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EFFECT OF ADVANCED TECHNOLOGY AND A FUEL-EFFICIENT ENGINE ON A SUPERSONIC-CRUISE EXECUTIVE JET WITH A SMALL CABIN

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SUMMARY

An arrow-wing supersonic-cruise executive jet was configured incorporating a minimum practical size cabin for eight passengers and baggage, only one pilot, and advanced, fuel efficient engines. This concept is capable of performing the same mission as previously studied configurations with significant decreases in ramp weight and fuel used. At a ramp weight of 51,000 pounds with eight passengers, the range is about 3,350 nautical miles (New York to Paris) at Mach 2.3 cruise, and 2,700 nautical miles at Mach 0.9. Transcontinental missions were also investigated for both supersonic and subsonic cruise speeds.

A method to reduce sonic boom overpressure for overland flight was evaluated. With a modification to the optimum flight profile during the climb and acceleration segment, a reduction in sonic boom overpressure from 1.3 psf to 1.0 psf can be achieved. This reduction is possible with only a slight increase in fuel load for the transcontinental mission.

N83-33876#

INTRODUCTION

Continued interest in the development of supersonic cruise aircraft has prompted the NASA Langley Research Center to further assess the impact of technologies identified by the Supersonic Cruise Research (SCR) Program. The SCR Program focused on technology improvements for supersonic cruise transport aircraft with emphasis on identifying solutions to performance, economic, and environmental problems. Two prior NASA studies applied concepts and technologies generated by the SCR Program to small eight-passenger supersonic-cruise executive jets. The earlier study (ref. 1) utilized Mach 2.2 design arrow-wing configurations and resulted in concepts with takeoff gross weights of 74,000 to 80,000 pounds for 1976 state-of-the-art titanium manufacturing technology. A recent study (ref. 2) utilized a Mach 2.7 design arrow-wing configuration based on an extensively studied NASA transport configuration wing planform (refs. 3 - 5). The resulting concept exceeded range requirements at a takeoff gross weight of 64,000 pounds. The latter configuration serves as the baseline concept for the current study.

Modifications to enhance the performance of the supersonic cruise executive aircraft concept of reference 2 were to incorporate an improved fuel-efficient turbofan engine and to reduce the cabin size as much as practical. Cabin size was minimized by reducing seat size, seat pitch, clearances, and using a single pilot. With current and anticipated improvements in electronics and automated controls, pilot work load should be reduced sufficiently so that single pilot operation could be certified for intercontinental and transcontinental operation. The engine used is a scaled version of the Boeing 701S turbine bypass turbojet designed for a maximum cruise Mach number of 2.7 at an altitude of 65,000 feet under standard day atmospheric conditions. During this study, all supersonic missions were flown at Mach 2.3 and standard day conditions; subsonic missions were flown at Mach 0.9. The supersonic design mission is to carry eight passengers from New York to Paris. In addition, a maximum range subsonic mission and two transcontinental New York to Los Angeles missions, one subsonic and one at Mach 2.3, were analyzed. An additional study was performed to evaluate a flight profile designed to reduce sonic boom overpressure for overland flight.

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PART I. - CONFIGURATION DESCRIPTION

E. E. Swanson

The supersonic executive aircraft concept of reference I-1 has been modified to incorporate an improved fuel-efficient turbine-bypass turbojet engine and a change in aircraft design philosophy resulting in a substantial reduction in aircraft gross weight for a 3,200 n.mi. transatlantic mission. The engine used is a scaled version of the Boeing 701S turbojet designed for a cruise Mach number of 2.7. The payload capacity of 8 passengers plus baggage of reference I-1 has been retained, but with a reduced comfort level. Minimum-size light-weight seats are installed at a seat pitch of 32 inches, with an aisle width of 9 inches between arm rests and a maximum ceiling height of 57.5 inches on the aircraft centerline. The fuselage cross section in the cabin area, shown in figure I-1, is elliptical with the major axis vertical and provides a minimum of one inch clearance between the passenger's head and the cabin side wall for a 90th percentile man. A combined lavatory and baggage area with 50.4 cubic feet of space allocated for passenger and crew baggage is behind the passenger section.

A single pilot was chosen based on the assumption that automated controls would reduce pilot work load during flight, and that a single pilot could be certified for intercontinental and transcontinental supersonic operation. Passenger and crew arrangement, with space allocation for the various aircraft subsystems, is shown in the interior arrangement (fig. I-2).

The resulting aircraft has a takeoff gross weight of 51,000 pounds and a wing loading of 62.9 lb/ft^2. Fuselage length is 103 ft, and the reference wing area is 811 sq ft. The aircraft general arrangement is shown in figure I-3. Table I-I lists the airplane geometric characteristics.

Horizontal and vertical tail areas for previous studies have been determined by analysis of the stability and control requirements of the individual configurations. Due to the extensive data base available, and to the similarity of the configurations, tail volume coefficients from those studies were used to determine tail areas. No detailed stability and control analysis was performed. The configuration of reference I-1 has wing fins located at approximately 72.5 percent of the semi-span for increased directional stability. For the present concept, the close proximity of these fins to the aircraft center-of-gravity results in a minimal contribution to stability; therefore, they have been removed. The vertical tail-volume coefficient has been increased slightly and a dorsal fin has been added to offset the loss of stability incurred by the removal of the wing fins. The "T" tail arrangement with a fixed horizontal stabilizer and geared elevator has been retained with the volume coefficient in the same range as in the previous studies.

The main landing gear is fuselage mounted and consists of two single struts with one 31 x 11.5-16 tire per strut. The nose landing gear is a single strut single wheel arrangement with an 18 x 5.7-8 tire, and it retracts forward into the nose section of the fuselage forward of the crew-compartment pressure bulkhead. Fuselage pressurization is provided from the crew forward pressure bulkhead to the section behind the lavatory and baggage area. Figure I-4 shows the normal area distribution curve and the volume utilization by the subsystem. Fuel tanks in the wing and fuselage provide appropriate center of gravity control throughout the aircraft flight envelope. Full span leading and trailing edge flaps are provided for lift and drag control. Flap areas are shown in figure I-5.

The droop nose fairing and retractable visor provide adequate pilot vision for takeoff, landing, and ground handling. After takeoff, the droop nose is retracted to provide a smooth aerodynamic shape for supersonic cruise.

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TABLE I-I. - GEOMETRIC CHARACTERISTICS.

Geometry		Wing	Horizontal	Vertical
Area (ref)	ft^2	811	71	55.59
Area (gross)	ft^2	895	71	55.59
MAC (ref)	ft	27.449	7.034	9.680
MAC (gross)	ft	31.923	7.034	9.680
Span	ft	39.297	11.305	6.094
Aspect Ratio (ref)		1.904	1.8	.667
Aspect Ratio (gross)		1.727	1.8	.667
Sweep, Λ_{LE}	deg	74, 70.835, 60	60	65
Root Chord	ft	52.218	10.049	13.032
Tip Chord	ft	5.031	2.512	5.213
Root t/c	%		2.996	2.996
Tip t/c	%		2.996	2.996
Taper Ratio			.250	.400
Tail Volume Coefficients	$\bar{V}$		.187	.083

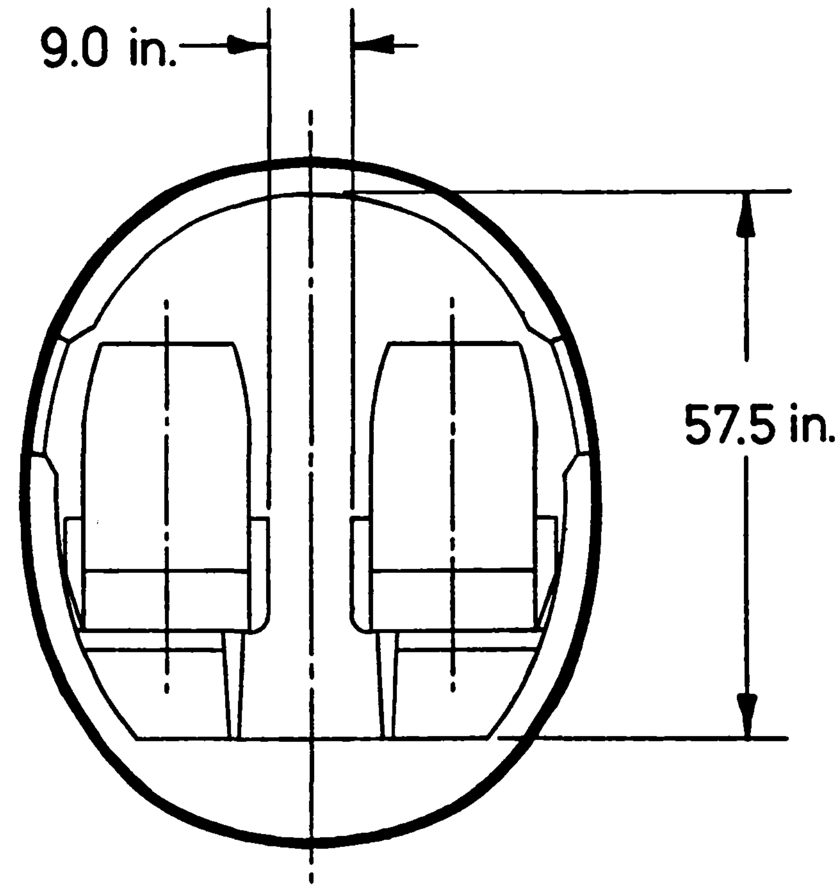

Figure I-1. - Fuselage passenger-area cross section.

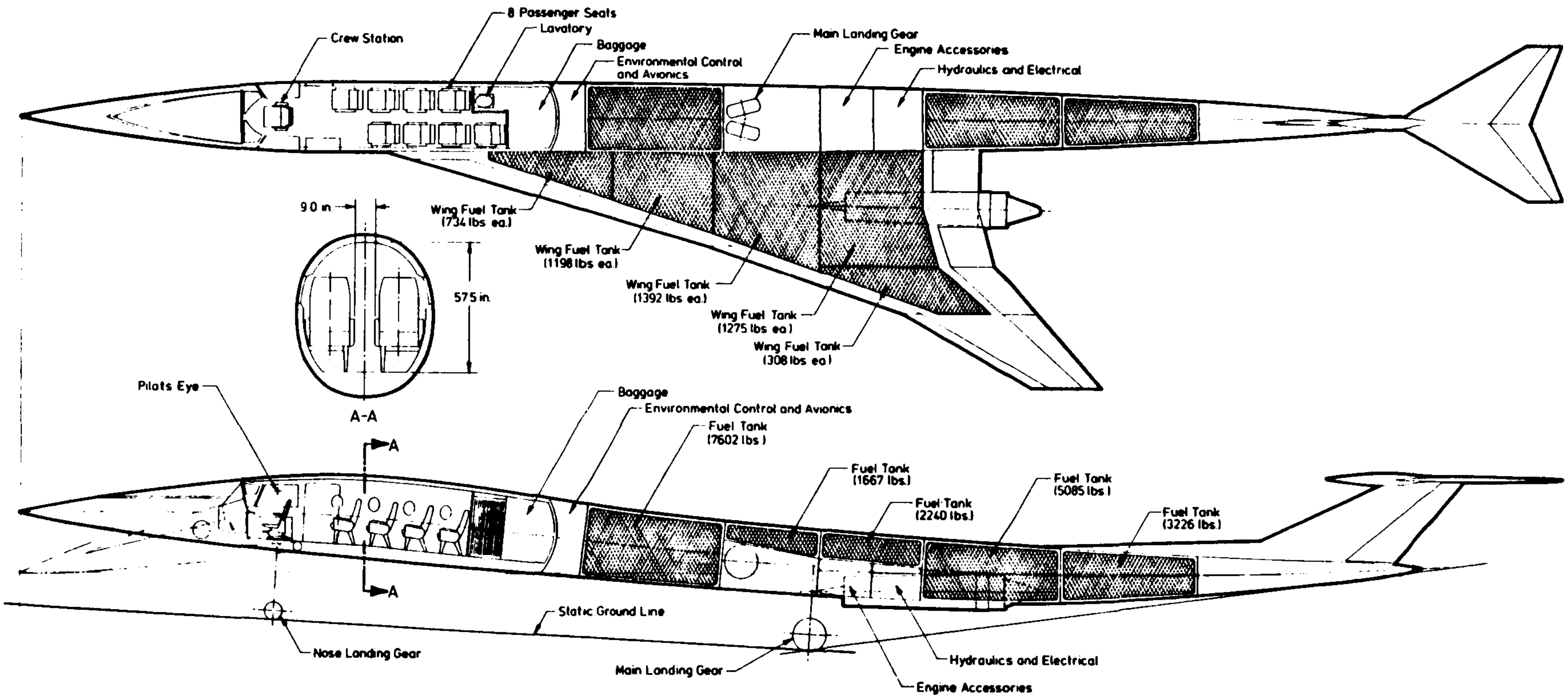

Figure I-2. - Interior arrangement.

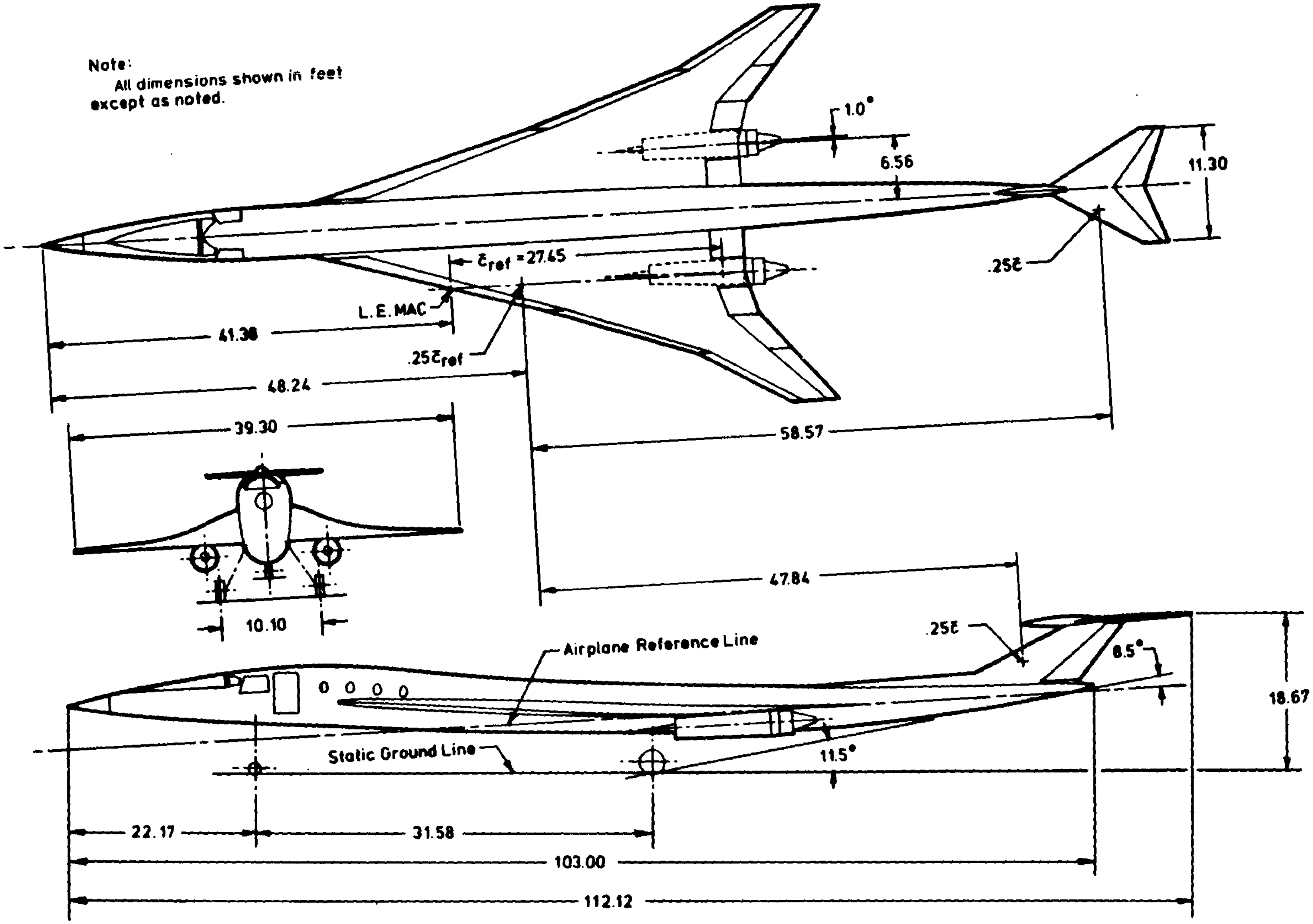

Figure I-3. - Aircraft general arrangement.

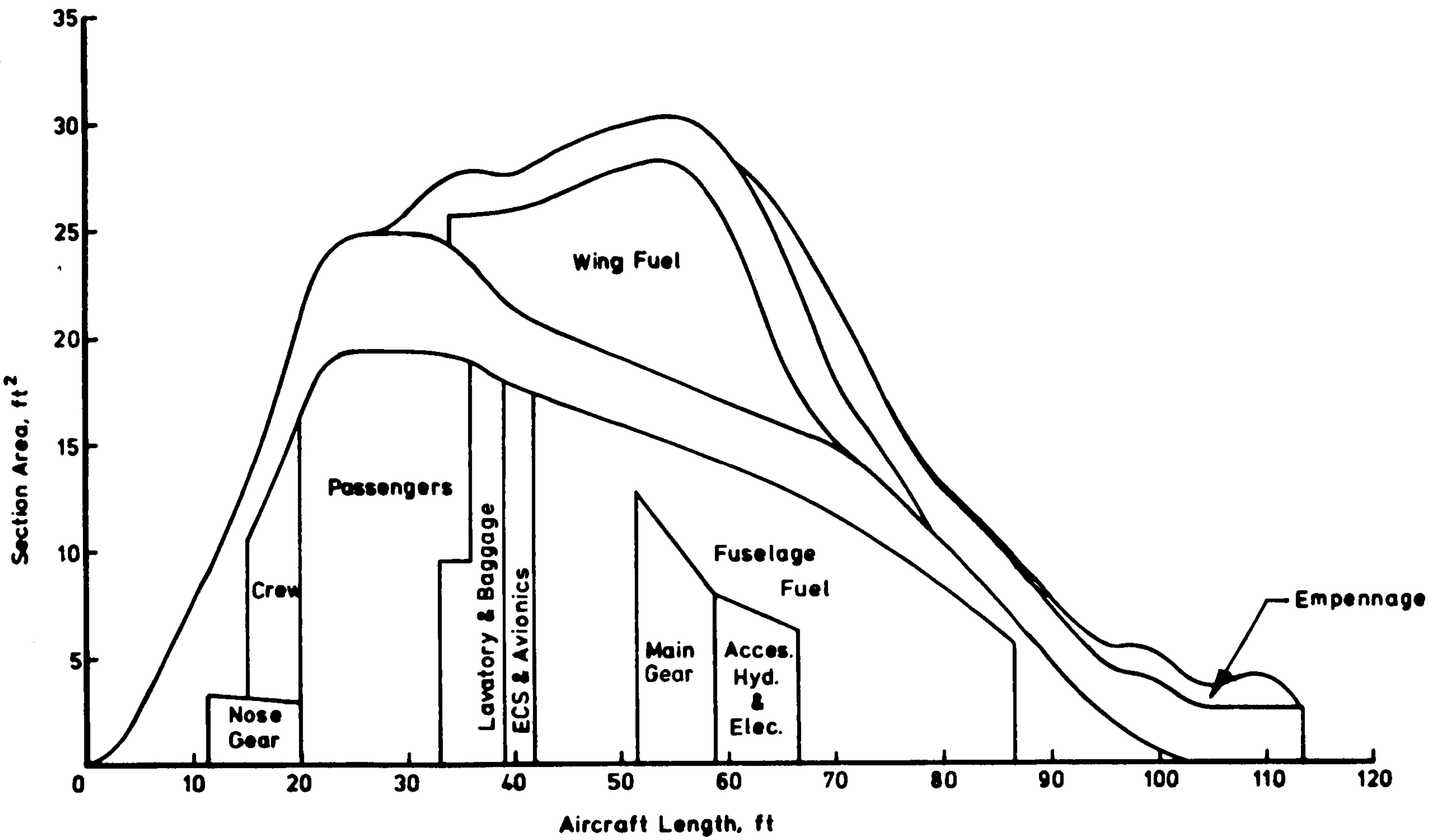

Figure I-4. - Aircraft volume utilization.

Flap No.	Area, ft^2
L-1	11.31
L-2	9.60
L-3	7.19
L-4	6.72
T-1	9.17
T-2	9.31
T-3	9.29
T-4	9.52

Areas are per side
S_w=811 ft^2

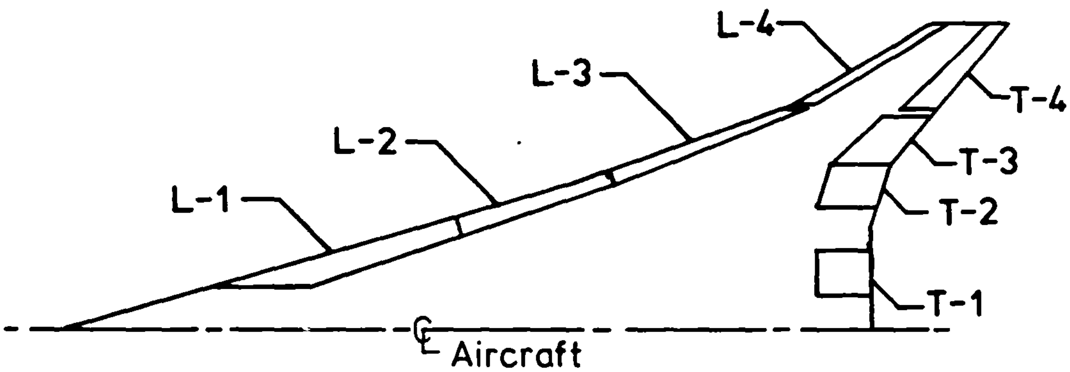

Figure I-5. - Wing flap areas.

PART II. - HIGH SPEED AERODYNAMICS

A. Warner Robins

The supersonic aerodynamic design of the configuration depended on the thorough wing-design effort of reference II-1 and the non-dimensionalized geometry of the wing is identical to that in the reference. Thorough coverage of the wing-design techniques employed is also provided in section V of reference II-1. The configuration is trimmed by center-of-gravity control, with the horizontal tail optimally set at 2 3/4 degrees incidence at cruise. Significant amounts of leading-edge thrust are indicated at cruise, with increasing amounts obtained as Mach number is reduced. The remaining components were developed and assembled in such a way as to retain the drag-due-to-lift characteristics of the basic, highly-developed wing (see refs. II-2 and II-3) while substantially reducing configuration wave drag.

SYMBOLS

A_x	cross section area
b	wing span
c	wing chord
$\bar{c}$	wing mean chord
C_D	drag coefficient
C_L	lift coefficient
h	altitude
L/D	lift/drag ratio
M	freestream Mach number
x, y, z	Cartesian coordinates
α	angle of attack
Δ	increment

Subscripts:

F	friction
form	form
i	lift-induced
LET	leading-edge thrust
0	at zero lift
R	roughness
W	zero-lift wave drag

METHODS AND APPLICATION

Configuration Aerodynamics

Aerodynamic characteristics were obtained by a variety of methods, only one of which is applicable throughout the Mach-number range. This process, known as the Sommer and Short T' method (see ref. II-4), provides for the calculation of skin-friction drag. Form drag, a subsonic-flow phenomena which arises from the increased viscous shear stresses associated with the increased local velocities caused by the form of the vehicle components, is found by application of geometry-dependent factors to these basic skin friction values. USAF DATCOM methods (ref. II-5) were used for this purpose. Roughness drag was estimated from previously-developed empirical data. Figure II-1 provides a sample of the buildup of these elements of zero-lift drag at the tropopause (at $h \approx 36{,}100$ feet).

Drag buildup at supersonic speeds is illustrated in figure II-2. Supersonic wave drag, which is determined through the use of a far-field analysis method described in reference II-6, is added to the friction and roughness drags. The values of wave-drag coefficient calculated for the configuration have been added to the remaining zero-lift drag values of figure II-1 to produce the variation of zero-lift-drag with Mach number shown in figure II-3.

A feature of the wave-drag program is the ability to define a least-drag fuselage area-distribution through a set of constraining fuselage stations for a given assemblage of components and for a given Mach number. This feature was utilized, and careful tailoring was employed to alleviate sharp local changes in area development, such as at the junctures of the thick upper elements of the vertical and horizontal tails, and at the empennage and body juncture. The empennage pod and dorsal fin resulted from such tailoring. The final fuselage area distribution with the specified constraints is shown in figure II-4. The Mach 2.4, average-equivalent-body area curves, are shown in figure II-5.

The linear-theory methods described in references II-7 through II-10 were used to compute the supersonic, lift-dependent drags (C_{D_i} and $C_{D_{LET}}$) illustrated in figure II-2. (Angle of attack and longitudinal stability characteristics were also obtained by these methods.) Note that the final supersonic drag polar differs from a no-leading-edge-thrust polar by an increment, $\Delta C_{D_{LET}}$, which contains not only the leading-edge-thrust attainable, but also the unattainable thrust which is manifested as vortex lift (see ref. II-11). No effects of fuselage volume on the lifting system were accounted for in the supersonic analysis, since, in aerodynamic design, the fuselage and wing integration provided that the rate of change of cross-section area above and below the wing camber plane remained equal (see refs. II-2 and II-3). Typical supersonic drag polars are shown in figure II-6. Maximum lift-drag ratio and the lift-drag ratios corresponding to specific operating points are shown in figure II-7. Maximum lift-drag ratio at the begin-cruise altitude is seen to be 7.45 while the operating value is 7.27.

Subsonic drag polars were obtained by the vortex-lattice method of reference II-12, supplemented by the method described in reference II-13 which provides the increments due to leading-edge thrust and vortex lift. While increased subsonic aerodynamic performance was realized with the use of simple, twenty-percent leading-edge flaps on the outboard wing panels, no flap optimization was undertaken. Ten degrees of leading-edge flap deflection is reflected in the data from Mach number 0.60 to 0.95. Some deflection of these leading-edge flaps might help at transonic and low supersonic speeds, particularly in combination with small amounts of trailing-edge flap deflection. Figure II-8 shows the $M = .9$, $h \approx$ 36,100 feet, drag polar reflecting the appropriate attainable leading-edge thrust and vortex lift, compared to the corresponding no-thrust and full-thrust polars. Drag polars for Mach numbers from .6 to .95 are shown in figure II-9.

Sonic Boom

Estimates of sonic-boom overpressure characteristics were made using the simplified process described in reference II-14. Rather than the simple, shape-factor charts, however, equivalent cross-section areas due to both volume and lift were combined for four flight conditions to provide the characteristic shape-factor curve for this specific study configuration. The results are shown in figure II-10 in which sonic-boom overpressures as a function of altitude and wing loading are plotted for Mach numbers 1.2 and 2.3. The effects of various boom-alleviation flight profiles on both sonic boom and range are shown in the section covering aircraft performance.

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II-12. Miranda, Louis R.; Elliott, Robert D.; and Baker, William M.: A Generalized Vortex Lattice Method for Subsonic and Supersonic Flow Applications. NASA CR-2865, December 1977.

II-13. Carlson, Harry W.; and Walkley, Kenneth B.: A Computer Program for Wing Subsonic Aerodynamic Performance Estimates Including Attainable Thrust and Vortex Lift Effects. NASA CR-3515, 1982.

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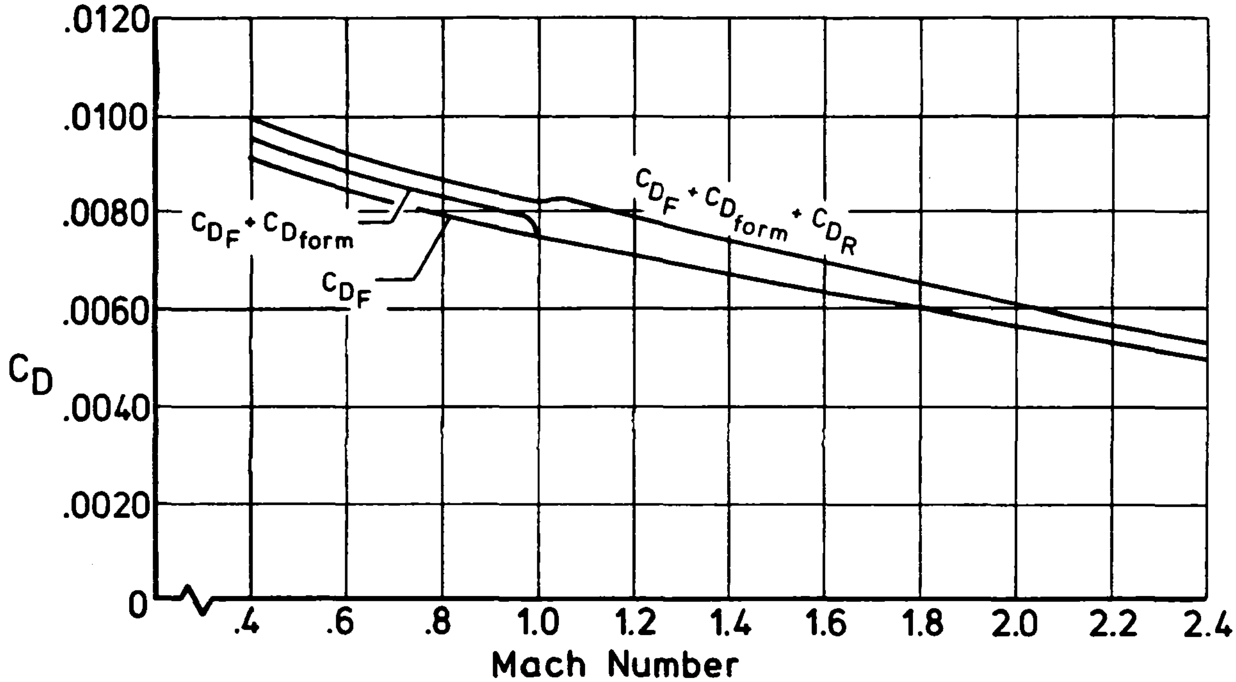

Figure II-1. - Buildup of zero-lift drag coefficient as a function of Mach number. Wave drag excluded. h = 36,100 feet.

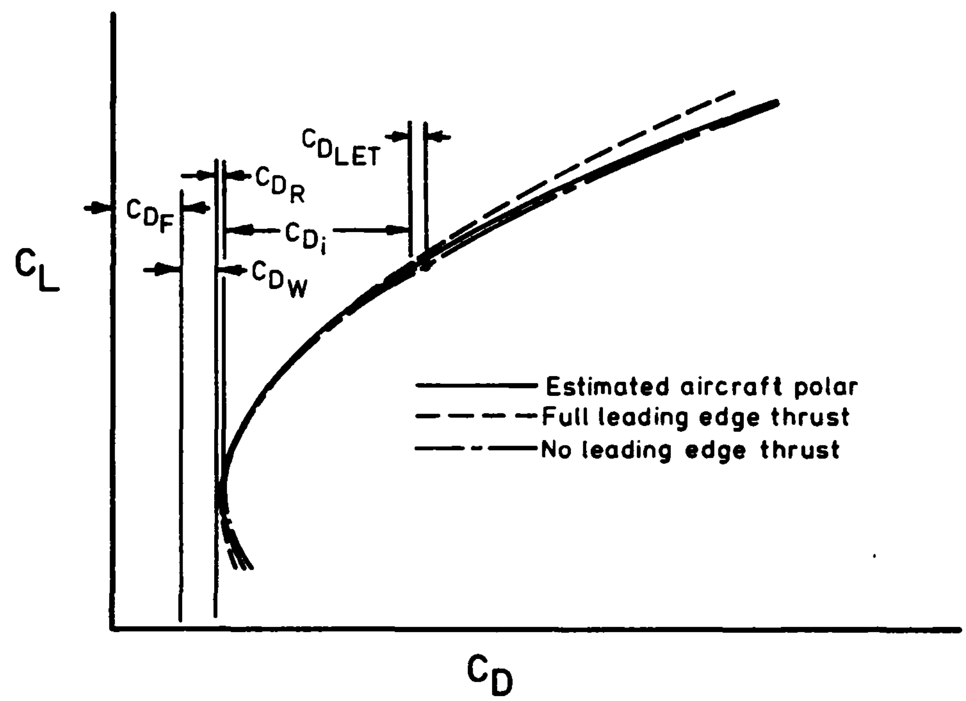

Figure II-2. - Buildup of supersonic drag polars.

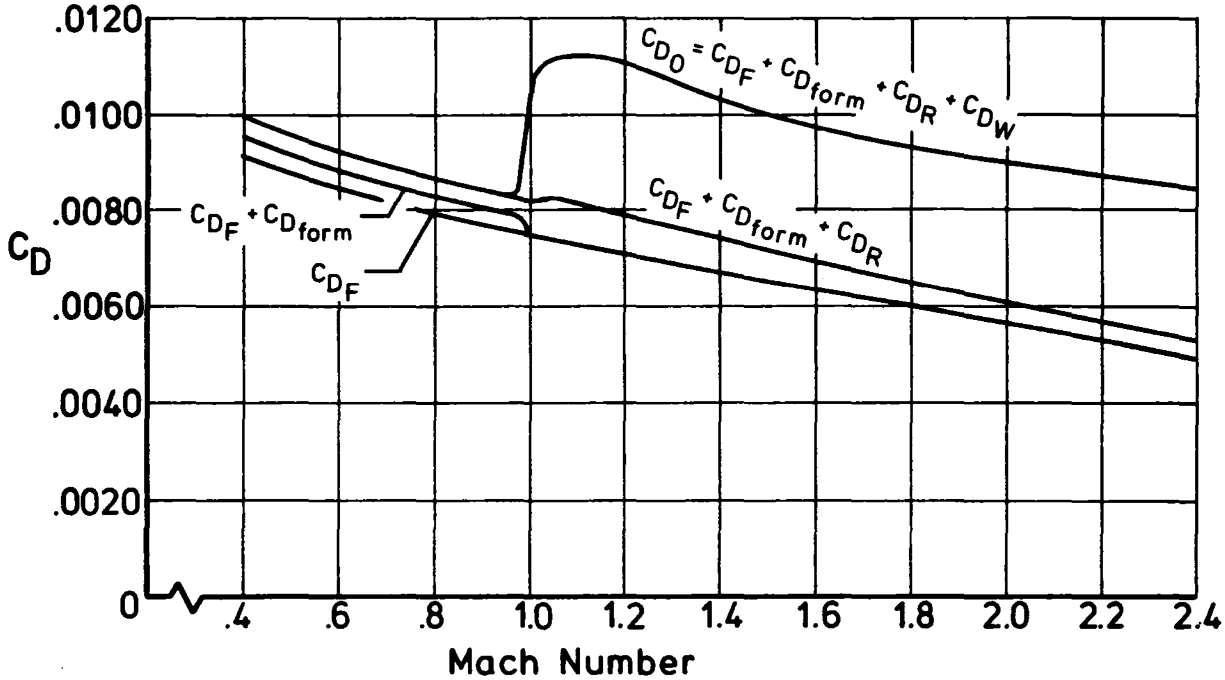

Figure II-3. - Buildup of zero-lift drag coefficient of the complete configuration as a function of Mach number. h = 36,100 feet.

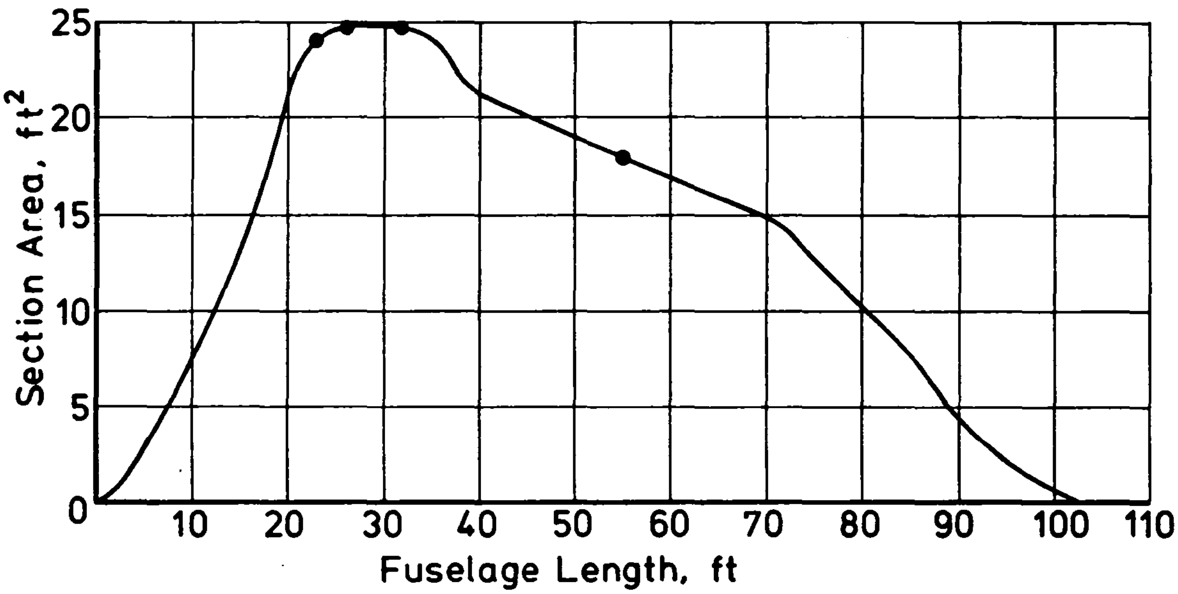

Figure II-4. - Optimized fuselage area distribution. Cross-section-area constraint points indicated by the solid symbols. Design Mach number = 2.4.

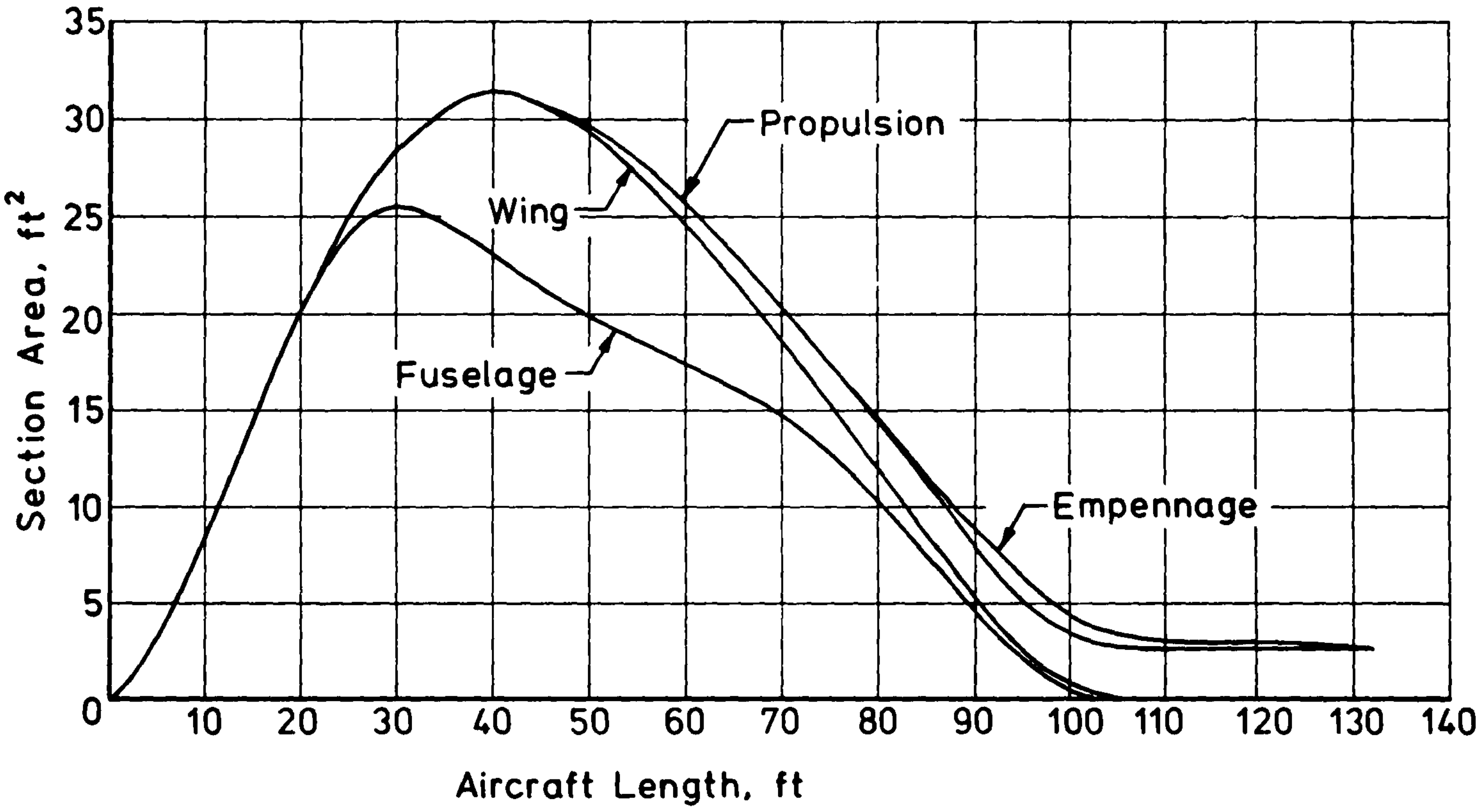

Figure II-5. - Average equivalent-body area-distribution buildup at M = 2.4.

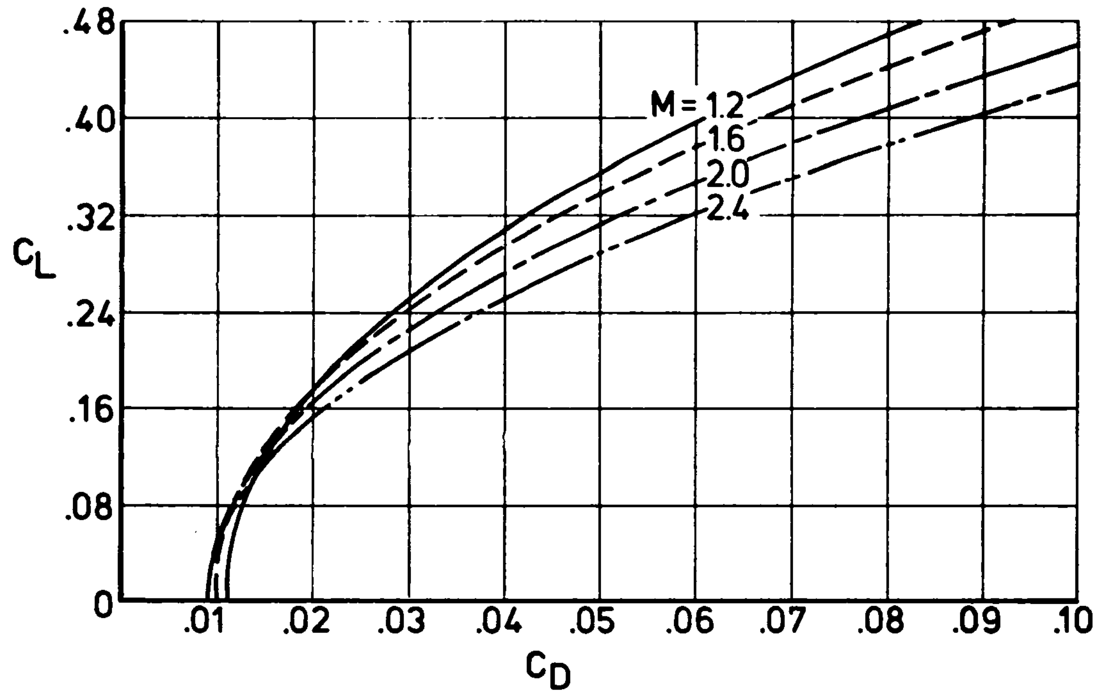

Figure II-6. - Supersonic drag polars. h = 36,100 feet.

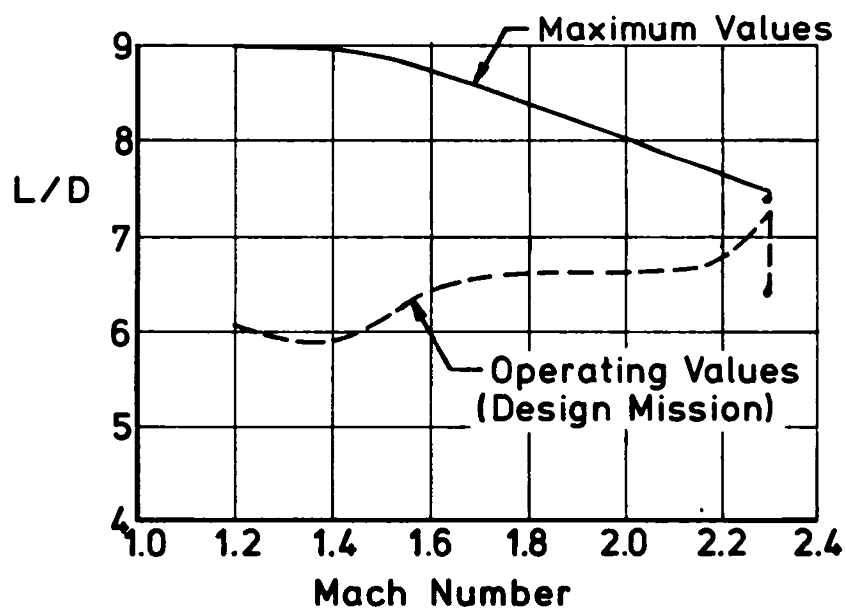

Figure II-7. - Maximum attainable and operating lift-drag ratios for the design mission.

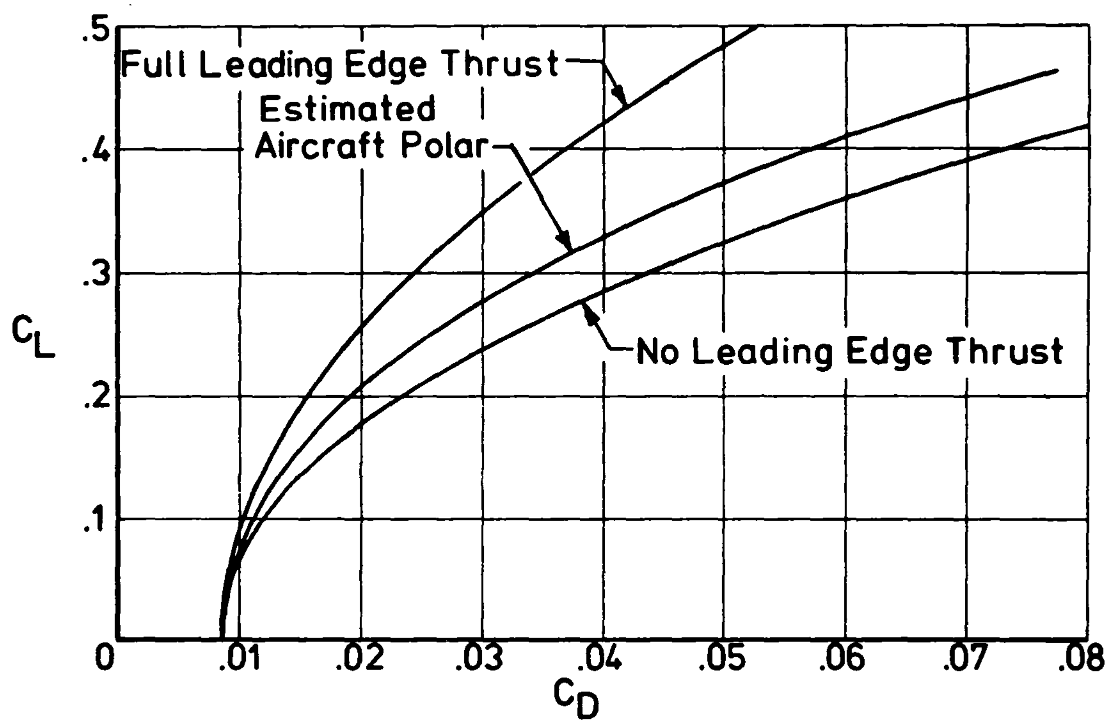

Figure II-8. - Comparison of estimated subsonic drag polar with drag polars reflecting no leading-edge thrust and full leading-edge thrust. M = 0.9 and h = 36,100 feet.

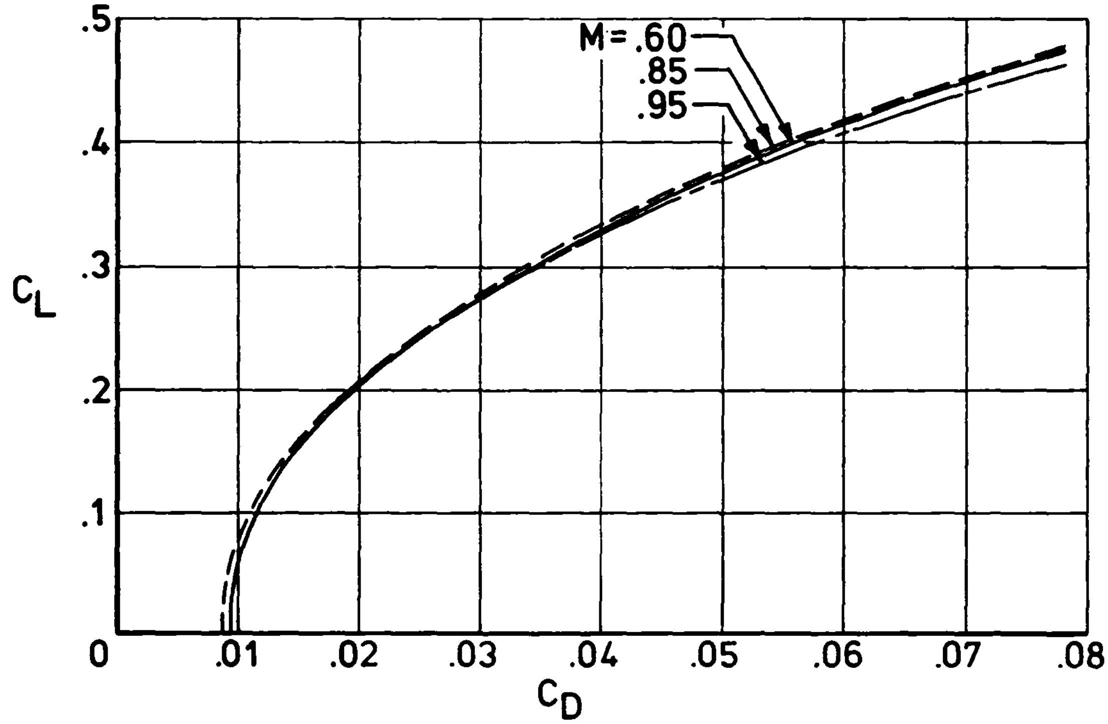

Figure II-9. - Subsonic drag polars for complete configuration. h = 36,100 feet.

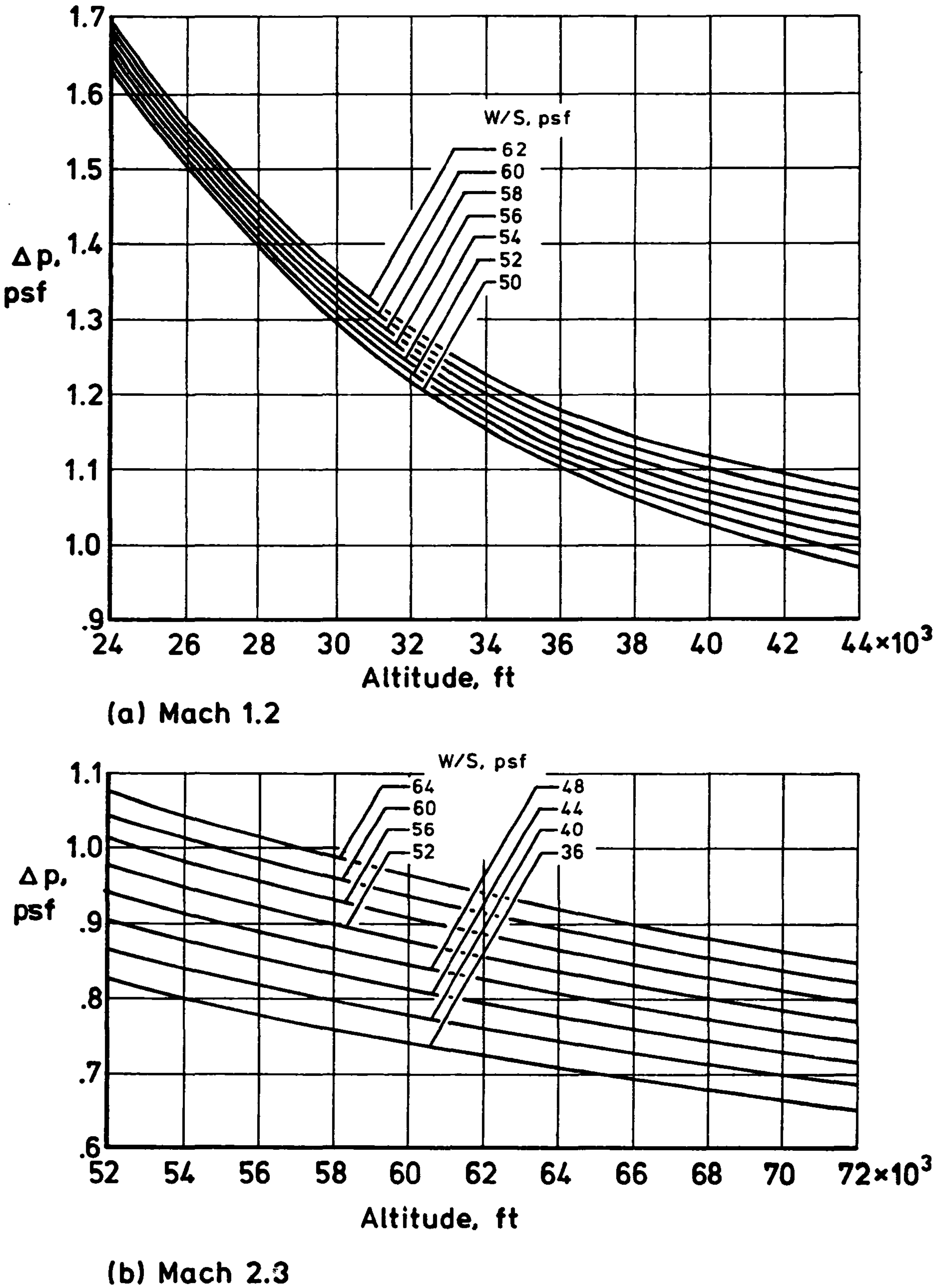

Figure II-10. - Transonic and supersonic-cruise sonic-boom overpressures as a function of altitude and wing loading.

PART III. - PROPULSION

W. A. Lovell

The engine selected for this airplane study is a scaled version of the Boeing 701S study engine. This engine is designed for cruise at a Mach number of 2.7 at an altitude of 65,000 ft at standard day atmospheric conditions. The technology level of this engine should be available in the early to mid 1990's. For this study, the engine has been sized for a two-engine airplane configuration with a maximum take-off gross weight of 51,000 lbf and an installed thrust to weight ratio of 0.39. Installed engine performance data for the resized Boeing 701S engine at standard day atmospheric conditions are provided.

Installed engine performance was developed using the NASA-Ames "P" inlet recovery, an ejector nozzle, 200 HP power extraction for aircraft accessories, and 1.0 lbm/sec service airbleed. The installation losses also include the effects of inlet spillage and bleed drag, nozzle boattail drag, and nozzle over and under expansion losses.

BASELINE ENGINE

The Boeing 701S engine, as designed by Boeing, is a 750 lbm/sec airflow turbine-bypass turbojet engine without thrust augmentation. The engine is designed for cruise at Mach 2.7 at 65,000 ft altitude at standard day atmospheric conditions. The exhaust system consists of a convergent-divergent ejector nozzle with a thrust reverser and a thermal acoustic shield for sound suppression. This engine is described in detail in reference III-1.

Performance

Baseline engine performance is based on the following conditions and installation effects:

- o 1962 U.S. Standard Atmosphere
- o NASA-Ames "P" inlet recovery
- o 1.0 lbm/sec high pressure airbleed for customer services

- o 200 HP power extraction for aircraft accessories
- o Afterbody drag of an isolated nacelle

Isolated nacelle afterbody drag is determined from the customer connect point on the engine to the end of the exhaust nozzle (see fig. III-1).

Baseline (as designed) engine characteristics at maximum power, sea level static standard day atmospheric conditions are tabulated below:

Total corrected engine airflow	750 lbm/sec
Cycle pressure ratio	13.5
Net installed thrust	67,633 lbf
Net installed specific fuel consumption	1.05 lbm/hr/lbf
Estimated weight (including nozzle, thrust reverser and thermal acoustic shield)	12,662 lbf
Maximum envelope diameter	80.5 in
Length of engine plus nozzle	308.8 in

Weight and Sizing

Baseline engine weight and size may be scaled based on relative size by means of the following equations.

$$D_2 = D_1 \left(\frac{F_{n_2}}{F_{n_1}}\right)^{.5}$$

$$L_2 = L_1 \left(\frac{F_{n_2}}{F_{n_1}}\right)^{.5}$$

$$WE_2 = WE_1 \left(\frac{F_{n_2}}{F_{n_1}}\right)$$

where:

WE = weight of the engine

D = engine diameter

L = engine length

F_n = net engine thrust

Subscripts:

1 = baseline engine parameter

2 = desired engine parameter

Relative size is defined as the ratio of F_{n_2}/F_{n_1}. Gross thrust, ram drag, fuel flow, and engine airflow are scaled in the same proportion as relative engine size.

STUDY ENGINE

Installation of this engine on the aircraft necessitated adjustment of the engine performance for inlet spillage and bleed drag, and nozzle-over-under-expansion losses. These adjustments were made to the baseline engine performance data at standard day atmospheric conditions. Corrections for nacelle skin friction, interference and wave drag are not accounted for in the engine performance data since these are included in the airplane drag polars.

The aircraft, as determined by this study, has a design gross weight of 51,000 lbf and an installed thrust-weight ratio of 0.39. The Boeing 701S engine was scaled in both size and performance to meet the study requirements for a twin engine configuration.

Weight

The baseline engine weight of 12,662 lbf, when adjusted to the aircraft size, is 1,865 lbf. Each of these weights includes the base engine, nozzle, thrust reverser, and thermal acoustic shield.

Nacelle and Inlet

The inlet selected for this study is a scaled version of the NASA-Ames "P" inlet (ref. III-2). It is a typical axisymmetric mixed compression design with a translating centerbody sized for supersonic cruise conditions. A nacelle concept layout to house the scaled Boeing 701S engine, incorporating the scaled NASA-Ames "P" inlet and a typical ejector nozzle, was prepared for use in determining nacelle drag and weight. A sketch of the resulting nacelle is shown in figure III-1.

Performance

The Boeing 701S engine, when scaled to meet the study aircraft requirements, produces 9,960 lbf thrust with a corrected engine airflow rate of 110 lbm/sec at sea level static standard day atmospheric conditions at maximum power. Installed standard day engine performance adequate for aircraft mission performance analysis have been adjusted to the study aircraft requirements. These data are presented on figures III-2 - III-6 for maximum climb and maximum and part power cruise ratings. Take-off thrust is shown on figure III-7.

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Note:
Dimensions shown are in inches.

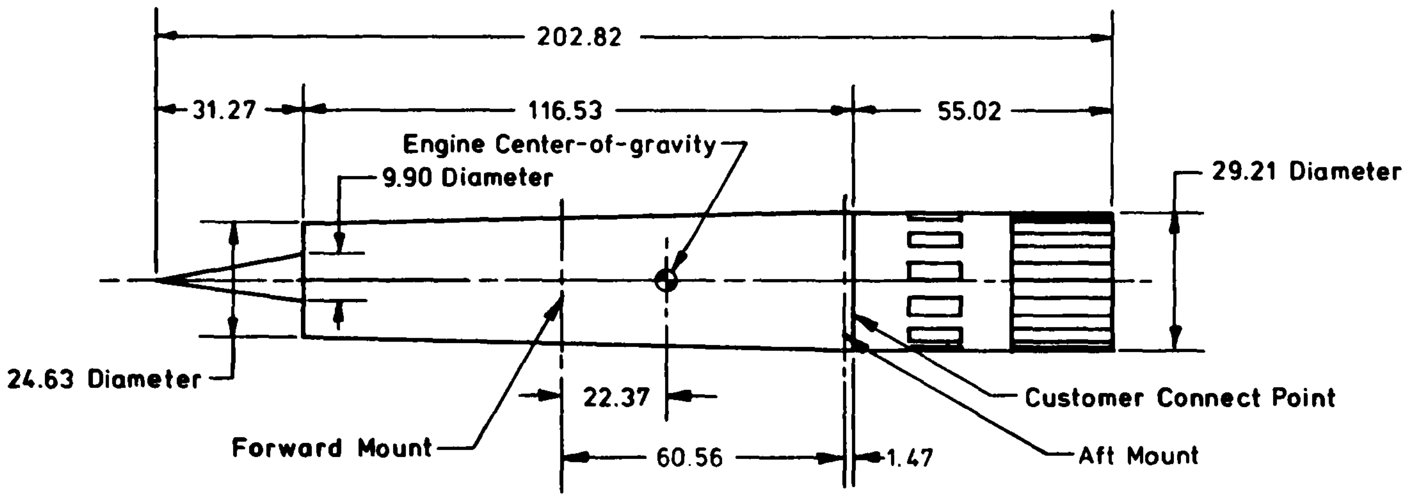


Figure III-1. - Nacelle geometry.

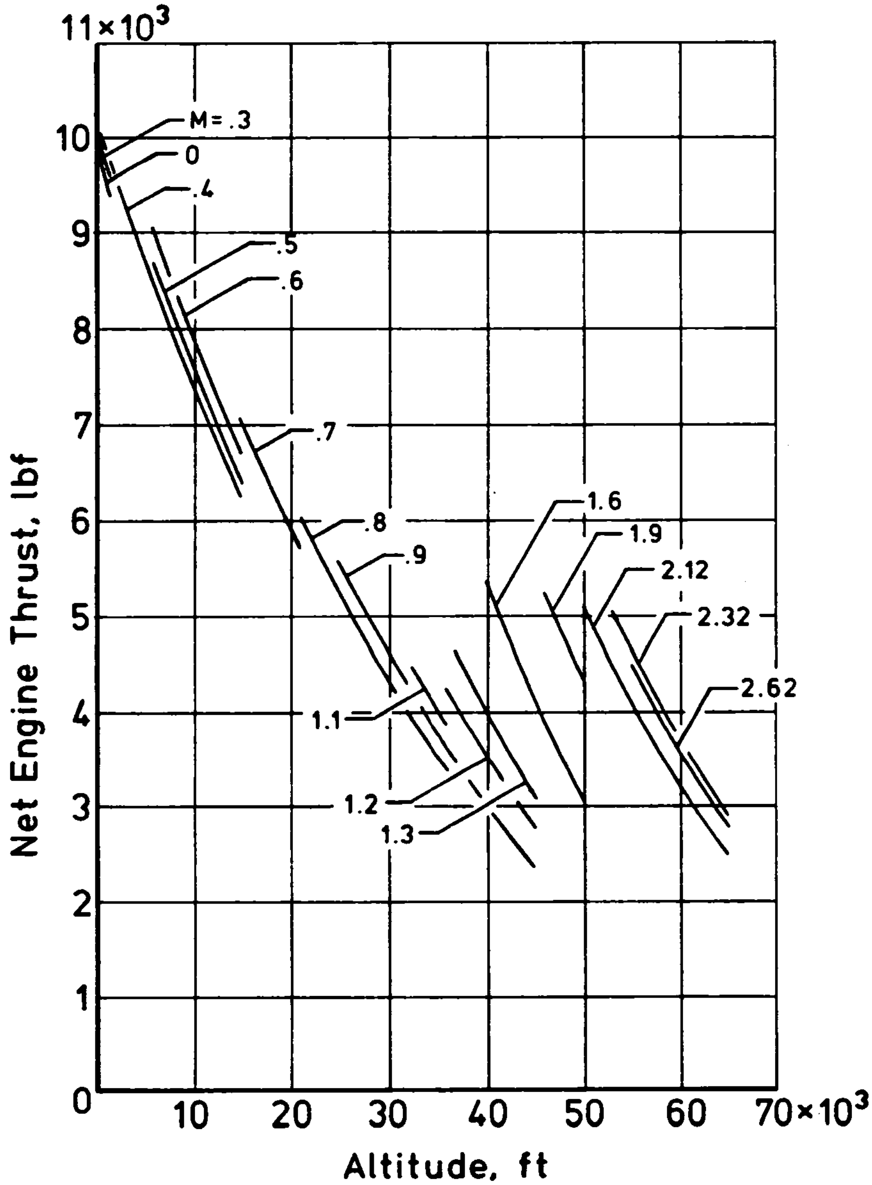


Figure III-2. - Engine maximum climb rated thrust.

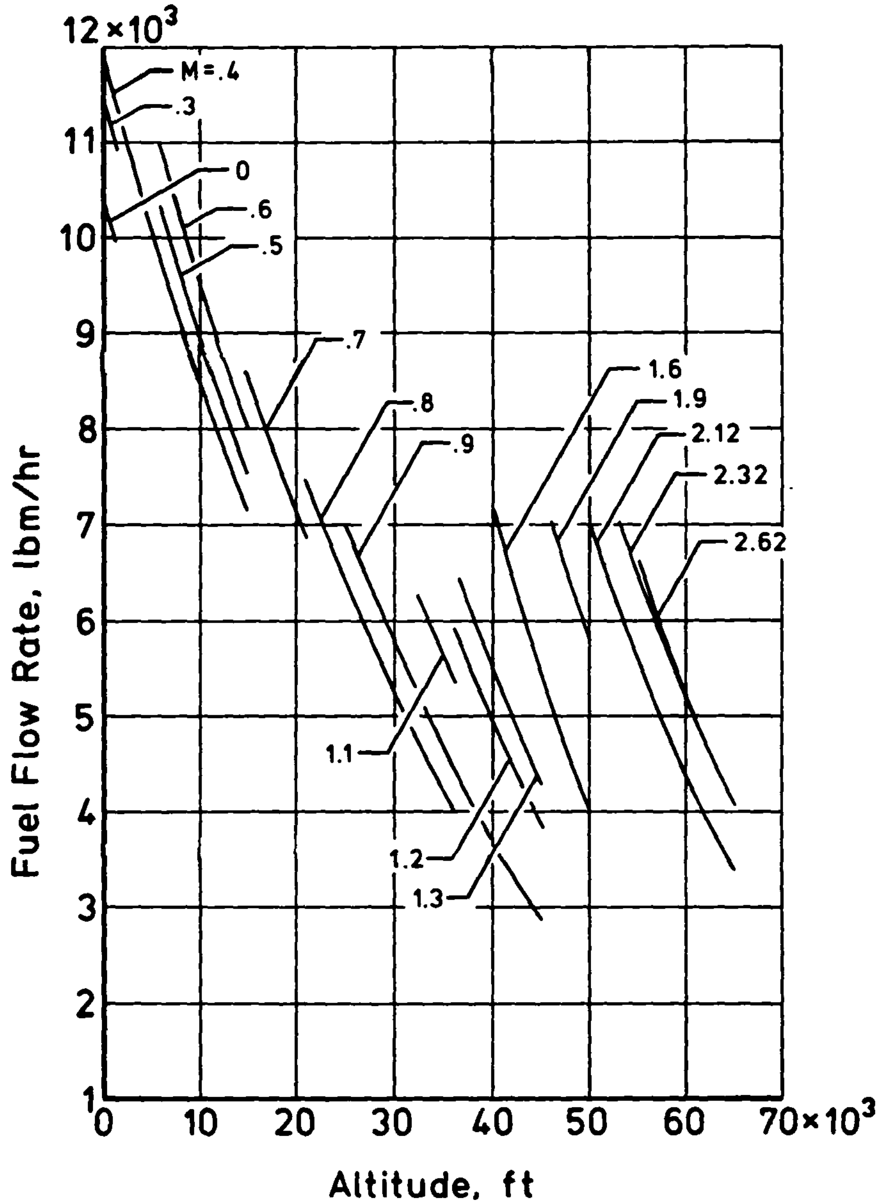


Figure III-3. - Engine fuel flow at maximum climb rated thrust.

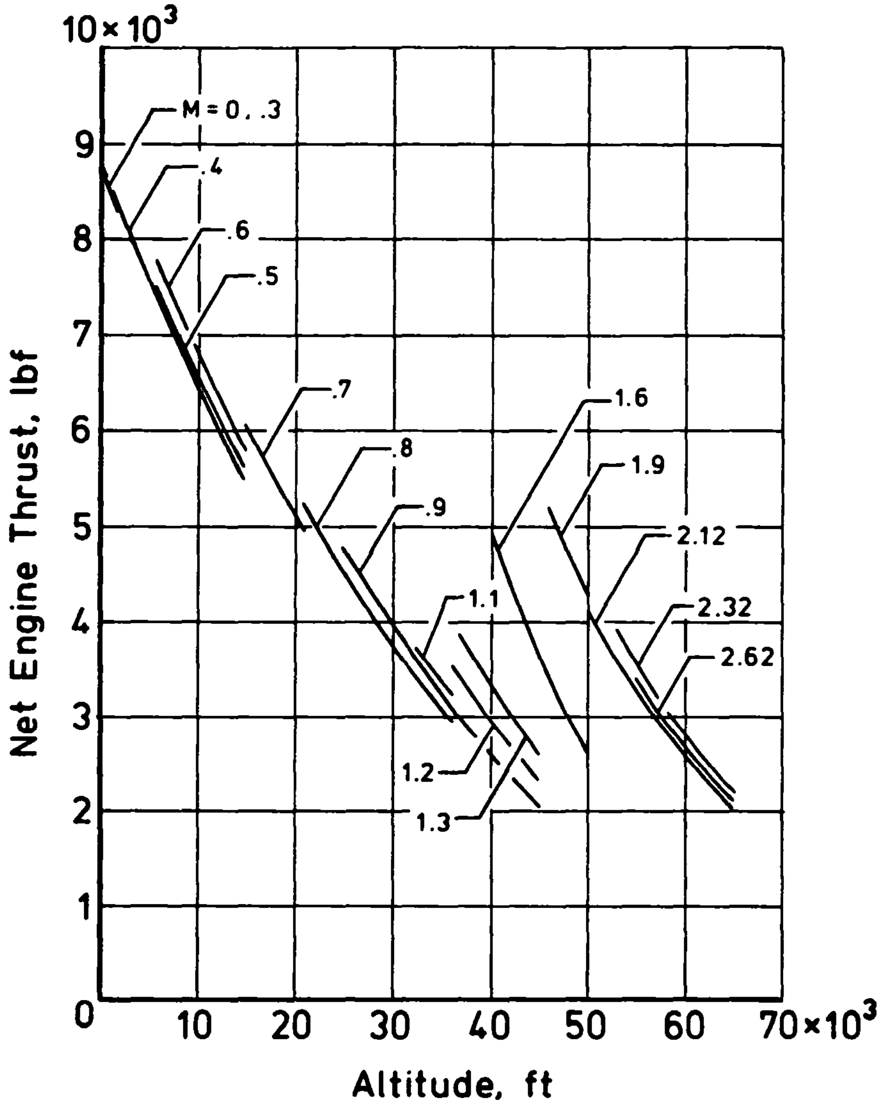


Figure III-4. - Engine maximum cruise rated thrust.

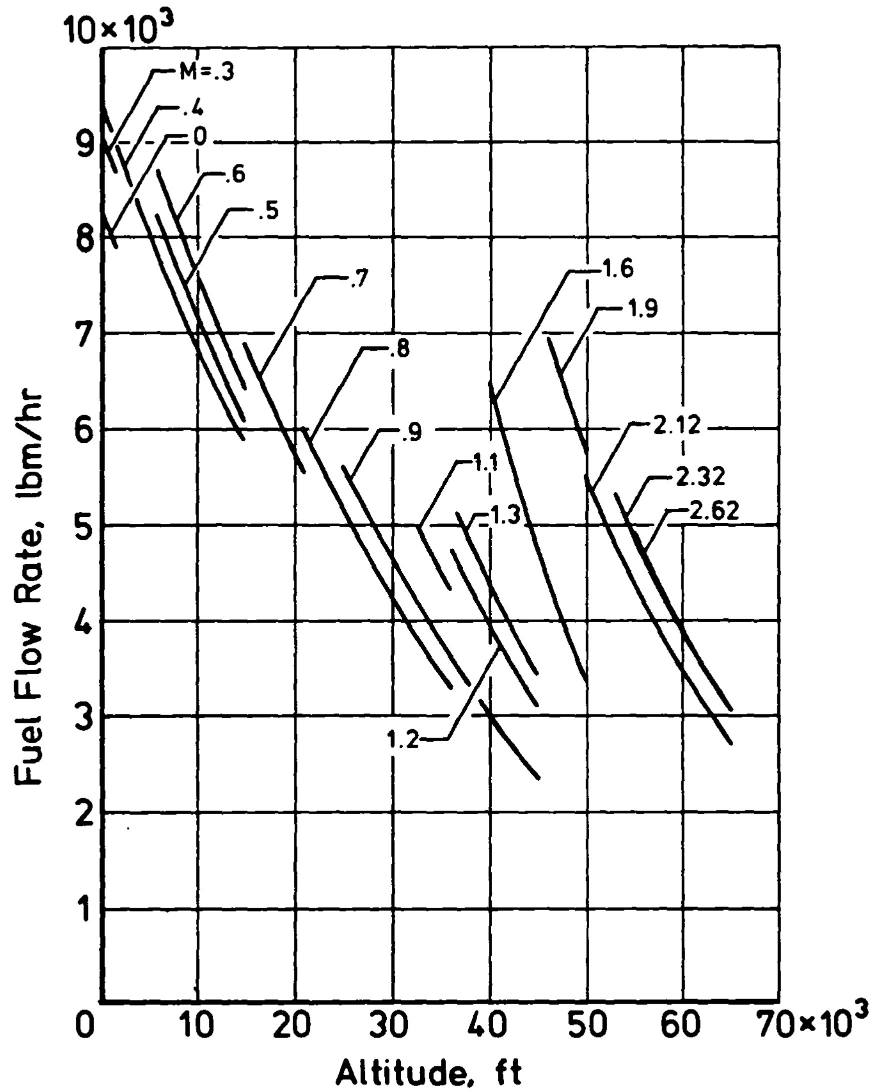


Figure III-5. - Engine fuel flow at maximum cruise rated thrust.

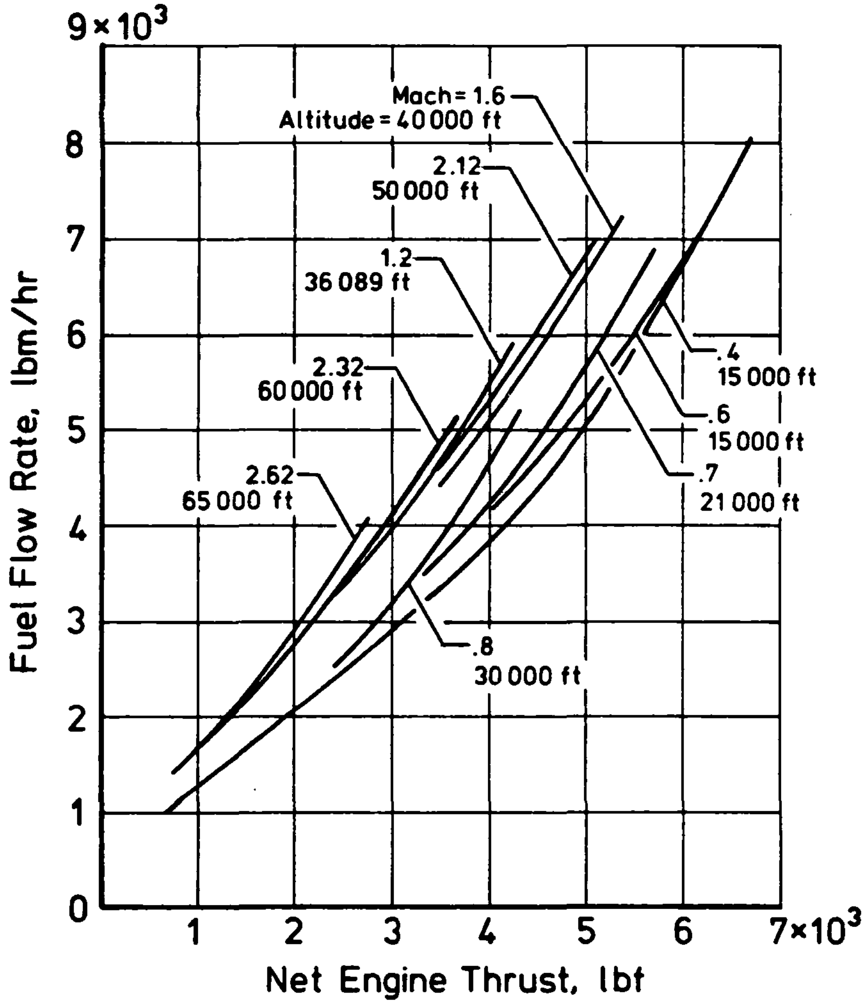


Figure III-6. - Engine fuel flow for maximum and part power thrust.

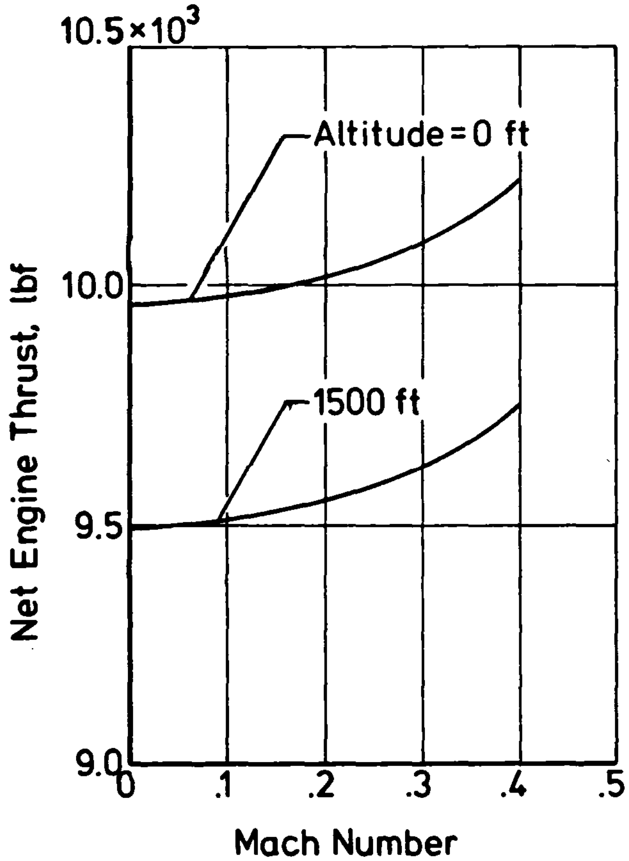


Figure III-7. - Engine takeoff thrust.

PART IV. MASS PROPERTIES

E. E. Swanson

The mass properties analysis of the study configuration was performed using the weight module of the Flight Optimization System (FLOPS) computer program developed by Kentron International, Inc., which is described in the Appendix. The structural weight estimates are based on utilizing 1980 technology level superplastic formed/diffusion bonded (SPF/DB) titanium throughout all primary and secondary airframe structure. Using this technology, the following weight reductions were anticipated when applied to the 1971 titanium technology level used in previous studies.

Wing, empennage, etc.	-7%
Fuselage	-22%
Nacelle, inlet, cowling	-19%

The resulting mass breakdown for this configuration is detailed in table IV-I. As previously mentioned in the configuration description section, the pressurized cabin area is elliptical in cross section. Due to the relatively high cruise altitude, 65,000 ft, cabin pressure differential will be 9.5 to 11.0 psi depending upon the pressure altitude selected. This pressure level may cause a weight penalty for a non-circular section that was not accounted for in this study.

The study aircraft was configured to insure that the balance characteristics would be such that the takeoff, cruise, and landing centers-of-gravity lie within limits prescribed by stability and control criteria. These limits are:

	Percent $\bar{c}_{ref}$	
Flight Condition	Aft Limit	Forward Limit
Takeoff	55.0	43.0
Landing	55.0	43.0

Combinations of fuel loading and transfer sequencing were investigated to determine the most forward and aft attainable center-of-gravity (c.g.) boundaries.

These limiting boundaries are shown in figure IV-1. With the wing apex located 18.33 ft aft of the aircraft nose, the center-of-gravity boundaries fall within the desired path and are attainable with proper fuel management.

The aircraft inertia characteristics were computed using the DATCOM method of reference IV-1 which is incorporated in the FLOPS weight analysis module. Inertias of the individual components and subsystems are computed about the respective centroids of each, transferred to the aircraft overall center-of-gravity locations, and then summed. Two conditions were analyzed, design takeoff gross weight and normal landing weight. A summary of the inertias is shown in table IV-II.

REFERENCES

IV-1. USAF Stability and Control DATCOM. Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, October 1960, Revised April 1978.

TABLE IV-I. - GROUP WEIGHT SUMMARY

	lbf
WING	5,784.
HORIZONTAL TAIL	372.
VERTICAL TAIL	270.
FUSELAGE	3,710.
LANDING GEAR	1,412.
NACELLE	896.
STRUCTURE TOTAL	(12,444.)
ENGINES	3,754.
MISCELLANEOUS SYSTEMS	206.
FUEL SYSTEM-TANKS AND PLUMBING	709.
PROPULSION TOTAL	(4,669.)
SURFACE CONTROLS	1,292.
INSTRUMENTS	110.
HYDRAULICS	483.
ELECTRICAL	1,067.
AVIONICS	500.
FURNISHINGS AND EQUIPMENT	980.
AIR CONDITIONING	655.
ANTI-ICING	121.
SYSTEMS AND EQUIPMENT TOTAL	(5,209.)
WEIGHT EMPTY	22,322.
CREW AND BAGGAGE - FLIGHT, 1	225.
UNUSABLE FUEL	343.
ENGINE OIL	83.
PASSENGER SERVICE	103.
OPERATING WEIGHT	23,076.
PASSENGERS, 8	1,320.
PASSENGER BAGGAGE	352.
ZERO FUEL WEIGHT	24,748.
MISSION FUEL	26,252.
TAKE-OFF GROSS WEIGHT	51,000.

TABLE IV-II. - MASS DATA SUMMARY

ITEM DESCRIPTION	CONDITION: TAKEOFF GROSS WEIGHT	CONDITION: NORMAL LANDING WEIGHT
WEIGHT, lbf	51,000	29,772
HORIZONTAL CENTER-OF-GRAVITY, in.	660.6	646.9
PERCENT OF $\bar{c}_{ref}$	49.8	45.7
ROLL INERTIA, slug-ft^2	3.94×10^4	2.90×10^4
PITCH INERTIA, slug-ft^2	49.38×10^4	36.88×10^4
YAW INERTIA, slug-ft^2	52.22×10^4	38.77×10^4
PRODUCT OF INERTIA, slug-ft^2	$.82 \times 10^4$	$.80 \times 10^4$
PRINCIPAL AXIS ANGLE OF INCLINATION, θ, DEG.	1.0	1.3

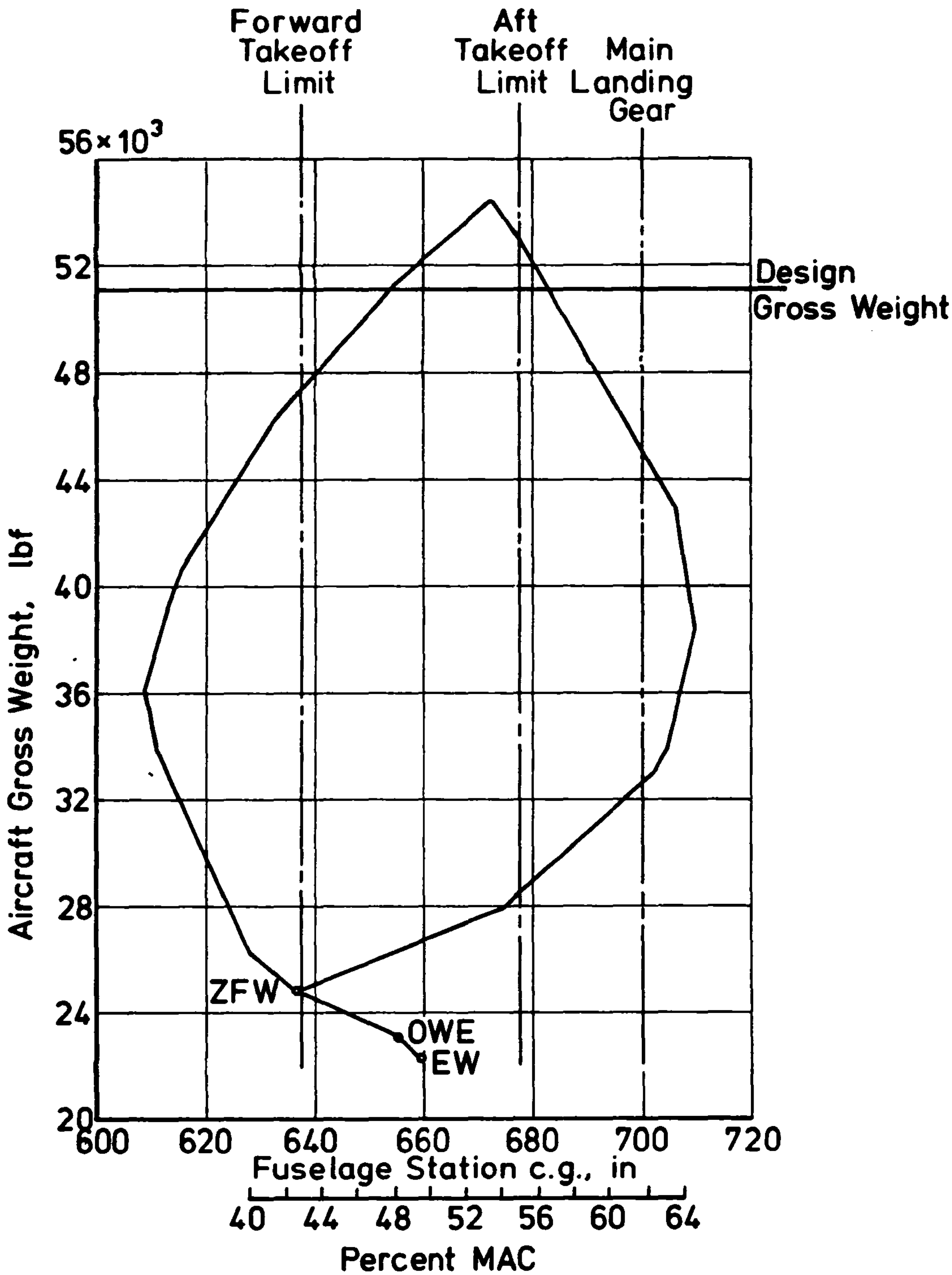


Figure IV-1. - Aircraft center-of-gravity envelope.

PART V. PERFORMANCE

F. L. Beissner, Jr.

The Boeing 701S engine, as described in the Propulsion section of this report, is a turbine-bypass turbojet engine designed to operate at Mach 2.7 and 65,000 ft cruise altitude. During this study, all supersonic missions were flown at Mach 2.3. The engines were sized for minimum ramp weight for the primary mission.

The aircraft is capable of a 3,355 n.mi. range supersonic flight (New York to Paris) when loaded to the design gross weight of 51,000 lbf. This flight performance includes taxi-out and takeoff allowances, FAA climb ($V \leq$ 250 KCAS up to 10,000 ft), optimum path climb and acceleration to cruise at Mach 2.3, a maximum end cruise altitude of 65,000 ft, and descent to destination. Reserves consist of a missed approach allowance, climb and subsonic cruise at 30,000 ft for 250 n.mi., a 30 minute hold, and descent to the alternate airport as shown in figures V-1 through V-4.

Four basic missions were evaluated using the Flight Optimization System (FLOPS) computer program described in the Appendix. The four missions are listed below and summarized in tables V-I through V-IV and figures V-1 through V-4.

- o Maximum range at Mach 2.3 cruise and 51,000 lbf gross weight.
- o New York to Los Angeles (2,130 n.mi.) at Mach 2.3, off-loading fuel to that required for the mission plus reserves.
- o Maximum range at Mach 0.9 cruise and 51,000 lbf gross weight.
- o New York to Los Angeles (2,130 n.mi.) at Mach 0.9, off-loading fuel to that required for the mission plus reserves.

The minimum weight aircraft capable of a Mach 2.3 New York to Los Angeles flight, table V-II, required fueling the aircraft to 42,730 lbf ramp weight. The sonic boom overpressure during acceleration for this case is 1.3 psf, which could be objectionable on an overland flight. Some measure of sonic boom reduction can be obtained by a minor deviation from the optimum flight path. It is accomplished by climbing higher than the optimum profile before accelerating through the transonic speed zone. A boom reduction profile resulting in an overpressure of 1.0 psf could be flown by climbing to 42,000 ft altitude before level transonic acceleration, and would require fueling the aircraft to 43,000 lbf ramp weight. The optimum profile and example alternates are shown on figure V-5. Figure V-6 shows the effect of acceleration altitude on sonic boom overpressure and ramp weight for the New York to Los Angeles flight. The low wing loading and high cruise altitude of this aircraft combine to pose no sonic boom problem on this mission.

TABLE V-I. - MISSION PERFORMANCE - SUPERSONIC RANGE AT MAXIMUM GROSS WEIGHT

MISSION: Supersonic Cruise @ Mach 2.3

MISSION SEGMENT	OPERATING WEIGHT (lbf)	ΔFUEL (lbf)	ΔRANGE (n.mi.)	ΔTIME (min.)
RAMP GROSS WEIGHT	51,000			
Warm-Up & Taxi Out		279		10
TAKEOFF GROSS WEIGHT	50,721			
Takeoff Segment		348		1
START CLIMB WEIGHT	50,373			
Climb & Accelerate		4,962	292	20
START CRUISE WEIGHT	45,411			
Cruise Segment		16,126	2,902	132
END CRUISE WEIGHT	29,285			
Descent & Decelerate		287	161	16
END DESCENT WEIGHT	28,998			
Reserve Fuel		4,250		
TOTAL FUEL		26,252		
TRIP FUEL, RANGE & TIME		21,375	3,355	168
BLOCK FUEL, RANGE & TIME		22,002	3,357	179

Reserve Fuel Breakdown	Weight (lbf)
1. Missed Approach	348
2. 250 n.mi. to Alternate Airport	2,190
3. 30 min. Hold at 30,000 ft.	1,712
TOTAL RESERVES	4,250

TABLE V-I. - Concluded.

Cruise Condition

	Begin Cruise	End Cruise
Lift/Drag	7.27	6.40
Altitude, (ft)	62,444	65,000

NOTES:

1. Taxi-in fuel weight of 279 pounds taken out of reserves at end of primary mission.

2. C.A.B. range corresponding to block time and fuel equals trip range minus traffic allowances as will be specified for supersonic aircraft.

TABLE V-II. - MISSION PERFORMANCE - NON-STOP NEW YORK TO LOS ANGELES

MISSION: Supersonic Cruise @ Mach 2.3

MISSION SEGMENT	OPERATING WEIGHT (lbf)	ΔFUEL (lbf)	ΔRANGE (n.mi.)	ΔTIME (min.)
RAMP GROSS WEIGHT	42,730			
Warm-Up & Taxi Out		279		10
TAKEOFF GROSS WEIGHT	42,452			
Takeoff Segment		348		1
START CLIMB WEIGHT	42,103			
Climb & Accelerate		3,902	246	17
START CRUISE WEIGHT	38,202			
Cruise Segment		8,917	1,722	78
END CRUISE WEIGHT	29,285			
Descent & Decelerate		287	162	16
END DESCENT WEIGHT	28,998			
Reserve Fuel		4,250		
TOTAL FUEL		17,982		
TRIP FUEL, RANGE & TIME		13,106	2,130	111
BLOCK FUEL, RANGE & TIME		13,733	2,132	122

Reserve Fuel Breakdown	Weight (lbf)
1. Missed Approach	348
2. 250 n.mi. to Alternate Airport	2,190
3. 30 min. Hold at 30,000 ft.	1,712
TOTAL RESERVES	4,250

TABLE V-II. - Concluded.

Cruise Condition

	Begin Cruise	End Cruise
Lift/Drag	7.11	6.39
Altitude, (ft)	64,966	65,000

NOTES:

1. Taxi-in fuel weight of 279 pounds taken out of reserves at end of primary mission.
2. C.A.B. range corresponding to block time and fuel equals trip range minus traffic allowances as will be specified for supersonic aircraft.

TABLE V-III. - MISSION PERFORMANCE - MAXIMUM SUBSONIC RANGE

MISSION: Subsonic Cruise @ Mach 0.90

MISSION SEGMENT	OPERATING WEIGHT (lbf)	ΔFUEL (lbf)	ΔRANGE (n.mi.)	ΔTIME (min.)
RAMP GROSS WEIGHT	51,000			
Warm-Up & Taxi Out		279		10
TAKEOFF GROSS WEIGHT	50,721			
Takeoff Segment		348		1
START CLIMB WEIGHT	50,373			
Climb & Accelerate		1,645	39	6
START CRUISE WEIGHT	48,728			
Cruise Segment		19,486	2,572	298
END CRUISE WEIGHT	29,242			
Descent & Decelerate		244	87	11
END DESCENT WEIGHT	28,998			
Reserve Fuel		4,250		
TOTAL FUEL		26,252		
TRIP FUEL, RANGE & TIME		21,375	2,698	315
BLOCK FUEL, RANGE & TIME		22,002	2,700	326

Reserve Fuel Breakdown	Weight (lbf)
1. Missed Approach	348
2. 250 n.mi. to Alternate Airport	2,190
3. 30 min. Hold at 30,000 ft.	1,712
TOTAL RESERVES	4,250

TABLE V-III. - Concluded.

Cruise Condition

	Begin Cruise	End Cruise
Lift/Drag	10.72	10.40
Altitude, (ft)	32,445	43,086

NOTES:

1. Taxi-in fuel weight of 279 pounds taken out of reserves at end of primary mission.

2. C.A.B. range corresponding to block time and fuel equals trip range minus traffic allowances as will be specified for supersonic aircraft.

TABLE V-IV. - MISSION PERFORMANCE - NON-STOP SUBSONIC NEW YORK TO LOS ANGELES

MISSION: Subsonic Cruise @ Mach 0.90

MISSION SEGMENT	OPERATING WEIGHT (lbf)	ΔFUEL (lbf)	ΔRANGE (n.mi.)	ΔTIME (min.)
RAMP GROSS WEIGHT	45,682			
Warm-Up & Taxi Out		279		10
TAKEOFF GROSS WEIGHT	45,403			
Takeoff Segment		348		1
START CLIMB WEIGHT	45,054			
Climb & Accelerate		1,466	37	5
START CRUISE WEIGHT	43,589			
Cruise Segment		14,346	2,006	233
END CRUISE WEIGHT	29,242			
Descent & Decelerate		244	87	11
END DESCENT WEIGHT	28,998			
Reserve Fuel		4,250		
TOTAL FUEL		20,934		
TRIP FUEL, RANGE & TIME		16,056	2,130	249
BLOCK FUEL, RANGE & TIME		16,683	2,132	260

Reserve Fuel Breakdown	Weight (lbf)
1. Missed Approach	348
2. 250 n.mi. to Alternate Airport	2,190
3. 30 min. Hold at 30,000 ft.	1,712
TOTAL RESERVES	4,250

TABLE V-IV. - Concluded.

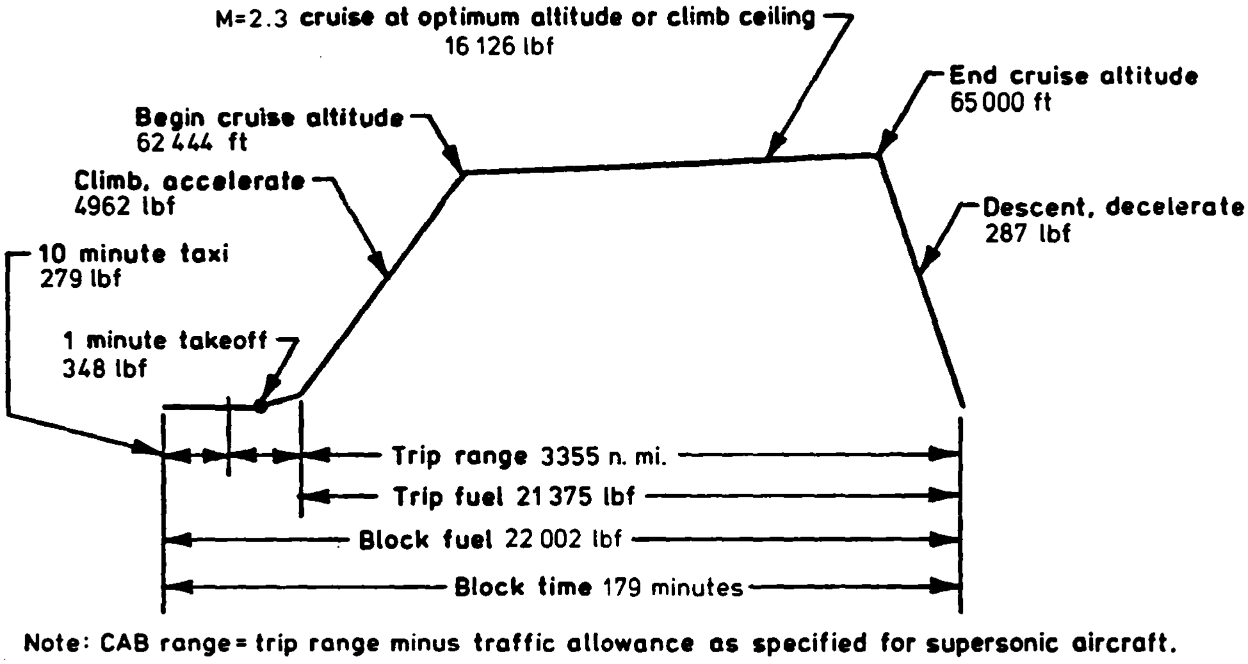
Cruise Condition

	Begin Cruise	End Cruise
Lift/Drag	10.65	10.40
Altitude, (ft)	34,829	43,086

NOTES:

1. Taxi-in fuel weight of 279 pounds taken out of reserves at end of primary mission.

2. C.A.B. range corresponding to block time and fuel equals trip range minus traffic allowances as will be specified for supersonic aircraft.



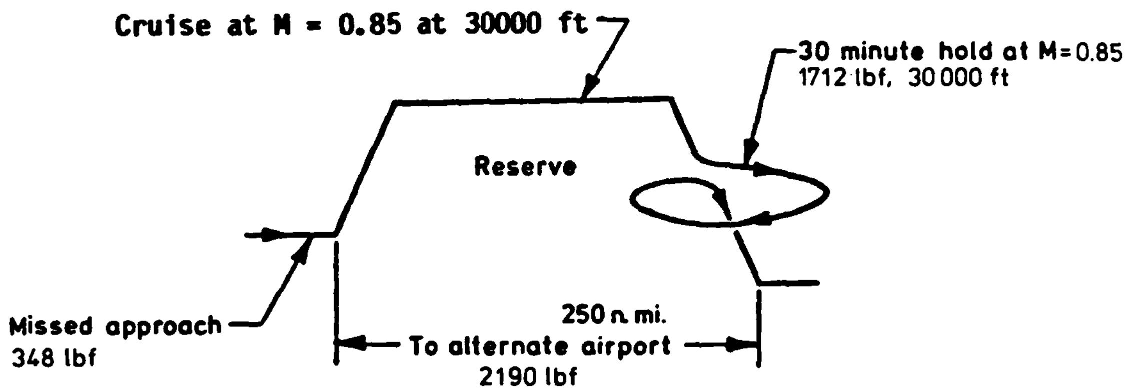


Figure V-1. - Design mission flight profile, supersonic cruise and maximum range, standard day, no-wind conditions.

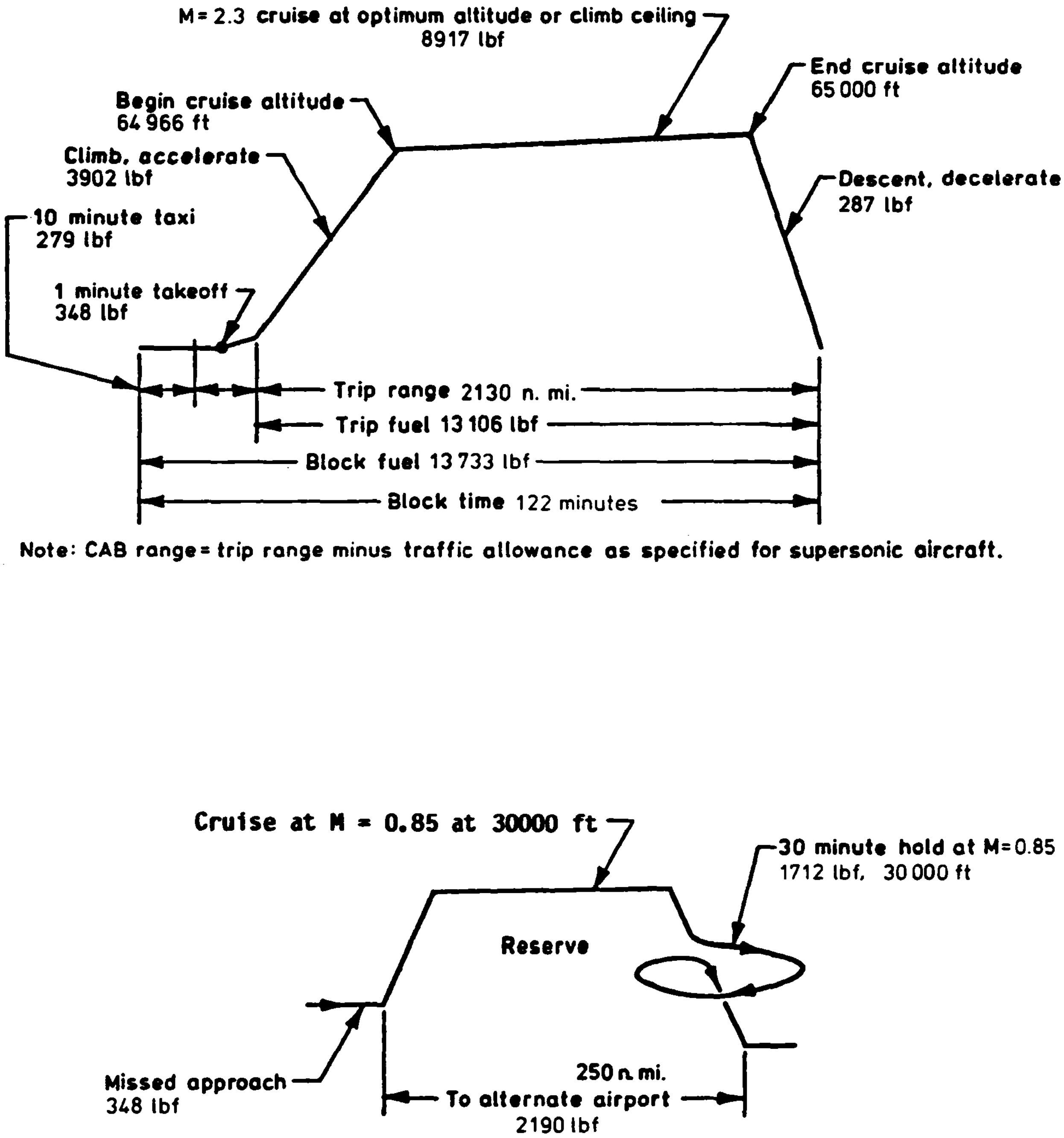


Figure V-2. - Mission profile, New York to Los Angeles at supersonic cruise, standard day, no-wind conditions.

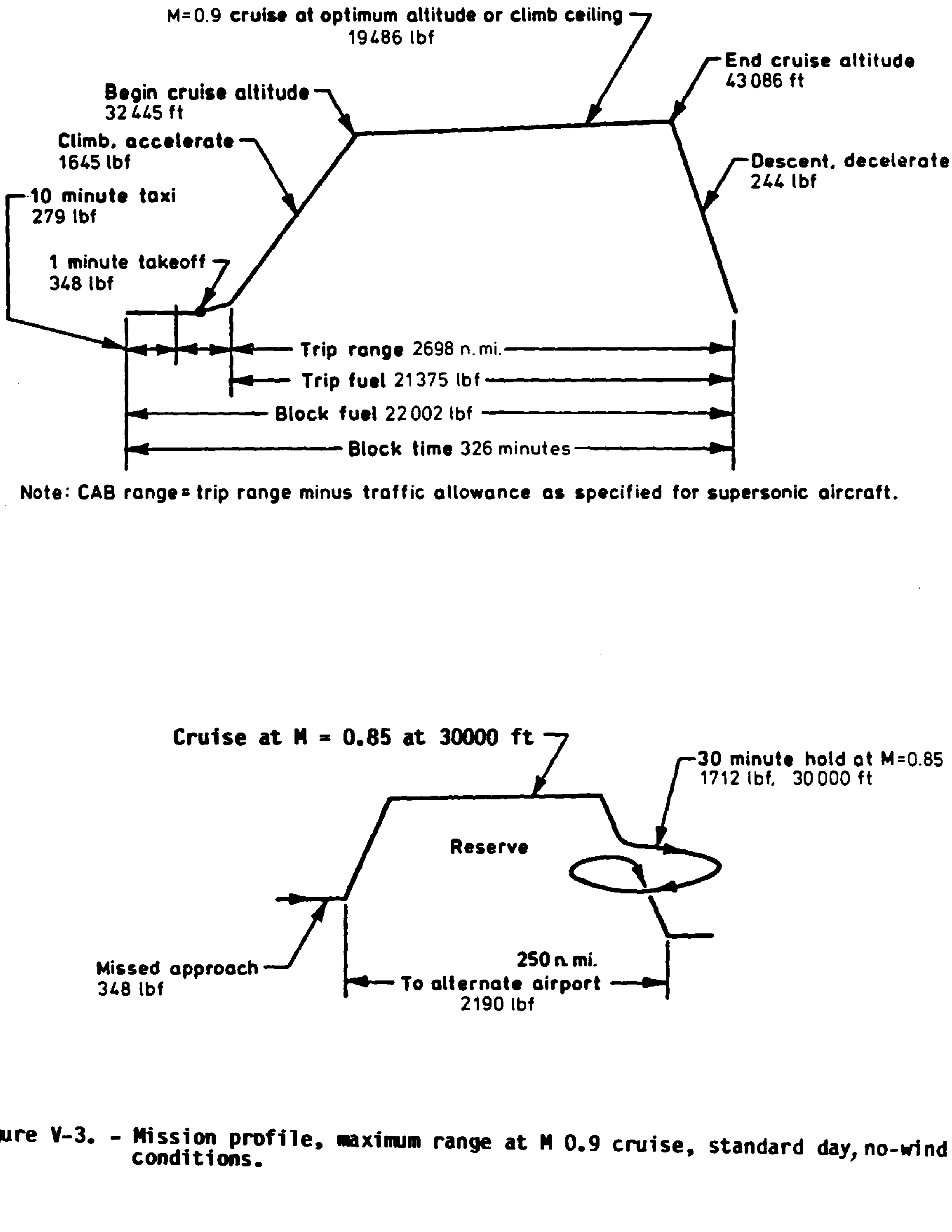


Figure V-3. - Mission profile, maximum range at M 0.9 cruise, standard day, no-wind conditions.

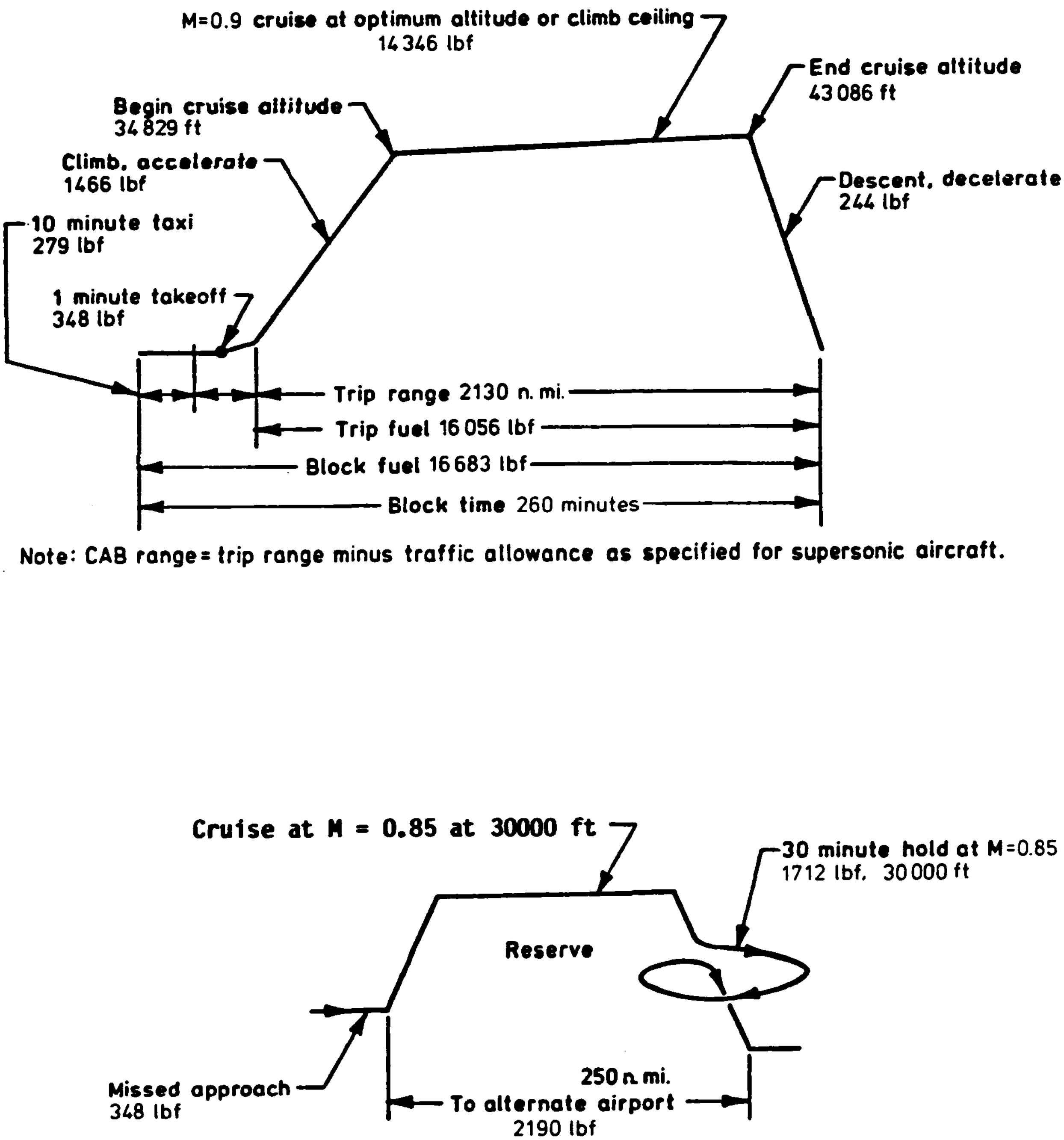


Figure V-4. - Mission profile, New York to Los Angeles at M 0.9 cruise, standard day, no-wind conditions.

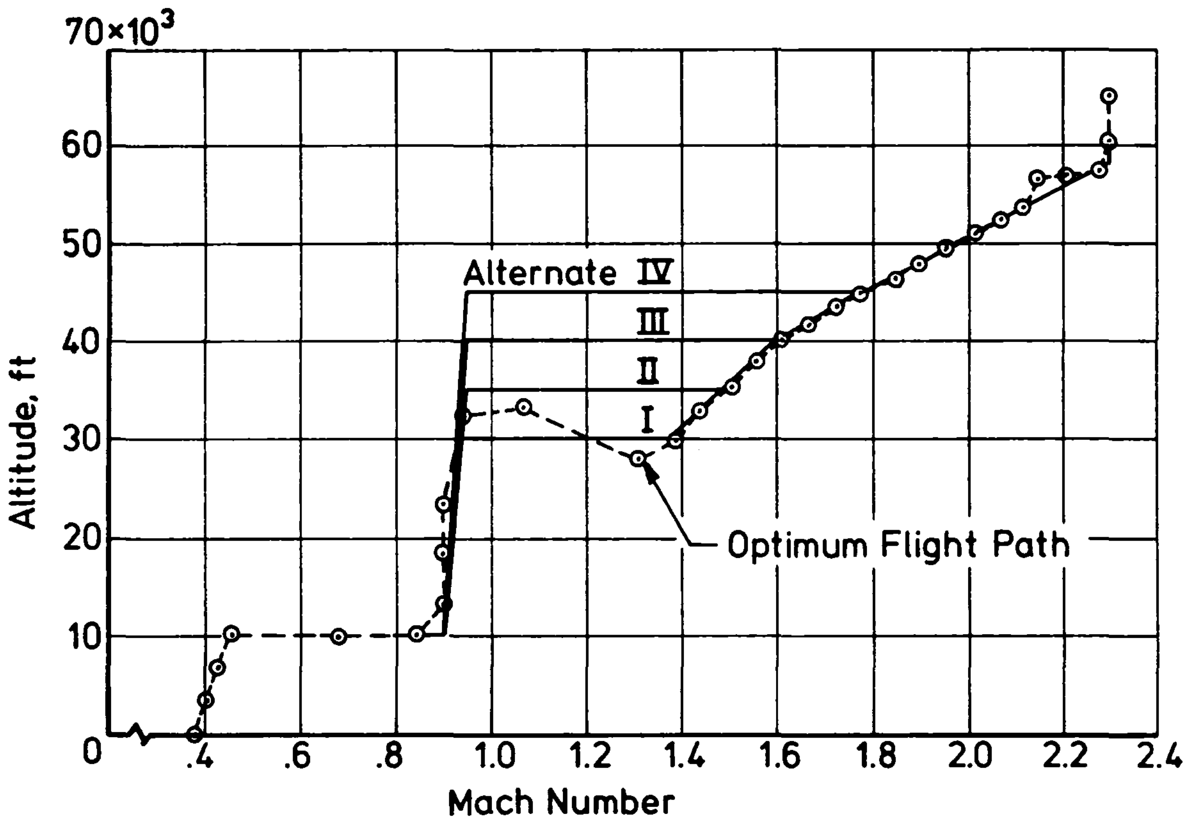


Figure V-5. - Alternate climb and accelerate profile, New York to Los Angeles flight.

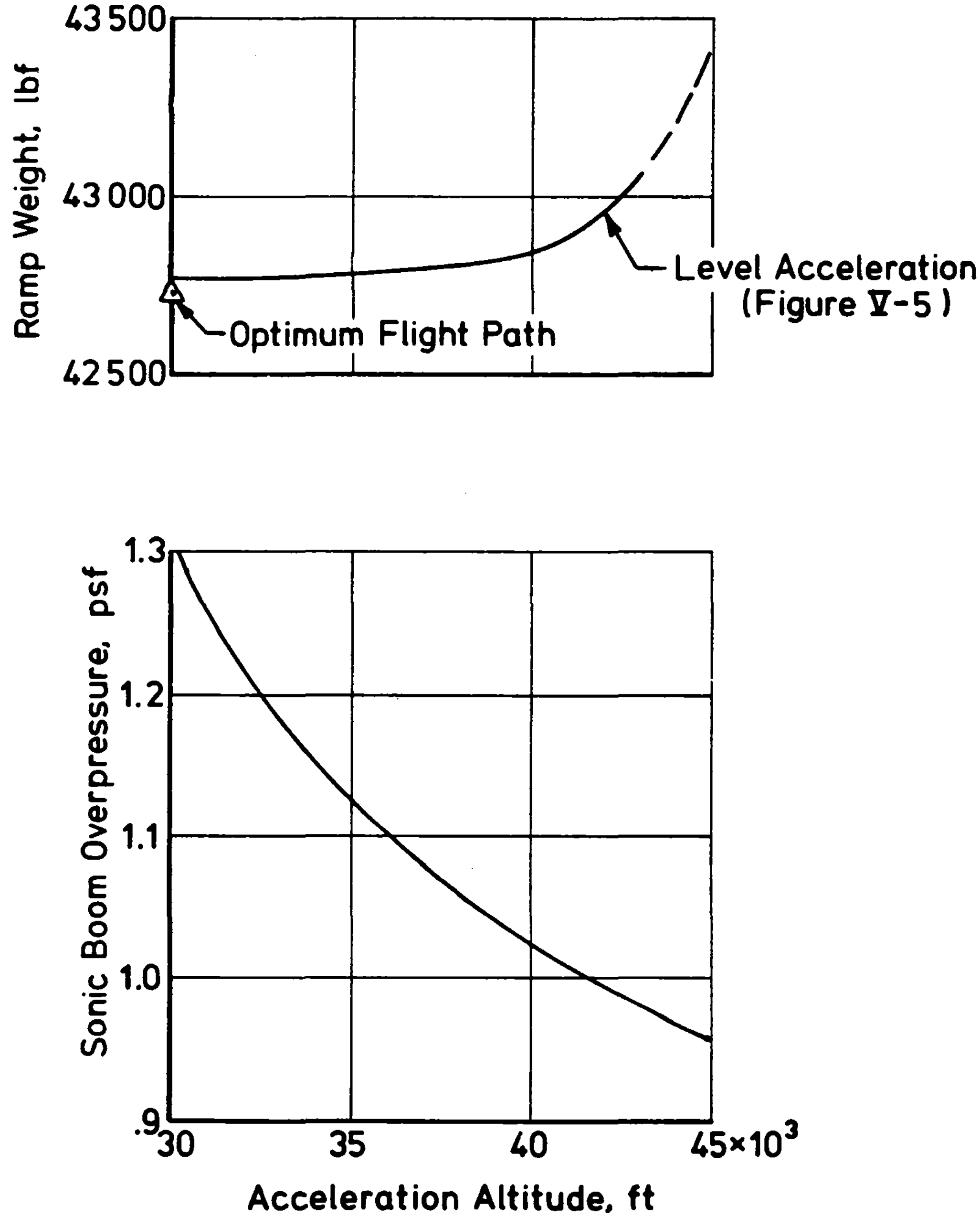


Figure V-6. - Effect of acceleration altitude on sonic boom overpressure and ramp weight at M 1.2.

APPENDIX

Flight Optimization System (FLOPS)

The Flight Optimization System (FLOPS) is a multidisciplinary system of computer programs for conceptual and preliminary design and evaluation of advanced aircraft concepts. It consists of four primary modules: 1) weights, 2) aerodynamics, 3) mission performance, and 4) takeoff and landing.

The weights module uses statistical and empirical equations to predict the weight of each item in a group weight statement. Centers of gravity and moments of inertia can also be calculated for multiple fuel conditions.

The aerodynamics module uses a version of the EDET (Empirical Drag Estimation Technique) (ref 1) program to provide drag polars for performance calculations. Alternatively, drag polars may be input and then scaled with variations in wing area and engine (nacelle) size.

The mission performance module uses the calculated weights and aerodynamics data and an engine deck to calculate performance. The engine deck consists of thrust and fuel flow data at a variety of Mach-altitude-power setting conditions. Based on energy considerations, an optimum climb profile may be flown to start of cruise conditions. The cruise segment may be flown at the optimum altitude for maximum range or at the optimum Mach number for maximum endurance. Reserve calculations include flight to an alternate airport and a specified hold segment.

The takeoff and landing module computes the all-engine takeoff field length, the balanced field length including one-engine-out takeoff and aborted takeoff, and the landing field length. The approach speed is also calculated, and the second-segment climb gradient and the missed approach climb gradient criteria are evaluated.

FLOPS may be used to analyze a point design, parametrically vary certain design variables, or optimize a configuration with respect to these design variables (for minimum gross weight or minimum fuel burned) using nonlinear programming techniques. The available design variables are wing area, wing sweep, wing aspect ratio, wing taper ratio, wing thickness-chord ratio, gross weight, thrust (size of engine), cruise Mach number, and maximum cruise altitude.

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A-1. Feagin, Richard C.; and Morrison, William D., Jr.: Delta Method, An Empirical Drag Buildup Technique. NASA CR-15171, December 1978.

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
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16. Abstract

An analytical study of a supersonic-cruise, executive, jet aircraft has indicated the effects of using advanced technology. The twin-engine, arrow-wing vehicle was configured with a cabin of minimum practical size to hold one pilot, eight passengers, and their baggage. The primary differences between this configuration and that of a previous report were the reduction in cabin size and the use of engines that are more fuel-efficient. Both conceptual vehicles are capable of performing the same mission. The current vehicle has a range of 3,350 nautical miles at Mach 2.3 cruise and 2,700 nautical miles at Mach 0.9. The concept description includes configuration definition, aerodynamic and propulsion-system characteristics, and mass properties.

Performance analyses are documented for intercontinental and transcontinental flight profiles. In the latter case, a reduction in sonic-boom overpressure from 1.3 to 1.0 pounds per square foot was achieved by varying the flight profile slightly from that for optimum performance.

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