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THE BOEING COMPANY

COMMERCIAL AIRPLANE DIVISION

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TITLE: EFFECT OF SONIC BOOM REQUIREMENTS ON AIRPLANE SIZING AND ECONOMICS

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MODEL ______ CONTRACT ______

ISSUE NO. 8 ISSUED TO: FAA

PREPARED BY J D Vachal 11/12/66

SUPERVISED BY R J Markey 11/12/66

APPROVED BY E Kowal 11/12/66

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REVISIONS

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FOREWORD

In a letter dated October 14, 1966, Reference SS-100 to Mr. M. L. Pennell, Major General J. C. Maxwell, USAF, Director, Supersonic Transport Development, asked that "certain additional investigations be completed in order to improve the SST Program analytical base." The purpose of this document is to satisfy that request.

Documentation was requested on the following important program issues:

1. If sonic boom considerations were completely removed and the SST design fully optimized for overwater operations, could significantly better economics be achieved?

   "In your conduct of such a parametric study, it is important that you re-establish the fundamental aircraft characteristics required for economic optimization in the total absence of sonic boom restrictions. It would not serve our purpose adequately if you simply determined how your existing design could be flown for optimum economics, or how it could be redesigned to approach interim economic optimization. Your result should evolve from a completely fresh approach to the parametric design solution, based on your current experience."

2. "A second program issue is the practicability of developing a domestic SST which would have an "acceptable" boom level and could be operated profitably over inhabited areas. For the purpose of discussion, this criterion might indicate a maximum overpressure of 1.0 - 1.2 lbs/sq. ft. in cruise, based on the realistic atmosphere (Friedman method). As in the preceding study, we should like you to consider the widest range of design possibilities.

   That is, can there be a combination of size, weight, shape, Mach number, altitude, engine configuration and the like which could reduce overpressures significantly and still indicate profitable operation at ranges no greater than required in U.S. domestic routes? Your investigation might also consider, for example, whether there is any possibility of combining weight, altitude, and speed to fly within the boom-cutoff conditions."

3. "A third program issue is the status of any developments, pending or conceivable, within the subsonic transport domain which suggest significantly greater performance, improved service, or reduced operating costs. Possibilities might range from application of advanced materials and engine technology to transonic tailoring, leading into the "mildly supersonic" condition within the boom-cutoff regime."

It was requested that the depth of the analysis be consistent with completion of the parametric studies by November 14, 1966, and that a preliminary review of the results be given on October 28, 1966.

At the preliminary review on October 28, General Maxwell requested that no further work be done on questions 1 and 3 above, and that emphasis be given to expanding the study of question 2. Accordingly, this document summarizes the October 28 data relative to questions 1 and 3 and the subsequent work relative to question 2.

TABLE OF CONTENTS

1.0 SUMMARY

A brief parametric study of the effect of sonic boom constraints on the design of the supersonic transport has been completed. For simplification, the economic effects have been evaluated in terms of Direct Operating Cost, computed in accordance with the "Supersonic Transport Economic Model Ground Rules", SST 66-3, June 30, 1966. This simplification assumes equal marketability for all airplanes, and thus only provides a technical figure of merit. A discussion of this limitation is presented in paragraph 1.2.

The major portion of the study is concerned with low sonic boom, domestic supersonic transports and the probable competing subsonic transports in the 1974 time period. This is summarized in paragraph 1.1. The study assumptions for level of technology in aerodynamics, propulsion, and weights for the domestic supersonic transports is given in the Appendix and the study details are given in Section 3. The corresponding assumption and study results for 1974 subsonic transports are given in Section 4.

Several very preliminary base-point configurations have been evaluated in order to establish the overall level of technology used in the study. The objective has been to use optimistic, but potentially attainable characteristics, in order to establish a rough lower bound for Direct Operating Costs. A simple evaluation of the sensitivity to assumptions is shown in Section 3 for the M = 2.7 airplane designed for a maximum sonic boom overpressure of 1.2 psf.

A further cautionary note is to remind the reader of the inherent limitations of any parametric airplane design study. Several cycles

in the study are necessary with intermediate configuration evaluations in order to achieve a reasonable level of confidence in the results. A proper design evaluation has not been completed in the present study because of the imposed time limitations. A cursory comparison of the results with the rough, base-point configurations indicates that the M = 2.7 airplanes designed for sonic boom overpressure of 1.2 psf or greater are optimistically attainable; the smaller airplanes for lower sonic boom are too optimistic, requiring changes such as fuel volume increases with corresponding degradation of drag and weight.

The parametric examination of the potential gains to be realized through elimination of sonic boom constraints on the intercontinental supersonic transport is summarized in paragraph 1.3 and discussed in Section 2.

1.1 Domestic Airplane

The Direct Operating Costs for the domestic transports considered in this study are summarized in Figure 1.1. The airplane characteristics are tabulated in Table 1A.

Supersonic transports limited to maximum sonic boom overpressures of 1.2 psf are evaluated to have DOC's about 60 percent greater than subsonic transports in the 1974 time period. Relaxing the sonic boom requirement to 1.5 psf shows DOC's 35 percent greater. The corresponding DOC's for the Model 747 transport, a potential M = 1.2 transport, the Concorde, and the domestic B-2707 are shown for comparison.

The study indicates that the choice of supersonic design Mach number has an effect on DOC. Temperature effects on fuel, systems, and structure cause an increase at speeds greater than 2.7 Mach number.

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The higher DOC's near a Mach number of 2.0 are caused by the cruise sonic boom limit. Optimum cruise altitudes increase with increasing Mach number, providing sonic boom alleviation at higher speeds.

All of the airplanes shown in Figure 1.1 are of titanium construction and are powered with study engines which are variations of the GE4/J5P. The effects of assuming aluminum structure and a higher pressure ratio engine, comparable to the Bristol BS 593, were individually assessed at a Mach number of 2.2. These are shown in Table 1A. The degradation in weight ratio using aluminum offset the advantage of cheaper construction. The greater engine weight of the higher pressure ratio engine more than offsets the fuel saving at the design range of 2500 nautical miles.

It should be noted that the sonic boom values quoted are the nominal maximum values in climb. A super boom region with peak values of twice the nominal will exist at the beginning of acceleration to cruise for all of the SST's.

1.2 <u>Effects of Airplane Marketability and Operational Factors on Direct Operating Costs of the Domestic Study Airplanes</u>

On the assumption that each of the study airplanes has equal marketability, both from the airlines and the manufacturers viewpoint, it can be seen in Figure 1.1 that DOC's of subsonic and transonic airplanes can be 30% to 60% less than supersonic airplanes designed for low sonic boom.

In order to develop a true perspective, appraisal of the factors that are not equal must be undertaken.

D.O.C. COMPARISON

PHASE IIC RULES
DOMESTIC (80% T 20% F)
JFK - SFO

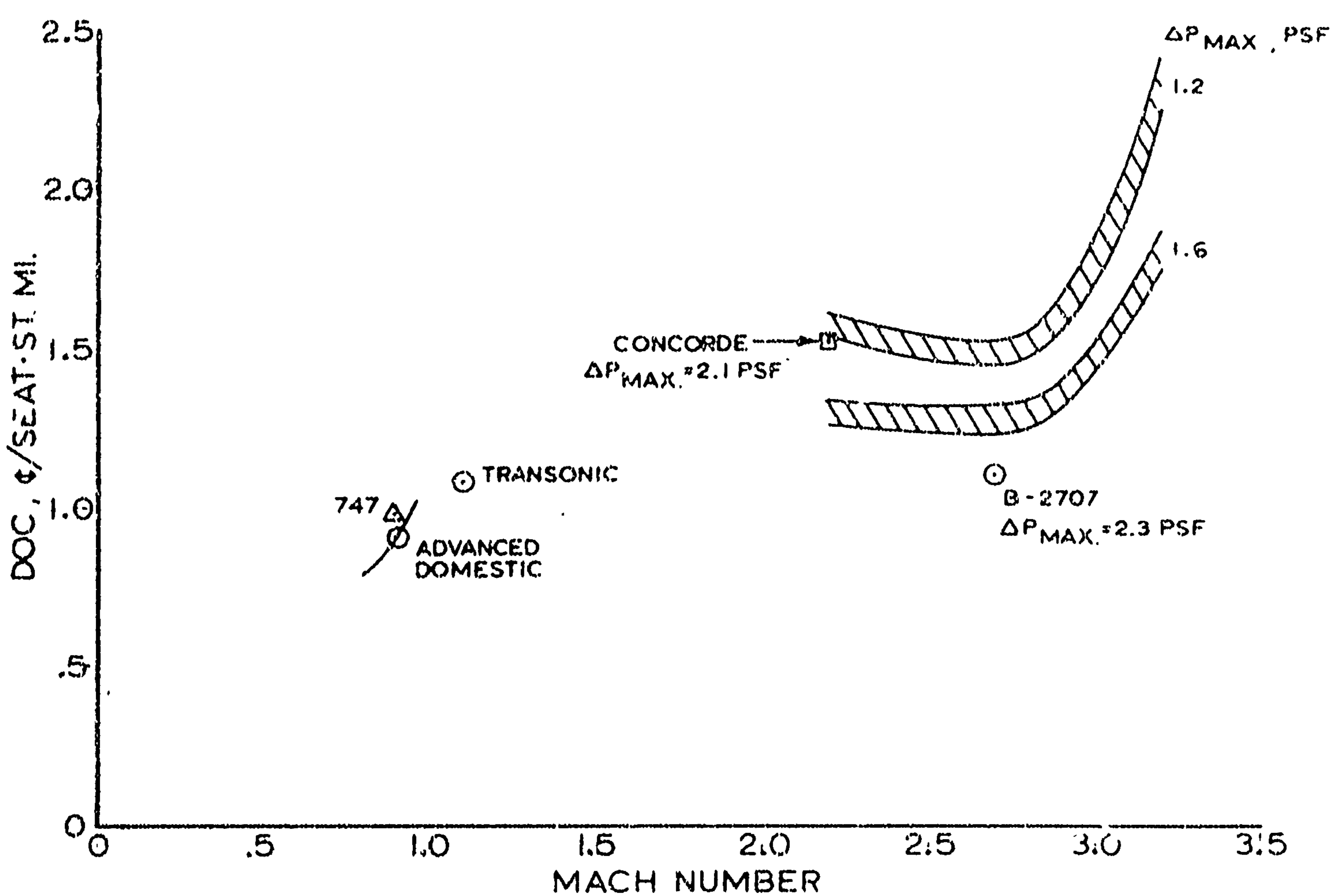

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Fig 1-1

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TABLE IA

SUMMARY
DOMESTIC AIRPLANES

	747	755-300 (Advanced Subsonic)	742-240	Parametric Airplanes				
Cruise Mach No.	.89	.90	1.05-1.15	2.2	2.2	2.2	2.7	3.2
* $\Delta P_{Climb}/\Delta P_{Cruise}$, PSF	None	None	None	1.2/1.2	1.2/1.2	1.2/1.2	1.2/1.2	1.2/1.04
Gross Weight, Lb.	546,000 P	350,000	434,800	100,000	90,000	168,000	2[illegible]5,000	200,000
Range, N.Mi.	2,400 P	2,400	2,400	2,500	2,500	2,500	2,500	2,500
Number of Passengers	374	261	261	42	40	73	100	77
Airplane Price, $M (Including Development)	20.1	13.7	20.1	7.8	7.6	10.7	16.2	16.4
D.O.C., ¢/Seat S.Mi.	.98	.91	1.08	1.80	1.89	1.53	1.44	2.24
Operating Wt. Empty, Lb.	334,400	198,400	238,500	49,250	41,400	70,450	104,750	92,500
Airframe Material	Aluminum	Aluminum and Titanium	Aluminum and Titanium	Aluminum	Titanium	Titanium	Titanium	Titanium
Takeoff Wing Loading, Lb/Ft^2	124	130	110	70	68	65.5	65	65
$(L/D)_{Max}$	16.6	15.2	12.8	8.8	8.7	8.7	8.6	7.5
Engine Type	JT9D-1	JT9D-1	.6 Bypass Ratio	High Press. Ratio Turbojet	High Press. Ratio Turbojet	Med. Press. Ratio Turbojet	Med. Press. Ratio Turbojet	Med. Press. Ratio Turbojet
Engine Airflow, Lb/Sec	1,444	1,444	789	250	210	310	320	500
Number of Engines	4	3	3	2	2	2	2	2
Static Thrust, Lb.	41,000	41,000	49,500	19,250	16,000	22,250	52,000	50,000

* Calculated using near-field theory corrected to U.S. Standard Atmosphere, 1962.

P Max. Gross Weight = 683,000 lb., Max. Range = 4600 N. Mi.

One major factor is that from the passenger viewpoint, enroute times and departure schedules would not be equal. Therefore, the trip value to the travelers would vary. From the airline view this increased value of the SST could be represented by an increased revenue potential which could be as much as .4 cents per seat miles at Mach 2.7 over the subsonic airplane. Even the transonic airplane would have some advantage. A factor which would tend to offset this revenue advantage of the SST is that the higher Mach number airplanes would require a greater investment per unit of productivity. On the airplanes limited to sonic booms of 1.2 psf, this could add as much as .2¢/seat mile to the costs of the Mach 2.7 airplane. Also start-up and introductory costs could add .05¢/seat mile.

In the case of the supersonic airplanes, some boom damage claims could be expected. Additional costs of .05¢/seat mile could be experienced.

The DOC effect of variations in productivity due to size, trip times, and utilization requiring different numbers of aircraft of each type would be small. It is estimated that the effect of increased speed on reducing indirect costs of passenger service in the supersonic airplanes would be offset by the increase due to landing fees and aircraft servicing.

In summary, because of the large difference in DOC between the subsonic airplanes and the most optimistic estimate for the supersonic airplanes, there is only a small probability of there being a market for supersonic airplanes of this quality.

1.3 International SST Optimized Without Sonic Boom Limitations

The elimination of sonic boom constraints on the design of a long range supersonic transport would allow the use of higher wing loadings and lower thrust loadings than have been selected for the B-2707 proposal airplane. These changes would improve the economics through a reduced gross weight for a given payload-range design choice.

An example of the potential improvement is shown in Table 1B. The characteristics of an 825,000 pound airplane, using the same engine size and wing area as the B-2707, are compared to those of the B-2707. The range increase of 500 nautical miles for hot day conditions would allow non-stop operation between more city pairs, thus increasing the utility of the airplane. Conversely, the payload could be increased at less range by designing a larger body, thus improving economics.

A review of the design decisions leading to the B-2707 has shown that no specific compromises were made for sonic boom. The philosophy of using variable sweep to meet low speed performance objectives has allowed the configuration to be optimized for cruise with a highly swept planform and aerodynamically contoured body; this simultaneously provides a configuration with optimum sonic boom characteristics. The elimination of sonic boom limits allows this configuration concept to be more fully exploited. The increases in wing loading and thrust loading which could be made would require the use of variable-sweep to achieve acceptable low speed performance. This trend is identical to that which has evolved in the design of subsonic transports where very powerful high-lift systems have been developed so that cruise-optimized airplanes could achieve the required low speed performance.

INTERNATIONAL SST PERFORMANCE SUMMARY* TABLE 1-B

- Reference Wing Area = 9,000 sq. ft.
- GE4/J5P Engine Airflow = 633 lb/sec.
- M_{MO} Climb Schedule (see Figure 2.3)

		B-2707		Parametric	
Maximum Taxi Weight - Lb.		675,000		825,000	
Operating Empty Weight - Lb.		287,500		330,590	
Payload - Lb.		50,000		50,000	
Range and Transonic Thrust Margin:		Range - N.Mi.	$\left(\frac{T-D}{D}\right)_{Min.}$	Range - N.Mi.	$\left(\frac{T-D}{D}\right)_{Min.}$
Standard Day, Cruise M = 2.7		3927	.73	4480	.58
Std. +10°C Day Cruise, M = 2.61; Std. +15°C Climb		3650	.54	4160	.45
Takeoff	Flap Setting, Deg.	20/40		20/40	
Max. Design	Thrust Setting	Max. Aug.		Max. Aug.	
Taxi Weight	F.A.R. Field Length, Ft.	5,700		8,900	
S.L. Std. Day	Lift Off Speed, Knots	162		180	
C.G. @ .615 C_R	Airport Noise, PNdb	117.5		117.5	
	Community Noise, PNdb	93		111	
Std. +15°C	F.A.R. Field Length, Ft.	6,800		10,500	
Landing	Normal Landing Wt., Lb.	384,500		434,000	
S.L. Std. Day	Flap Setting	30/50		30/50	
Dry Runway	F.A.R. Field Length, Ft.	5,800		6,340	
C.G. @ .615 C_R	Approach Speed, Knots	125		133	
	Approach Noise, PNdb	104		107	

* Based on September 6, 1966 Proposal Inputs

2.0 INTERNATIONAL SST OPTIMIZED IN THE ABSENCE OF SONIC BOOM RESTRICTIONS

2.1 Design Approach

This section presents parametric data on fundamental aircraft characteristics required for economic optimization in the total absence of sonic boom overpressure restrictions. It is recalled that fundamentally to minimize airplane D.O.C., the payload to gross weight ratio must be as high as possible. In order to obtain a high payload to gross weight ratio, airplane economics are maximized at a given design range by:

- developing a dense configuration with a high wing loading
- high gross weight, limited only by technology

Use of this design approach leads to:

- an airplane optimized for cruise performance
- use of variable sweep and high lift systems in order to meet low speed and community noise objectives
- avoidance of sonic boom limitations

The starting point for the parametric study was to select a cruise optimized wing-body configuration as a base point. A wing planform with a subsonic leading edge at supersonic cruise Mach numbers was selected because its use results in the lowest drag-due-to-lift and, therefore, the highest cruise lift-drag ratios. This planform in combination with a well-tailored body was selected in order to minimize cruise drag. Although not pertinent for the purpose of this study, use of these design features also results in a configuration with optimized sonic boom characteristics.

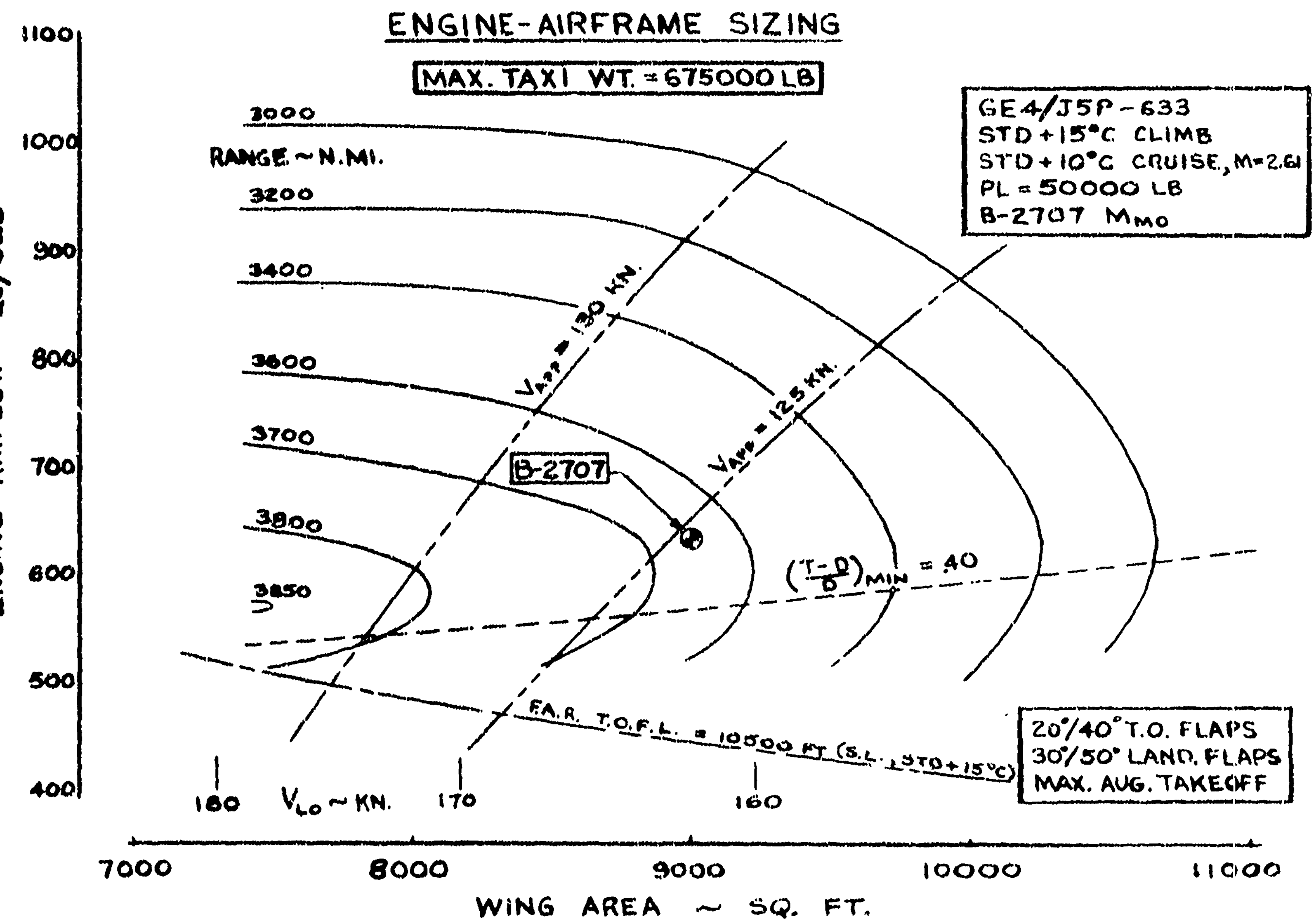
Use of the variable-sweep feature provides the capability to meet low speed performance and community noise objectives without compromising cruise efficiency. Thus, the configuration selection for the parametric study was in no way constrained for sonic boom. Based upon the cruise configuration selection, aerodynamic and structural data were generated. These inputs, in combination with the General Electric GE4/J5P engine characteristics, formed the technology basis for the parametric study. The technology basis is consistent with the level achieved in design of the B2707 and thus has a firm base point.

The choice of climb schedule for least fuel penalty in climb and acceleration as well as structural weight effects on overall performance has been examined and is discussed in Paragraph 2.3.

2.2 Engine-Airframe Sizing

The effect of parametrically varying the wing area and engine size on range, transonic thrust margin and low speed characteristics for a 675,000 pound gross weight airplane with fixed payload is shown in Figure 2.1. Figure 2.1 is sometimes called a "thumbprint" because of the characteristic shape of the range contours thus produced. The range contours can be explained by considering the range trade with wing area (for a constant powerplant size) and vice versa.

Consider first the range trade versus wing area. The lift-drag ratio improves with increases in wing area because the wetted area ratio improves. However, wing weight also increases as area increases which is adverse, so that a range maximum will occur at some wing area. Next, consider the range trade with increases in powerplant



D. BENNETT 10-25-66

ENGINE-AIRFRAME SIZING
675,000 LB. AIRPLANE

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Fig. 2-1

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size; this involves a trade between powerplant weight (which increases with increasing airflow) and improved fuel consumption which leads to a range optimization.

Other trades such as low speed performance and transonic thrust margin are also shown. For a constant takeoff field length, for example, as wing area increases, the power plant size can be reduced as shown in Figure 2.1.

A standard + 15°C day for climb and a standard + 10°C day for cruise was the basis for the range calculations which are representative of airline operation. The climb schedule followed in all cases was the schedule labeled "1" as shown in Figure 2.3.

The wing area of the B-2707 of 9000 square feet, and the airflow of the offered GE4/J5P engine is 633 lb/sec. This point is noted in Figure 2.1 The range loss relative to the maximum obtainable is about 200 miles Notice that to obtain maximum range, the powerplant size would be about 575 lb/sec. and the wing area would be 7500 square feet. This confirms that for maximum range a high wing loading and low takeoff thrust to weight ratio is desirable, provided all other practical design requirements and considerations can be met concurrently. As noted in Figure 2.1, takeoff field length and approach speed would increase but would be within the objectives if the wing area and engine size were selected for maximum range.

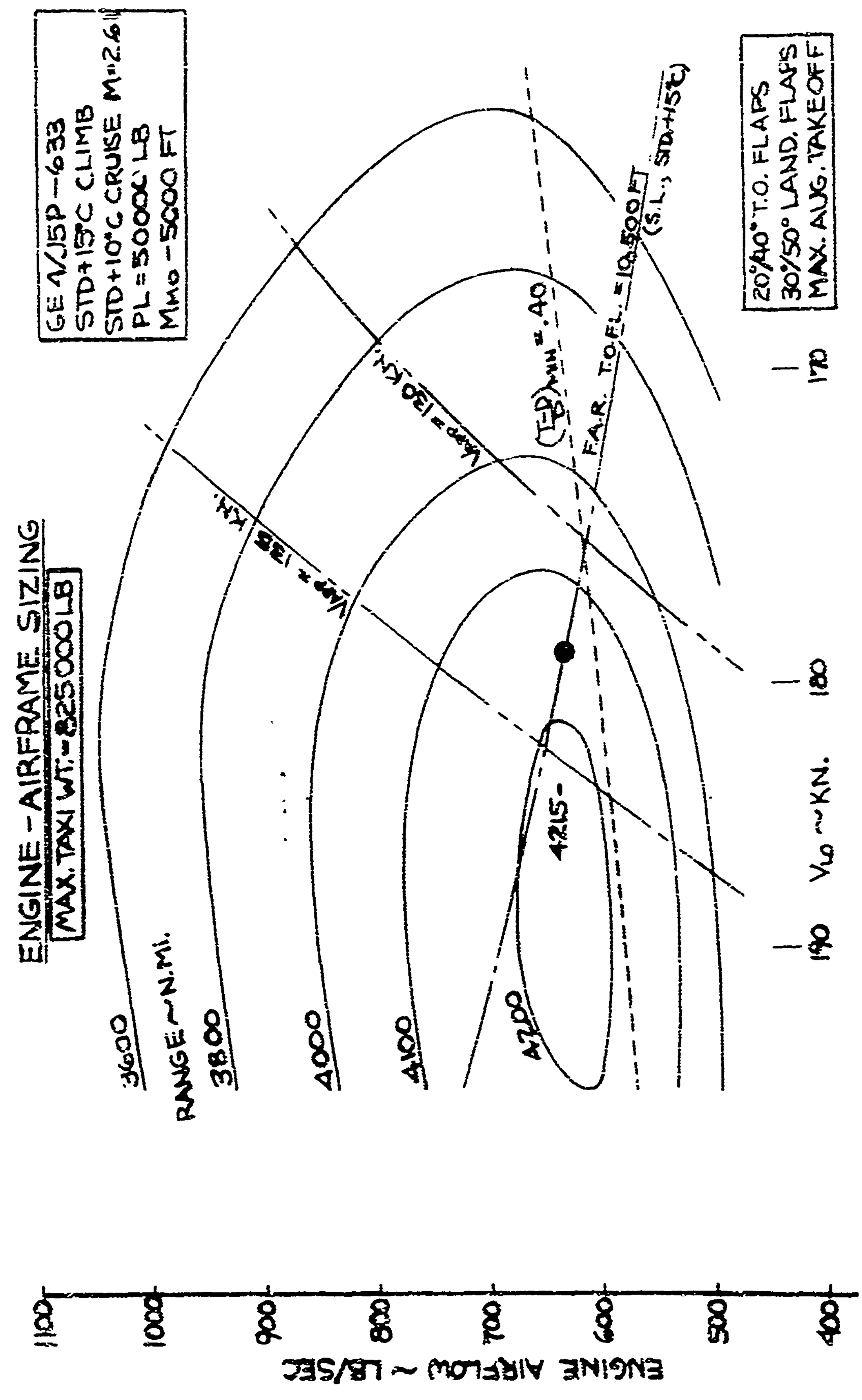
It is similarly true that more range can be achieved by increasing the gross weight for the conditions shown, i.e., where the offered powerplant size and wing area are oversized. To further clarify this point, a further example, given in Figure 2.2 is presented which shows the effect on range performance of increasing the gross weight to 825,000 pounds. The wing-body configuration selected for this study was basically the same as for the 675,000 pound airplane study except it had slight changes in order to get more fuel volume. The wing was thickened slightly and the body volume was increased for this purpose.

As expected, with the 9000 square foot wing and 633 lb/sec engine, the maximum range is increased substantially (500 nautical miles) and the takeoff field length is increased to 10,500 feet (on a hot day) and the approach speed increased to about 133 knots. As discussed above, this range increase is due to holding the wing area constant at 9000 square feet and the engine size constant at 633 lb/sec resulting in a higher wing loading and lower thrust to weight ratio for the 825,000 pound airplane. Because the 825,000 pound airplane has 500 miles greater range, it would be able to provide service to more city pairs and would therefore produce a greater return of revenue. Alternatively, the body could be redesigned to provide greater payload at less range. This example illustrates how performance and, thus, economics, could be improved with no sonic boom restrictions.

A different climb schedule was used for the 825,000 pound airplane and it is shown in Figure 2.3. This schedule was a preliminary selection prior to completion of the Placard Study discussed in Section 2.3 but is representative from a trend standpoint.

2.3 Placard Study

Determination of the climb schedule yielding maximum range capability



G. ECKARD 10-25-66

ENGINE-AIRFRAME SIZING
825,000 LB. AIRPLANE

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Fig. 2-2

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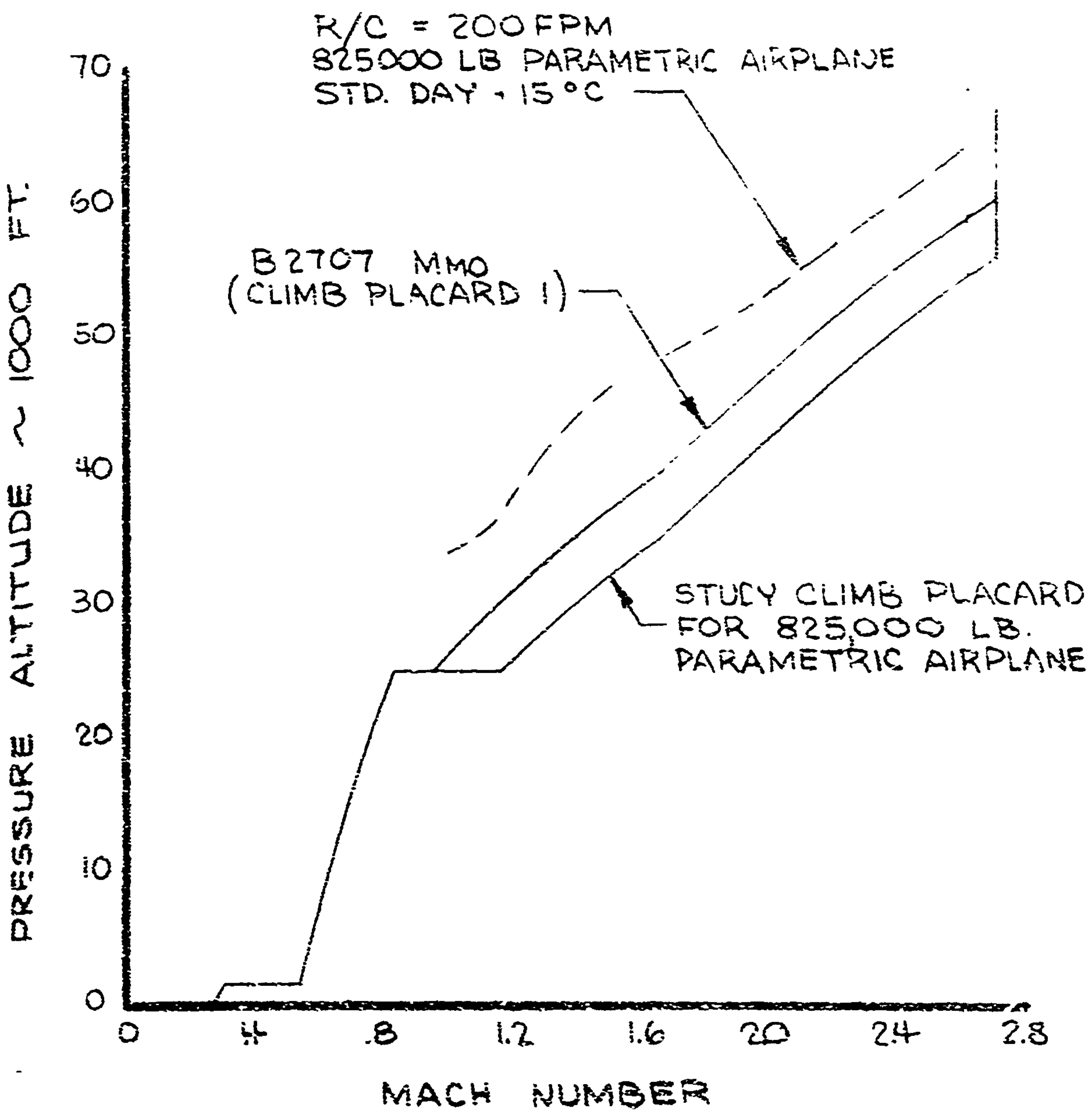
CLIMB SCHEDULES

GE 4/JP5 - 633 ENGINES



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CLIMB SCHEDULES

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Fig. 2-3

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involves a careful study of possible paths with respect to structural and flutter considerations as well as fuel saved during climb. Classical climb path studies have been calculated in the past showing that for least range loss a fairly high dynamic pressure ("q") schedule must be followed in order to minimize the fuel burned. Basically, the minimum fuel climb path corresponds to the altitude schedule for maximum range factor and maximum thrust margin in climb. This type of schedule generally is contrary to the requirements for least structural weight. Transonic and supersonic "q", flutter and upset maneuvers usually determine the design loading conditions and, hence, the amount of structure required for a given schedule.

Placard studies have been carried out on the B-2707 for gross weights of 675,000 pounds and 825,000 pounds. Nine climb paths were selected based upon past experience with trade studies of this type. Altitude and Mach numbers for the study placards are given in Table 2-A. Schedule Number 1 is the highest "q" placard which can be used based on the strength and stiffness which are provided in the structure from [illegible] er critical design points in the flight envelope such as maximum gross weight takeoff, landing loads, cruise, etc. Increases in structural weight are necessary as various parts of the placard become design points with placard changes to higher "q". For this study, the M = 1.2 and 2.7 altitude reference points were used. The net changes in range considering both structural weight and fuel usage are given in Table 2-A. The range tabulation shows that the schedule selected for the B-2707 at 675,000 pound taxi weight results in maximum range; hence, may be considered the optimum. At 825,000 pounds

TABLE 2-A

B-2707 M_{MO} STUDY

GE4/J5P ENGINES W_a = 633 LB/SEC.

S_{REF} = 9000 FT^2

STD. DAY M = 2.7 CRUISE

M_{MO} CLIMB PLACARD	ALTITUDE AT M = 1.2	ALTITUDE AT M = 2.7	△ OEW - LB.	△ RANGE FROM B-2707 M_{MO} - NAUT. MI.	
				MAX. TAXI WT. = 675,000 LBS.	MAX. TAXI WT. = 825,000 LBS.
1*	31,200	60,500	0	0	0
2	30,400	57,500	+1,300	-10	+45
3	31,750	54,300	+2,900	-33	+43
4	28,750	60,500	+1,500	-19	-6
5	28,100	57,500	+2,800	-31	+36
6	27,500	54,300	+4,600	-58	+29
7	26,500	60,500	+5,200	-86	-60
8	25,600	57,500	+7,000	-111	-30
9	25,200	54,300	+9,400	-	-50

* Basic B-2707 M_{MO} Climb Placard

selection of a climb M_{sched} in which the altitude at $M = 2.7$ is lowered about 3000 to 5000 feet (schedules 2 and 3) results in maximum range. This occurs because the higher altitude placard limits the start of cruise to altitudes above best cruise altitude at takeoff gross weights above about 750,000 pounds. Other schedules (5 and 6) result in nearly the same range improvements. Hence, it can be concluded that only very minor range improvements are possible for the 825,000 pound airplane and none for the 675,000 pound airplane.

3.0 DOMESTIC SST DESIGNED TO MEET ACCEPTABLE SONIC BOOM OVERPRESSURES

Parametric airplane sizing and Direct Operating Cost data for Domestic SST's designed to meet acceptable sonic boom overpressures are presented in this section. Sizing and economic data for airplanes constrained to meet climb overpressures of from 1.0 to 1.6 psf are presented. The basis for the sonic boom overpressure calculations is near field theory, with corrections applied for the 1962 U.S. Standard Atmosphere. This is the most realistic theory and calculation basis developed to the present time. Detailed aerodynamic, sonic boom, weights, and powerplant input data used for the study are given in Appendix A.

As a result of the briefing and interim report on these studies to the FAA on October 28, 1966, the scope of the Domestic Airplane Parametric study was extended to include a range of cruise Mach numbers. Specifically, the study plan was extended to include airplanes designed for cruise Mach numbers of 1.7, 2.2, 2.7, and 3.2. (Subsequently, the M = 1.7 design point study was deleted.) Because the scope of the study was enlarged to include Mach numbers in cruise lower than 2.7, one comparative study of aluminum versus titanium airframe materials was made. Also, a brief comparative study was made of engine cycles for the same reason, and results of these studies are given in Paragraph 3.4.

3.1 Design Approach

The design approach to this study required careful selection and consideration of the following major items:

1) Choice of the best possible and practicable baseline configuration,
2) Performance requirements of range and engine-airframe matching criteria.

To provide verification of the inputs selected, a preliminary baseline configuration was selected. Only a minimum amount of engineering analysis has been made on this baseline configuration due to the time limitations. Because of the time restriction, a brief examination of the sensitivity of changes in the inputs to the study results has been made and is discussed in Paragraph 3.2.

3.1.1 Baseline Configuration

The choice of the baseline configuration emphasized features which would result in low sonic boom characteristics and low supersonic drag. The wing sweep selected was 74° and average thickness ratio 2.75%. Supersonic wing aspect ratio selected was 1.6. This latter choice was a compromise between wave drag and low drag due to lift considerations. Nacelles were placed well aft to provide favorable interference affects. A highly area-ruled body sized for 85 passengers was selected. These features were combined into a M = 2.7 baseline configuration using a twin engine arrangement with a wing area of 6000 square feet and a preliminary drawing was made. The maximum lift-drag ratio of this configuration is estimated to be 9.5. The maximum lift-drag ratio versus Mach number, an area distribution plot, and sonic boom characteristics of the baseline configuration are given in Appendix A.

3.1.2 Economic Model and Mission Ground Rules

The Supersonic Economic Ground Rules (SST 66-3) dated June 30, 1966, were followed in calculating the range performance, fuel reserves, and Direct Operating Costs. Sonic boom overpressures were calculated using the near field theory corrected for U.S. Standard Atmosphere, 1962. (The September 6, 1966 proposal data used the far field solution for calculating sonic boom overpressures using the $\sqrt{P_g/P_a}$ atmospheric correction.)

3.1.3 Performance Requirements and Engine-Airframe Sizing Criteria

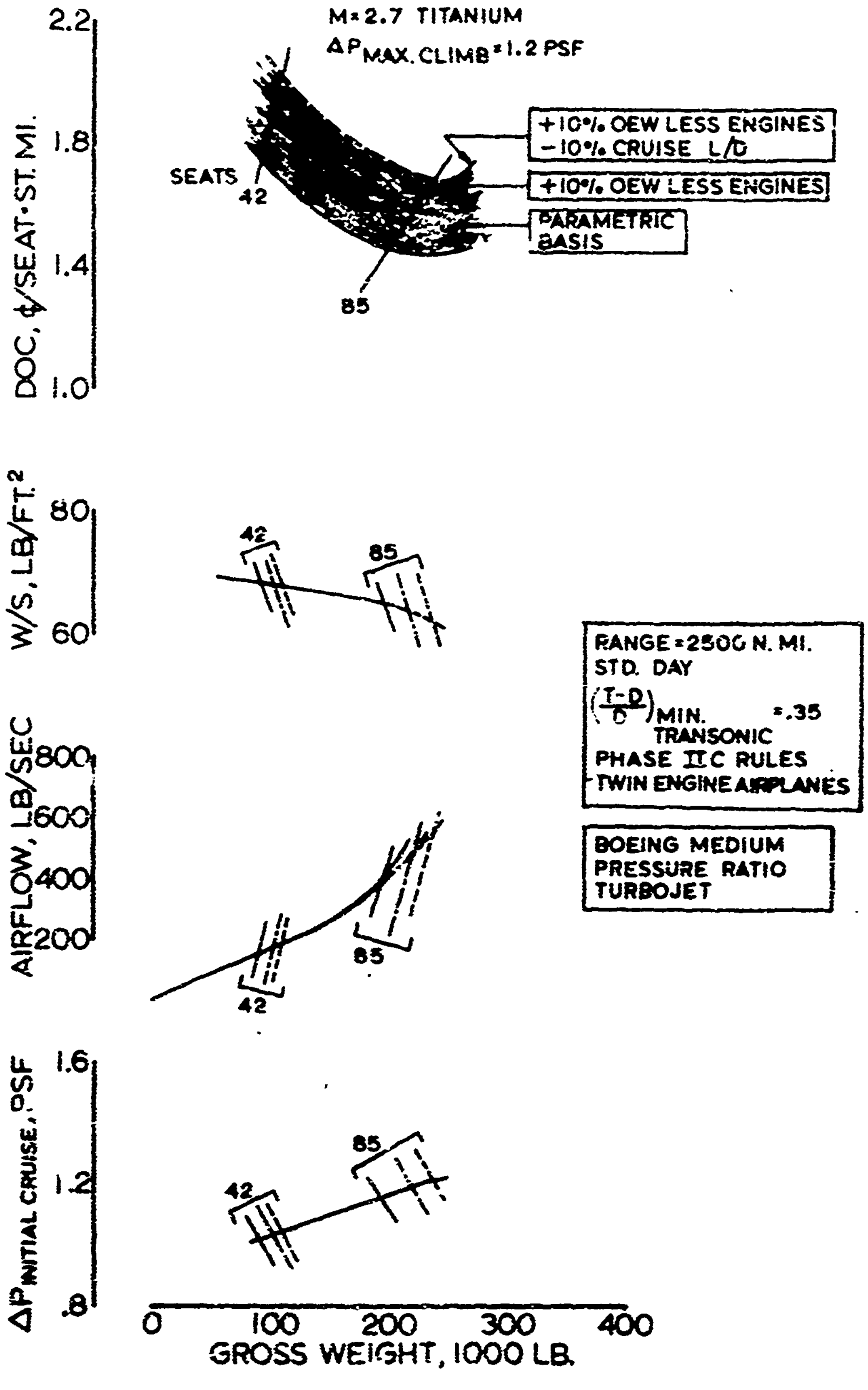
The range requirement selected was 2500 nautical miles under standard day conditions. The basis for this selection was that it would provide New York to San Francisco range capability under the temperature conditions of standard +15°C for climb and standard for cruise. Engine sizes selected provide for a minimum transonic thrust margin of .30 on a standard +15°C day. This provides a minimum climb corridor of about 4000 feet. Wing areas were selected to provide the minimum gross weight for the particular payload considered, thus maximizing the payload to gross weight ratio. Since relatively low wing loadings and large engine sizes were expected as a result of the sonic boom requirements, no restrictions were made on engine or wing sizing for low speed considerations.

3.2 Gross Weight Parametric Study Results

D.O.C., engine airflow, wing loading and initial cruise sonic boom overpressures versus gross weight are shown in Figures 3.1, 3.2, 3.3, and 3.4. All airplanes were sized to meet the 2500 nautical mile range requirement together with a minimum transonic thrust margin of .35 as noted in paragraph 3.1.3.

Figure 3.1 shows the sensitivity of the study results to changes in inputs of cruise lift to drag ratio and airframe operating empty weight. It is felt that both the lift to drag ratio and operating empty weights are optimistic. The baseline configuration was an 85 passenger airplane with a 6000 square foot wing area. The selected airplanes for this body size have about half of this wing area. The much smaller wing area probably will result in lower lift to drag ratios and higher OEW's than assumed by the parametric extrapolation used in this study. Body drag and drag due to lift were assumed constant versus wing area and fuel volume problems were only roughly considered at the smaller wing areas. A study would be necessary to determine the characteristics of a new baseline configuration which is more representative of the optimum airplanes shown in Figures 3.1 to 3.4.

It is noted that an increase of 10% in operating empty weight results in about a 10% increase in gross weight for a given payload and, hence, approximately a 10% increase in DOC. If, in addition, there is a loss of 10% in cruise lift to drag ratio the DOC's would increase a further 6%, resulting in a total increase in direct operating costs of about 16%.



SENSITIVITY TO DESIGN INPUTS: CRUISE L/D & OEW LESS ENGINES

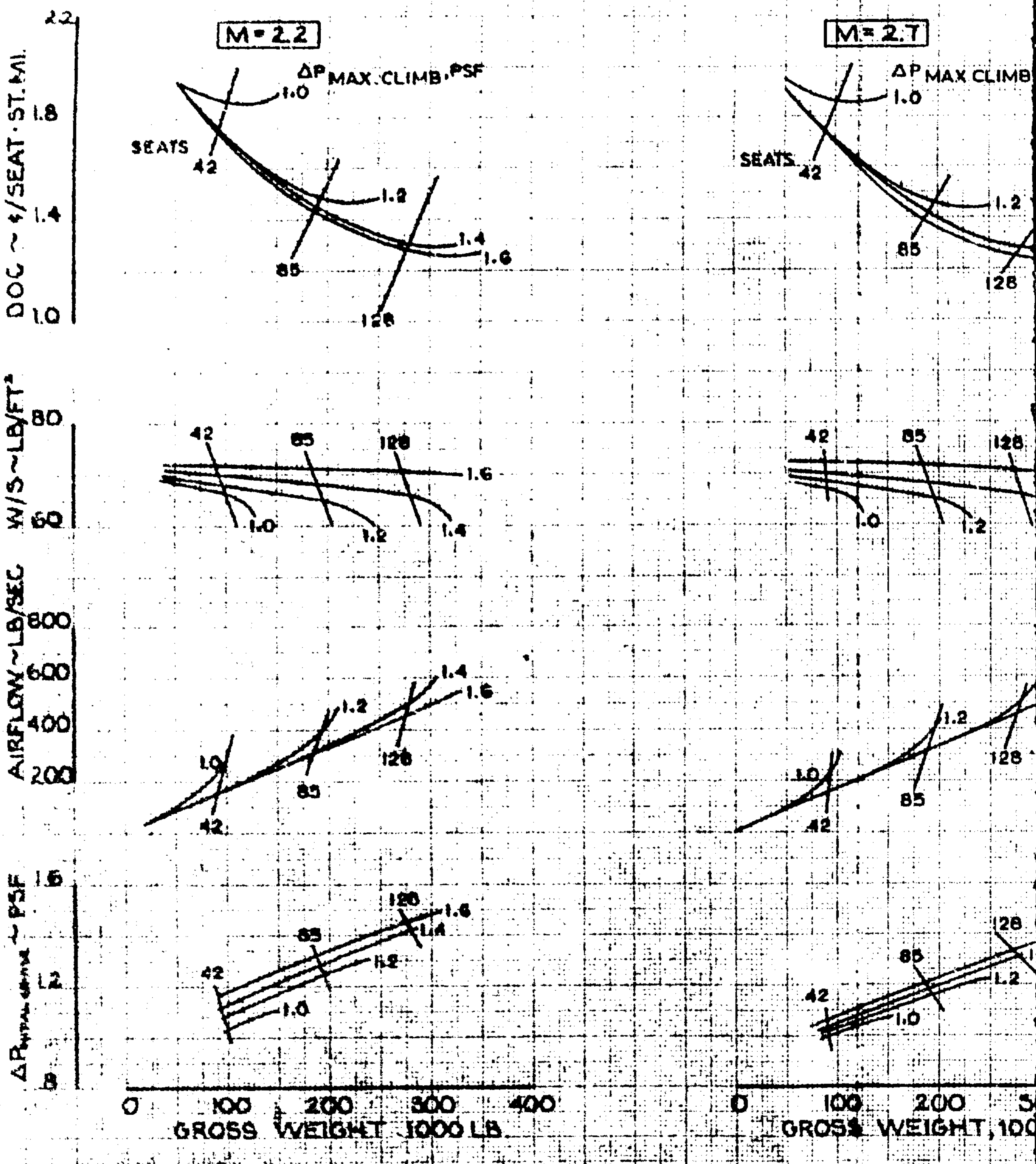
THE BOEING COMPANY

D6A10483-1

Fig. 3-1

3.3 Effect of Cruise Mach Number

Study results showing the effect of cruise Mach number are presented in Figure 3.2. From the data of Figure 3.2 a minimum DOC airplane meeting the design objectives of maximum climb and cruise overpressure of 1.2 PSF was selected for each Mach number. The characteristics of these airplanes are summarized in Table 1A, page 7. The lowest DOC's occur at a cruise Mach number of 2.7 because the cruise range factor is at its maximum resulting in the lowest gross weight for a given payload. Moreover the initial cruise overpressure limitation penalizes the M = 2.2 airplane since optimum cruise altitude decreases with decreasing Mach number. The airplanes are sized for transonic climb so that for the same gross weight, payload, and climb overpressure the wing loading is approximately constant. In addition, the engine size is independent of cruise Mach number because the engine characteristics are unchanged in the transonic region. The OEW increases only slightly, about 3%, from M = 2.2 to M = 3.2. Hence, cruise range factor (lift to drag ratio times true velocity divided by engine SFC) is the chief parameter determining the lowest gross weight for a given payload. The true velocity increases linearily from M = 2.2 to M = 3.2. SFC increases about 8% from M = 2.2 to M = 2.7 and about 25% from M = 2.7 to M = 3.2. L/D decreases 20%, nearly linearly, from M = 2.2 to M = 3.2. The net result is that from M = 2.2 to M = 2.7, the L/D decrease and SFC increase are more than counterbalanced by the true velocity increase. From M = 2.7 to M = 3.2 the very large SFC increase together with the L/D decrease more than compensates the true velocity increase. Hence, the range factor decreases. In addition, some of the costs, particularly fuel, are greater for the airplanes cruising at M = 3.2 resulting in still higher DOC's compared to M = 2.7 and 2.2.



CALC			REVISED	DATE	BOEING MED. PRESSURE RATIO TURBOJET , TITANIUM	
CHECK						FIG. 3.2
APR					THE BOEING COMPANY	PAGE 27
APR						

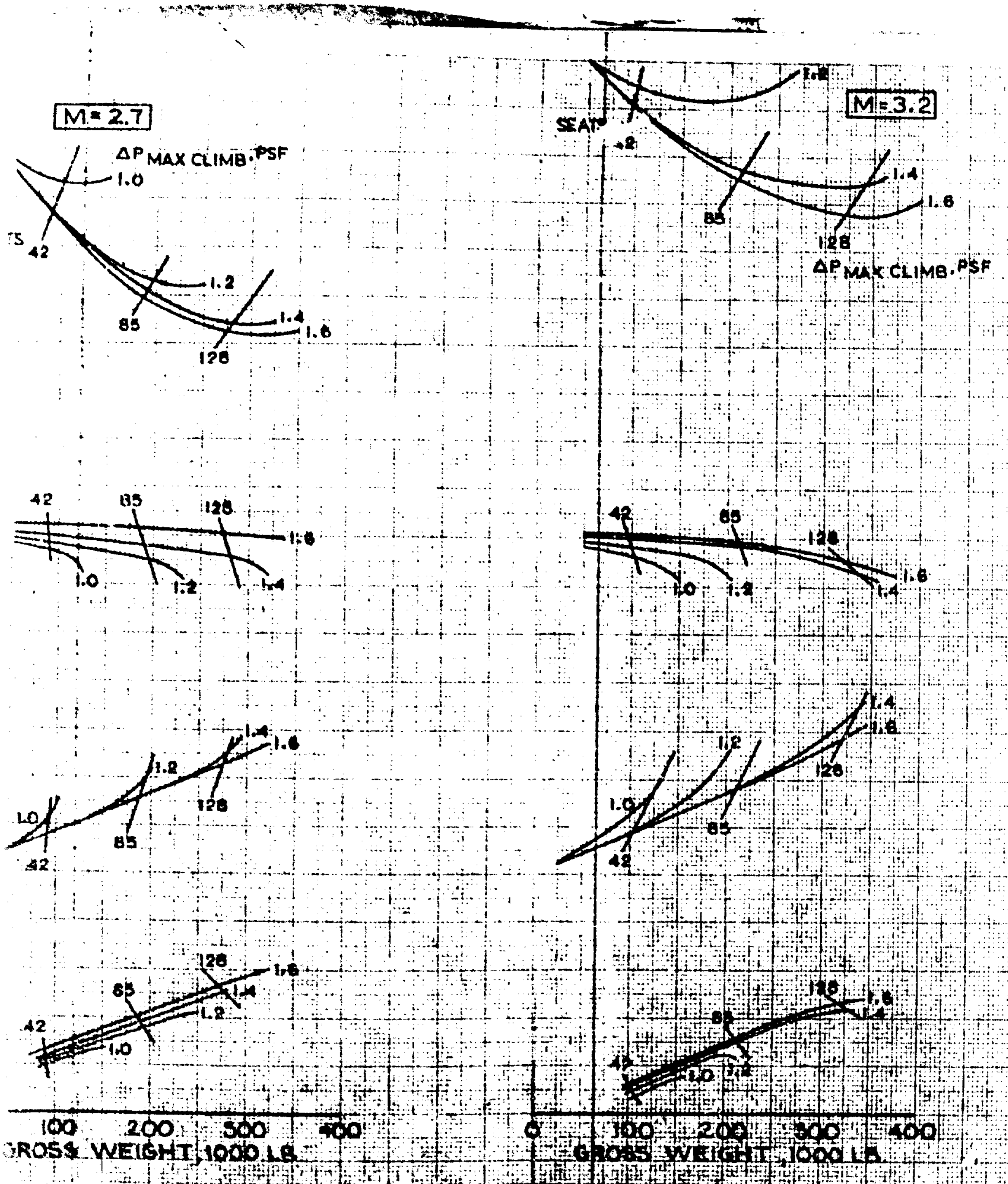
RANGE = 2500

$\left(\frac{T-D}{D}\right)_{\substack{MIN. \\ TRANS}} = .35$

PHASE IIC

TWIN ENGIN

1



RANGE = 2500 N. MI. STD. DAY

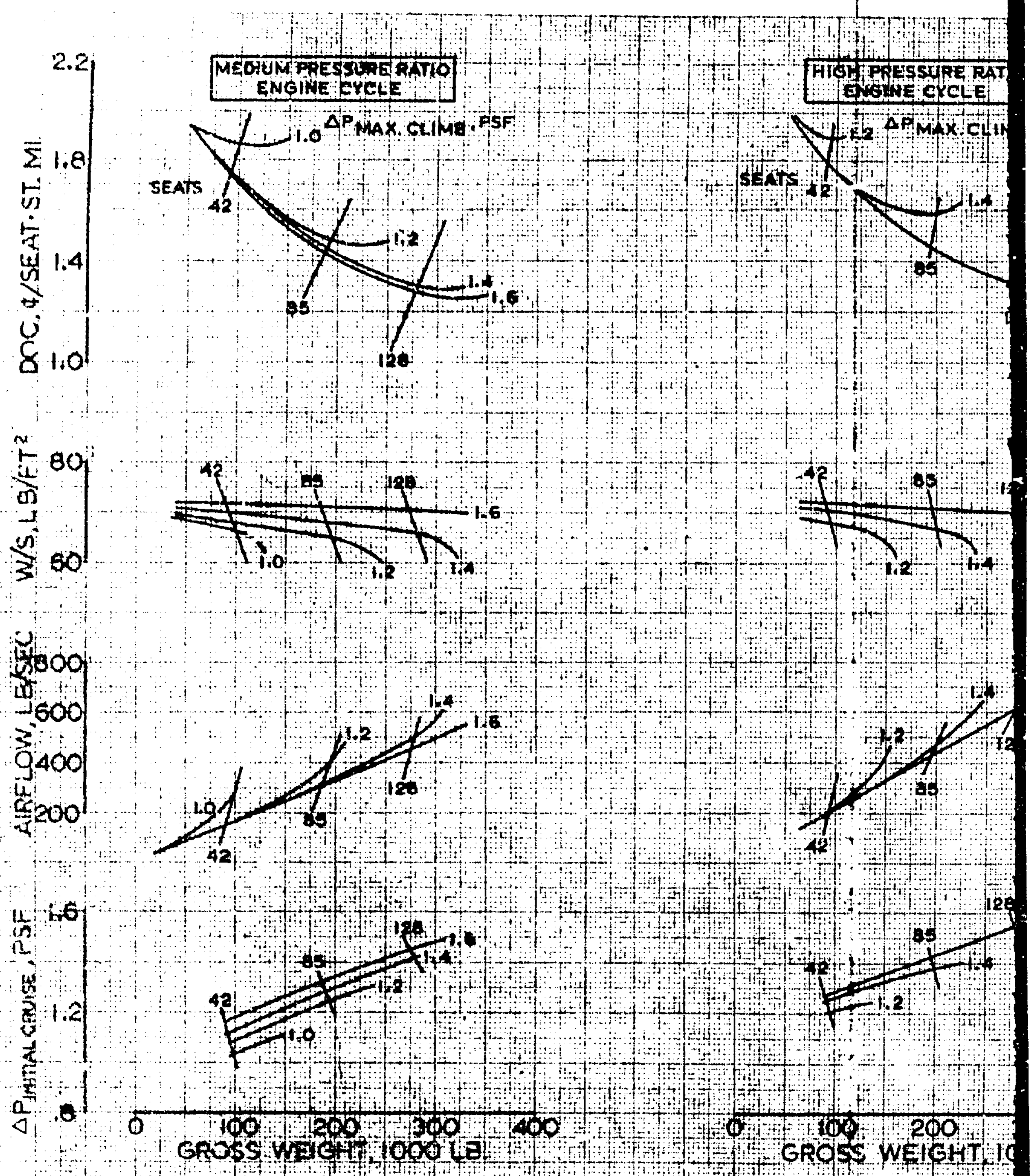
$\left(\frac{T-D}{D}\right)_{\substack{MIN\\TRANS}} = .35$

PHASE II C RULES
TWIN ENGINE AIRPLANES

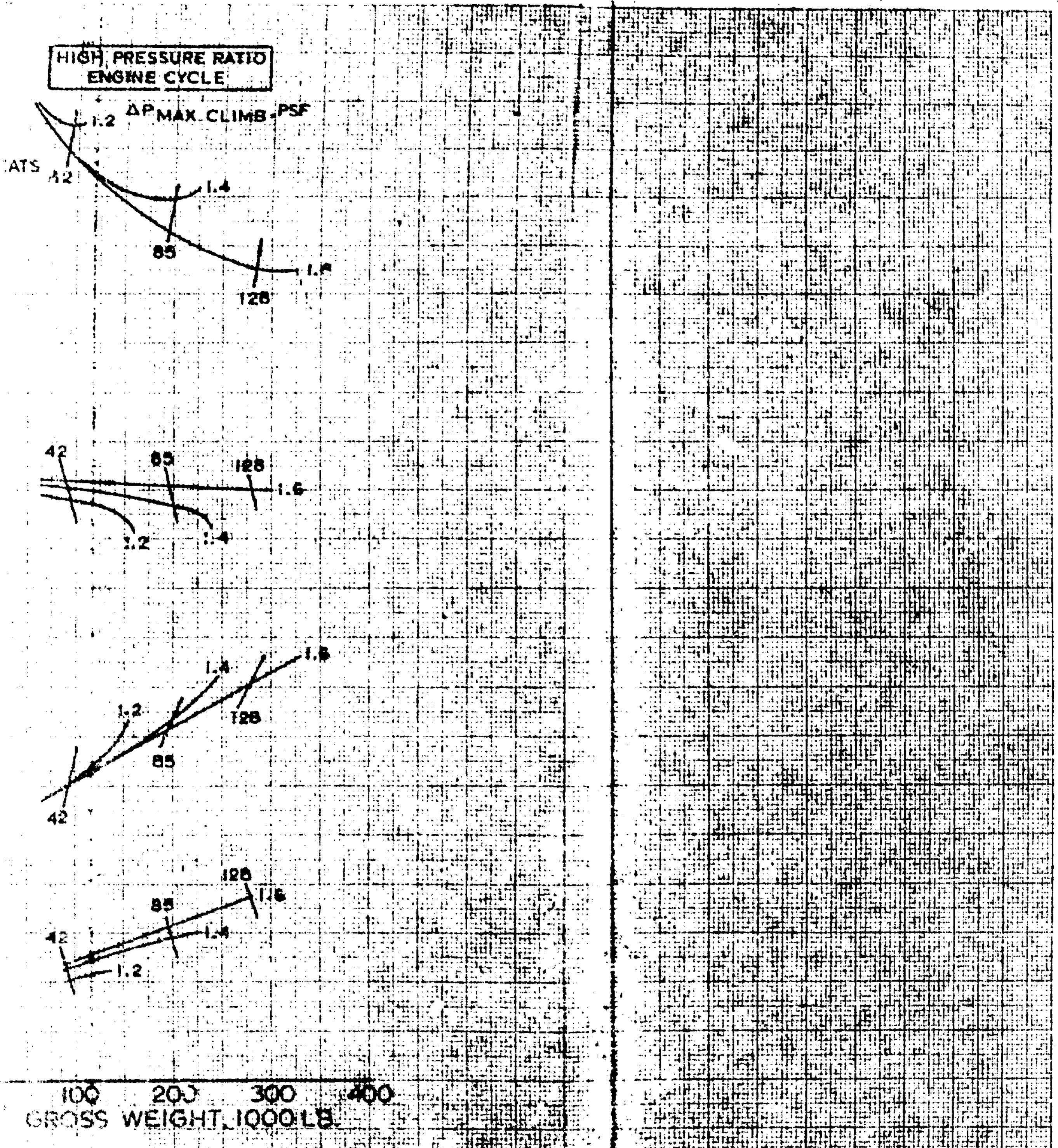
3.4 Engine Cycle Choice for M = 2.2 Cruise Airplane

The effect of engine cycle choice for a M = 2.2 cruise airplane is shown in Figure 3.3. A high pressure ratio turbojet designed for cruise at M = 2.2 was compared with a medium pressure ratio turbojet designed originally for M = 2.7. The high pressure ratio (HPR) engine operates at relatively low turbine inlet temperatures with only partial augmentation. The medium pressure ratio (MPR) engine operates at high turbine inlet temperatures and has a full augmentor. Since the engines are sized for the same transonic thrust margin, the HPR turbojet must be sized 50% larger in airflow than the MPR turbojet. This results in a 100% engine weight increase due to the greater weight per unit airflow of the HPR engine and, hence, in a large range loss. The SFC advantage of the HPR turbojet in the subsonic and transonic regions results in a range improvement which very nearly compensates for the range loss due to increased engine weight. Cruise is performed at partial agumentation where both engines have comparable SFC's. The net effect is a slight range loss for a given gross weight and payload when using the HPR turbojet. Since in this study range is held constant, this results in a higher gross weight for a given payload. If the study were continued, an obvious step would be to investigate the possible improvement resulting from increased augmentation on the HPR engine.

It is important to note that the initial cruise overpressure is higher for the HPR turbojet. This is due to the fact that the initial cruise weight is higher because the SFC of the HPR engine is lower in climb. Thus, the minimum DOC airplanes selected from Figure 3.3 sized to the 1.2 PSF objectives in cruise and climb tend to greatly favor the medium pressure ratio engine. These airplanes are summarized in Table 1A, page 7.



CALC	RHF	11/12/	REVISED	DATE	CRUISE MACH NUMBER=2.2 TITANIUM	FIG. 3.3	RANGE=2500 $\left(\frac{T-D}{D}\right)_{MIN.\ TRANS.}$ =.35
CHECK					THE BOEING COMPANY	PAGE 89	PHASE TWIN



RANGE = 2500 N.MI. STD. DAY

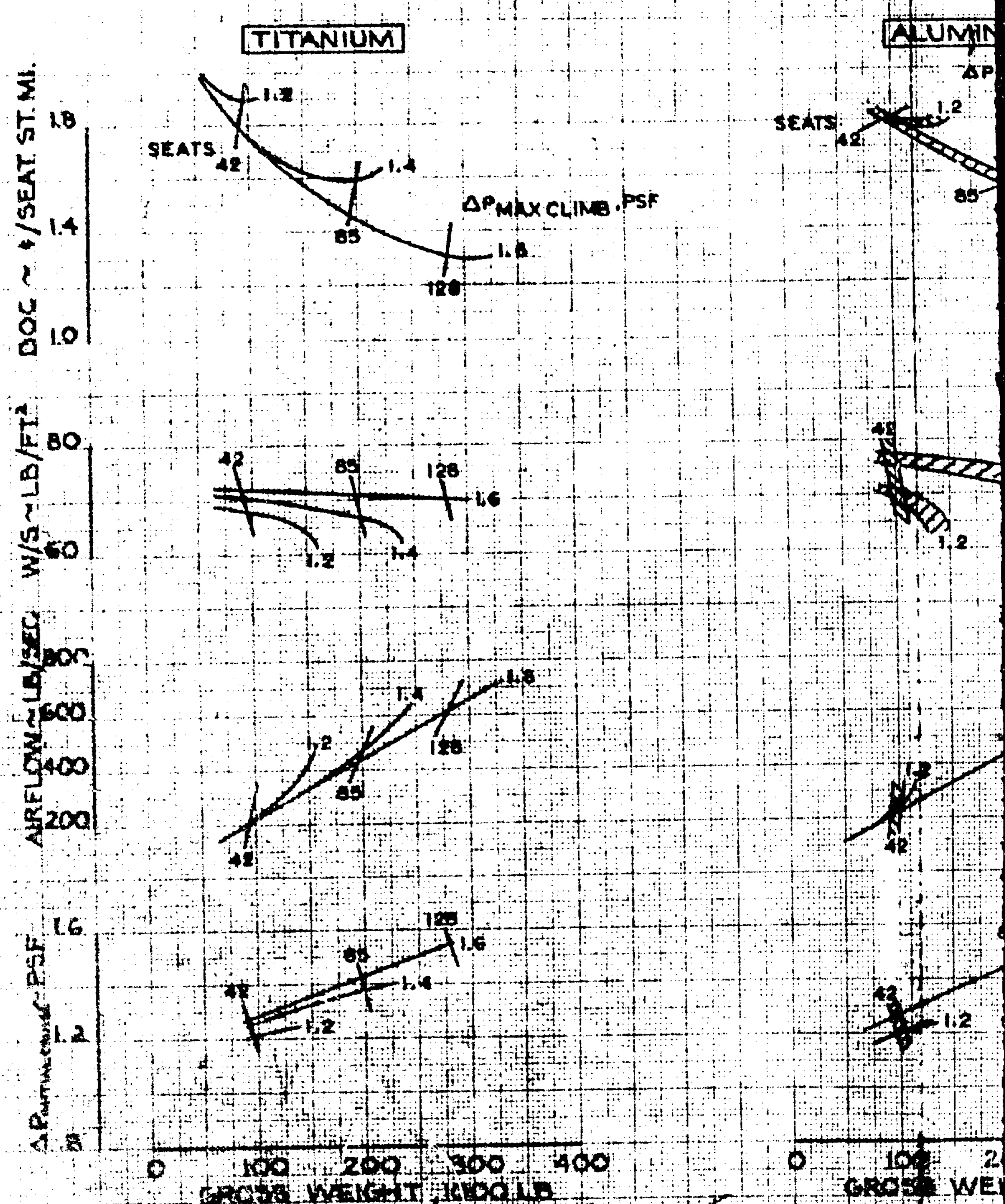
$\left(\frac{T-D}{D}\right)_{\substack{MIN.\\TRANS.}} = .35$

PHASE II C RULES

TWIN ENGINE AIRPLANES

3.5 Aluminum Versus Titanium Airframe for M = 2.2 Cruise Airplane

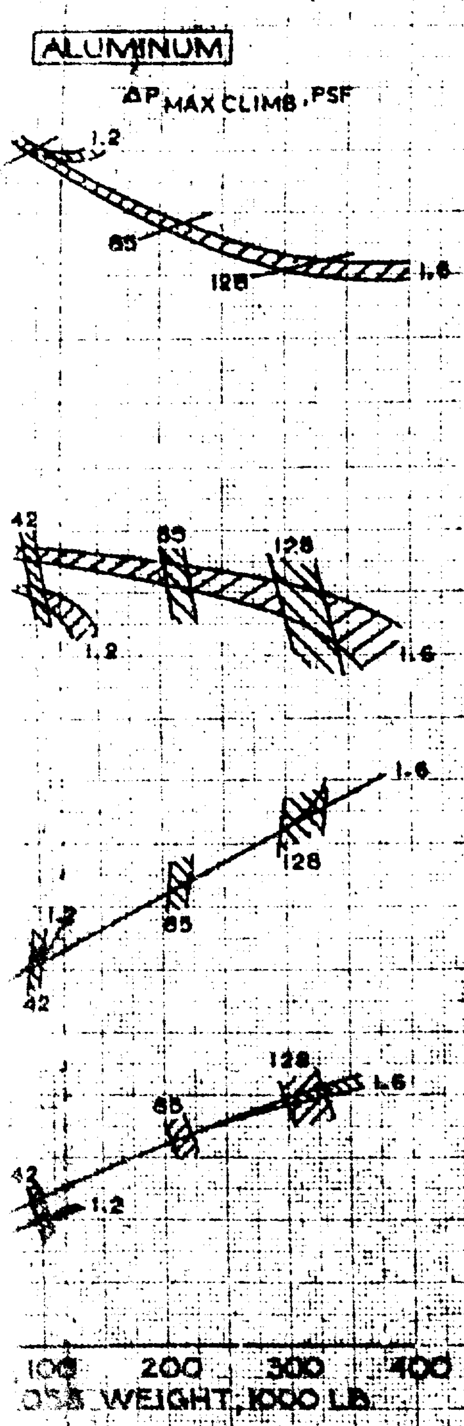
The M = 2.2 cruise airplane was analyzed with regard to material selection. Aluminum and Titanium airframes were studied and the results are presented in Figure 3-4. The powerplant used for this study was the high pressure ratio turbojet. The estimated structural weights for the aluminum airplanes are 5 to 15% greater than those for titanium airplanes. As shown in Figure 3-1, sensitivity to design inputs, these weight increases would increase the DOC's 5 to 15% due to larger gross weights for a given payload. The compensating effect on DOC of lower aluminum costs is not sufficient to offset the increase due to weight except for very small airplanes. In general, the results indicate an aluminum airframe would result in 5 to 10% greater DOC for comparable conditions. It is expected that this would also be the case if the medium pressure ratio turbojet were used for the comparison.



CALC		11/12/6	REVISED	DATE	BOEING HIGH PRESSURE RATIO TURBOJET, M=2.2	
CHECK						FIG.
APD					THE BOEING COMPANY	PAGE

RANGE (T-D)

1



ALUMINUM OEW LESS ENGINES ASSUMED 5 TO 15% GREATER THAN TITANIUM VALUES

RANGE = 2500 N. MI. STD. DAY

$\left(\frac{T-D}{D}\right)_{\substack{MIN.\\TRANS.}} = .35$

PHASE IIC RULES

TWIN ENGINE AIRPLANES

2

4.0 NO BOOM DOMESTIC TRANSPORTS DESIGNED FOR 1974 OPERATION

The following topics relating to domestic transports which could fly overland without producing a sonic boom are discussed in this section.

(1) A general discussion of the sonic boom cut-off Mach number, and weather factors influencing its choice, as related to a transonic cruise Mach number airplane.

(2) Subsonic cruise airplanes and the projected technology base available for use in design are reviewed and compared with the Model 747.

(3) The point design characteristics of a transport designed to cruise at transonic Mach numbers below the sonic boom cut-off Mach number.

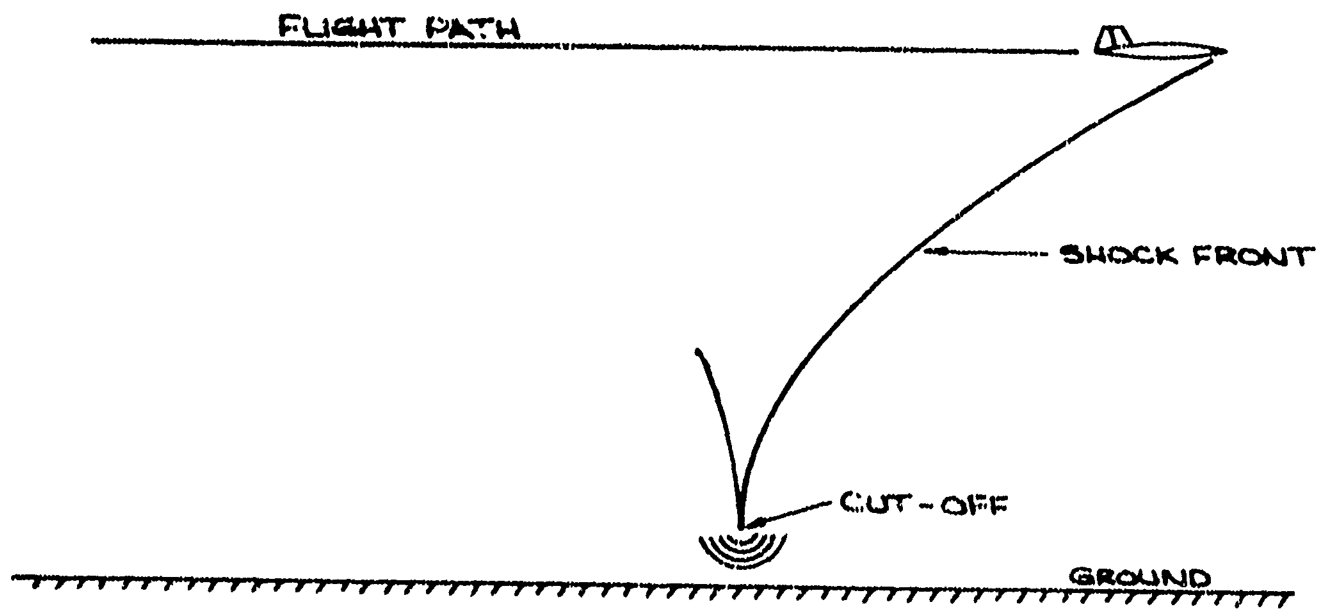
Finally, a brief economic comparison is made among these airplane types for the San Francisco to New York route.

4.1 Possible Transonic Cruise Mach Numbers For No Sonic Boom

Shock waves produced by supersonic airplanes are refracted away from the ground as they travel through the atmosphere. The refraction is due to variations of wind velocity and temperature between the airplane and the ground. Complete refraction of the shock waves is known as cut-off, and is possible for supersonic flight near Mach 1.0 (as sketched in Figure 4.1). This phenomenon has been observed experimentally and has been reported in NASA TND-3520. The airplane Mach number at which cut-off occurs is known as the cut-off Mach number. No boom would reach the ground for flight at Mach numbers less than the cut-off Mach number.

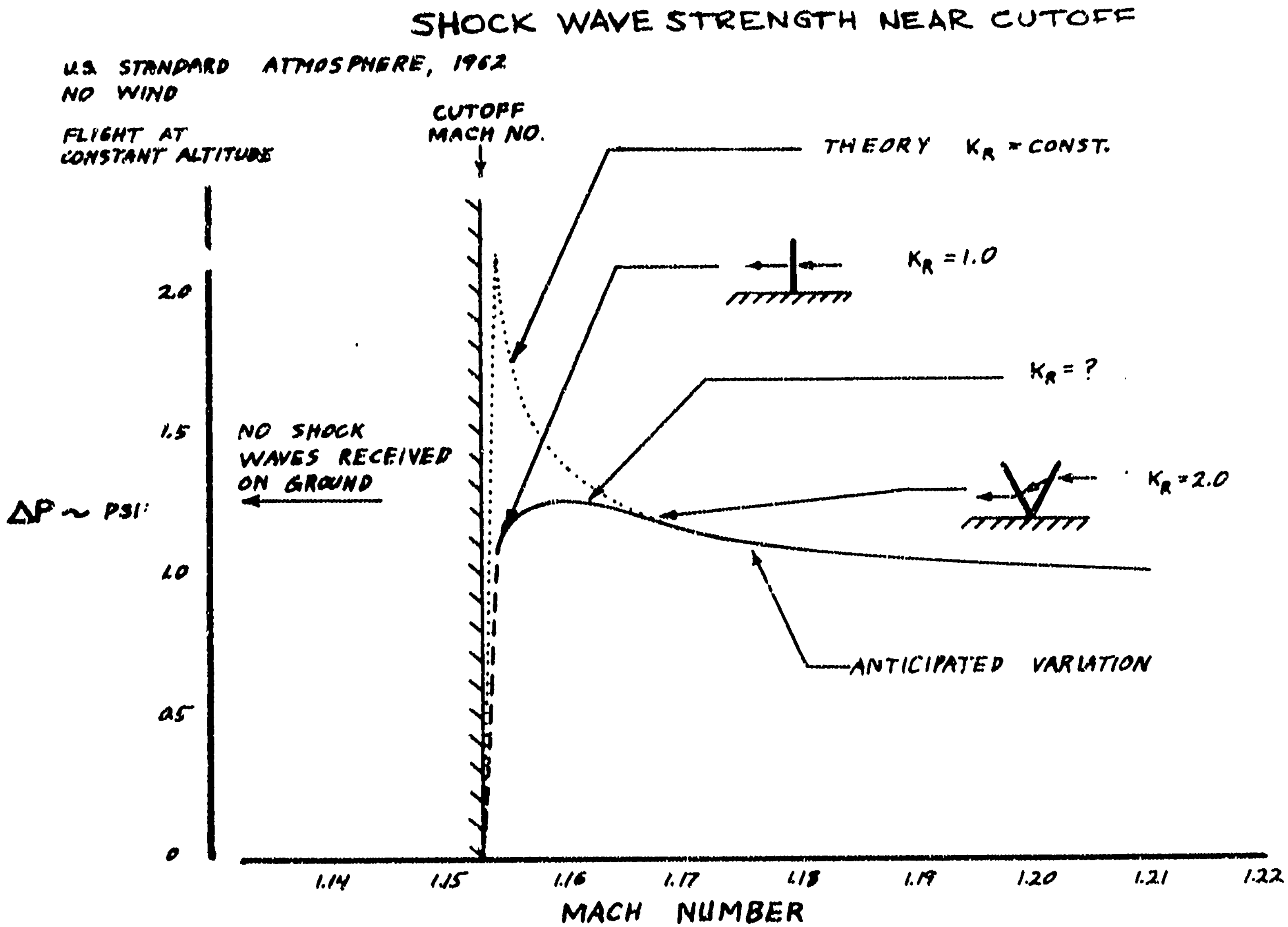
SONIC BOOM CUT-OFF ABOVE GROUND

STEADY FLIGHT



The shock wave strength as measured on the ground is influenced by atmosphere refraction and by reflection from the ground. When an oblique shock wave intersects the ground, a reflected wave of equal strength and angle is produced. The total pressure jump across the system of the incident and reflected waves is twice that of the incident wave, and the factor accounting for this reflection, K_R, is 2.0. If the shock wave is normal to the ground, the reflected wave does not exist so that K_R is 1.0. Theoretical calculations of the variation of shock wave strength due to atmospheric refraction (assuming $K_R = 2.0$) indicates an increase in overpressure for flight very near the cut-off Mach number as indicated by the dashed line in Fig. 4.2. However, the incident shock wave angle approaches 90° for flight very near the cut-off Mach number. The combination of the effects of atmospheric refraction and change in K_R is indicated by the solid line in Fig. 4.2. Evidence of these self cancelling effects has been observed and is discussed in NASA TND-3520.

The effect on the cut-off Mach number of variations in atmospheric conditions from those which are defined for a standard day are shown in Fig. 4.3. The effect of winds is shown in the upper curve. Tailwinds at the airplane reduce the cut-off Mach number while headwinds increase it. The effect of variations in the temperature gradient is shown in the lower curve. Temperatures lower than standard at the ground reduce the cut-off Mach number while temperatures higher than standard increase it.



SHOCK WAVE STRENGTH NEAR CUTOFF

THE BOEING COMPANY

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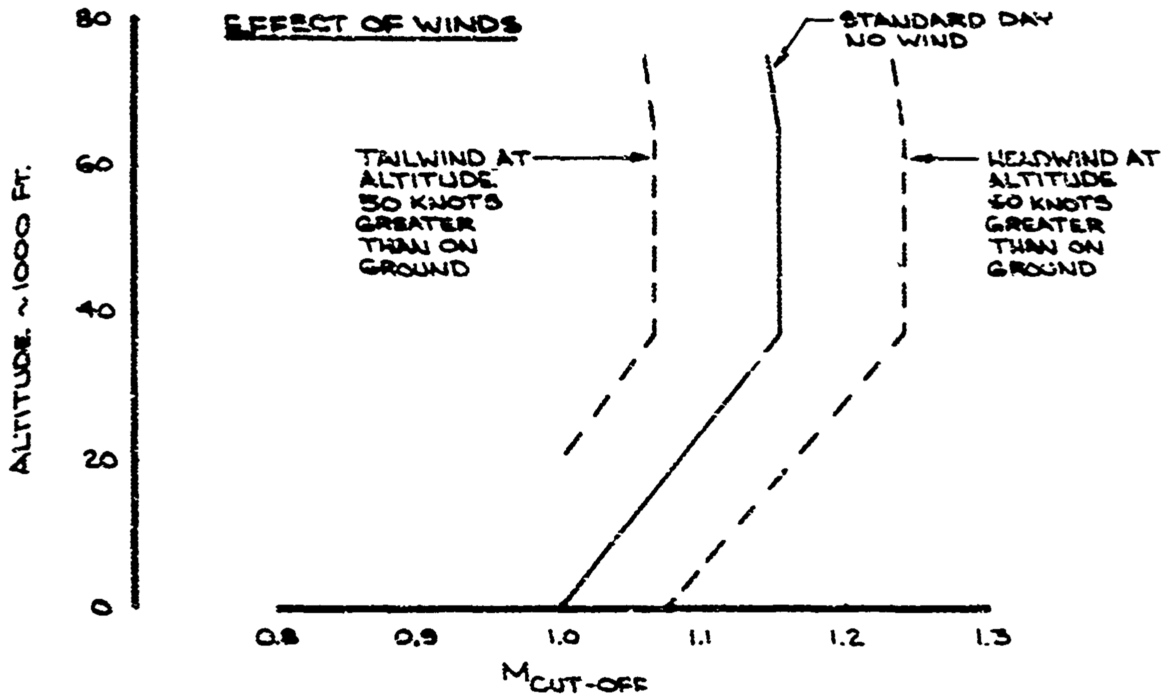
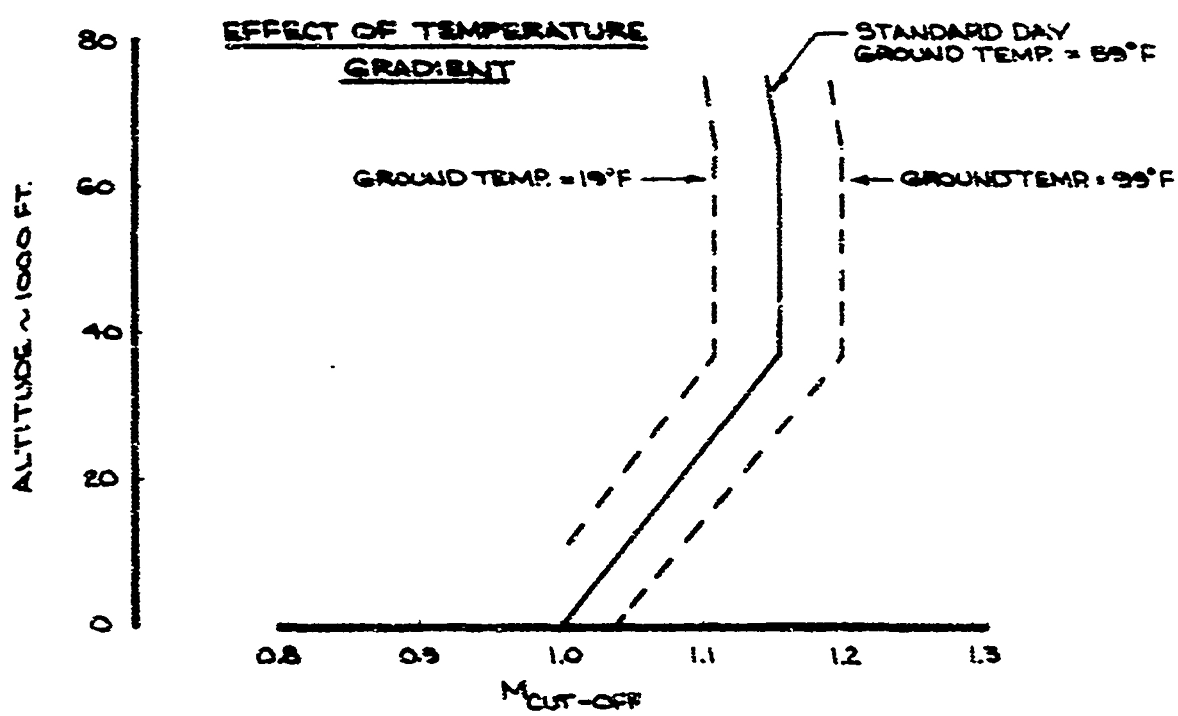
Fig. 4-2

EFFECT OF WIND AND TEMPERATURE ON CUT-OFF MACH NUMBER





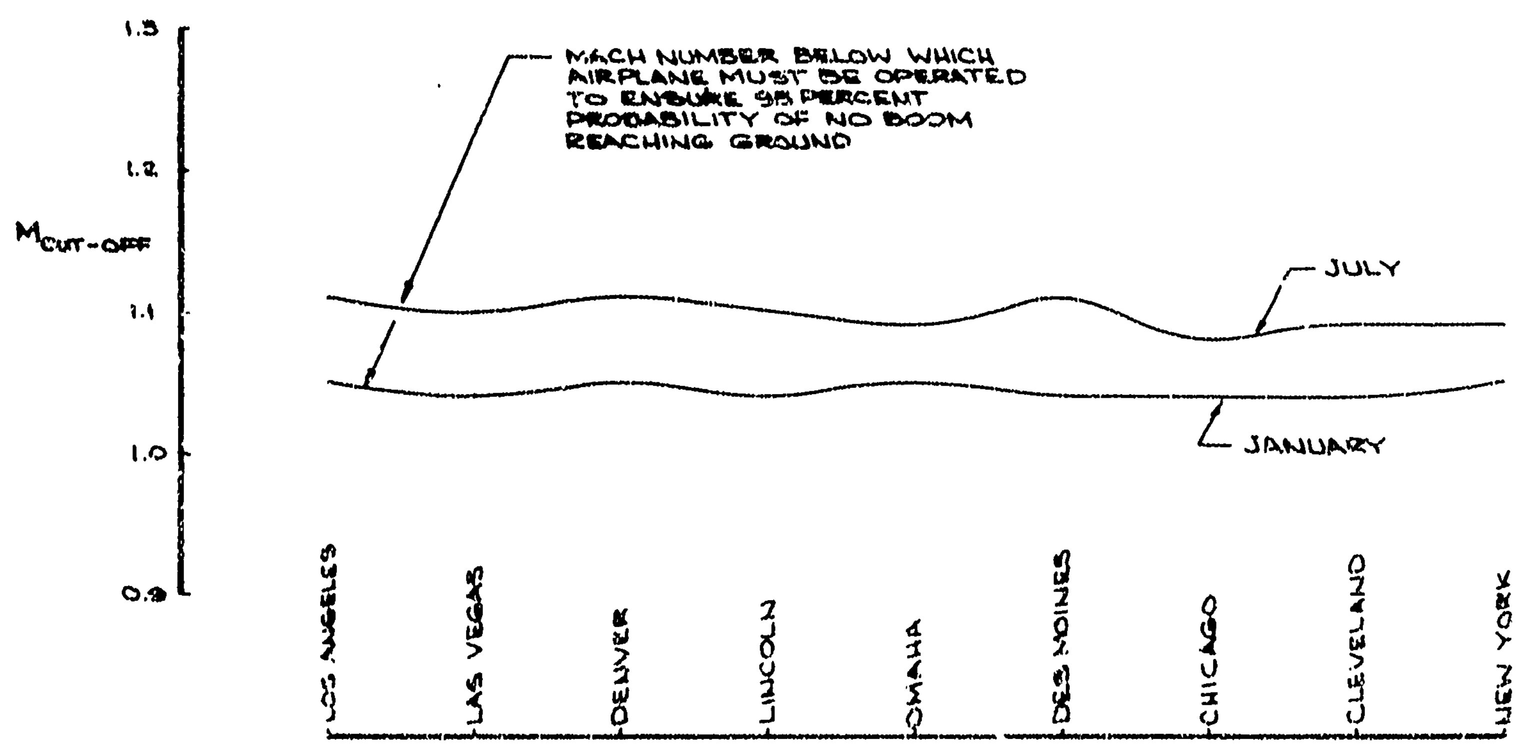
A statistical study of weather patterns over the continental U.S. was conducted to determine the variation of the cut-off Mach number with season and location. These results are shown in Fig. 4.4 for a flight from Los Angeles to New York. For flights in the summer the operating Mach number would have to be about 1.08 to ensure a 95% probability of no boom reaching the ground. In the winter, this value is about 1.04. Subsonic flight is necessary to assure 100% reliability that no boom reaches the ground.

4.2 Technology Base for 1974 Time Period Airplanes

A significant technical factor, the airplane cruise lift to drag (L/D) ratio, is plotted versus Mach number in Fig. 4.5. The two current designs represent the Boeing 707-320B at Mach .80 ($\frac{L}{D}$ = 19) and the Boeing 747 at Mach .86 ($\frac{L}{D}$ = 16.6). The three points identified as 1974 airplanes are at an $\frac{L}{D}$ = 15.2 at Mach .90, $\frac{L}{D}$ = 12.8 at Mach 1.2 and $\frac{L}{D}$ = 8.2 at Mach 2.7. These airplanes have been designed to maximize overall operational efficiency considering the trades between takeoff and landing performance, cruise efficiency, and airplane size, weight, and price.

The 1974 airplanes take advantage of the promising results of recent high speed wing design research. Advanced airfoil technology will enable a wing of given sweepback and thickness to operate at speeds 3 to 5% faster than present wings before encountering critical Mach number effects. The slotted airfoil (Figure 4.6), which has been given considerable study by NASA and also by Boeing, is one concept which shows such promise. The benefits of advanced airfoil technology

VARIATION OF CUT-OFF MACH NUMBER FOR LOS ANGELES NEW YORK FLIGHT AT 45,000 FT



BASED ON D6-14024 TN WEATHER DATA

CRUISE EFFICIENCY

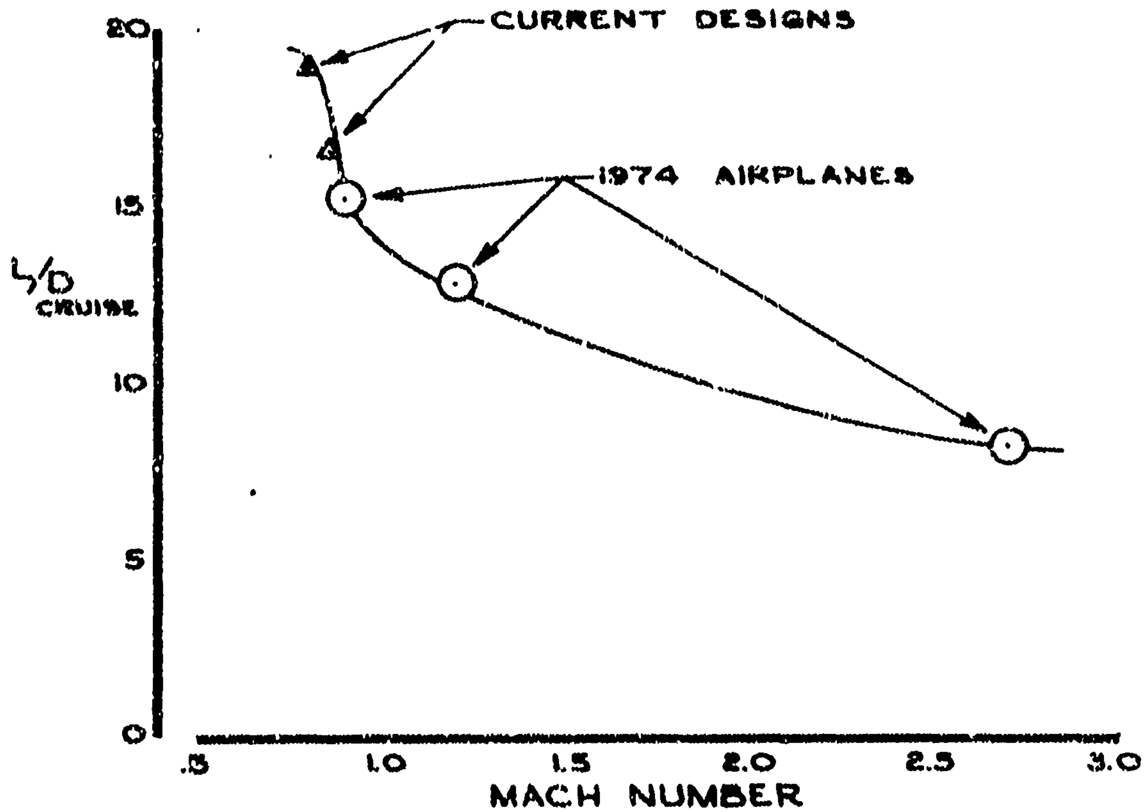


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Fig. 4-5

can also be realized in other ways. The speed advantage can be turned into an empty weight advantage by applying the advanced technology in the form of a substantially thicker wing. The improved structural efficiency which results can reduce the wing weight about 17 percent. Alternatively, the speed advantage can be turned into improved takeoff capability. For a given wing weight and cruise speed, the advanced technology permits a wing of less sweepback and higher aspect ratio -- both beneficial for higher allowable takeoff gross weights.

Turning specifically to the Mach 1.2 regime, it is essential that the wave drag of the wing-fuselage combination be minimized. Configurations such as that shown in Fig. 4.7 have been under study and test by Boeing for about 5 years. The pronounced body contouring and highly swept (55°-60°) arrow wing results in the $\frac{L}{D}$ performance shown in Fig. 4.8. It may be noted that an $\frac{L}{D} = 13$ was demonstrated at Mach 1.2.

In addition to aerodynamic efficiency, the question of propulsion system performance is important to mid-1970 airplane performance. The state-of-the-art of gas turbine engines has advanced markedly over the last twenty years-- an increase in engine thrust-to-weight ratio of 90 percent and a decrease in specific fuel consumption of 30 percent.

SLOTTED AIRFOIL

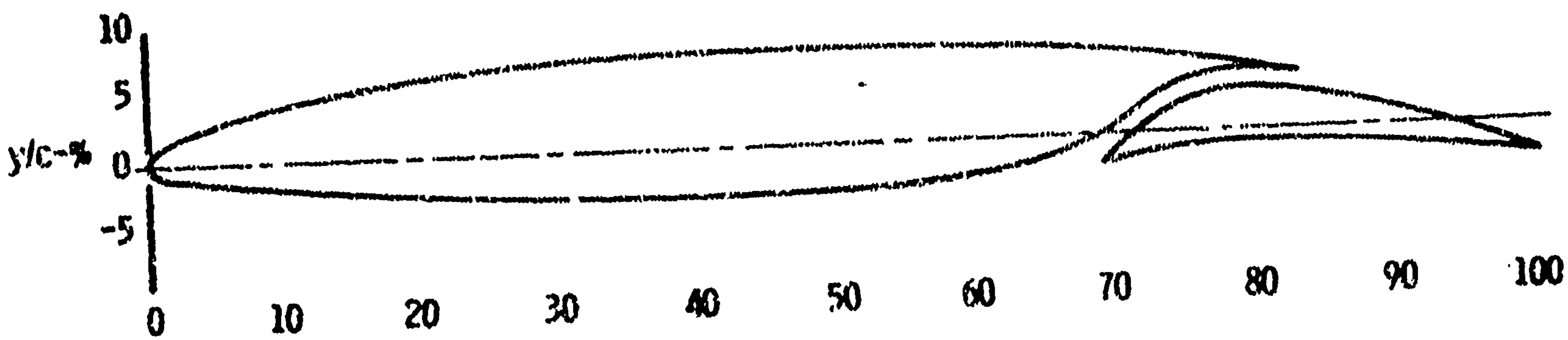

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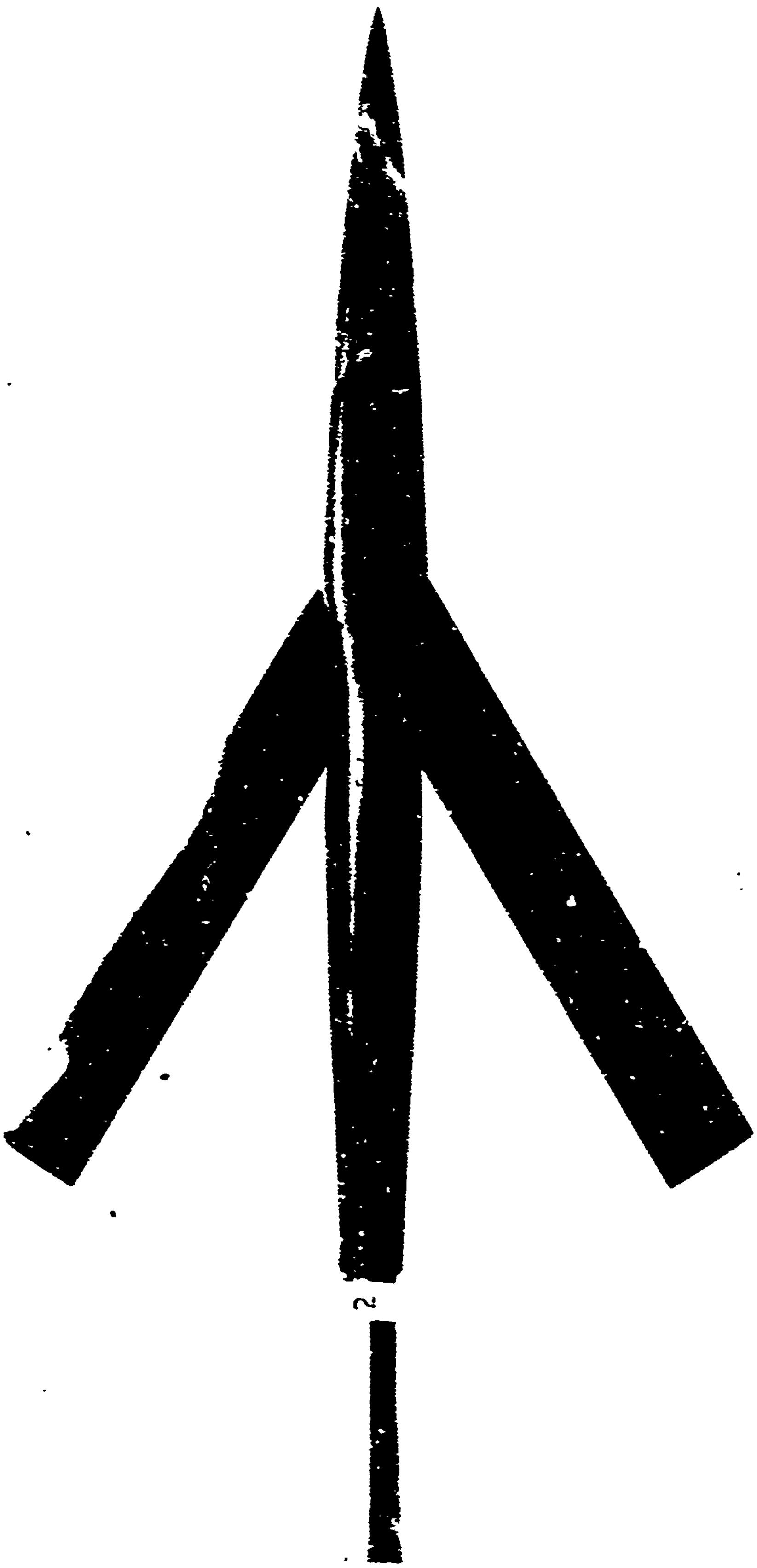
Fig. 4-5



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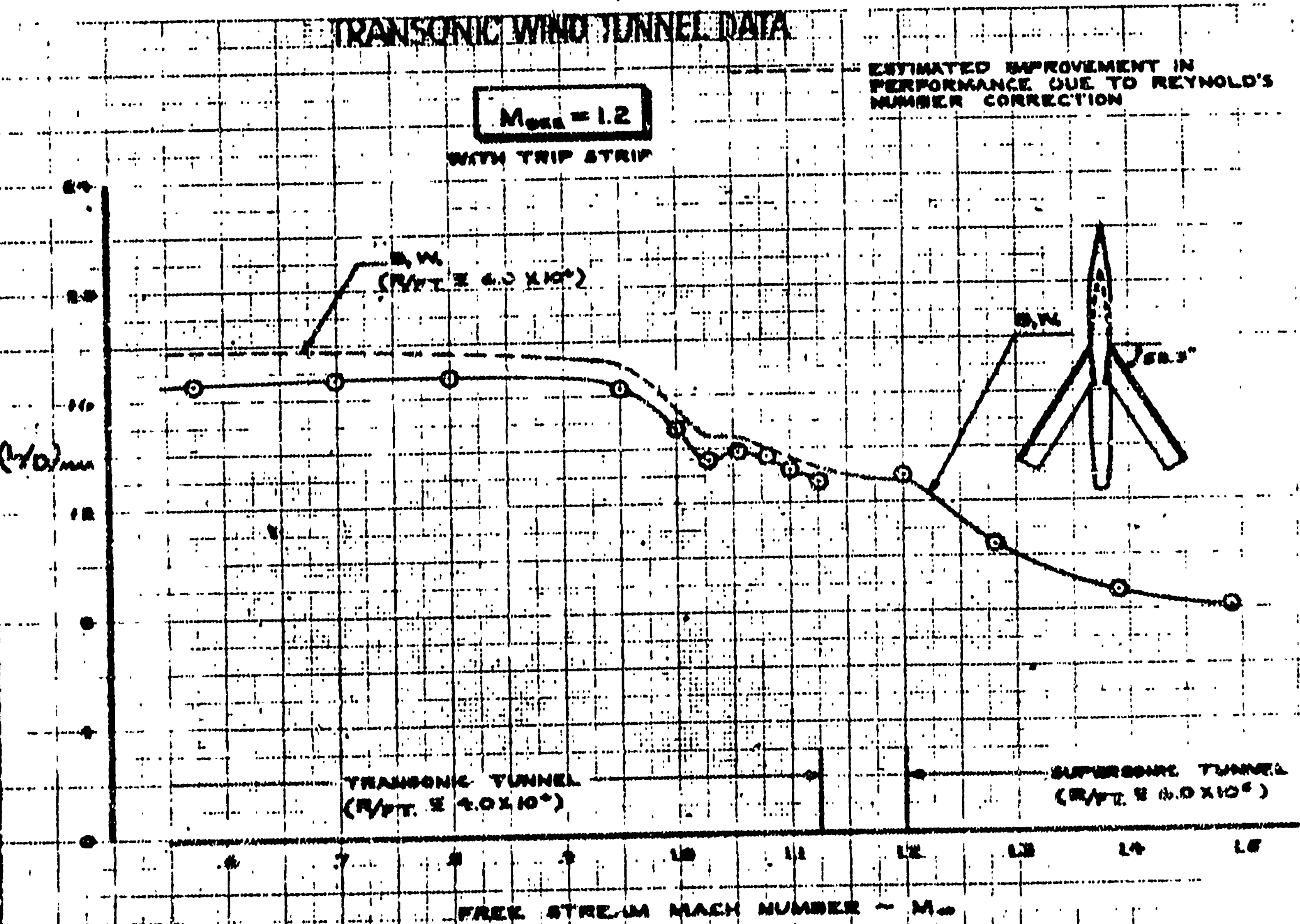
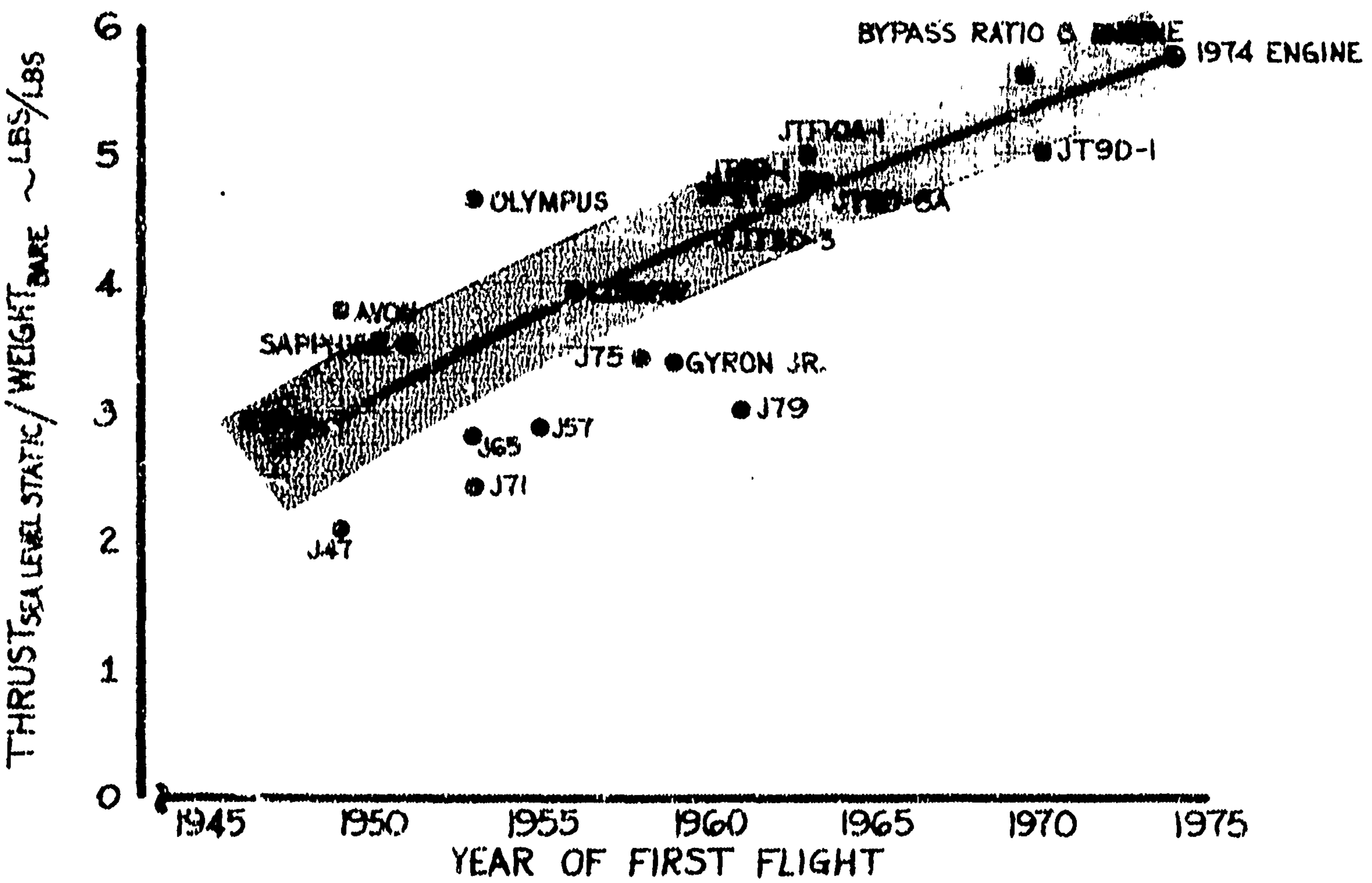


CALC	A.V.B...	...-2-...	REVISED	DATE	PERFORMANCE CHARACTERISTICS OF A WING - BODY CONFIGURATION DESIGNED FOR M = 1.2	D6A10483-1
CHECK						Fig. 4-5
APR					THE BOEING COMPANY	PAGE 43
APR						

The improvement in engine sea level static thrust-to-weight ratio (T/W) versus year of first flight is shown in Fig. 4.9. In the early 1950's, the engine T/W ratio was approximately 3.0. These turbojet engines provided power for some successful airplanes such as the 707-120, 707-320, KC-135, and B-52. In the early 1960's, the turbofan engines entered commercial service with T/W ratios from 4.0 to 5.0. These higher engine T/W's were a result of higher turbine temperatures brought about by metallurgical advances and improved propulsion efficiency of the bypass engine. These engines have replaced in many cases the turbojet and resulted in more economical airplanes such as the 720B, 707-320B, and B-52H. Looking ahead, the General Electric TF-39 engine (C-5A) and Pratt & Whitney JT9D-1 engine (747) will be flying about 1970 with T/W ratios of approximately 5.3. It is estimated that by 1975, a T/W of 5.75 will be typical of a new engine, based on the current rate of advancing the state-of-the-art.

The improvement in specific fuel consumption (SFC) versus year of first flight is shown in Fig. 4.10. The condition chosen for comparison purposes is the minimum cruise SFC at 35,000 feet and Mach 0.8. A remarkable decrease in SFC is observed over the last 20 years--30 percent. The improvements are due mainly to an increase in bypass ratio, and therefore propulsion efficiency, advancing from the turbojet, bypass zero, in the early 1950's, to the high-bypass-ratio turbofans, 5 to 8, in 1970. Figure 4.11 shows the minimum SFC versus bypass ratio, indicating the large effect of bypass ratio on SFC.

ENGINE THRUST TO WEIGHT RATIO



ENGINE S.F.C.

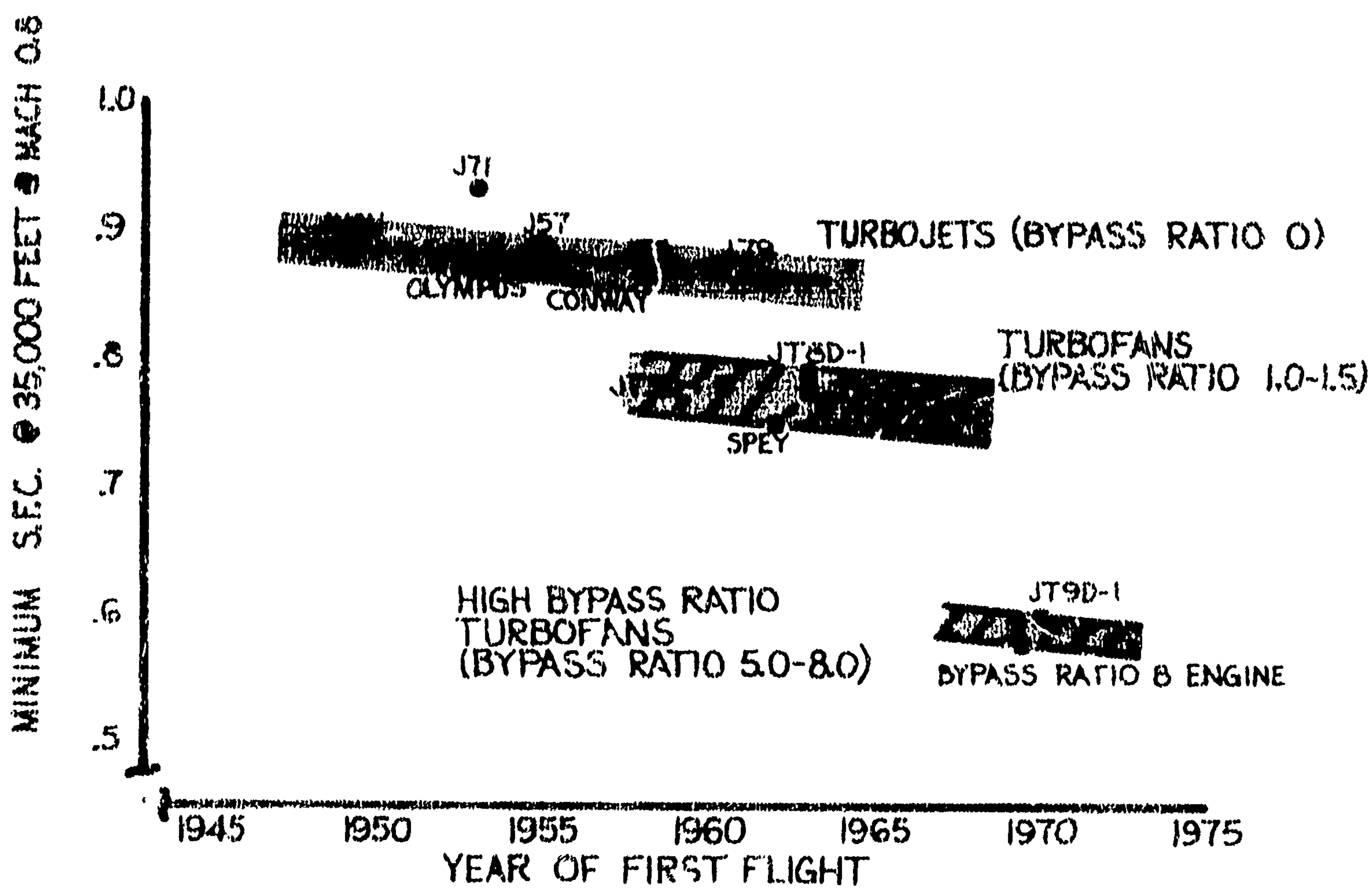


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Fig. 4-10

THE BOEING COMPANY

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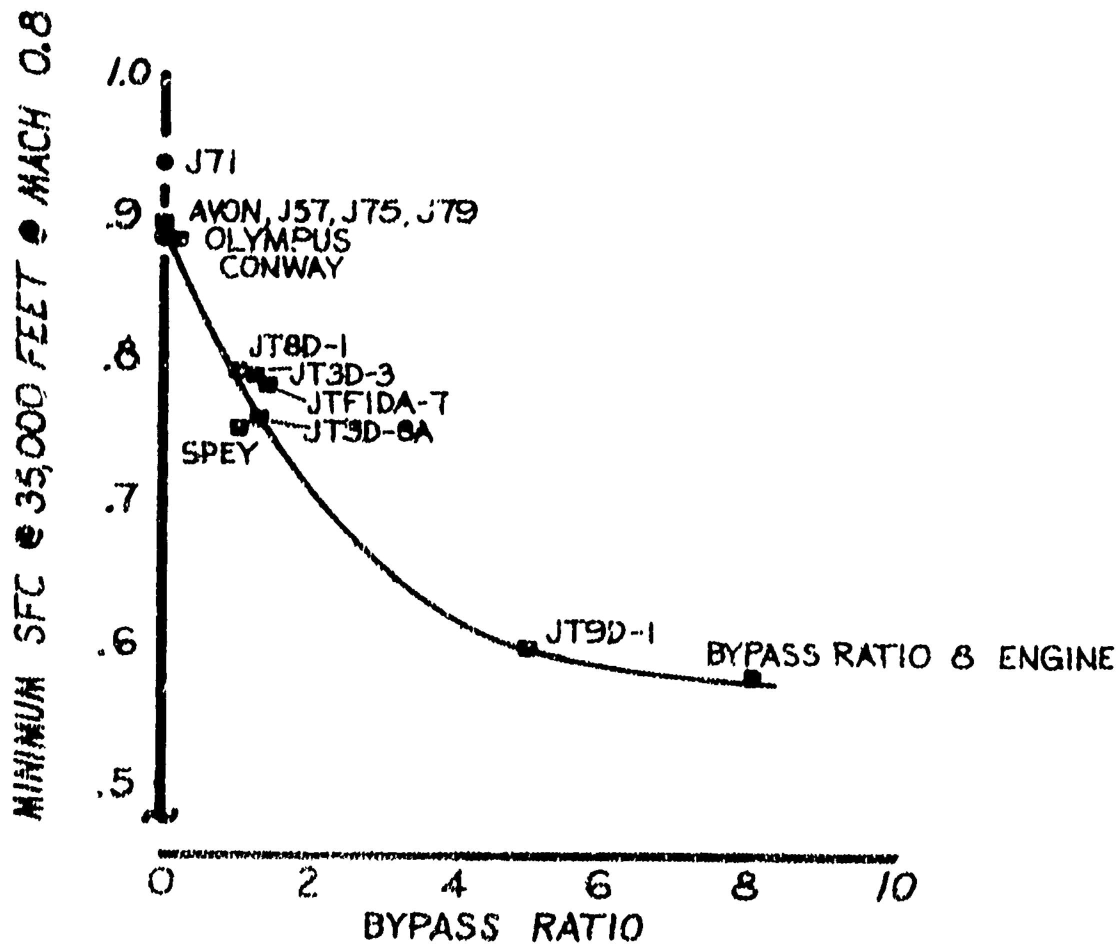
EFFECT OF BYPASS RATIO ON S.F.C.



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THE BOEING COMPANY

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Fig. 4-11

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Component technology improvements have made the high-bypass-ratio engine possible since it is not economical without high turbine inlet temperature and high compressor pressure ratio. Turbine temperatures have increased from 1400°F in the early engines to about 2300°F in some of the later high-bypass-ratio engines. Compressor pressure ratios have increased from about 10 to 25 in this period. It is expected that the SFC will decrease some in the 1975 engine, possibly by 2 percent. As indicated in Fig. 4.11, further increases in bypass ratio do not have a significant effect on the SFC's of a high-bypass-ratio, subsonic commercial engine. Therefore, the expected improvement will be largely due to improved component efficiency. Since weight and SFC are traded in the development and design of an engine, specific improvements will be a function of the engine's application.

4.3 1974 Time Period Point Airplane Design Characteristics

Based on the aerodynamic and propulsion technology just described, as well as improved structural weights due to higher allowables and the use of titanium, airplanes were designed at Mach .90 and 1.2. These airplanes were designed to make them comparable on a transcontinental U.S. segment with the SST. The four principal design rules are noted below:

- Design range--------------------JFK - SFO
- Passengers----------------------261 (20% First, 80% Tourist)
- Maximum Approach Speed----------135 Knots
- Technology Level----------------1970 Go-ahead, 1974 Airline Operation

4.3.1 Design Cruise Mach Number .90 Airplane

The Mach .90 airplane (Model 755-300) is shown in Fig. 4.12, along with some principal characteristics. The three-engine design is about 215 feet long, 132 feet wing span, and has a takeoff gross weight of 350,000 pounds. The low aspect ratio (6.5) wing is swept 42.5°, and incorporates sophisticated high lift devices. The fuselage has a circular cross section, suitable for eight abreast seating in the tourist section. The three high bypass ratio (5-8) engines are installed with two on the wings, similar to the Boeing 747 arrangement, and one at the base of the vertical tail. In order to assess the technology level of the Model 755-300, the technical improvements incorporated in the design are noted below:

ITEM	IMPROVEMENT	COMPARED TO
$(\frac{L}{D})$ Cruise	0.5	747
$M_{Critical}$	.03	747
$C_{L_{Max}}$	20%	727
OEW	5%	747 Technology
$(\frac{T}{W})$ SLS	15%	JT9D
SFC_{Cruise}	2%	JT9D

These advancements are believed achievable in the time period specified, and can be realistically forecast for airline operation in the mid-1970's.

4.3.2 Design Cruise Mach Number 1.20 Airplane

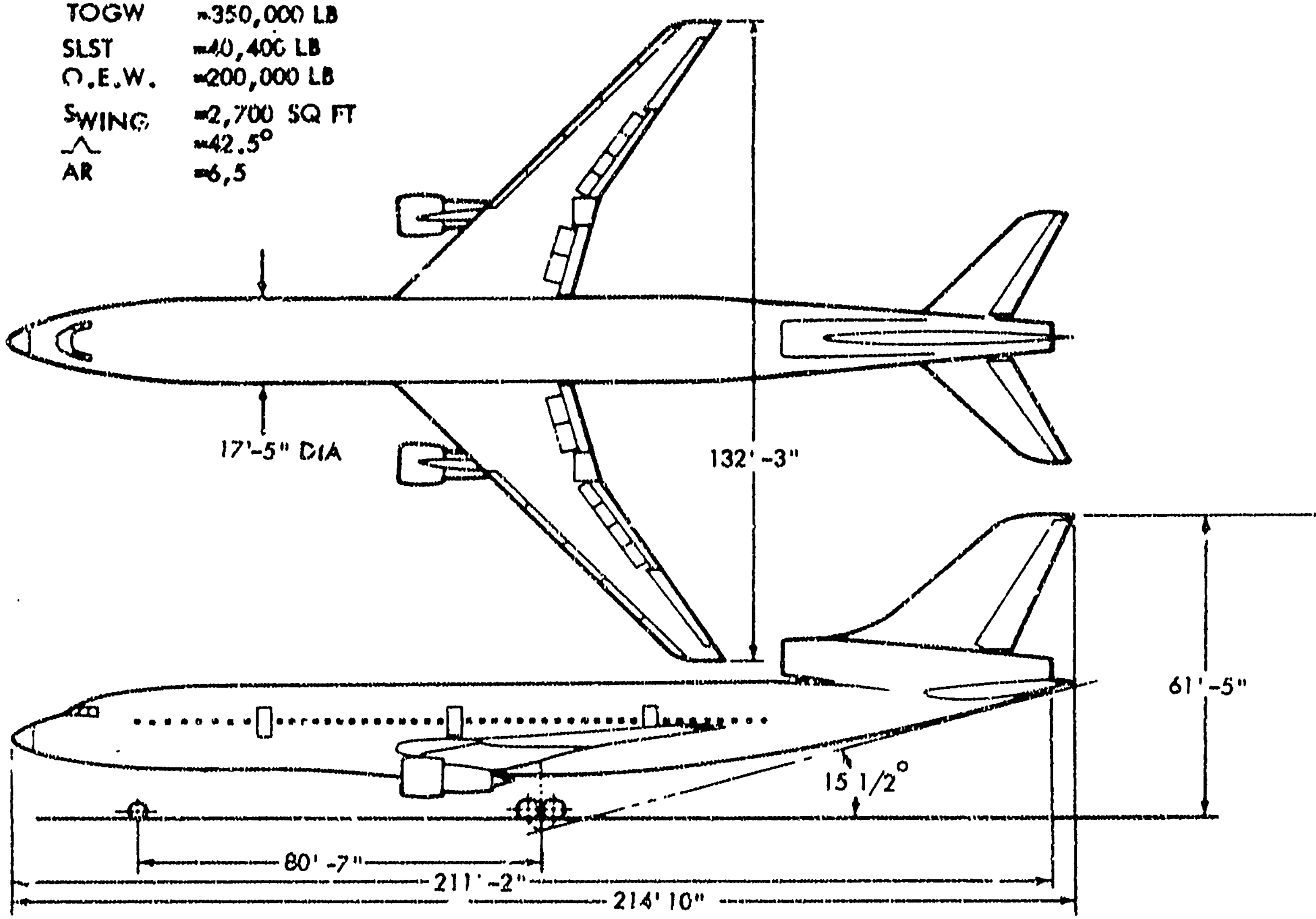
The Mach 1.2 airplane is presented in Fig. 4.13. While designed to Mach 1.2 cruise capability, it would normally operated in the sonic

9/11

MODEL 755 - 300

MACH CRUISE = 0.9

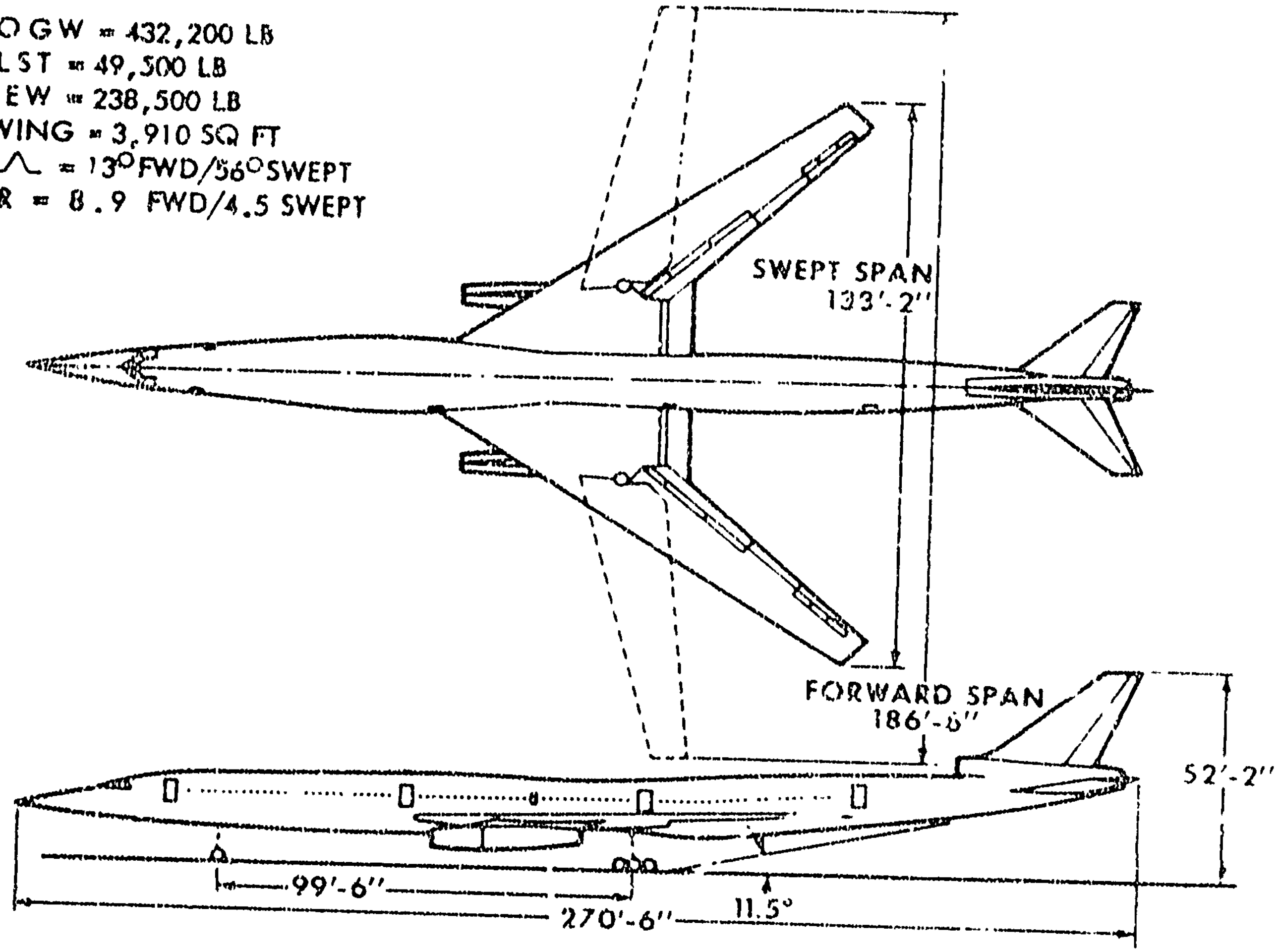
TOGW	=350,000 LB
SLST	=40,400 LB
O.E.W.	=200,000 LB
S_{WING}	=2,700 SQ FT
Λ	=42.5°
AR	=6.5



MODEL 742 - 240

MACH CRUISE = 1.2

TOGW = 432,200 LB
SLST = 49,500 LB
OEW = 238,500 LB
S_{WING} = 3,910 SQ FT
Λ = 13° FWD/56° SWEPT
AR = 8.9 FWD/4.5 SWEPT



boom cut-off region at speeds between Mach 1.05 and 1.15 for the reasons discussed in Para 4.1. This three-engine, variable-sweep aircraft has an overall length of about 271 feet and a wing span of 138 feet in the cruise configuration. With the wings swept forward for takeoff and landing, the span increases to 187 feet. The aspect ratio 4.5 wing has a sweep of 56° and incorporates advanced, double-slotted flaps. The fuselage is area ruled to minimize Mach 1.2 volume wave drag, resulting in a minimum cross section identical to the 707 body and a maximum cross section appreciably greater. Six abreast tourist seating is possible throughout the entire passenger cabin. The three 0.6 bypass ratio engines are installed in a similar manner to the 755-300. Technical improvements incorporated in the design are consistent with those used for the 755-300, with performance substantiated by the wind tunnel results presented in Fig. 4.8.

4.4 Economic Comparison

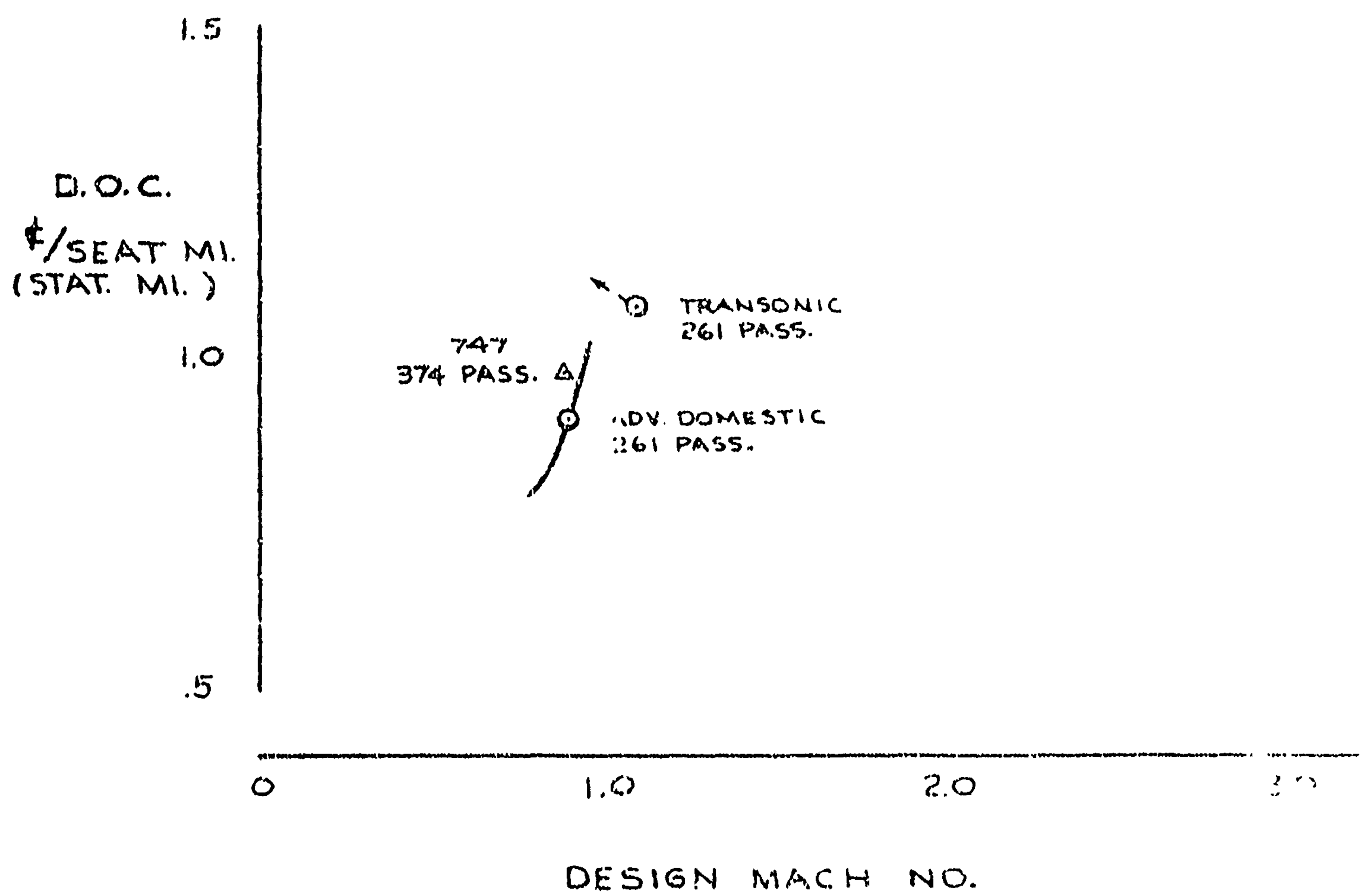
As shown in Fig. 4.14, an economic comparison of the Mach .90 and 1.20 airplanes with the 747 airplane shows the following:

- An advanced technology airplane designed at Mach .90 will operate under the conditions shown for about 7% less than the 747, while the Mach 1.2 airplane is 10% above.

- Relaxing the cruise speed requirement from Mach .90 to Mach .80 results in an airplane approximately 20% lower in operating cost than the 747.

D.O.C. COMPARISON

DOMESTIC (80% T 20% F)
JFK - SFO



These airplanes have been designed for 261 passengers; the I.O.C. of the Mach .90 airplane could be improved somewhat if designed for a larger payload.

APPENDIX A

DESIGN INPUTS FOR DOMESTIC SST DESIGNED TO MEET ACCEPTABLE SONIC BOOM OVERPRESSURES

TABLE OF CONTENTS

1.0 AERODYNAMIC AND SONIC BOOM CHARACTERISTICS

The requirement of low sonic boom characteristics for a domestic airplane dictates a number of configuration features. A long, slender body combined with a wing that distributes the lift over a long length help alleviate sonic boom overpressures. These two features are also compatible with the requirement of low drag in the supersonic flight regime. A high aspect ratio wing provides good climb characteristics that allow the airplane to fly higher at a given Mach number, thus reducing sonic boom overpressures at the ground, but the choice of an aspect ratio for the wing must be tempered by wave drag considerations. Nacelles should be placed well aft to provide favorable interference effects.

These features were combined into a $M = 2.7$ baseline configuration, using a twin-engine arrangement with a 6000 $ft.^2$ wing having a leading edge sweep angle of 74°, an average thickness ratio of 2.75 percent, and an aspect ratio of 1.6. The body, sized to carry 85 passengers, was highly area-ruled for low drag and sonic boom.

1.1 Aerodynamic Characteristics

The parametric study was based on initial estimates of aerodynamic characteristics for a low-boom domestic airplane. These initial estimates were later confirmed by a detailed analysis of the baseline configuration.

Maximum lift-drag ratio as a function of Mach number, is shown for the baseline configuration in Figure A-1. The design is estimated to have an $(L/D)_{max}$ of 9.5 at the cruise Mach number of 2.7 and an altitude of 65,000 feet. Estimated values for 42 and 128 passenger versions

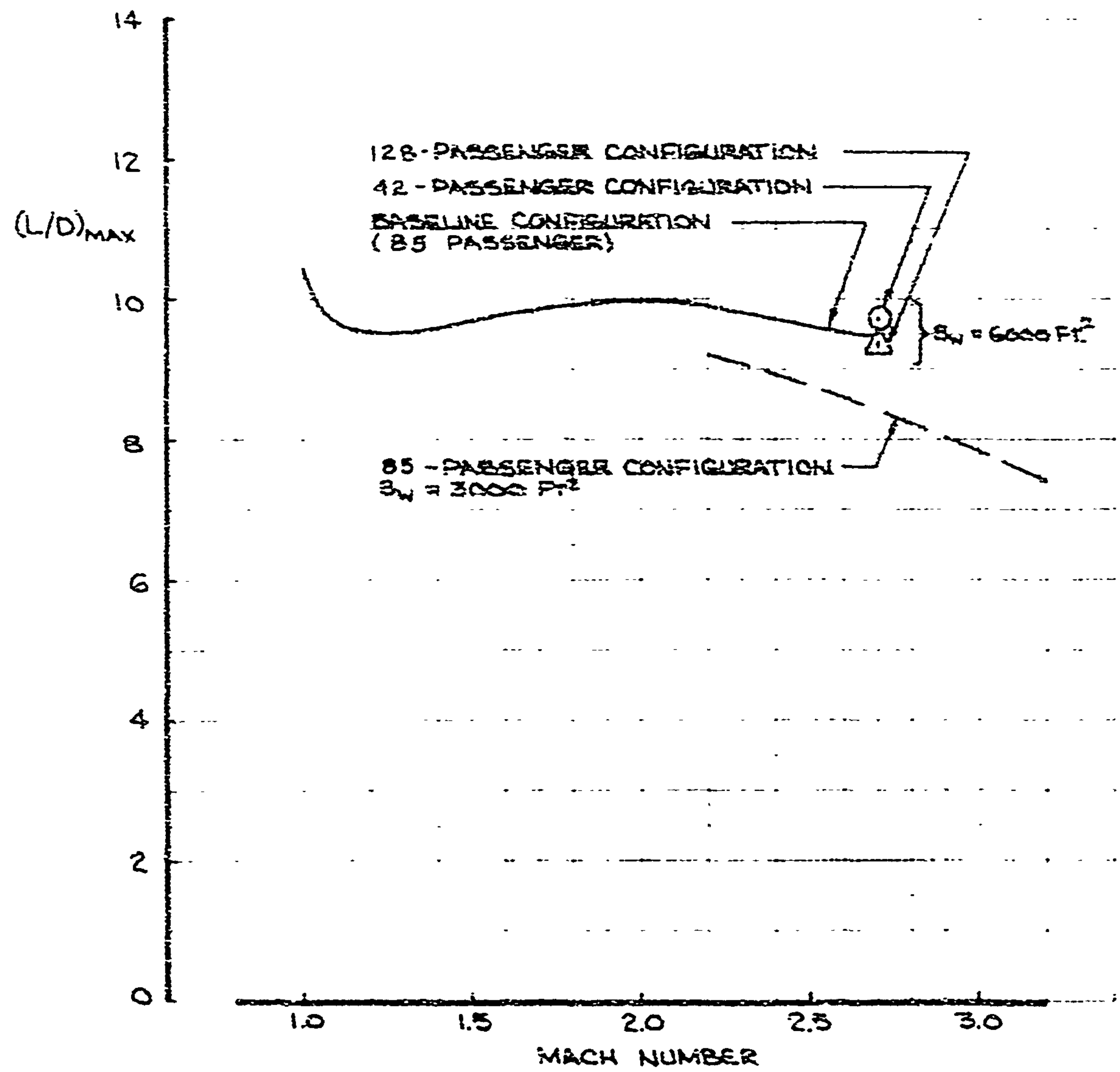


FIG. A-1 MAXIMUM LIFT-DRAG RATIOS

of the baseline design are also shown for M = 2.7. These two alternate configurations were developed by changing body geometry while holding wing geometry and area (6000 ft.^{2}) constant. The drag increment for the 42-passenger body was assumed to be one-half the drag of the baseline body. The 128-passenger body was assumed to be the same as the 85-passenger body with the addition of a 21-foot cylindrical section. The body drag for the larger body included the additional friction and air conditioning drags.

Maximum lift-drag ratios for a 85-passenger configuration with a wing area of 3000 square feet are also shown in Figure A-1. The aerodynamic characteristics for this configuration were obtained by adjusting the 6000 ft.^{2} configuration characteristics to the smaller wing area. This configuration is representative of an airplane meeting the range and sonic boom requirements of this study. The greater loss in $(L/D)_{max}$ with increasing Mach number for the 3000 ft.^{2} configuration is caused by the assumed variation of wave drag with Mach number and the relative sizes of the wing and body.

A Mach 2.7 area distribution for the baseline configuration is shown in Figure A-2.

1.2 <u>Sonic Boom Characteristics</u>

The sonic boom characteristics of the domestic airplanes were estimated at the beginning of the parametric studies by the following procedure: The sonic boom characteristics of a number of previously studied configurations were examined. These characteristics were all adjusted to a common wing area and a "typical" curve of sonic boom

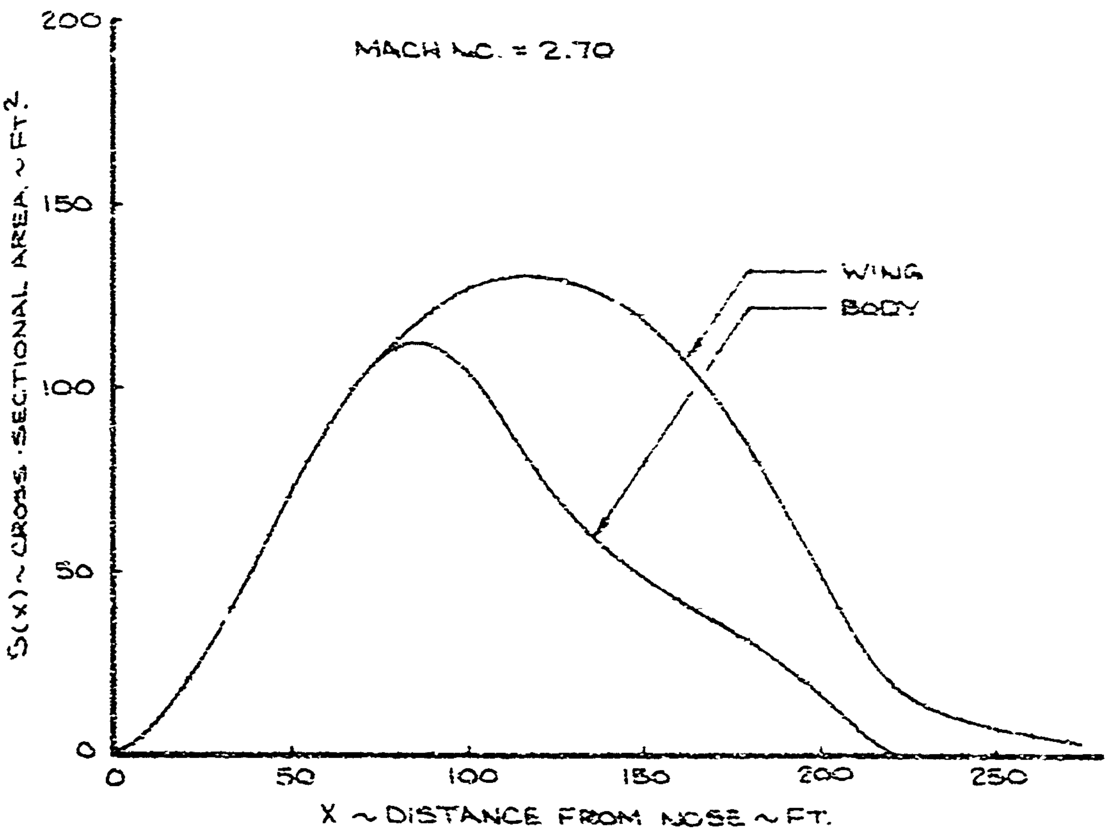


FIG. A-2 BASELINE CONFIGURATION MACH 2.7 AREA DISTRIBUTION overpressure parameter versus lift coefficient was determined. The typical curve was then adjusted by reducing the value of the overpressure by 5 percent at zero lift coefficient; leaving it unchanged at high lift coefficients, where the boom would be lift dominated; and fairing a variation between these two extremes. This established the sonic boom characteristics for one wing area. In this study the initial wing area was 6000 square feet. Characteristics for other wing areas were obtained by adjusting the boom, by wing area to the three-eighths power at high lift coefficients, and by total frontal area to the one-half power at zero lift. The estimate, thus obtained, served as the basis for the parametric studies and is compared with the sonic boom characteristics of the B-2707 and SCAT-15F in Figure A-3. From this comparison it can be seen that the characteristics assumed for the parametric study are somewhat optimistic but not to an excessive degree.

In order to confirm the level of sonic boom characteristics established for the parametric studies, the characteristics of the baseline configuration and a refinement of the baseline were determined by a theoretical analysis. Both configurations were assumed to have a wing area of 6000 square feet. The refined baseline configuration represented a point-design that was optimized to produce a minimum overpressure at Mach 1.3, an altitude of 40,000 ft. and for a weight of 250,000 pounds. This configuration included a more favorable body shape and a different longitudinal positioning of the wing with respect to the body. Maximum transonic sonic boom overpressures as a function of altitude for the baseline and refined baseline configurations are compared with values

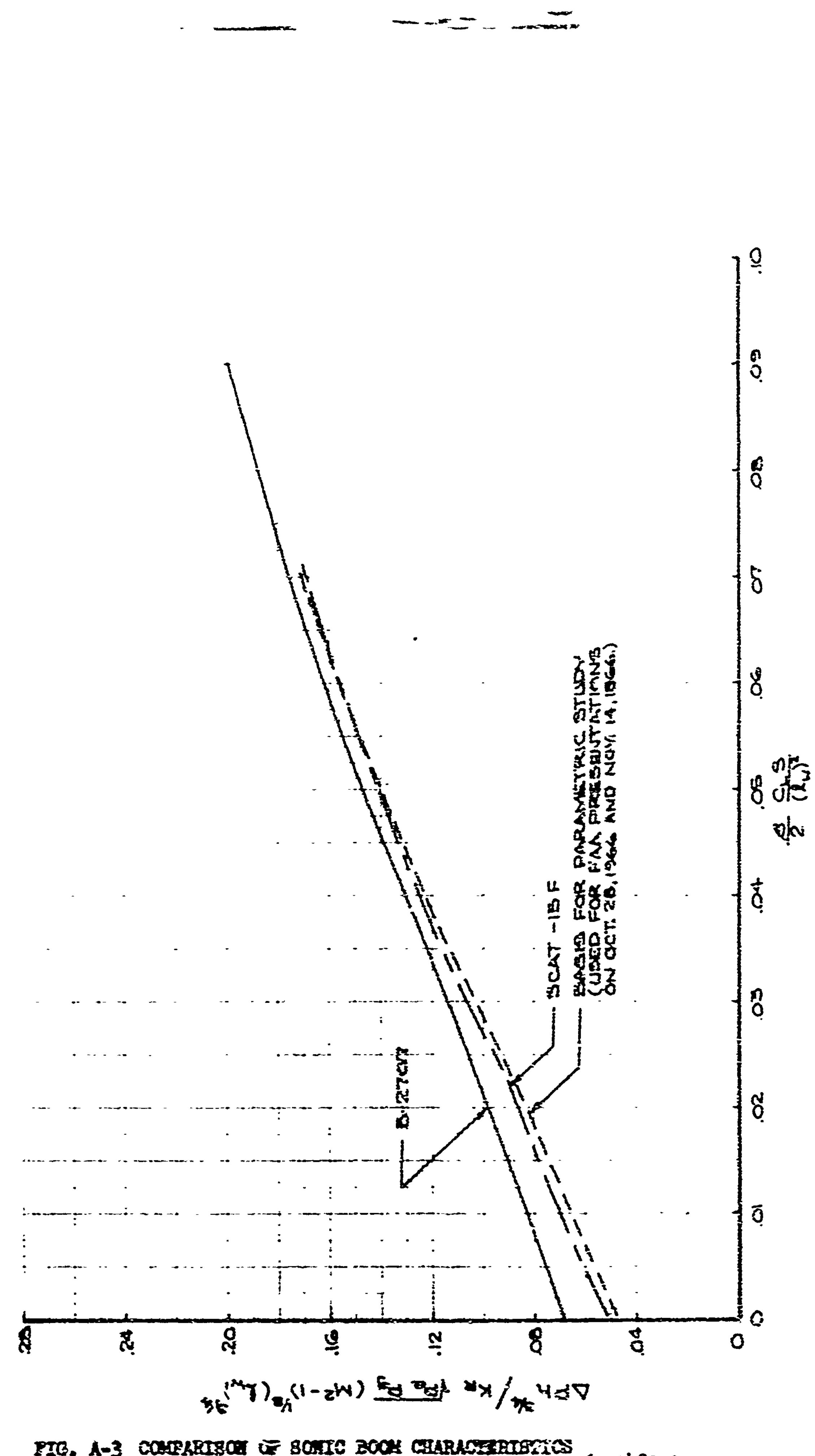


FIG. A-3 COMPARISON OF SONIC BOOM CHARACTERISTICS obtained from the characteristics established for the parametric studies in Figure A-4. The range of M = 1.3 altitudes covered during the parametric studies is also shown. This comparison illustrates the validity of the characteristics assumed for the parametric studies as well as the significant improvements that can be achieved with a point-design. The Mach 1.3 area distributions used in the sonic boom analyses of the baseline and refined baseline airplanes are shown in Figures A-5 and A-6.

In the preceeding discussion, the approach to the problem of minimizing the overpressures produced on the ground by a domestic airplane during transonic climb and acceleration was to optimize the geometry for minimum boom at Mach 1.3. An alternate approach would optimize the airplane geometry for best cruise sonic boom, and use specialized flight procedures to minimize or eliminate ground overpressures during climb and acceleration. These transonic flight procedures would involve combinations of altitude, Mach number, and flight path angle that prevent the airplane shock waves from reaching the ground. Under these conditions the ray paths of the shock wave are turned to a horizontal direction, i.e. "cut off", before they reach the ground. Increasing the flight path angle increases the cut-off Mach number for a given altitude and thereby reduces the ground area subjected to transonic overpressures. A different flight procedure could be employed on flights where the takeoff is made near ocean areas. For these flights the problem of transonic overpressures on the ground could be avoided entirely by conducting the climb and

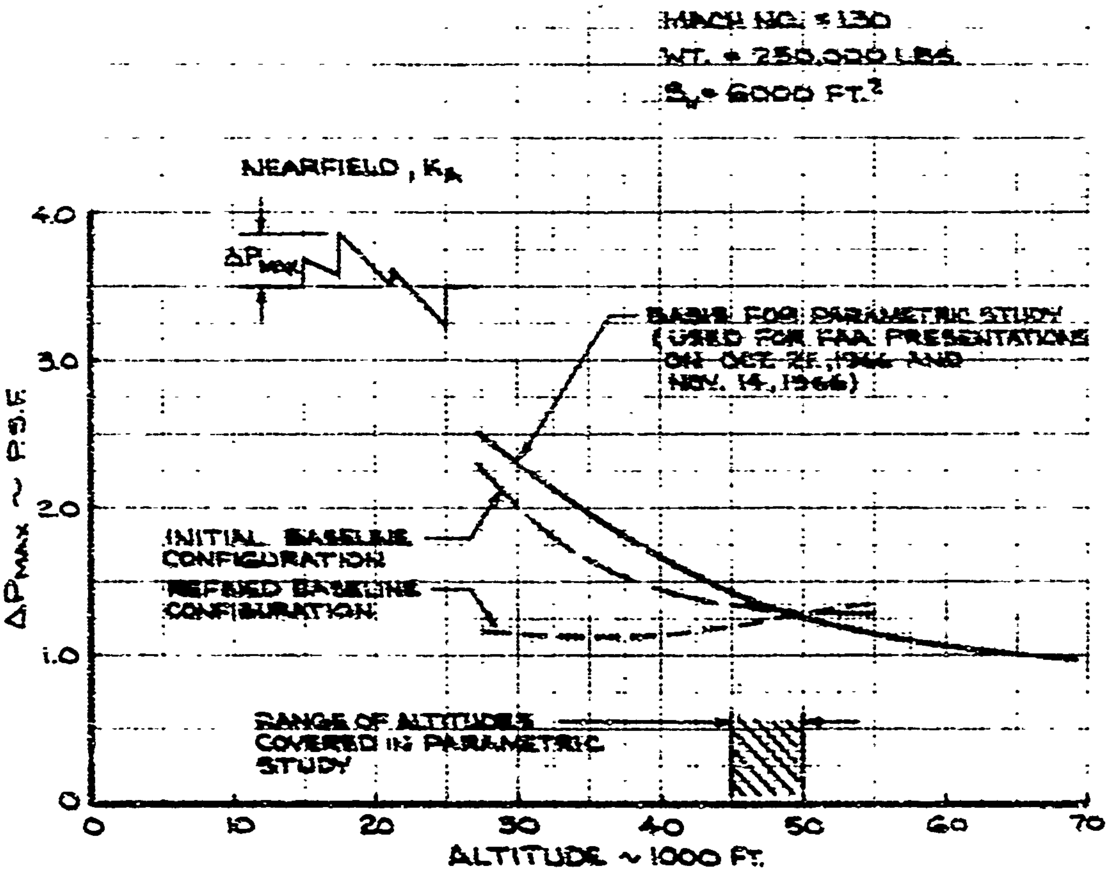


FIG. A-4 COMPARISON OF MAXIMUM TRANSONIC OVERPRESSURES

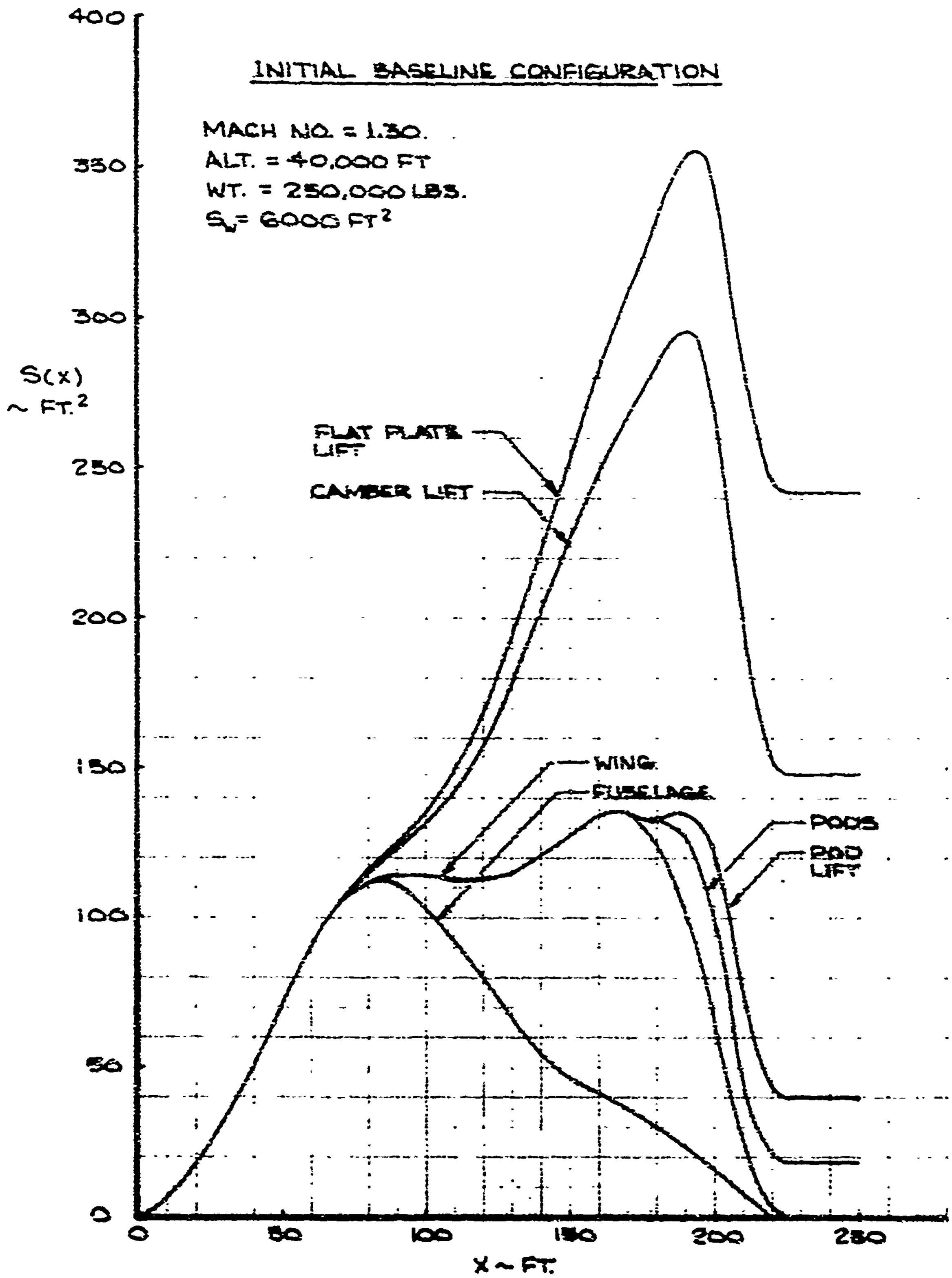


FIG. A-5 INITIAL BASELINE MACH 1.3 AREA DISTRIBUTION

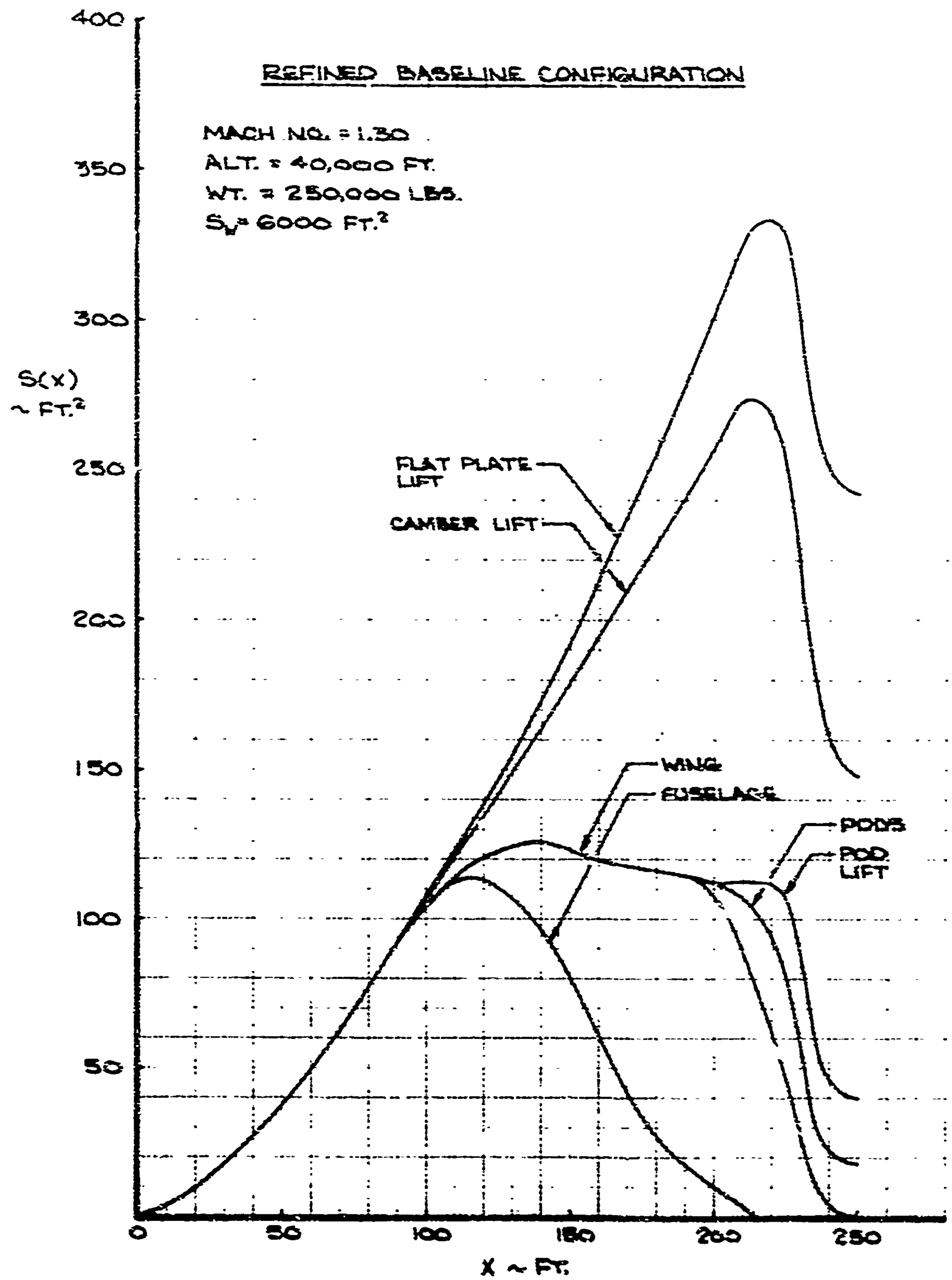


FIG. A-6 REFINED BASELINE CONFIGURATION
MACH 1.3 AREA DISTRIBUTION

acceleration phase over water. Additional study would be required to establish the trades for these alternate approaches.

2.0 POWER PLANT CHARACTERISTICS

2.1 Cycle Configuration

Power plant performance studies were based upon two engine cycles used in current supersonic transport engine design.

(a) A medium pressure ratio, high maximum turbine inlet temperature, fully augmented single spool turbojet.

(b) A high pressure ratio, low maximum turbine inlet temperature, partially augmented two-spool turbojet.

These cycles are designated the Boeing Medium Pressure Ratio Cycle and Boeing High Pressure Ratio Cycle, respectively.

2.2 Mach Variations

Engine performance was generated for both engines over the following range of airplane operating Mach numbers:

Medium Pressure Ratio Engine	M = 3.2	at	65,000	ft Altitude
" " " "	M = 2.7	"	"	" "
" " " "	M = 2.2	"	"	" "
High Pressure Ratio Engine	M = 2.2	"	60,000	" "
" " " "	M = 1.7	"	50,000	" "

Summary engine characteristics and installed performance are given in Table A-A and Figures A-7 and A-8. Engine weights include the augmentor, thrust reverser and exhaust nozzle.

2.3 Engine Cycle

The cycle pressure ratios used in the engine airplane matching studies provide maximum performance over the range of Mach numbers considered. Turbofan cycle investigations were not pursued due to time limitations though it was apparent that this type of engine would also be suitable over the range of Mach numbers considered.

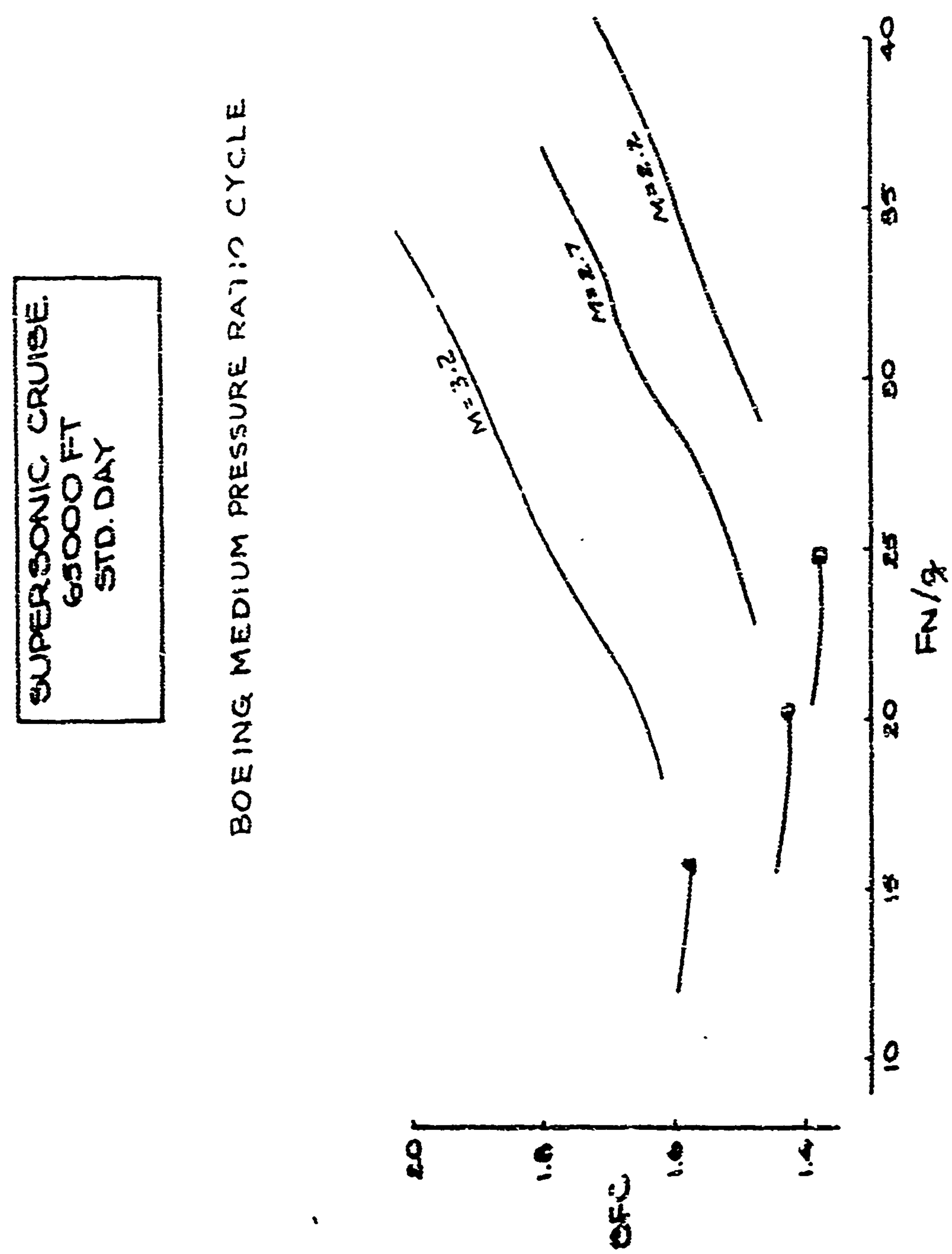
Of the two cycles, that with the high pressure ratio, two-spool compressor was found to be more suitable for the two lowest Mach number applications. The inherent high flow capability of the two-spool machine at the cruise condition where the inlet is sized allows better inlet/engine matching during transonic climb and acceleration. With regard to the lower turbine inlet temperature of this cycle, an improvement of 100-200°F could be applied, which would still allow the turbine to operate within the current supersonic engine state of the art. Such an increase may require some weight increase and increased cooling flows, but the end result could allow the Mach 1.7 and 2.2 airplanes to cruise without augmentation.

The medium pressure ratio high temperature cycle was found to be very suitable for the Mach range 2.7 - 3.2. At the Mach 2.2 condition, it was found necessary to high-flow the single spool engine for engine/inlet matching considerations. A 10% flow increase at cruise was obtained by engine overspeeding, the accompanying weight increase being compensated by the weight reduction resulting from the lower environmental temperature. For the Mach 1.7 condition, the engine demand became difficult to match with an inlet having external compression; therefore, the M 1.7 condition for this cycle was not included in the study.

TABLE A-A

Engine Type	Single Spool Medium Pressure Ratio			Two Spool High Pressure Ratio	
Design Mach No.	3.2	2.7	2.2	2.2	1.7
$W\sqrt{\theta_2}/\delta_2$ lb/sec	633	633	633	432	432
Takeoff					
Max. Aug. F_N	63,200	63,200	63,200	36,200	36,300
Transonic					
M = 1.2 Alt. = 45,000 ft.					
F_N/q	76.5	76.5	76.5	36.5	36.5
SFC	1.848	1.848	1.848	1.445	1.445
Supersonic Cruise					
W_a lb/sec	470	291	203	187	195
Altitude - ft.	65,000	65,000	65,000	60,000	50,000
F_N/q Min. Aug.	18.3	22.8	28.7	15.4	23.3
SFC Min. Aug.	1.62	1.475	1.466	1.46	1.315
Subsonic Cruise					
M = 0.8 Alt. = 36,150 ft.					
F_N	5000	5000	5000	5000	5000
SFC	1.08	1.08	1.08	.89	.89
Inlet					
Type	(2)	(2)	(1)	(1)	(3)
Length/Dia.	2.0	2.0	1.8	1.8	1.5
Weight					
Engine lb.	11,800	11,237	11,090	8,450	8,050
Nozzle Size Ins.	83.5	74.2	71.0	55.0	48.5
% Augmentation	100	100	100	30	30

(1) Axisymmetric External-Internal Compression Inlet with Translating Centerbody
(2) Axisymmetric External-Internal Compression Inlet with Variable Centerbody
(3) Axisymmetric External Compression with Translating Centerbody



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CHECK			J.H	11/10/66		Fig. A-7
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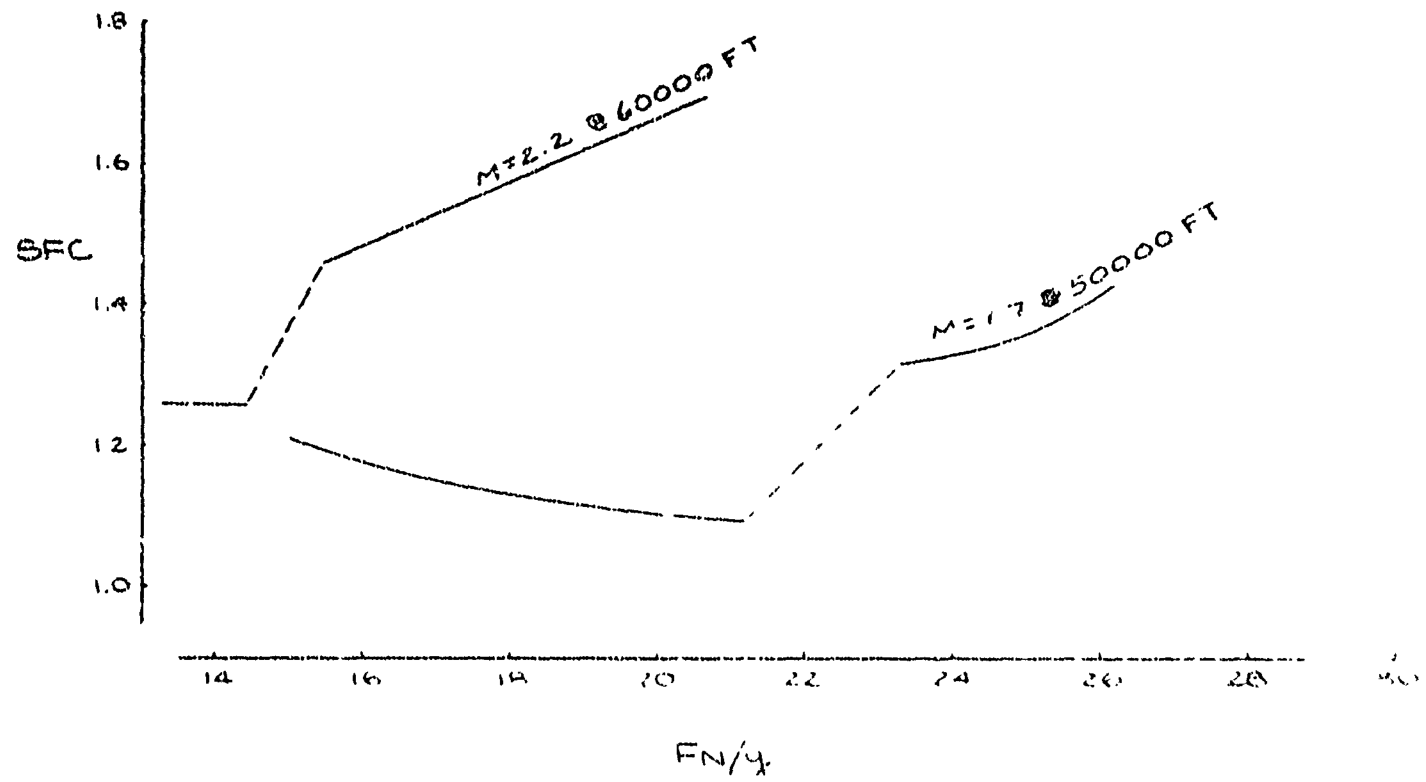
SUPERSONIC CRUISE
STANDARD DAY

BOEING HIGH PRESSURE RATIO CYCLE



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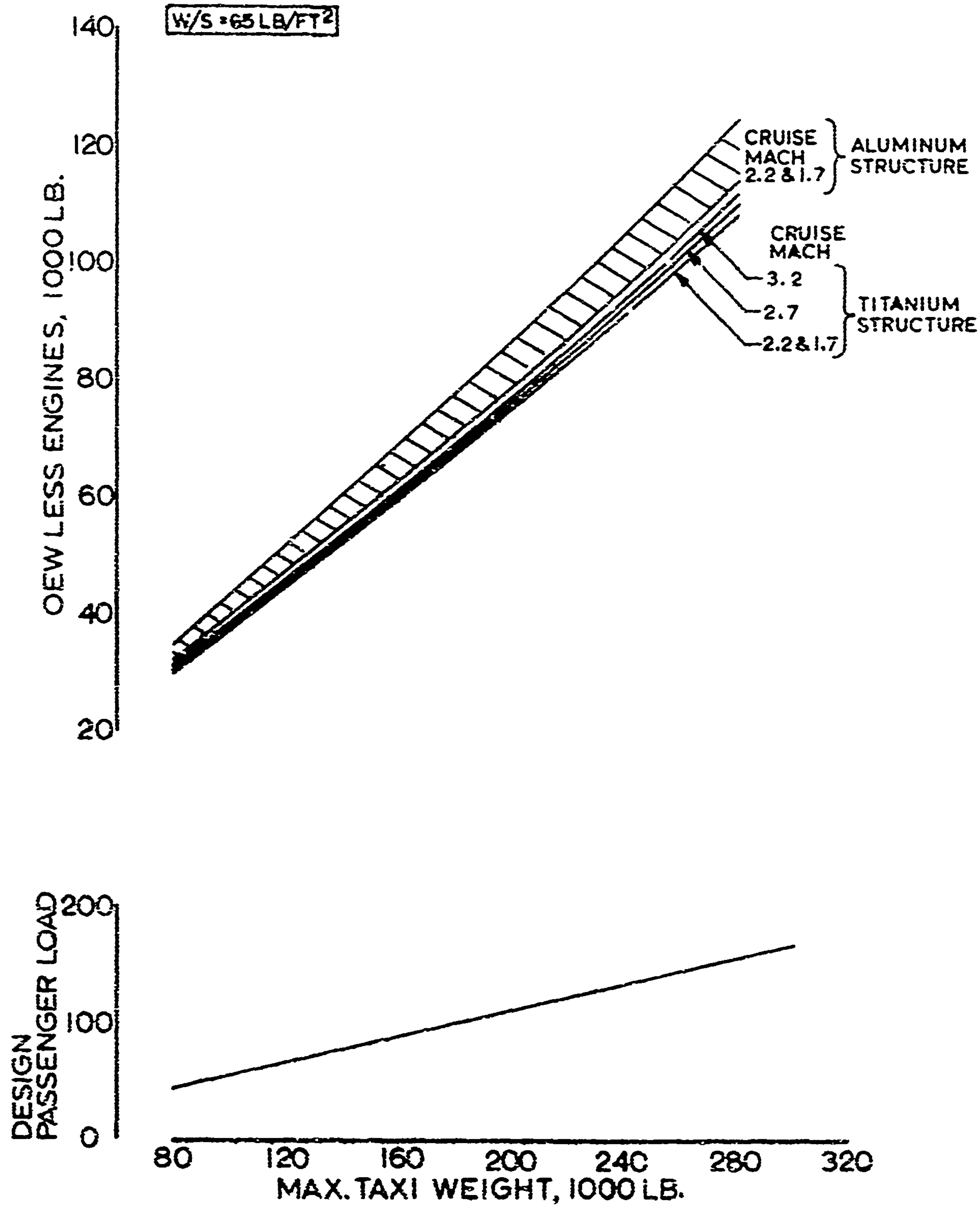
3.0 WEIGHTS DATA

The weights used for this study are parametric extrapolations of B-2707 information. Weights are shown in Figure A-9 for a representative wing loading of 65 psf. These weights include estimates for body fuel increments as required. The primary structure for all Mach numbers is titanium.

The increments shown for design cruise Mach number variations include allowances for the changes in structure, systems, and power plant.

Figure A-10 shows the total propulsion pod weight versus airflow for both study engine cycles. Parametrically selected airplanes will be analyzed in more detail to refine the weight estimates.



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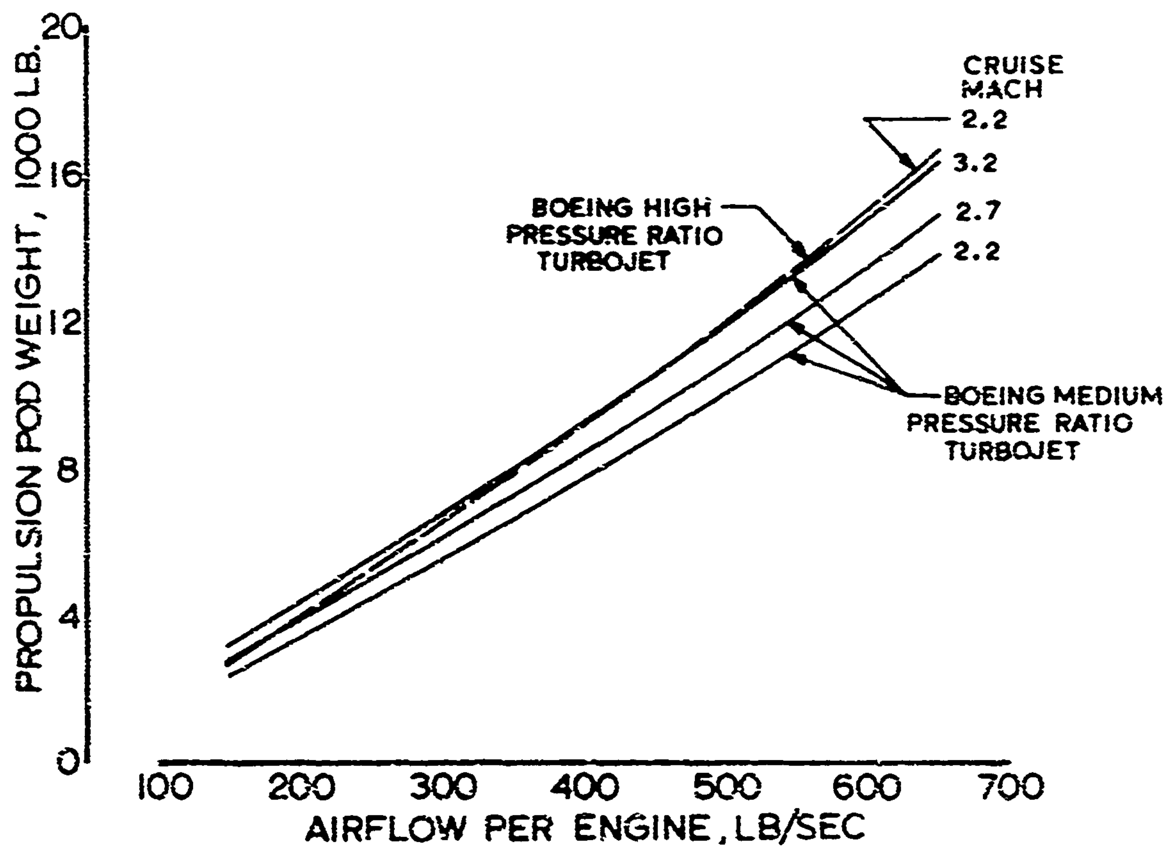

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Fig. A-10

THE BOEING COMPANY

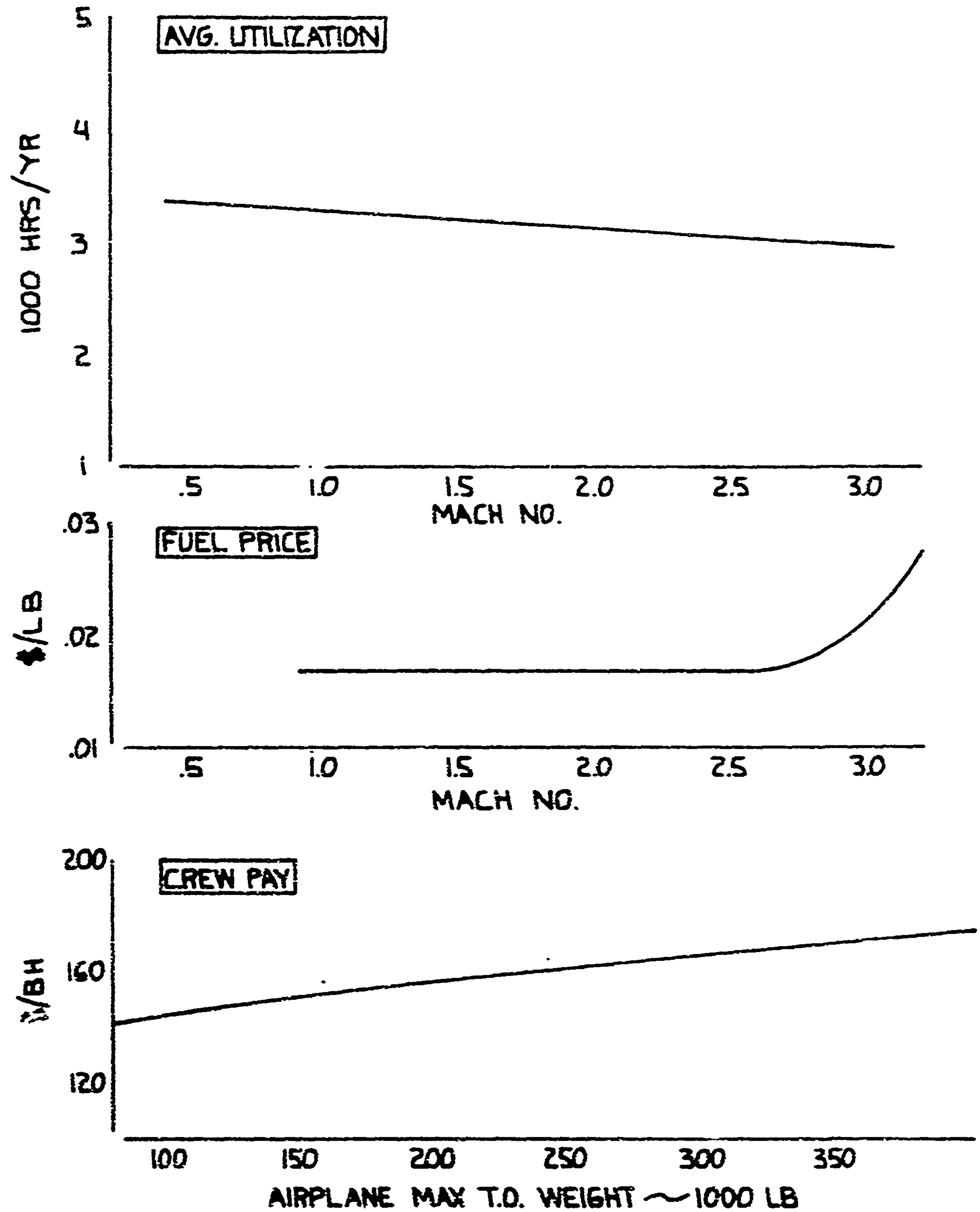
4.0 DIRECT OPERATING COST GROUND RULES

The direct operating costs of the study airplanes were calculated in accordance with the FAA Supersonic Transport Economic Model Ground Rules (SST 66-3, June 30, 1966) with the exception of crew pay.

Sales prices were based on production of 200 airplanes and development costs amortized over 300 airplanes. The prices of subsonic airplanes shown for comparison have been adjusted to reflect a development program similar to that of the SST. However, development costs were included in airframe and engine price to determine maintenance costs. While the design ranges used to determine maximum gross weight included the effect of wind and temperature, the operating costs are based on a standard day, no wind.

Annual utilization was varied from 3300 block hours per year for subsonic airplanes to 2920 block hours per year at Mach 3.2 as shown on Fig. A-11. Fuel price was constant for all designs Mach numbers up to 2.7, and increased 63 percent for M 3.2 as shown on Fig. A-11. Crew pay was raised with design gross weight as shown.

DOMESTIC OPERATION COST FACTORS



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