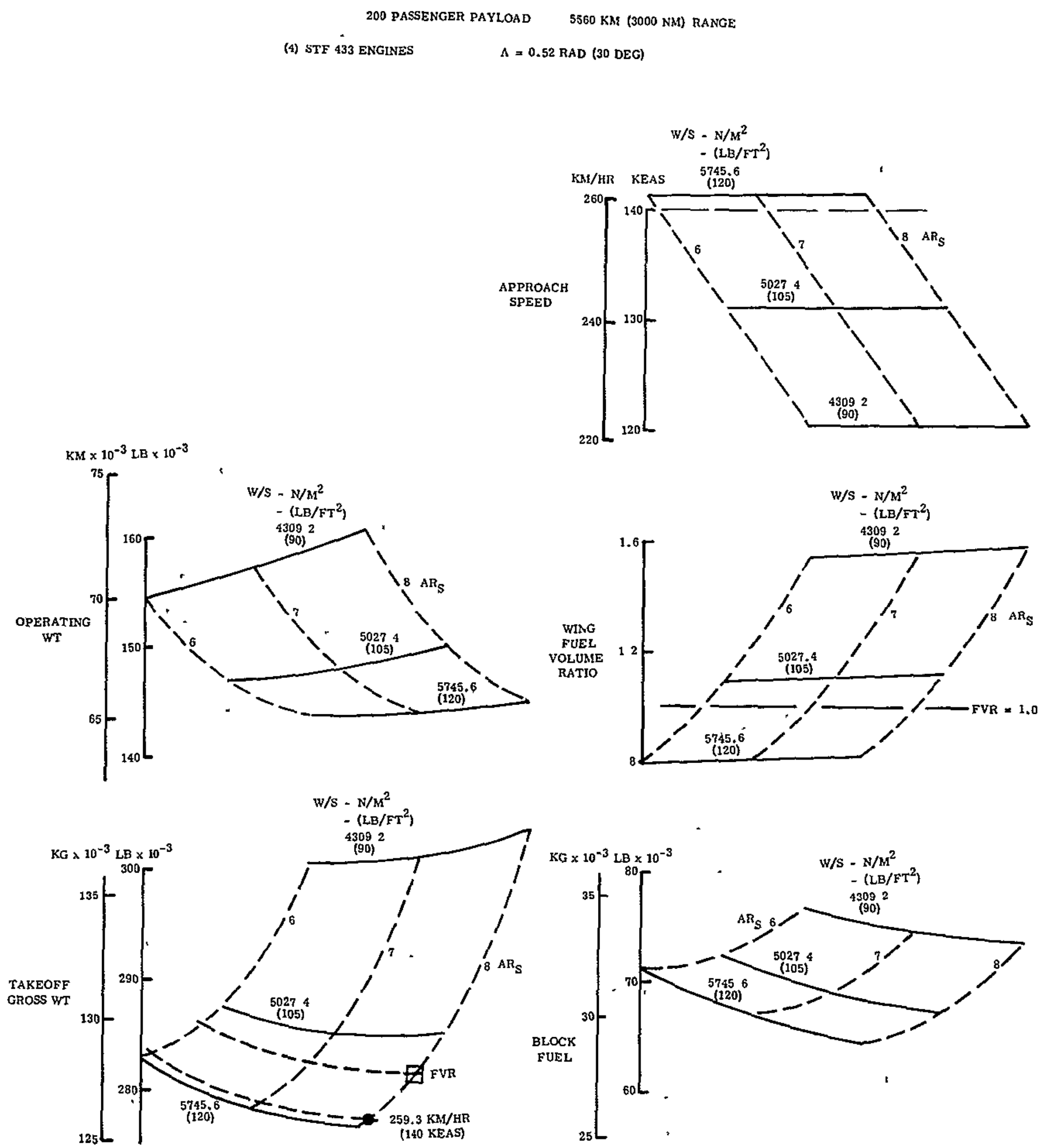

FIGURE A29 COMMERCIAL PASSENGER TRANSPORT CONVENTIONAL CONFIGURATION SIZING CHART - CRUISE AT M = 0.85

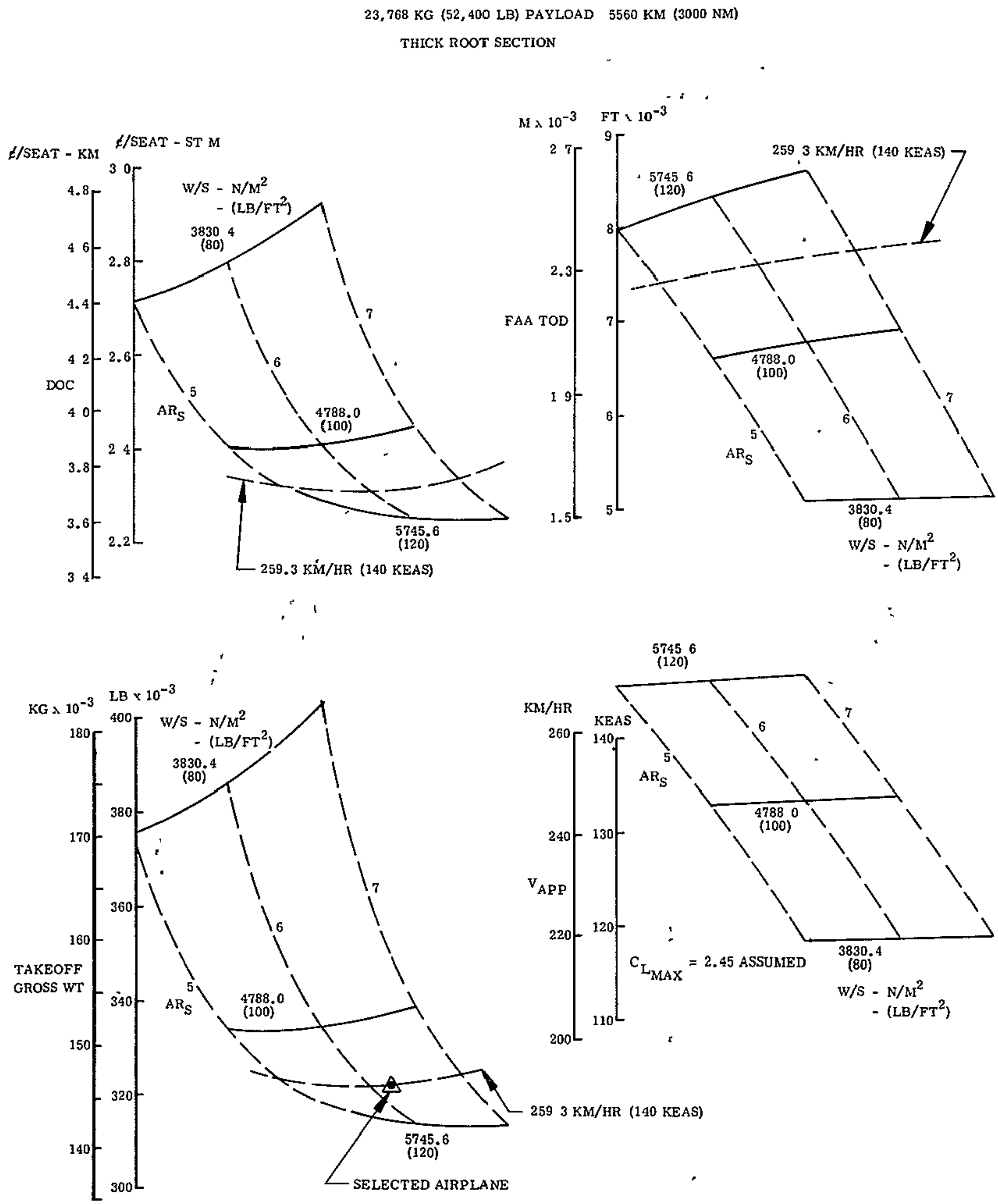

FIGURE A30 COMMERCIAL PASSENGER TRANSPORT CONVENTIONAL CONFIGURATION SIZING CHART - M = 0.95

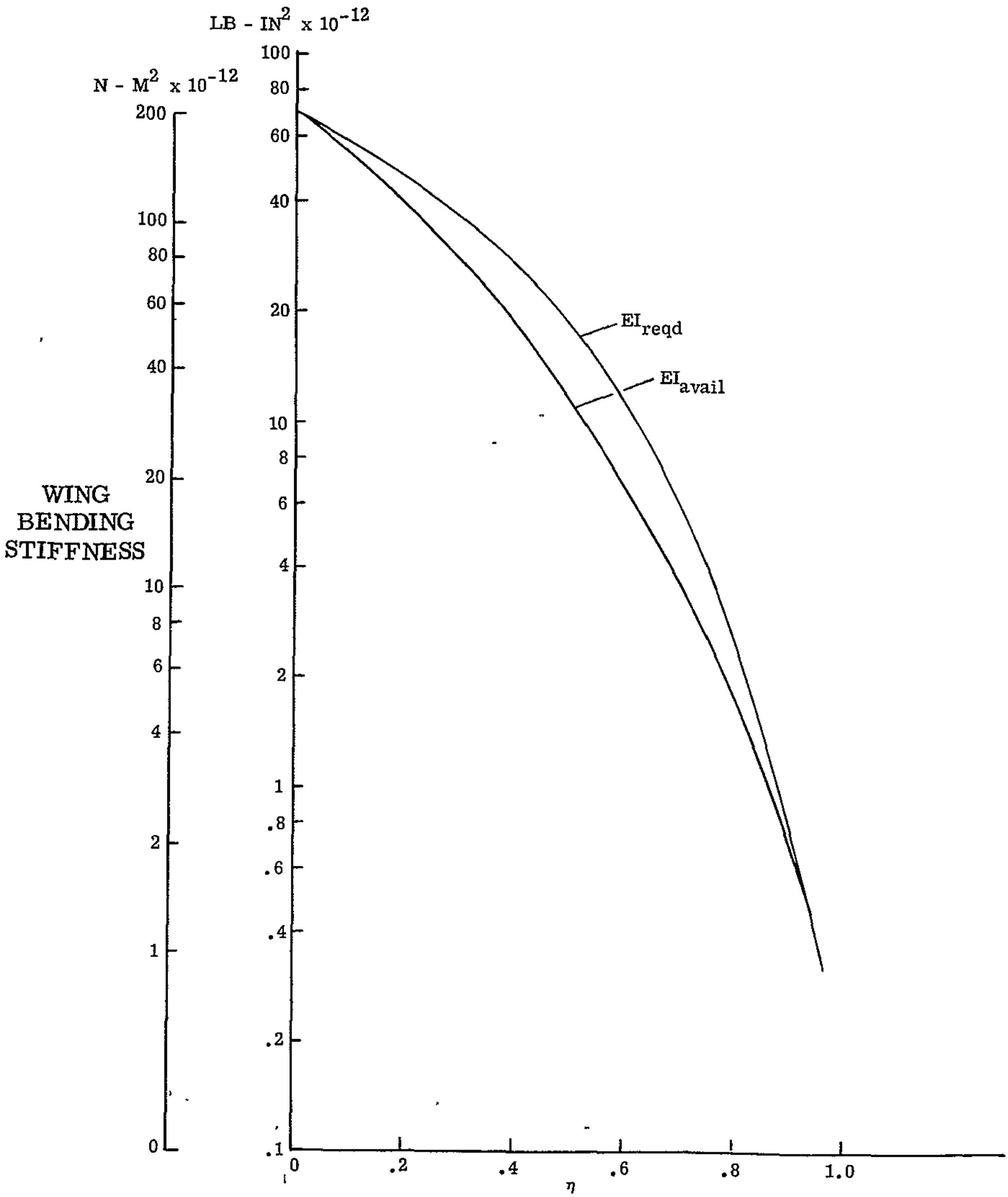

FIGURE A31 WING BENDING STIFFNESS

C-3

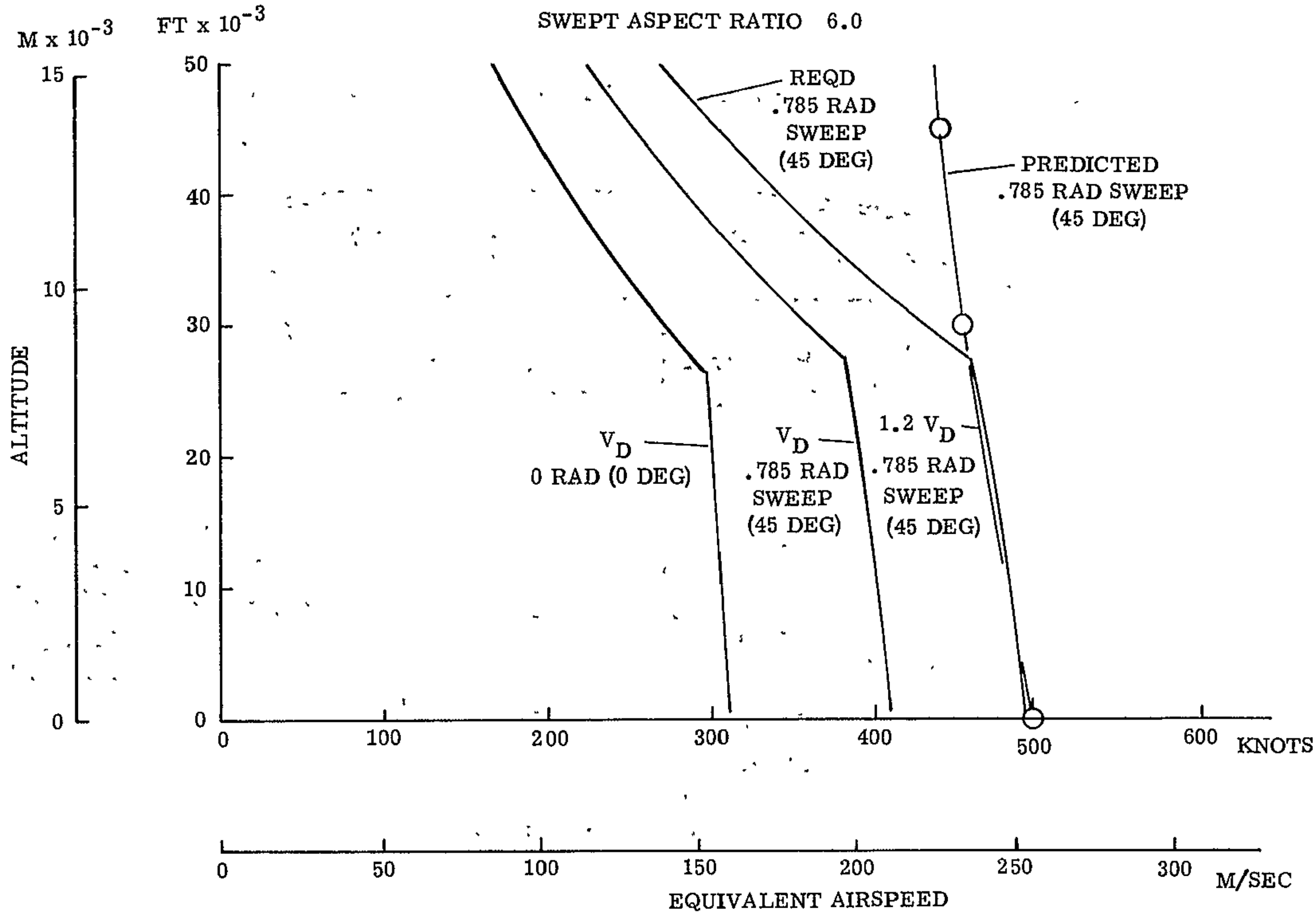

FIGURE A32 FLUTTER BOUNDARIES

DIMENSIONAL DATA 177,929 N (40,000 LB)

ENGINE	CM	(IN)
OVERALL LENGTH	355.6	(140.0)
FAN CASE LENGTH	152.4	(60.0)
FAN CASE DIA	218.4	(86.0)
FAN EXIT INNER DIA	132.0	(52.0)
FAN EXIT OUTER DIA	203.2	(80.0)
LOW TURBINE OUTER DIA	127.0	(50.0)
FAN NOZZLE OUTER DIA	182.2	(71.7)
FAN/PRIMARY NOZZLE INNER/OUTER DIA	127.0	(50.0)
PRIMARY PLUG MAX DIA	93.3	(36.7)

NACELLE	CM	(IN)
OVERALL LENGTH	714.0	(281.1)
INLET LENGTH	285.5	(112.4)
FAN DUCT LENGTH	276.1	(108.7)
PRIMARY DUCT LENGTH	71.1	(28.0)
PRIMARY PLUG LENGTH	218.4	(86.0)
MAXIMUM DIA	238.0	(93.7)
HIGHLIGHT DIA	205.2	(80.8)
INLET THROAT DIA	194.1	(76.4)
INLET RAKE ANGLE	.2793 RAD	(16 DEG)

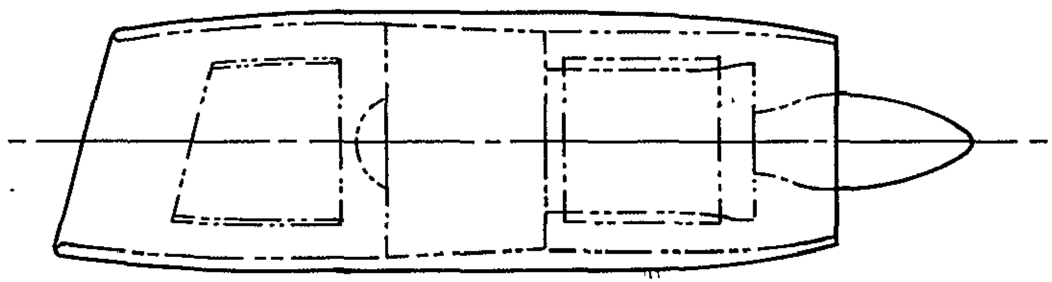

FIGURE A33 PRATT AND WHITNEY STF 433 NACELLE PROPORTIONS

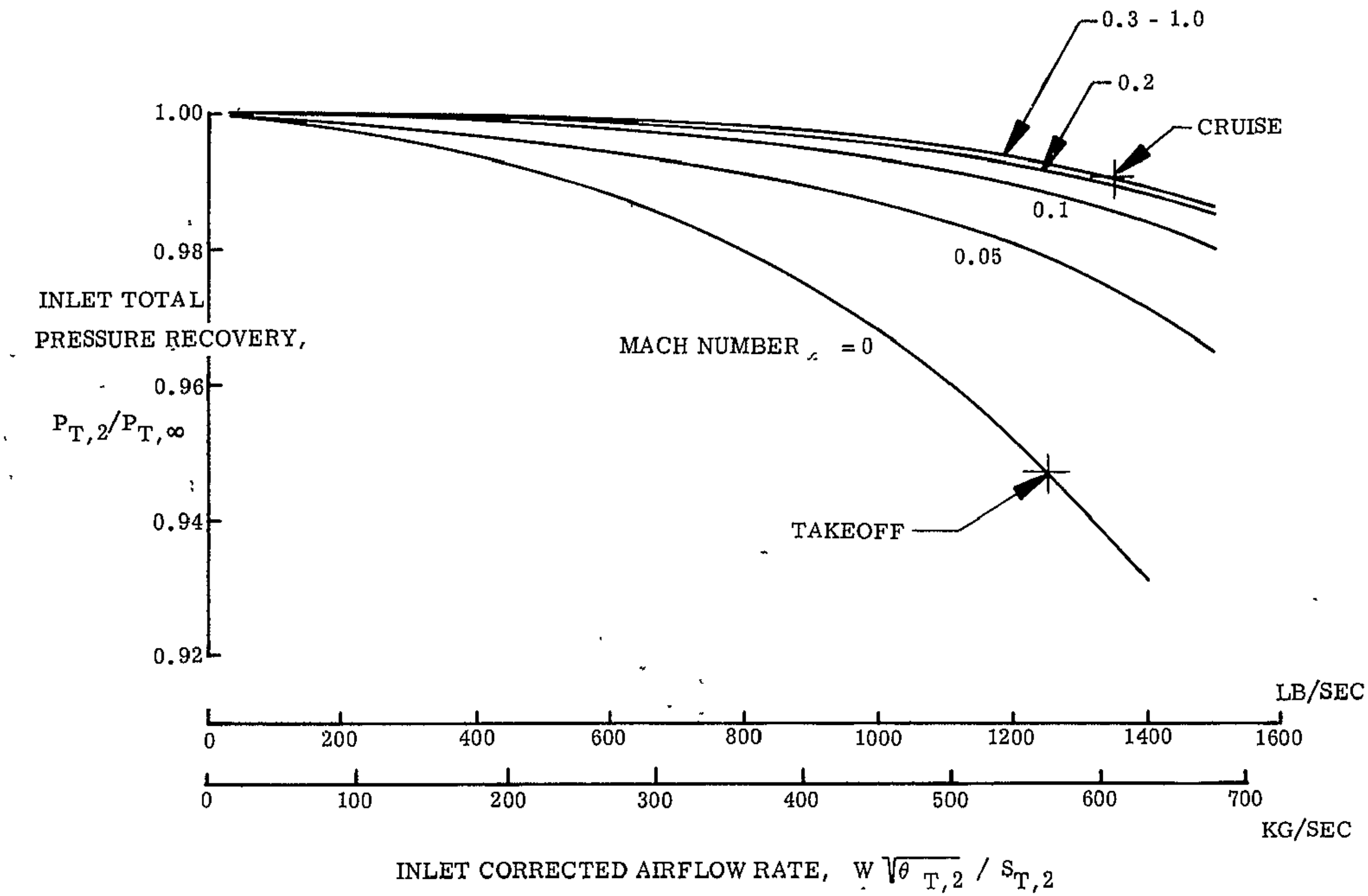

FIGURE A34 INLET TOTAL PRESSURE RECOVERY

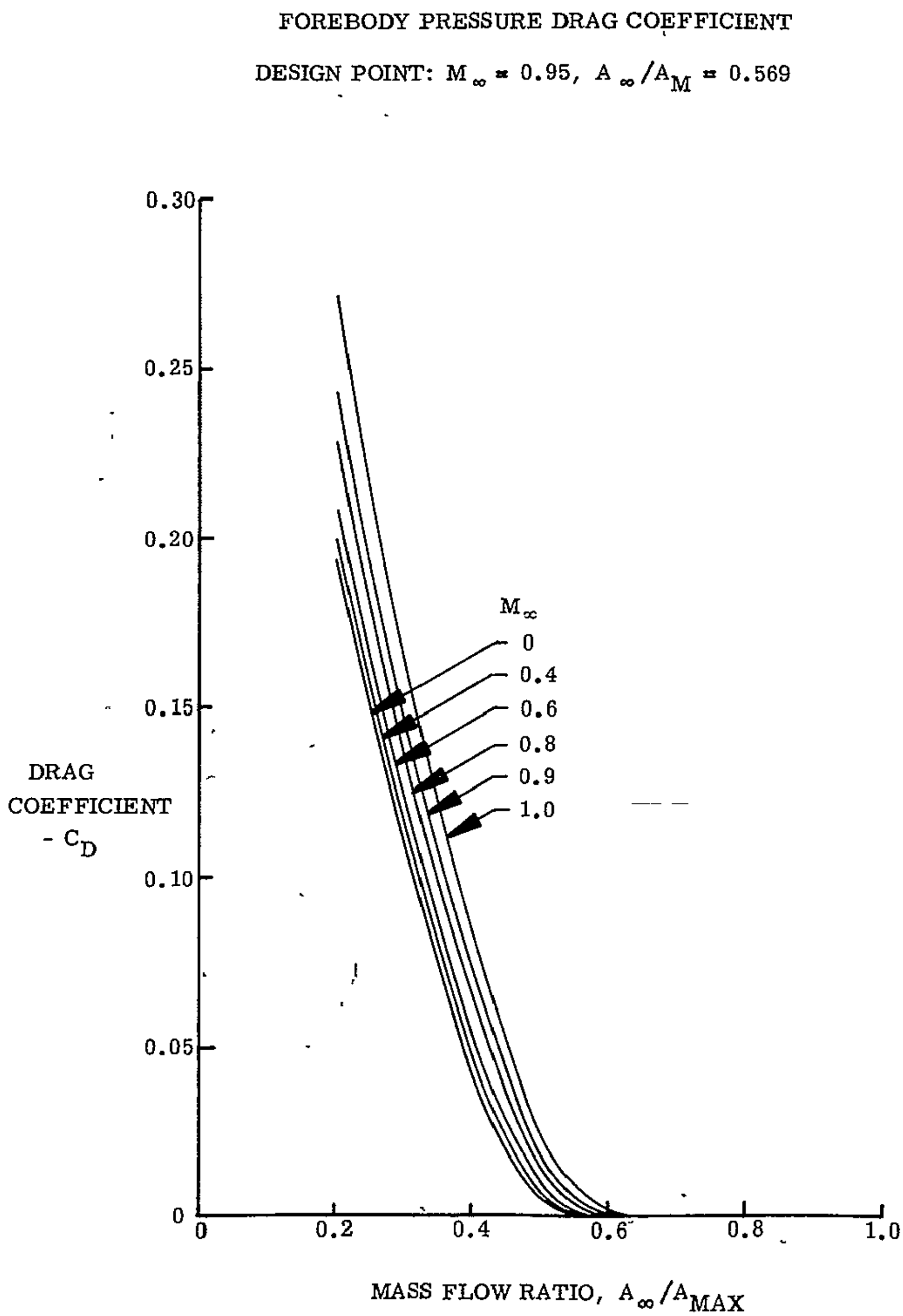

FIGURE A35 ADDITIVE DRAG

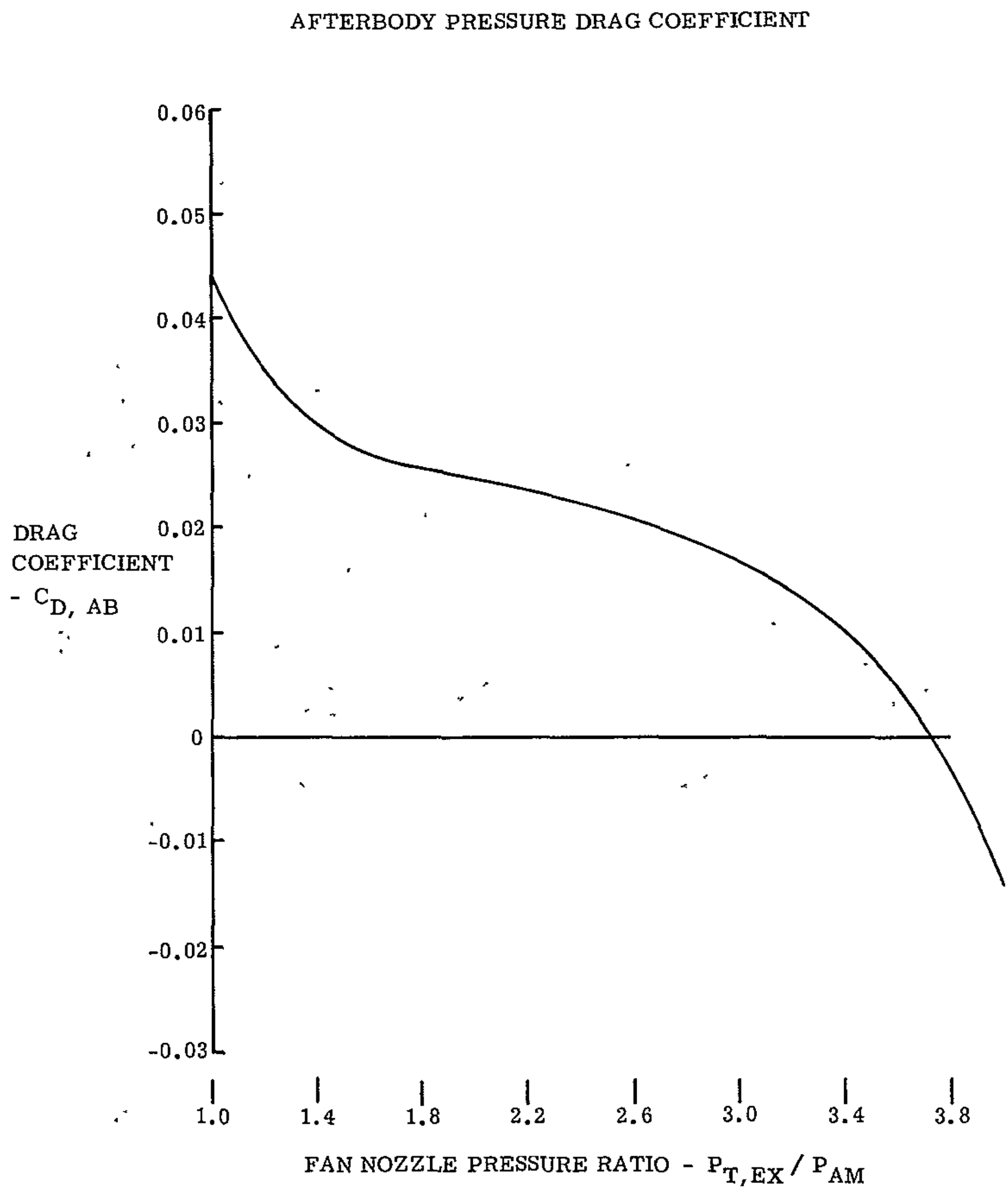

FIGURE A36 AFTERBODY PRESSURE DRAG

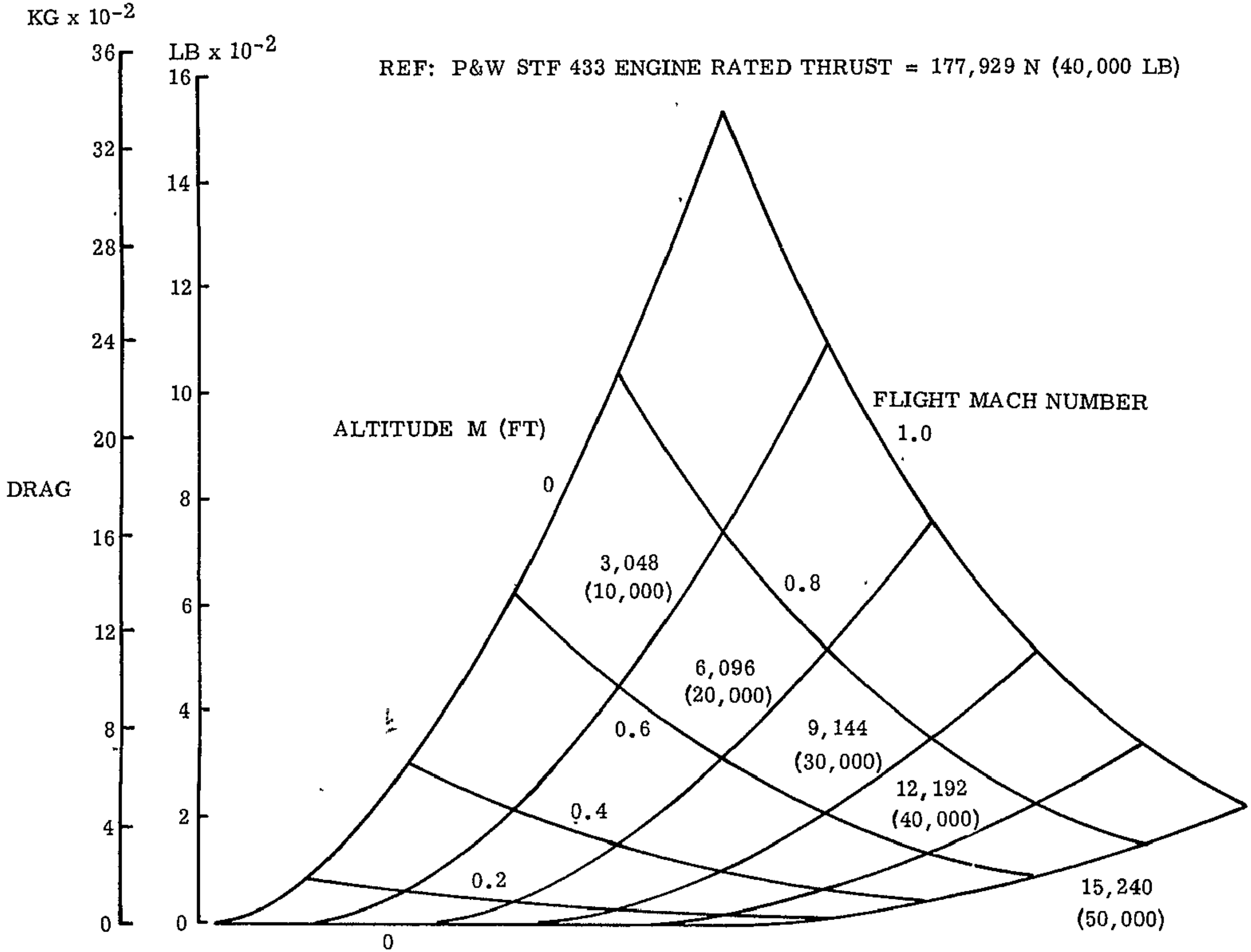

FIGURE A37 NACELLE SKIN FRICTION DRAG

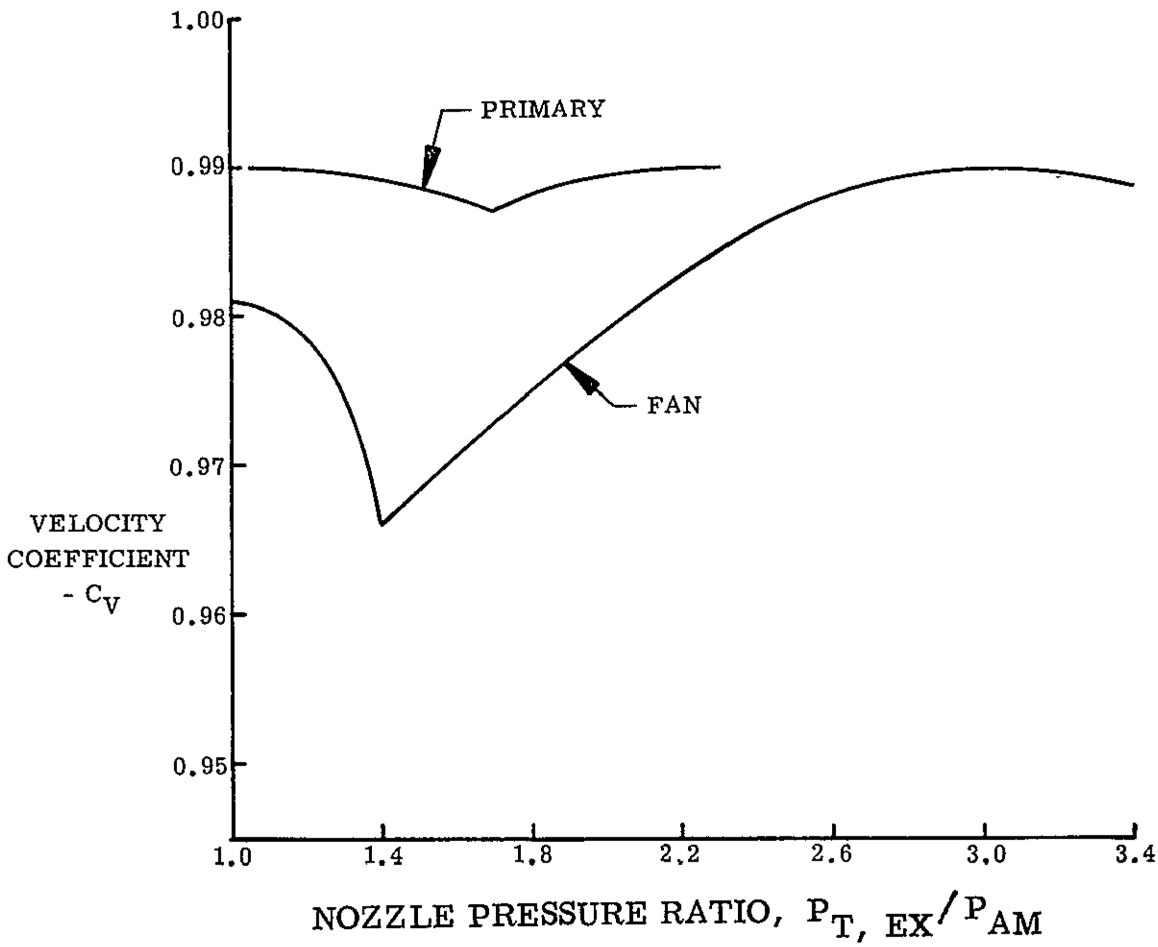

FIGURE A38 NOZZLE VELOCITY COEFFICIENTS

NACELLE COOLING DRAG

REF: P&W STF 433 ENGINE, RATED THRUST 177,929 N (40,000 LB)

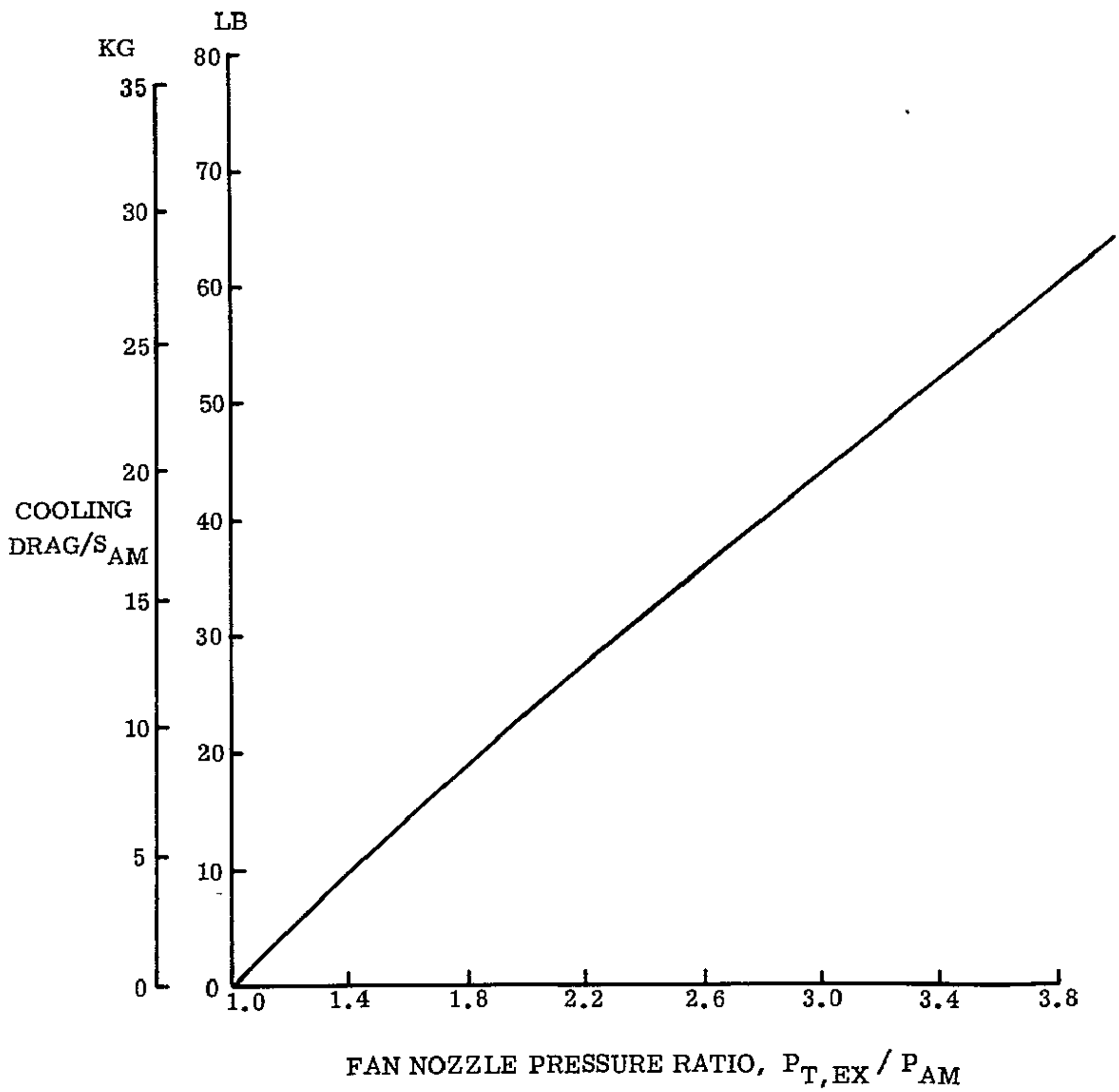

FIGURE A39 COOLING DRAG

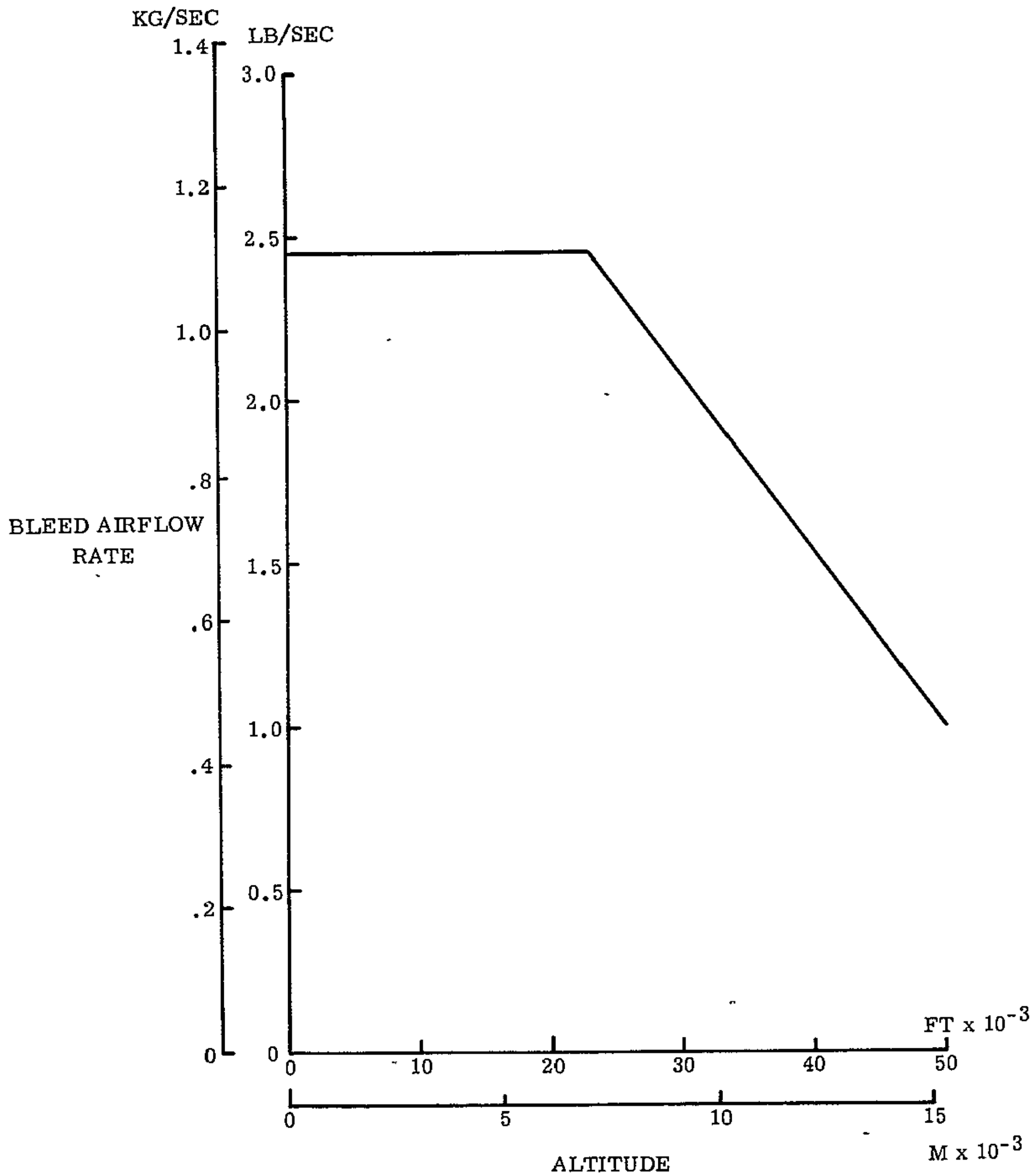

FIGURE A40 BLEED AIRFLOW

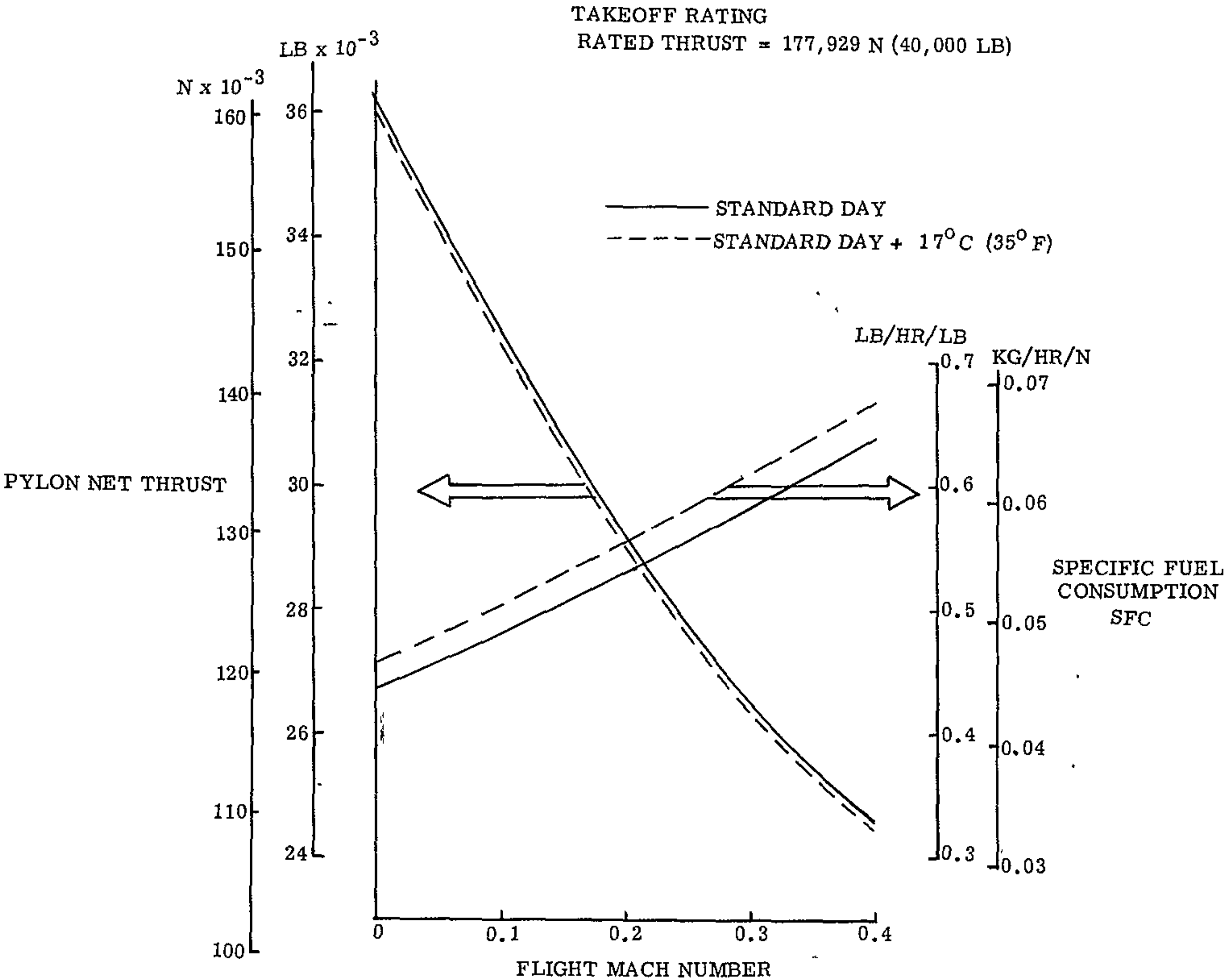

FIGURE A41 ENGINE INSTALLED PERFORMANCE - SEA LEVEL

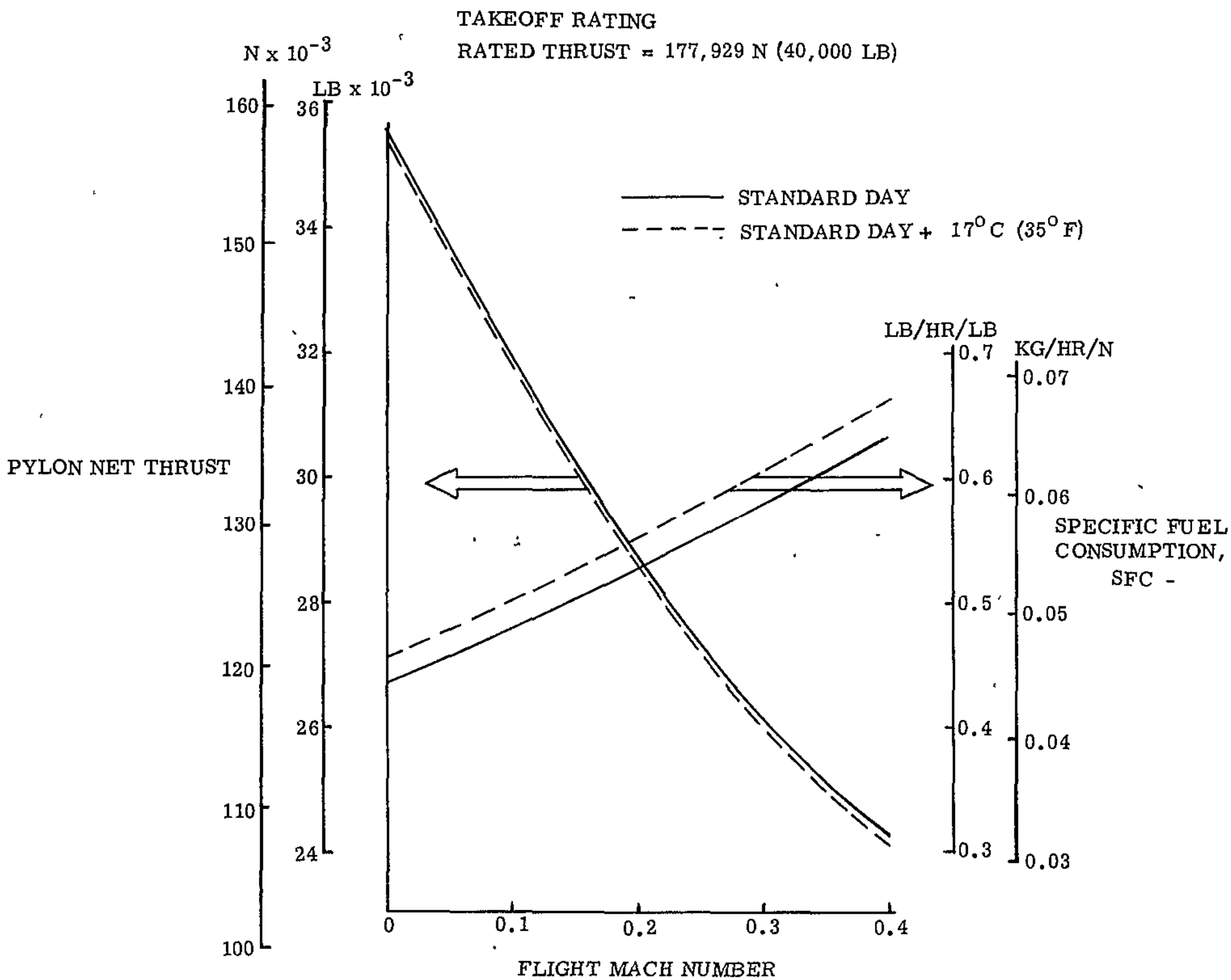

FIGURE A42 ENGINE INSTALLED PERFORMANCE - 305 M (1000 FT)

STANDARD DAY
RATED THRUST = 177,929 N (40,000 LB)

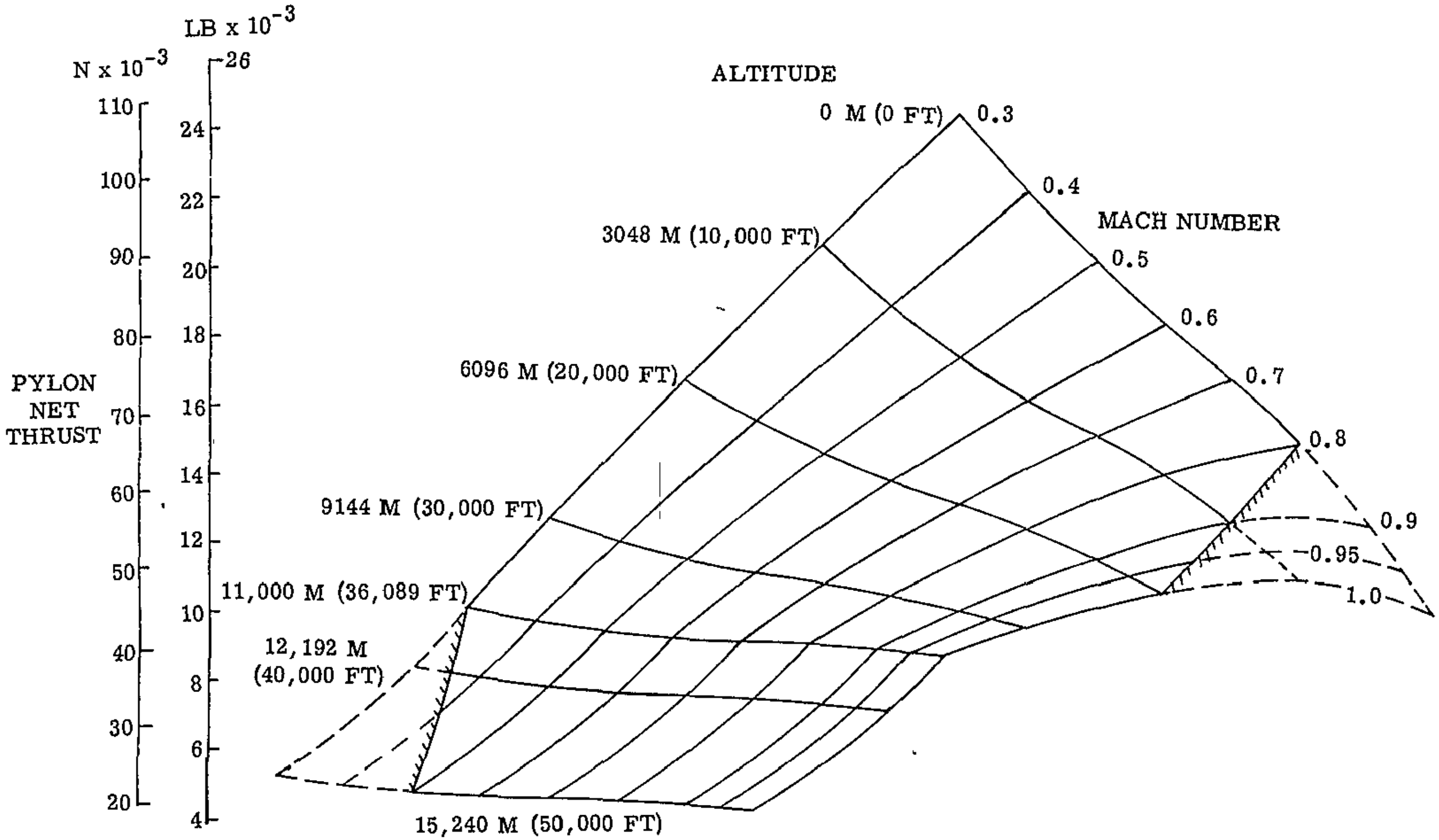

FIGURE A43 ENGINE MAX CRUISE RATED NET PYLON THRUST

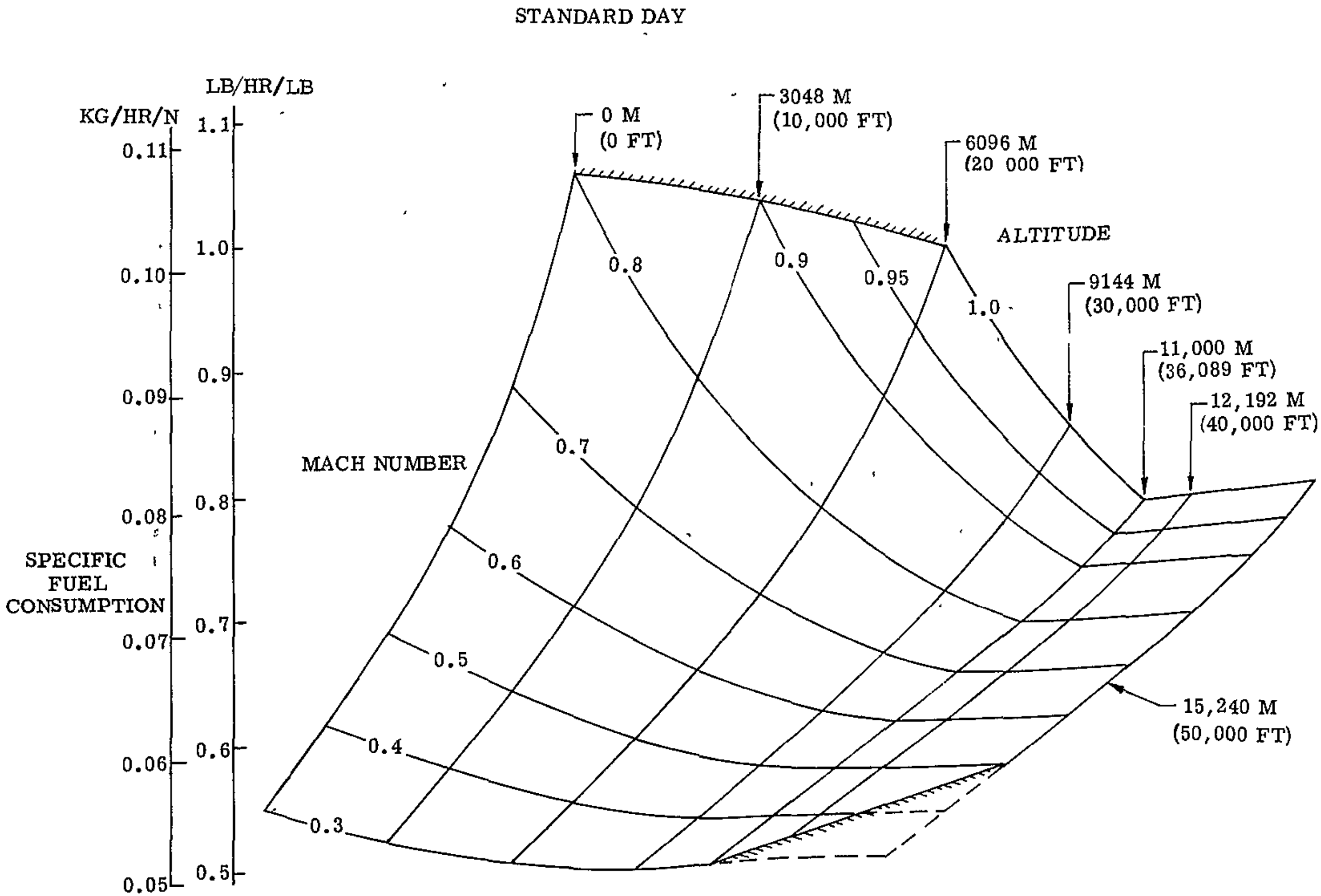

FIGURE A44 ENGINE MAX CRUISE RATED NET PYLON SFC

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P&W STF 433 ENGINE

STANDARD DAY

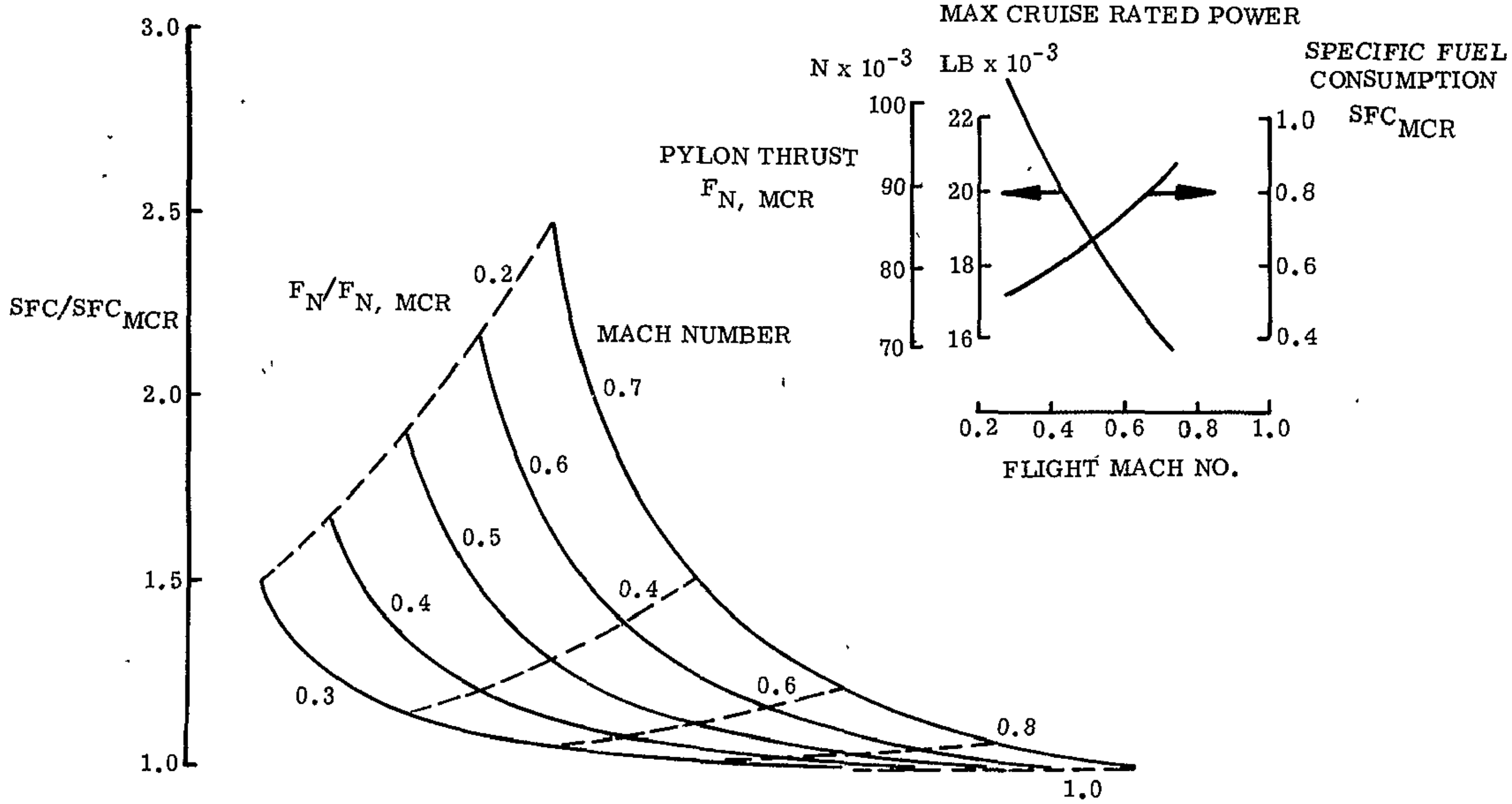

FIGURE A45 PART POWER CRUISE SFC - 1524 M (5000 FT)

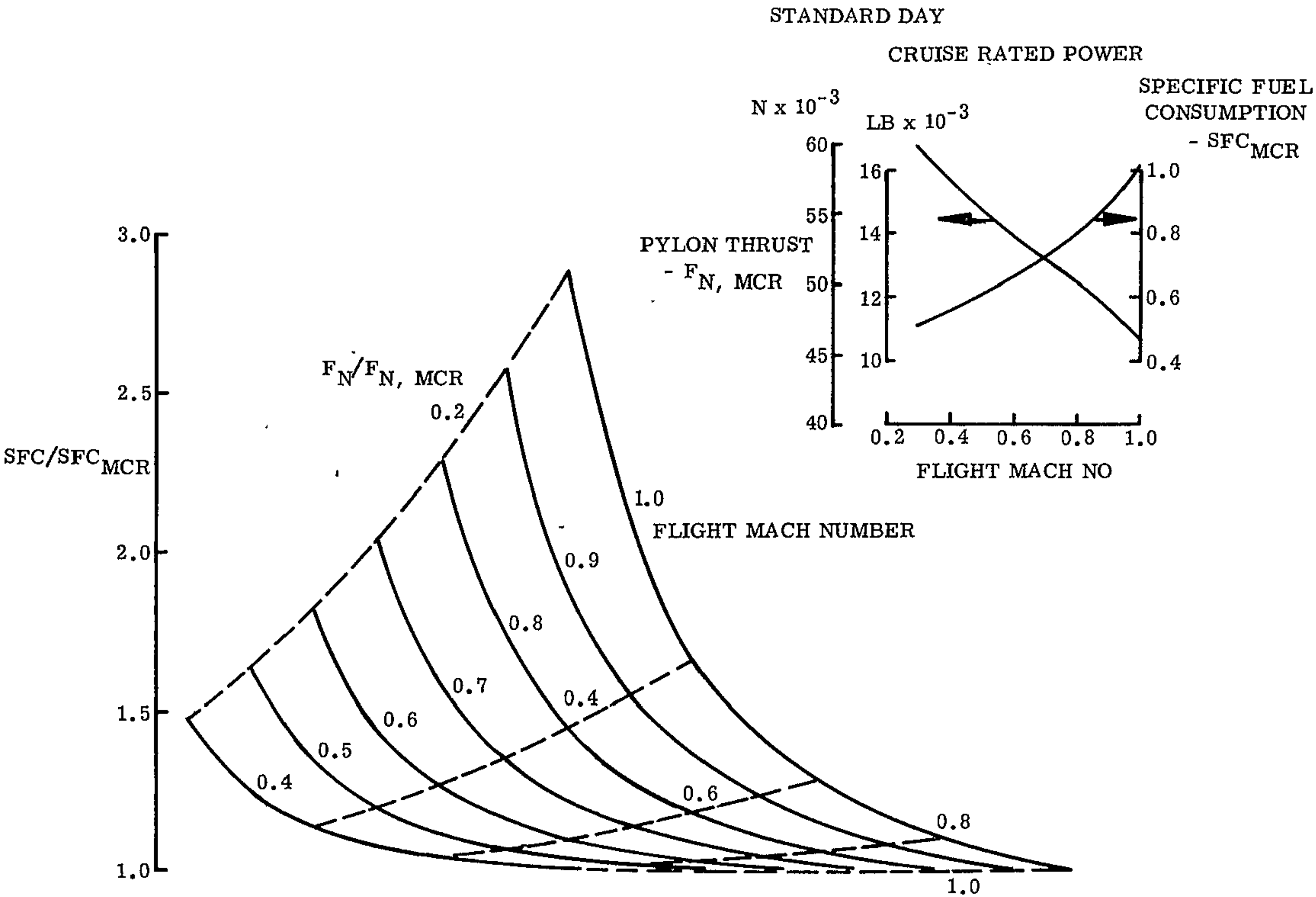

FIGURE A46 PART POWER CRUISE SFC - 6096 M (20,000 FT)

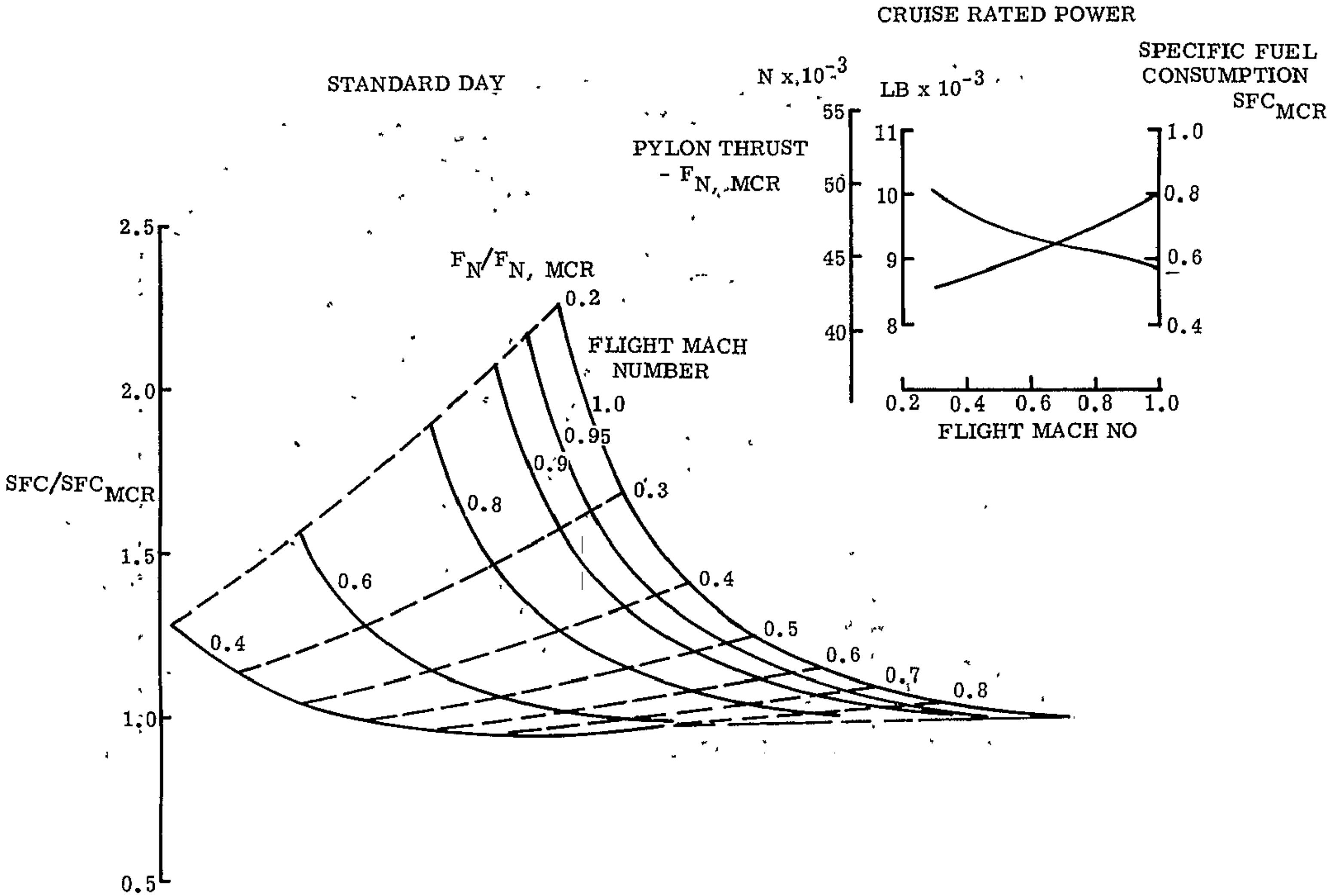

FIGURE A47 PART POWER CRUISE SFC - 11,000 M (36,089 FT)

- RATED THRUST SCALING RANGE - 44,482 TO 289,134 N (10,000 TO 65,000 LBF)

- $$\frac{\text{WEIGHT}}{\text{WEIGHT}_{(\text{REF.})}} = \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right]^{1.15}$$

- $$\frac{\text{LENGTH}}{\text{LENGTH}_{(\text{REF.})}} = 0.3901 + 0.7722 \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right] - 0.1623 \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right]^{2}$$

- $$\frac{\text{DIAMETER}}{\text{DIAMETER}_{(\text{REF.})}} = 0.3629 + 0.7599 \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right] - 0.1228 \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right]^{2}$$

- $$\left[\frac{\text{SFC}}{\text{SFC}_{(\text{REF.})}}\right] = 1.1999 - 0.7949 \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right] + 0.7900 \left[\frac{\text{THRUST}}{\text{THRUST}_{(\text{REF.})}}\right]^{2}$$
  (THRUST < 88,964 N (20,000 LBF))

- $$\left[\frac{\text{SFC}}{\text{SFC}_{(\text{REF.})}}\right] = 1.0$$
  (THRUST ≥ 88,964 N (20,000 LBF))

WHERE:

$\text{THRUST}_{(\text{REF.})}$ = 177,928 N (40,000 LBF)

$\text{WEIGHT}_{(\text{REF.})}$ = 3447 KG (7600 LBM)

$\text{LENGTH}_{(\text{REF.})}$ = 3.6 M (140 IN)

$\text{DIAMETER}_{(\text{REF.})}$ = 2.2 M (86 IN)

$\text{SFC}_{(\text{REF.})}$ = SFC FOR ENGINE RATED THRUST 88,694 N (20,000 LBF) AT SPECIFIC ALTITUDE, AIRSPEED, POWER SETTING, ETC.

FIGURE A48 PRATT AND WHITNEY STF-433 ENGINE SCALING DATA

- (NACELLE DIAMETER)$_{MAX}$/(ENGINE DIAMETER)$_{MAX}$ = 1.09
- NACELLE LENGTH/(ENGINE DIAMETER)$_{MAX}$ = 3.27
- INLET LENGTH/(ENGINE DIAMETER)$_{MAX}$ = 1.30
- PYLON LENGTH/ (ENGINE DIAMETER)$_{MAX}$ = 3.27
- PYLON HEIGHT/(ENGINE DIAMETER)$_{MAX}$ = 0.22
- THRUST REVERSER WEIGHT/ENGINE WEIGHT = 0.2

FIGURE A49 NACELLE SCALING DATA

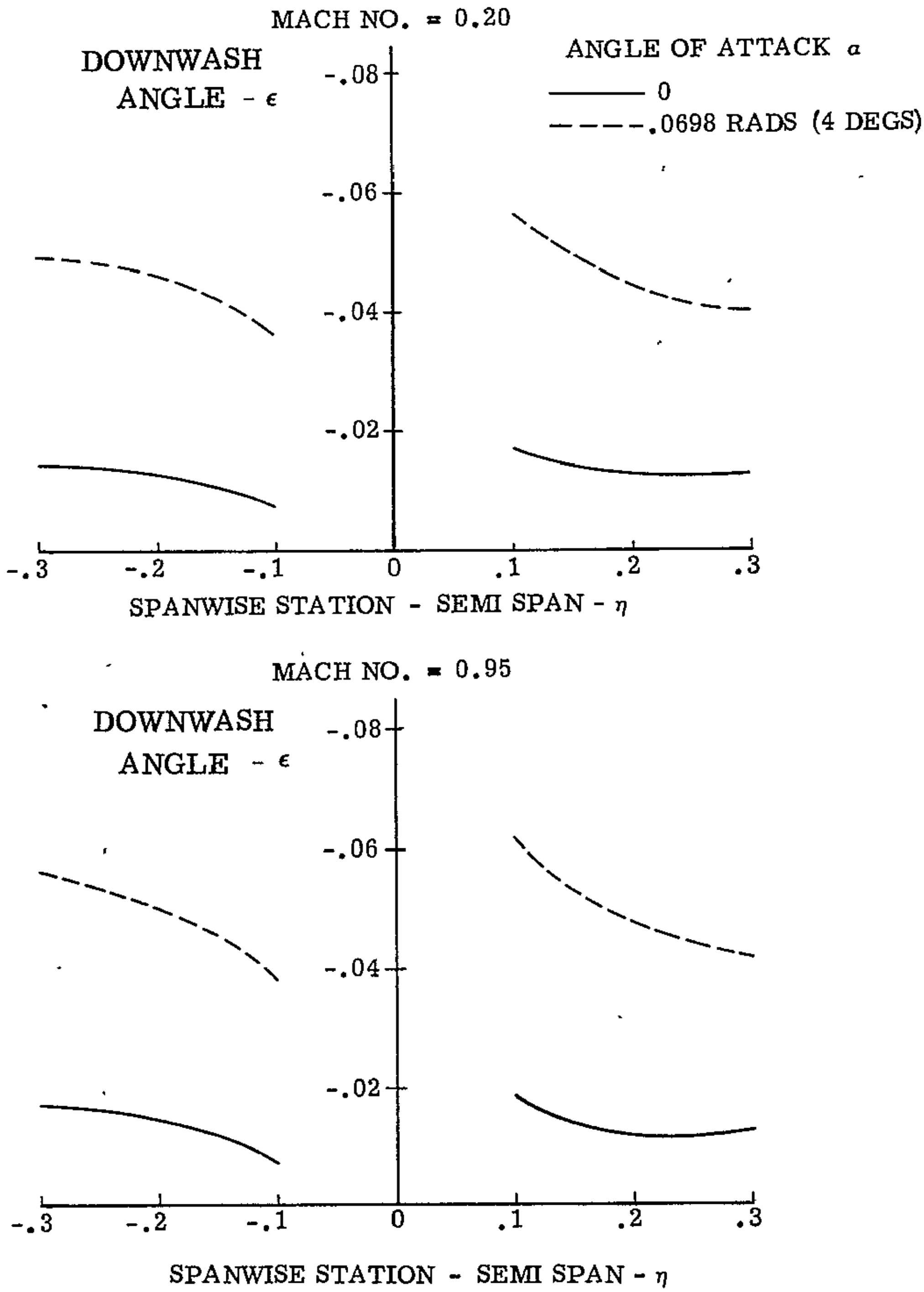

FIGURE A50 DOWNWASH ANGLES AT INLET

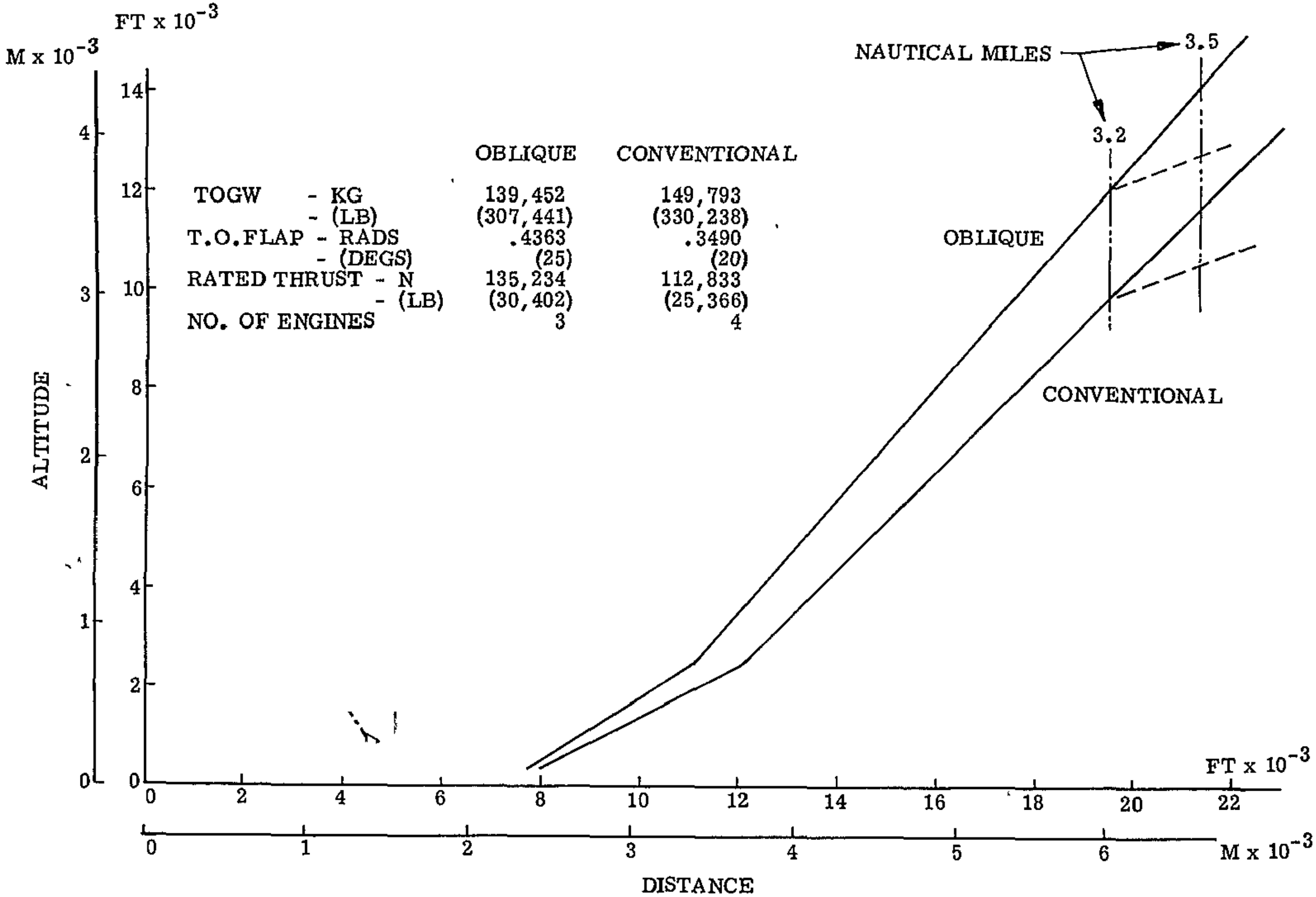

FIGURE A51 TAKEOFF PROFILES

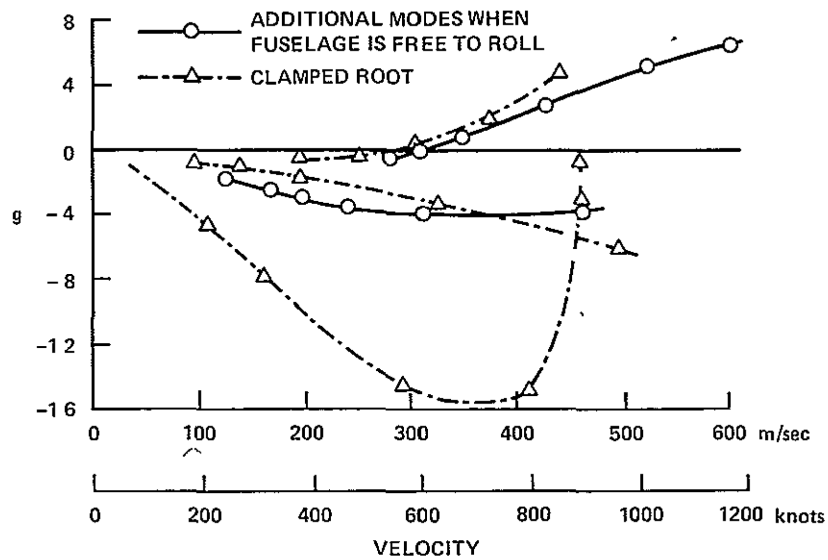

FIGURE B1 V-g DIAGRAM FOR A SWEEP ANGLE OF 0 RAD (0 DEG)

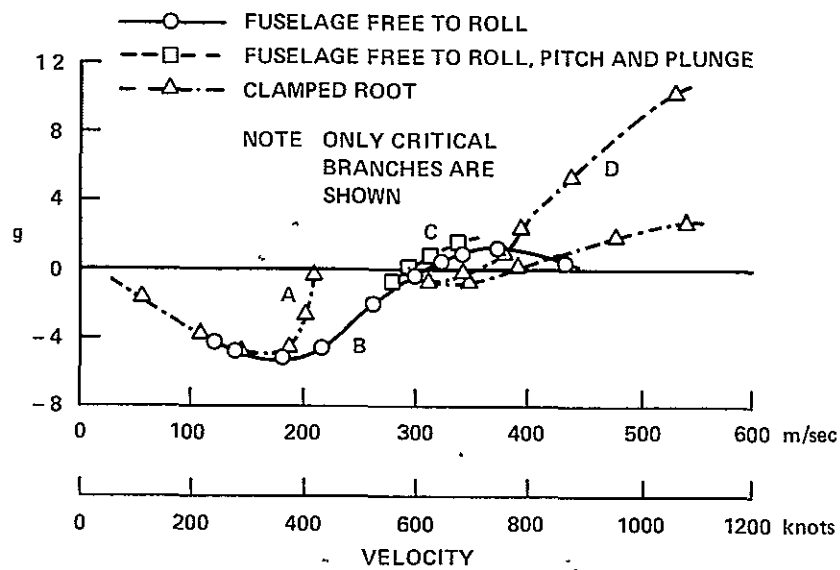

FIGURE B2 V-g DIAGRAM FOR A SWEEP ANGLE OF 0.523 RAD (30 DEG)

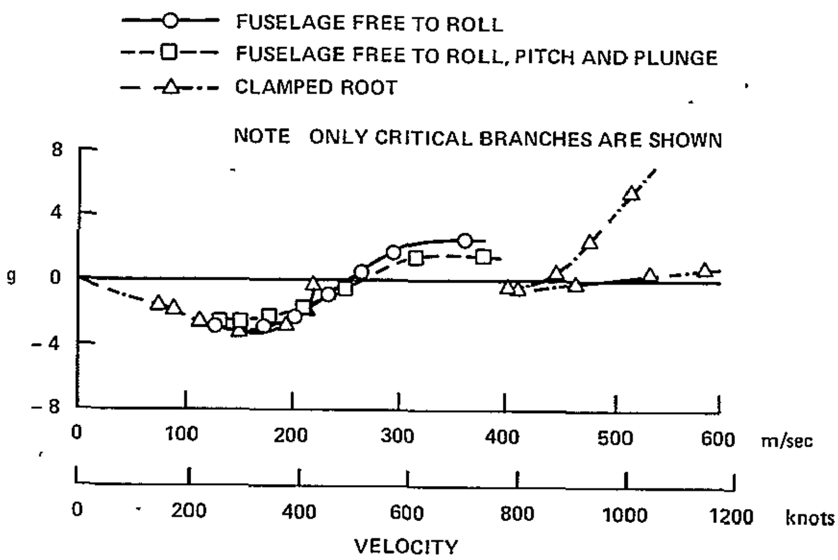

FIGURE B3 V-g DIAGRAM FOR A SWEEP ANGLE OF 0.785 RAD (45 DEG)

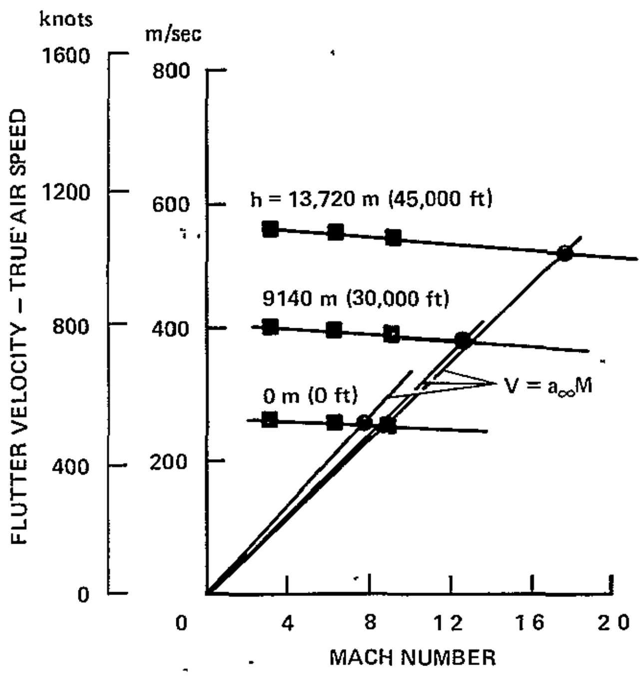

FIGURE B4 MATCHED FLUTTER POINT AT ALTITUDES
OF 0, 9140, AND 13,720 M (0, 30,000 AND 45,000 FT)

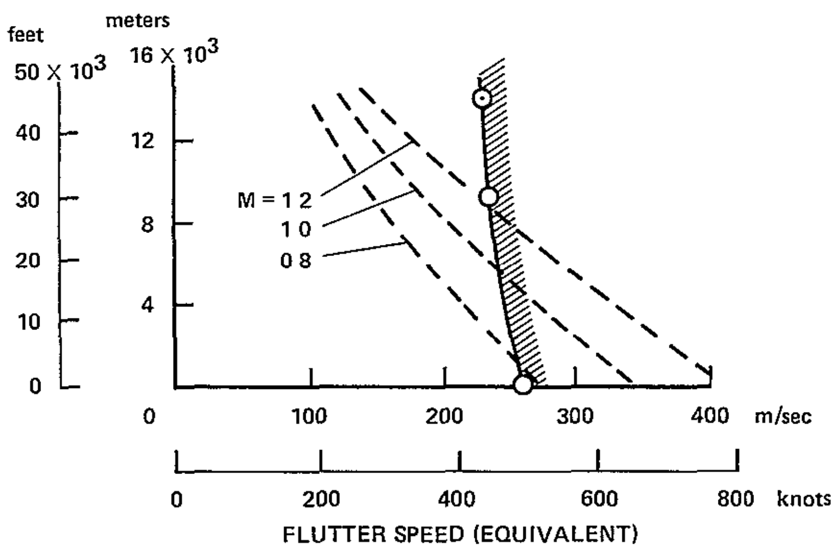

FIGURE B5 FLUTTER BOUNDARY