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PRELIMINARY STUDY OF SUPERSONIC-TRANSPORT CONFIGURATIONS WITH LOW VALUES OF SONIC BOOM

by James F. Dugan, Jr.

Lewis Research Center

Cleveland, Ohio 44135

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16. Abstract

A parametric study of low-boom supersonic-transport airplanes with conventional configurations was made to identify the features of specific configurations that promise relatively low sonic-boom overpressures (less than 47.9 N/m^2 (1.0 lb/ft^2)). The range of values considered was gross weight from 28 300 to 170 000 kg (62 500 to 375 000 lb); cruise Mach numbers of 2 to 3.2; and wing loadings of 1436, 2870, and 4309 N/m^2 (30, 60, and 90 lb/ft^2). Fuselage length was varied from 49.1 to 102.4 m (161 to 336 ft) and fuselage diameter from 2.75 to 3.98 m (9.03 to 13.05 ft). A nominal Mach 2 configuration weighing 56 700 kg (125 000 lb) and having a wing loading of 2870 N/m^2 (60 lb/ft^2) was selected; and its gross geometric, aerodynamic, and structural features were estimated. At a cruise altitude of 18 300 m (60 000 ft), lift-drag ratio was estimated to be 7.35, while sonic-boom overpressure calculated from the real-field equations of NASA TN D-4935 was 41.7 N/m^2 (0.87 lb/ft^2). It was not necessary to assume any radical improvements in the sonic-boom characteristics of the airframe configuration in order to achieve this low value. Takeoff thrust loading using four afterburning turbojet engines at maximum dry thrust was 0.32. Payload for a 4440-km (2400-n mi) range was 16.7 percent of gross weight, giving a direct operating cost of 0.82 cent per seat statute mile. A domestic supersonic transport having these characteristics might be acceptable to the general public and economically feasible. Because of the simplified nature of the study, the results should be considered as speculative and subject to verification by more detailed studies.

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SUMMARY

A parametric study of low-boom supersonic-transport airplanes with conventional configurations was made to identify the features of specific configurations that promise relatively low sonic-boom overpressures (less than 47.9 N/m^2 (1.0 lb/ft^2)). The range of values considered was gross weight from 28 300 to 170 000 kilograms (62 500 to 375 000 lb); cruise Mach numbers of 2 to 3.2; and wing loadings of 1436, 2870, and 4309 N/m^2 (30, 60, and 90 lb/ft^2). Fuselage length was varied from 49.1 to 102.4 meters (161 to 336 ft) and fuselage diameter from 2.75 to 3.98 meters (9.03 to 13.05 ft).

A nominal Mach 2 configuration weighing 56 700 kilograms (125 000 lb) and having a wing loading of 2870 N/m^2 (60 lb/ft^2) was selected; and its gross geometric, aerodynamic, and structural features were estimated. At a cruise altitude of 18 300 meters (60 000 ft), lift-drag ratio was estimated to be 7.35 while sonic boom calculated from the real-field equations of NASA TN D-4395 was 41.7 N/m^2 (0.87 lb/ft^2). It was not necessary to assume any radical improvements in the sonic-boom characteristics of the airframe configuration in order to achieve this low value since it is primarily a result of the low gross weight. Takeoff thrust loading using four afterburning turbojet engines at maximum dry thrust was 0.32. Payload for a 4440-kilometer (2400-n mi) range was 16.7 percent of gross weight, giving a direct operating cost of 0.82 cent per seat statute mile. A domestic supersonic transport having these characteristics might be acceptable to the general public and economically feasible. A more detailed study of this class of vehicle seems warranted.

The methods used in this preliminary study to calculate aerodynamics, weights, and sonic-boom overpressure are quite simple but did give results quite close to published results on particular supersonic transport designs. Because of the simplified nature of the study, however, the results should be considered as speculative and subject to verification by more detailed studies.

INTRODUCTION

The first generation of commercial supersonic transports will be flown primarily on over-water routes because their estimated cruise sonic-boom overpressure of over 95.7 N/m^2 (2 lb/ft^2) is deemed unacceptable to the general public. From an annoyance viewpoint, the lower the boom overpressure the better. However, reducing the boom overpressure below the order of 95.7 N/m^2 (2 lb/ft^2) tends to hurt airplane performance and economy. Therefore, it is relevant to debate just how much boom will be acceptable to the populace.

The acceptable level of sonic-boom overpressure is not well defined since the lower the boom overpressure is made, the fewer the people that are likely to be annoyed. In reference 1, the acceptable boom overpressure during cruise over the sea is claimed to be 14.4 N/m^2 (0.3 lb/ft^2) while that for cruise over land is estimated at 4.8 to 9.6 N/m^2 (0.1 to 0.2 lb/ft^2). Other estimates claim a sonic-boom overpressure of 43.1 N/m^2 (0.9 lb/ft^2) might be acceptable. The inability of the first-generation SST's to meet boom levels like those just discussed precludes their use over land.

It is reported in reference 2 that the United States scheduled airline passenger traffic in 1977 is forecast to be 266 billion revenue passenger miles. Of this total 200 billion is domestic (i.e., over land). Since over 20 percent of this market is estimated to be for distance greater than 2414 kilometers (1500 miles), the potential market for a domestic low-boom supersonic transport is very large.

Some studies for such a domestic SST have already been performed. For example, North American Aviation, Inc., working under contract to the FAA, performed a feasibility study in which the NASA SCAT 15F design concept, which evolved from analytic and wind tunnel studies of supersonic transports, was retained and an attempt was made to develop an economically sound airplane for scheduled airline operation on domestic coast-to-coast routes. Some design compromises were found to be necessary, and the original NASA weight estimates were found to be overly optimistic, the net result being that peak overpressures in the region of 67 to 72 N/m^2 (1.4 to 1.5 lb/ft^2) were estimated at the conclusion of the study (p. 398 of ref. 3). Thus, in the North American study, a configuration highly refined for peak aerodynamic efficiency was modified somewhat for airline operation and found to have too high a sonic-boom overpressure.

A similar study of the SCAT 15F is presented in reference 4. While the direct operating cost (DOC) was estimated to be 1.1 cents per seat statute mile for a sonic-boom overpressure of 67 N/m^2 (1.4 lb/ft^2), it rose to a value of 3 cents per seat mile for a sonic-boom overpressure of 45.5 N/m^2 (0.95 lb/ft^2). The implication of this is that with current airframe and propulsion technology a supersonic transport of this configuration could be made acceptable to the public but would not be economically feasible.

Reference 5, however, has shown that it is possible to generate airplane biplane configurations that can reduce substantially the strength of the front and tail shocks of sonic booms for airplanes designed for transatlantic operations and for cross-country operations with reduced gross weights. In the latter case, sonic-boom values as low as 19.2 and 14.4 N/m^2 (0.4 and 0.3 lb/ft^2) are possible. Reference 5 analyzed only superficially the consequences of utilization of such concepts on airplane performance.

The present study presents preliminary estimates of the performance and economy of low-boom domestic SST's with conventional configurations. A low-boom configuration was selected from a parametric study in which gross weight, cruise Mach number, wing loading, cruise altitude, and fuselage dimensions were varied. Then engines were added and estimates were made of its aerodynamic, structural, and propulsive characteristics so that airplane performance and DOC could be calculated.

A nominal Mach 2 low-boom supersonic transport was identified. Its lift-drag ratio, structural weight, wing size and planform, and fuselage length and maximum diameter were estimated. The initial cruise altitude and engine size were varied to maximize payload for a 4440-kilometer (2400-n mi) range (Miami to Seattle range is 4410 km (2381 n mi)) and to illustrate the tradeoff between payload and sonic boom. The DOC and maximum sideline noise during lift-off were also calculated.

The methods used in this preliminary study to calculate aerodynamics, weights, and sonic-boom overpressure are quite simple but did give results quite close to published results on particular supersonic transport designs. Because of the simplified nature of the study, however, the results should be considered as speculative and subject to verification by more detailed studies.

Although SI units are given as the primary system and U.S. customary units as the secondary system, the work was done in the U.S. customary units.

METHOD OF ANALYSIS

The sonic-boom overpressure produced by a supersonic object is a function of its shape, length, lift, Mach number, and altitude. It has been found convenient to express these relations by plotting $\Delta p/p_r(h/l)^{0.75}\beta^{-0.25}K_r^{-1}$ against $(\beta/2)C_L(S/l^2)$ (ref. 6). (All symbols are defined in appendix A.) In reference 6 this relation (see appendix B) is determined for the shape that yields the lowest possible Δp in the far field (i.e., the simplified situation that is asymptotically approached for very large values of h/l). Both lift and volume effects are included. The far-field lower bound is shown in figure 1.

In this study, the procedures and equations developed in reference 7 were used to calculate the values of sonic-boom overpressure. The fuselage was taken to be a minimum-drag body having 0.67 times the drag of a conical body of equal length and frontal area (see p. 225 of ref. 8). Reference 7 shows that the maximum static-pressure rise of a minimum-drag body is approximated at all distances from the body by the simple closed-form expression presented in appendix B. The results of this expression are shown in figure 1 and indicate that the minimum-drag body gives values of far-field sonic-boom overpressure slightly greater than the lower bound and values of real-field sonic-boom overpressure slightly less than the far-field lower bound.

The equations used to calculate airplane aerodynamics are presented in appendix C and those for airplane weight in appendix D. Details of the propulsion system are presented in appendix E. Engine performance as a function of power setting, Mach number, and altitude was calculated by using the component matching procedures of reference 9. A retractable noise suppressor of the type described in reference 10 is built into the exhaust nozzle. Engine weight was calculated according to reference 11.

The DOC was computed by the standard method described in reference 12 and maximum sideline noise after takeoff by the standardized methods of references 13 and 14. Airframe cost was \$265 per kilogram (\$120/lb), while engine cost was \$54.23 per kilogram (\$24.60/lb) of takeoff thrust and fuel cost was \$0.22 per kilogram (\$0.10/lb).

RESULTS AND DISCUSSION

Parametric Study of Configurations Having Low Sonic-Boom Overpressures

The results of a parametric study of sonic-boom overpressure as a function of aircraft configuration are shown in figure 2. The configuration employed has a minimum-drag fuselage and an arrow wing with a root chord equal to 75 percent of a similarly swept delta wing. Flight Mach number is 2 and wing sweep is 62° to give a slightly subsonic leading edge. In each case, the lift coefficient (and altitude) are selected to result in maximum lift-drag ratio. For the solid line of figure 2(a), the body length is held constant at 71.6 meters (235 ft) and wing loading is 2870 N/m^2 (60 lb/ft^2) while airplane gross weight is varied. Sonic-boom overpressure varies strongly with gross weight. To achieve a sonic boom of about 33.5 N/m^2 (0.7 lb/ft^2), or less, a gross weight of the order of 56 700 kilograms (125 000 lb) or less is required. At the lowest gross weight, altitude is 21 300 meters (70 000 ft); while at the highest gross weight, 19 800 meters (65 000 ft). For the dashed line in figure 2(a), fuselage length varied with gross weight. Such a trend would result from increasing gross weight and fuselage length to carry more passengers over a fixed range. At the lowest gross weight, a 34-percent decrease in length increased sonic boom 16.4 percent. At the highest gross weight, a 43-percent increase in fuselage length decreased sonic boom 12.4 percent. As shown in figure 2(b), wing loading has only a slight effect on sonic boom; hence, wing loading can be selected to maximize airplane performance. Since a low wing loading gives a high lift-drag ratio and a high wing loading gives a lower structural weight, an estimate of structural weight and performance with specific engines is required to select the proper wing loading.

In figure 2(c), body length and diameter were varied while gross weight was fixed at 56 700 kilograms (125 000 lb) and wing loading at 2873 N/m^2 (60 lb/ft^2). Sonic-boom overpressure decreased as l/d increased as a result of an increase in length but increased slightly with a decrease in fuselage diameter.

The effect of Mach number is shown in figure 2(d). Sonic-boom overpressure decreased slightly as cruise Mach number increased from 2 to 3.2.

The effect of cruise lift coefficient is shown in figure 3. The top curve illustrates the trend of sonic boom as C_L and, hence, altitude increase. An increasing altitude tends to reduce sonic boom; but in this case the increase in C_L, which tends to increase sonic boom, neutralizes the altitude effect and sonic boom is fairly constant. At a C_L of 0.09 (well below the C_L optimum for maximum L/D), sonic-boom overpressure is 33.5 N/m^2 (0.70 lb/ft^2). The corresponding altitude is 15 240 meters (50 000 ft). The corresponding L/D is only 6. At the maximum L/D of 7.8, the sonic boom has risen to 35.4 N/m^2 (0.74 lb/ft^2). At higher values of C_L, sonic boom is constant and L/D decreases. Therefore, the desirable cruise C_L appears to be a value close to that for maximum L/D.

Selected Mach 2 SST

Based on the previous parametric study, a configuration was selected for more detailed analysis. A planform sketch of the selected configuration is shown in figure 4(a). It has a cruise Mach number of 2.0, a gross weight of 56 700 kilograms (125 000 lb), and a takeoff wing loading of 2873 N/m^2 (60 lb/ft^2)). The fuselage has a length of 46.0 meters (150.8 ft) and a maximum diameter of 3.9 meters (12.8 ft), and it can accommodate 129 passengers.

The area variation (including the equivalent area due to lift) of the low-boom airplane is shown in figure 4(b). The solid line is the area variation of the fuselage alone. The dashed line is the area variation of the lifting wing - body combination before the fuselage is altered. The dash-dot line is the area variation of the final configuration for which the sonic-boom overpressure was calculated. This smooth area variation, which approximates that for a minimum-drag body, was achieved by altering the rear fuselage as shown in figure 4(c). A body maximum diameter of 3.9 meters (12.8 ft) was required. A smaller diameter would be too small to permit enough alteration to achieve a smooth area variation for the complete configuration. Other characteristics of the low-boom Mach 2 SST are as follows:

Takeoff gross weight, kg (lb)	56 700 (125 000)
Takeoff wing loading, N/m^2 (lb/ft^2)	2873 (60)
Takeoff thrust loading, N/kg (lbf/lbm)	3.14 (0.32)
Fuselage length, m (ft)	46.0 (150.8)
Fuselage maximum diameter, m (ft)	3.9 (12.8)
Wing area, m^2 (ft^2)	194 (2083)
Wing sweep, rad (deg)	1.12 (64)
Wing span, m (ft)	22.4 (73.5)
Wing root chord, m (ft)	17.2 (56.5)
Compressor pressure ratio at takeoff	9.3
Turbine inlet temperature, ^{o}C (^{o}F):	
Takeoff, climb, and acceleration	1090 (2000)
Cruise	1040 (1900)
Maximum afterburner temperature, ^{o}C (^{o}F)	1704 (3100)
Airflow per engine, kg/sec (lb/sec)	58 (127.5)
Operating weight empty, percent of gross weight	42.7
Fuel, percent of gross weight	40.6
Payload, percent of gross weight	16.7
Cruise altitude, m (ft)	18 300 (60 000)
Cruise lift-drag ratio	7.35
Cruise sonic boom, N/m^2 (lb/ft^2)	41.7 (0.87)
Sonic boom during climb and acceleration, N/m^2 (lb/ft^2)	<41.7 (<0.87)
Sideline noise during takeoff, PNdB:	
Unsuppressed	117.3
Suppressed	103.3
FAR 36	103.6

Noise, payload, and DOC. - To this configuration were added four afterburning turbojets having a design compressor pressure ratio of 9.3 and a maximum temperature at the turbine inlet of 1090^{o} C (2000^{o} F) to satisfy the noise constraint. Takeoff is at maximum dry thrust, giving a takeoff thrust loading of 0.32 and a maximum sideline noise (at 648 m (0.35 n mi)) of 117.3 PNdB at 199 knots. The suppressed noise level of 103.3 PNdB is slightly below the level required by Federal Air Regulation 36.

The effect of initial cruise altitude on the sonic-boom overpressure and payload fraction of the selected configuration is shown in figure 5 for a range of 4440 kilometers (2400 n mi). The sonic-boom overpressure is insensitive to initial cruise altitude, but the payload maximizes for an initial cruise altitude of about 18 300 (60 000 ft) at a value of 16.7 percent of gross weight. At this altitude, the sonic-boom overpressure is 41.7 N/m^2 (0.87 lb/ft^2). The calculated value of steady-flight sonic-boom overpressure during climb and acceleration to cruise conditions is less than the cruise value, but booms of two to four times the steady-flight values may result from airplane acceleration. The maximum payload fraction for the 4440-kilometer (2400-n mi) range corresponds to 105 passengers, although the fuselage can accommodate 129 passengers, as was noted previously.

The following estimates of direct operating cost and some major input assumptions were made for the selected airplane:

Cost of airframe	$5.6 million
Cost per engine	$246 000
Passengers	105
Direct operating cost, cents/seat st. mi.	0.82

This DOC compares with a DOC of 0.68 cent per seat statute mile for the 440-passenger Boeing 747 and 1.06 cents per seat statute mile for the 298-seat Boeing SST B2707-300 (ref. 15). The trip times for a 4440-kilometer (2400-n mi) flight would be 318 minutes for the 747, 162 minutes for the Mach 2 airplane, and 141 minutes for the B2707-300.

The major features of two intercontinental SST's (the Concorde and the Boeing B2707-300) and two domestic SST's (the SCAT 15F and the low-boom SST of the present study) are compared in the following table:

Characteristic	Transcontinental		Domestic	
	B2707-300	Concorde	SCAT 15F	Low boom
Range, km (n mi)	6540 (3530)	5930 (3203)	4440 (2400)	4440 (2400)
Cruise flight Mach number, M_{cr}	2.7	2.05	3.0	2.0
Cruise altitude, m (ft)	----------	18 300 (60 000)	24 200 (79 500)	18 300 (60 000)
Sonic-boom overpressure, Δp, N/m^2 (lb/ft^2)	95.8 (2.00)	124.5 (2.60)	45.5 (0.95)	41.7 (0.87)
Number of passengers	298	168	50	105
Payload, percent	8.3	8.6	6.1	16.7
Gross weight, kg (lb)	340 000 (750 000)	176 400 (389 000)	77 100 (170 000)	56 700 (125 000)
Length, l, m (ft)	90.8 (298)	62.1 (203.75)	54.3 (178)	46.0 (150.8)
Ratio of weight to wing planform area at takeoff, W/S, N/m^2 (lb/ft^2)	4668 (97.5)	4788 (100)	2035 (42.5)	2873 (60)
Cruise lift-drag ratio, $(L/D)_{cr}$	7.7	------------	8.8	7.35
Thrust-weight ratio at takeoff, $(F/W)_{TO}$	*0.35*	*0.396*	*0.32*	*0.32*
Cruise specific fuel consumption, hr^{-1}	1.48	1.18	----------	1.69
Direct operating cost, cents per seat statute mile	1.06	1.81	3.00	0.82

The low-boom design has the unique advantages of the lowest boom and the highest payload fraction. These advantages result from several design features.

The low sonic boom results from the following factors: First, both domestic SST's have a very much lower gross weight than the intercontinental SST, primarily because of their shorter range and smaller passenger load. Further, the low-boom domestic SST of the present study (hereinafter called the low-boom SST) has a lower gross weight than the SCAT 15F type primarily because of its high payload fraction, a factor to be discussed later. Gross weight has a strong effect on sonic boom, as was shown in figure 2(a). Second, the airplane cruise lift-drag ratio is compromised slightly to shape the wing-fuselage combination to reduce the boom.

The high payload fraction also results from a combination of factors: First, compared with the intercontinental SST, the shorter range of the low-boom SST favors higher payload fraction. Second, compared with the SCAT 15F domestic airplane, the higher wing loading favors higher payload fractions. Third, the low-boom SST uses 1975 engine weight, which is estimated to be 25 percent lower than 1968 engine weight.

As indicated earlier, the low-boom airplane achieves its low sonic-boom overpressure chiefly because of its low gross weight. The low gross weight results from selecting a moderate payload and range and achieving a low operating weight empty (OWE) fraction. Similar results could be achieved by designing the Concorde for a 4440-kilometer (2400-n-mi) range. The second column in the following table shows the weight breakdown using scaled Olympus engines and indicates a payload fraction of 12.6 percent. With 1975 engine technology, third column, the payload fraction becomes 17.9 percent or 1 percent better than that of the low-boom airplane.

	Scaled Olympus engines	1975 Engines
OWE, percent of gross weight	47.0	41.7
Fuel, percent of gross weight	40.4	40.4
Payload, percent of gross weight	12.6	17.9
	100.0	100.0

Market for a domestic Mach 2 supersonic transport. - The market for a domestic Mach 2 supersonic transport was estimated for the year 1977 when U.S. domestic traffic is forecast to be 200×10^9 revenue passenger miles (ref. 2). Of this total 22.8 percent is estimated to be for trips of 2414 kilometers (1500 mi) and longer. Based on an average trip distance of 3704 kilometers (2000 n mi), a 55-percent load factor, and 3000-hours-per-year utilization, we estimate the market for a domestic supersonic transport such as the one discussed herein to be as high as 307 airplanes. If the supersonic transport were to capture only one-half the market, a fleet of 154 airplanes would be required.

Potential for improvement. - The space-limited passenger capacity is 129, whereas the calculated payload for the 4440-kilometer (2400-n mi) range, as limited by fuel consumption, is 105. The following table shows the increase in passengers that would result from a 10-percent improvement in the major features of the airplane. It is assumed that a passenger and his baggage weigh 91 kilograms (200 lb) and that passenger services require an additional 75.3 kilograms (166 lb) per passenger.

10-Percent improvement in major feature -	Resulting passenger increments
Weight of wing and tail	3.2
Weight of fuselage	1.0
Weight of engine	1.7
Weight of fixed equipment	5.9
Cruise specific impulse	6.2
Cruise lift-drag ratio	6.2

If each of the features in the table could be improved 10 percent, the number of passengers would increase to 129 and the DOC would become 0.63 cent per seat mile.

Improvements in many of these items are probably attainable now or in the near future. In reference 16, for example, it is indicated that the steel wing skin of the BQM-34E supersonic aerial target can be replaced by laminated graphite-epoxy, which provides equivalent strength and flutter behavior with a weight saving of approximately 50 percent. When boron epoxy doublers were added to the wing pivot structure of the F-111, the fatigue life was doubled. A removable graphite epoxy inspection panel was also installed at less cost than an aluminum honeycomb panel. The lower labor cost more than compensated for the higher material cost (about $110.23/kg ($50/lb)), thus giving a lower net cost. Currently, the Air Force is funding some promising work in the use of composites in fuselage structure (ref. 17).

Engine weight was calculated based on technology expected to be available in 1975; all the engine manufacturers are doing encouraging work in the use of composites (e.g., ref. 18). In reference 19 it is stated that an L/D of 10 at Mach 2.6 may be practical in the future although, undoubtedly, some penalty in L/D would result from tailoring such configurations for low boom values.

SUMMARY OF RESULTS

A parametric study of low-boom supersonic-transport configurations identified the kinds of configurations that promise low sonic booms according to the theory given in NASA TN D-4935. Long fuselages and low gross weights favor low sonic-boom overpressure levels. Sonic-boom overpressure decreases slightly as cruise Mach number increases from 2 to 3.2. Special biplane shapes such as those described in NASA SP-255 were not studied.

A nominal Mach 2 configuration having a gross weight of 56 700 kilograms (125 000 lb), a wing loading of 2870 N/m^2 (60 lb/ft^2), and a fuselage length of 46.0 meters (150.8 ft) was selected; and its gross geometric, aerodynamic, and structural features were estimated. It was powered by four afterburning turbojet engines sized to give a takeoff thrust loading of 0.32 at maximum dry power setting. At a cruise altitude of 18 300 meters (60 000 ft), the lift-drag ratio was 7.35. Fuel fraction for a range of 4440 kilometers (2400 n mi) was 40.6 percent of gross weight; and the calculated operating weight empty was 42.7 percent. This led to a very respectable payload fraction of 16.7 percent of gross weight and a direct operating cost of 0.82 cent per seat statute mile. Cruise sonic boom was 41.7 N/m^2 (0.87 lb/ft^2), primarily because of the low gross weight, while the calculated value of steady-flight boom overpressure during climb and acceleration was slightly lower. With 14 decibels of suppression, maximum sideline noise after takeoff at 102 meters per second (199 kt) was 103 PNdB, which is slightly below the Federal Air Regulation 36 requirement.

The methods used in this preliminary study to calculate aerodynamics, weights, and sonic-boom overpressure qre quite simple but did give results quite close to published results on particular supersonic transport designs. Because of the simplified nature of the study, however, the results should be considered as speculative and subject to verification by more detailed studies.

Lewis Research Center,
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APPENDIX A

SYMBOLS

A	body frontal area
AR	aspect ratio
A_L	equivalent base area due to lift
A_x	cross-sectional area distribution
B	body width at 50-percent wing root chord
B_x	equivalent area due to lift
C_{D_i}	induced drag coefficient based on wing planform area
C_{D_0}	zero-lift drag coefficient based on wing planform area
C_{Dp}	pressure drag coefficient based on body frontal area
C_f	friction coefficient, turbulent boundary layer, based on wetted area
C_L	airplane lift coefficient based on wing planform area
C_p	pressure coefficient of a cone
D	airplane drag
d	fuselage maximum diameter
F'_L	lifting force per unit length
f	wing weight factor
H	body depth at 50-percent wing root chord
h	altitude
K_r	reflection factor (K_r = 1.9 in this study)
K_s	body shape factor, $C_{Dp}/(\beta C_L S/\bar{l}^2)$
k	chord ratio
L	airplane lift
L_s	body structural length (excluding nose and tail cones, radomes, turrets, pitot tubes, and other protuberances of similar nature)
l	body length
$\bar{l}$	length from nose of fuselage to wing-tip station

l' length of forebody from nose to maximum cross section

M flight Mach number

N ultimate load factor at design gross weight (3.75)

Δp sonic-boom overpressure, N/m^2 (lb/ft^2)

p_r reference pressure for a 1962 U.S. standard atmosphere (ref. 20)

Q_u wing-weight landing-gear factor (0.95 for fuselage-mounted main landing gear, 1.00 for wing-mounted main landing gear)

q dynamic pressure

Re Reynolds number

S wing planform area

S_B body wetted area

t/c thickness ratio of wing in percent

V velocity

W airplane weight

W_A airflow

W_{AB} weight of afterburner

W_E weight of bare engine

W_F weight of fuel

W_{IN} weight of inlet

W_N weight of exhaust nozzle

W_R weight of fuel in wing at design gross weight plus weight of engines in wing

W_{WC} weight of wing and contents

W_{FFE} weight of furnishings and fixed equipment

x axial distance

α angle of attack

β Mach number function, $(M^2 - 1)^{0.5}$

$\Lambda_{0.5c}$ sweep angle at 50-percent chord

λ taper ratio

Subscripts:

B	body
cr	cruise
DG	design gross
f	friction
ff	far field
FS	fuel system
G	gross
g	ground
H	hydraulics
LG	landing gear
p	pressure
rf	real field
SC	surface controls
T	tail
TO	design takeoff
W	wing
0	free stream or incompressible

APPENDIX B

SONIC-BOOM CALCULATION PROCEDURE AND COMPARISON OF THEORIES

Far-Field Sonic-Boom Equation from Reference 6

According to the far-field analysis presented in reference 6, if an optimum combination of area and lift for an airplane is assumed with no restrictions on the minimum volume and with area at the airplane base equal to zero, the lower bound of sonic-boom overpressure can be expressed as

$$\frac{\Delta p}{p_r}\left(\frac{h}{l}\right)^{0.75}\beta^{-0.25}K_r^{-1} = 0.54\left(\frac{\beta}{2}C_L\frac{S}{l^2}\right)^{0.5} \tag{B1}$$

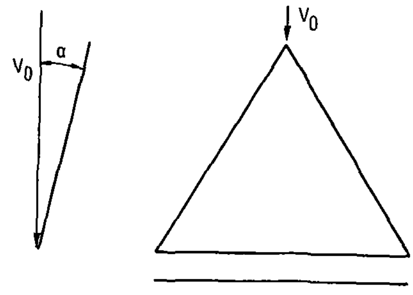
This equation was used to plot the line labeled reference 6 in figure 1. The value of K_r is estimated to be 1.9 and p_r was obtained from reference 20.

Sonic-Boom Equations from Reference 7

Reference 7 shows that the sonic boom of an axisymmetric body with an N-wave pressure trace is evaluated by

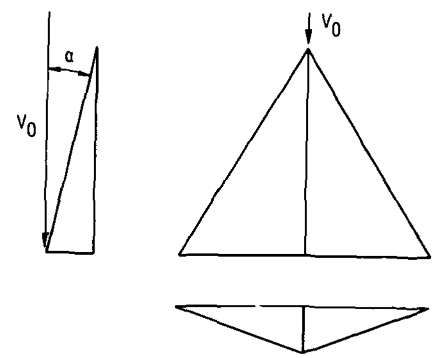
$$\frac{\Delta p_{rf}}{p_r} = \frac{0.631\,K_s^{0.5}M^2\,\frac{A}{l'^2}\,K_r}{\left(\frac{h}{l'}\right)^{0.5}\beta^{0.5}\left[1 + \frac{1.27\,K^{0.5}M^4\,\frac{A}{l'^2}\left(\frac{h}{l'}\right)^{0.5}}{\beta^{1.5}}\right]^{0.5}} \tag{B2}$$

Although this expression was developed for an axisymmetric body, it can be applied to the lifting case by relating an equivalent base area due to lift A_L to the lift coefficient and wing planform area. Consider a planar delta planform at an angle-of-attack α:



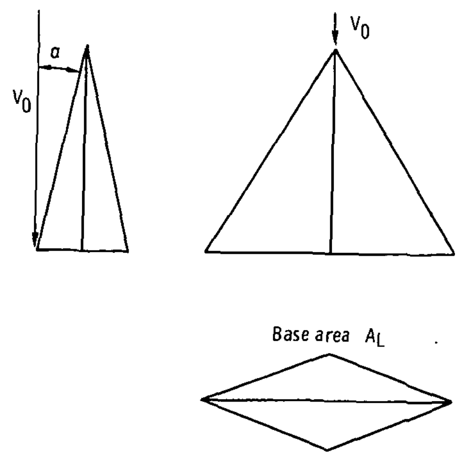
(a)

Equivalently, the lift is unchanged if each chord is translated upward but maintains the same angle of attack, so that the leading edge of the new wing lies in a horizontal plane, as shown in sketch b:



(b)

A second wing, a mirror image of the new wing in the horizontal plane containing the leading edge of that wing, is added to the new wing (sketch c).



(c)

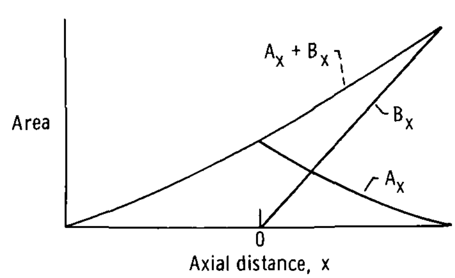
This results in a zero-lift body, with the diamond-shape base area A_L, that generates the same pressure field beneath it as the corresponding lifting wing (sketch a) and hence generates the same sonic boom if the leading edge is swept at or ahead of the Mach line. According to equation (B2) the sonic boom is a function of A_L. This base area due to lift A_L is related to the wing lift as follows:

$$A_L = 2S\alpha = 2S\frac{C_L}{\frac{dC_L}{d\alpha}} = \frac{2SC_L\beta}{4} = \frac{\beta}{2}C_LS \qquad \text{(B3)}$$

where

$$\frac{dC_L}{d\alpha} = \frac{4}{\beta} \qquad \text{for } 0 \leq \Lambda \leq \left(90^\circ - \arcsin\frac{1}{M}\right)$$

For a wing-body combination, the sonic-boom strength depends on the manner in which the actual cross-sectional area combines with a fictitious or equivalent area due to lift (see p. 3 of ref. 6). For a minimum-drag body, the actual cross-sectional area distribution A_x has the form shown in sketch d:



(d)

If a thin delta wing is added starting at the axial station where fuselage area is a maximum (x = 0), its equivalent cross-sectional area due to lift at station x is

$$B_x = \frac{\beta}{2q}\int_0^x F'_L\, dx$$

where F'_L is the lifting force per unit length. The cross-sectional area distribution A_x reduces to zero or nearly zero at the base of the airplane, depending on the jet exit and wake condition assumed. The equivalent area due to lift B_x, however, reaches a maximum at the base. This maximum value of B is equal to $(\beta/2)C_LS$, or in other terms, $(\beta/2)(W/q)$. Thus, in level flight the maximum value of B depends on the weight of the airplane and the flight conditions.

For a wing-body combination, the real-field sonic boom is expressed as

$$\Delta p_{rf} = \frac{0.631\, K_s^{0.5} M^2 \left(\frac{\beta}{2} C_L \frac{S}{\bar{l}^2}\right) K_r p_r}{\left(\frac{h}{\bar{l}}\right)^{0.5} \beta^{0.5} \left[1 + \frac{1.27\, K_s^{0.5} M^4 \left(\frac{\beta}{2} C_L \frac{S}{\bar{l}^2}\right)\left(\frac{h}{\bar{l}}\right)^{0.5}}{\beta^{1.5}}\right]^{0.5}} \tag{B4}$$

where $(\beta/2)C_L S$ has replaced body frontal area A and $\bar{l}$ has replaced l'. If the leading edge of the wing begins at the axial station where the minimum-drag fuselage area is a maximum and the $A_x + B_x$ variation approximates the A_x variation over the front half of the fuselage, then the value of K_s is that for a minimum-drag body. Generally, this requires some waisting of the rear fuselage, as is evident in figure 4. Equation (B4) is valid for a range of C_L for which the maximum value of B_x is greater than the maximum value of A_x; but equation (B4) is replaced by equation (B2) when the maximum value of A_x exceeds the maximum value of B_x, as it does for the limiting case of $C_L = 0$. Equation (B4) was used to calculate the real-field sonic-boom levels presented in figure 2.

For the far-field

$$1 \ll \frac{1.27\, K_s^{0.5} M^4}{\beta^{1.5}} \left(\frac{\beta}{2} C_L \frac{S}{\bar{l}^2}\right)\left(\frac{h}{\bar{l}}\right)^{0.5}$$

so far-field sonic-boom overpressure is expressed as

$$\Delta p_{ff} = \frac{0.558\, K_s^{0.25} \left(\frac{\beta}{2} C_L \frac{S}{\bar{l}^2}\right)^{0.5} K_r p_r \beta^{0.25}}{\left(\frac{h}{\bar{l}}\right)^{0.75}} \tag{B5}$$

Equation (B5) was used to calculate the far-field sonic-boom levels from reference 7 that are presented in figure 1. Notice that K_s is a function of M.

For purposes of comparison, the various calculation techniques in references 6 and 7 were applied to particular supersonic airplanes. According to Swihart (ref. 15), the heavy Boeing Mach 2.7 SST would have had a cruise sonic-boom overpressure of 95.8 N/m^2 (2 lb/ft^2). The sonic boom of the Boeing SST is calculated to be 88.1 N/m^2 (1.84 lb/ft^2) by using reference 7. For the YF-12, the real-field sonic-boom value of 40.2 N/m^2 (0.84 lb/ft^2) lies within the bound of measured values of 33.5 to 52.7 N/m^2 (0.7 to 1.1 lb/ft^2). For the SCAT 15F of reference 4, the far-field sonic boom of 57.0 N/m^2 (1.19 lb/ft^2) calculated by Luidens (ref. 7) compares with the far-field sonic-boom overpressure of 67.0 N/m^2 (1.40 lb/ft^2) reported in reference 4 and based on the theoretical sonic-boom characteristics of the SCAT 15F configuration obtained from the Langley Research Center. The values of sonic-boom overpressure calculated by the method of reference 7 assume the fuselage is sized and shaped to result in a smooth variation of the area plus equivalent cross-sectional area due to lift. Since this is not the case for these three airplanes, their sonic-boom overpressures would be expected to be greater than the values calculated by the method of reference 7.

Since the sonic-boom values obtained from reference 7 agree reasonably well with the experimental data and with those obtained by the more cumbersome procedures, the procedures of reference 7 were used in conducting a parametric study of configurations having low values of sonic boom.

APPENDIX C

AIRFRAME AERODYNAMICS

The lift-drag ratio was calculated from

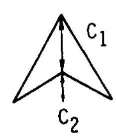
$$\frac{L}{D} = \frac{C_L}{C_{D_0} + \frac{C_{D_i}}{C_L^2} C_L^2} \tag{C1}$$

where

$$\frac{C_{D_i}}{C_L^2} = \frac{K}{2\pi}\left(\frac{\tan \Lambda}{2} + \frac{\beta^2}{\tan \Lambda}\right) \tag{C2}$$

for an arrow wing, and

$$k = \frac{c_1}{c_1 + c_2}$$



Equation (C2) is valid for an elliptic spanwise and chordwise load distribution on the wing and a subsonic leading edge with the leading edge slightly behind the Mach angle.

The zero-lift drag coefficient of the airframe is the sum of the body, wing, and tail pressure and the friction drag coefficients. Interference drag and trim were assumed to be zero. The pressure or wave drag coefficient on the body was calculated from

$$C_{D_0,p,B} = 2(0.67)C_p \frac{A}{S} \tag{C3}$$

where the cone pressure coefficient was obtained from reference 21. It is assumed that the drag of a minimum-drag body is 2/3 that of a cone having the same length and diameter. The body friction drag coefficient was calculated from

$$C_{D_0,f,B} = 2.4\, C_{f_0} \frac{C_f}{C_{f_0}} \frac{l}{d} \frac{A}{S} \tag{C4}$$

where $2.4\ l/d$ is the ratio of surface area to frontal area for a slender body whose radius varies as axial distance to the 2/3 power, the type of body used herein for the fuselage. The ratio of compressible friction drag coefficient to incompressible friction drag coefficient is

$$\frac{C_f}{C_{f_0}} = \left(1 + 0.123\ M^2\right)^{-0.65} \tag{C5}$$

which agrees with the experimental data of reference 22. The incompressible friction drag coefficient is given by

$$C_{f_0} = 0.455(\log_{10} Re)^{-2.58} \tag{C6}$$

the Prandtl-Schlichting formula (p. 34 of ref. 23) for incompressible turbulent boundary layer.

Wing friction drag coefficient was calculated from

$$C_{D_0,f,W} = 2C_{f_0} \frac{C_f}{C_{f_0}} \tag{C7}$$

The numeral 2 accounts for the surface area being twice the planform area. The wing pressure drag coefficient was read from the figure on page 190 of reference 22.

The tail friction and pressure drag coefficients were calculated from those of the wing. It was assumed that the characteristic length of the tail was 0.3 times that of the wing and that the vertical tail surface and planform areas were 0.08 times those of the wing.

The following table shows how the $(L/D)_{max}$ of two SCAT 15F configurations calculated by this simple method compare with the Langley and Boeing values:

Configuration	Maximum lift-drag ratio, $(L/D)_{max}$	
	Accurate method	Simple method
SCAT 15F, ref. 24	[a]9.2	9.6
SCAT 15F-B7, ref. 25	9.1	8.8

[a]Actual L/D; $(L/D)_{max}$ would be 0.1 to 0.3 higher.

The simple method gives fairly accurate results and is judged to be adequate for this preliminary study.

APPENDIX D

WEIGHT CALCULATIONS

Initially, empirical weight estimation methods are formulated by airplane manufacturers to provide consistency with the configuration selection matrix approach. After specific aircraft are selected, detailed weight estimates are made based on design drawings, load and stress analyses, and subsystem requirements. These detailed estimates are used to revise and update the earlier empirical weight estimation methods. The empirical equations used herein are valid for an aluminum structure and were presented at the University of Tennessee Space Institute short course, Modern Theory and Practice of Weight Optimization and Control for Advanced Aeronautical Systems, November 1968 (ref. 24).

$$W_W = 600f + 500$$

where

$$f = \frac{10^{-5} W_{DG}^{0.6} N^{0.6} S^{0.8} AR^{0.8} (1 + \lambda)^{0.4} \, 1.07 \sec \Lambda_{0.5c} Q_u \left(1 - \frac{W_R}{W_{DG}}\right)^{0.4}}{\left(\frac{t}{c}\right)^{0.4}}$$

$$W_B = 0.115(NW_{DG})^{0.25} S_B \left[\frac{W_{DG} - W_{WC}}{L_s(B + H)^2}\right]^{0.33}$$

$$W_T = 0.08 \, W_W$$

$$W_{FS} = 0.0209 \, W_F$$

$$W_{LG} = 0.0436 \, W_{DG}$$

$$W_H = 0.01485 \, W_{DG}$$

$$W_{SC} = 27.2 \, S^{0.5} + 5.74 \, l$$

$$W_{FFE} = 0.0348 \, W_{TO} + (166 \times \text{Number of passengers})$$

These equations were used to estimate the operating weight empty of the aluminum Concorde. The calculated OWE agrees quite well with the actual value:

Configuration	OWE, percent of W_{TO}
Concorde, ref. 27	43.00
Concorde, simple empirical equations	42.86

Hence, the empirical equations seem quite adequate for a preliminary study.

APPENDIX E

ENGINE CHARACTERISTICS

The axisymmetric inlet has a translating centerbody with a 15° half-angle cone. The inlet pressure recovery schedule is given in figure 6. The afterburning turbojet engine characteristics are

Compressor pressure ratio	9.3
Compressor efficiency	0.86
Turbine inlet temperature, $^{\circ}C$ ($^{\circ}F$):	
Maximum for takeoff (for low noise)	1090 (2000)
Maximum for cruise	1040 (1900)
Turbine efficiency	0.89
Maximum afterburner temperature, $^{\circ}C$ ($^{\circ}F$)	1704 (3100)
Airflow, kg/sec (lb/sec)	57.8 (127.5)

The exhaust nozzle is a variable-geometry auxiliary-inlet nozzle with a gross thrust coefficient of 0.985 at cruise. Each of the four nacelles for the selected Mach 2 supersonic transport is 7.01 meters (23 ft) long and has an inlet diameter of 0.66 meter (2.17 ft) and an exit diameter of 0.95 meter (3.12 ft). The calculated engine drag coefficient variation with flight Mach number is presented in figure 7. The drags accounted for are friction, excess flow, bleed, and wave. A retractable noise suppressor of the type described in reference 10 is built into the exhaust nozzle. It provides 14-PNdB suppression during takeoff with about a 11-percent loss in thrust. When the suppressor is retracted, there is no thrust loss. Engine weight was calculated according to reference 11 for a first flight in 1975. Installed engine weight is the sum of base engine weight, afterburner weight, exhaust nozzle weight, and inlet weight, where

$$W_{AB} = 1.11\ W_A$$

$$W_N = 890\left(\frac{W_A}{350}\right)^{1.02}$$

$$W_{IN} = 0.36\ W_E$$

Total mission range is 4440 kilometers (2400 n mi) with 235 kilometers (127 n mi) being letdown range. Reserve fuel includes 7 percent of mission fuel, fuel for a 483-

kilometer (261-n mi) flight to an alternate airport, and fuel for 30 minutes of hold at Mach 0.6 and 4572 meters (15 000 ft). Mission fuel includes fuel for 25 minutes at idle thrust and 1 minute at maximum dry thrust, fuel to climb and accelerate to initial cruise conditions, cruise fuel, and letdown fuel at idle thrust for 13.3 minutes. The engines operate at maximum dry thrust from takeoff to Mach 0.9 and 6100 meters (20 000 ft). Maximum afterburning thrust is used from here to the initial cruise conditions. The engines are throttled during cruise and are at idle thrust during letdown. The climb path is shown in figure 8.

Engine size was selected to give a takeoff thrust loading at maximum dry power of 0.32. This was found in prior SST studies to provide satisfactory takeoff and climb performance.

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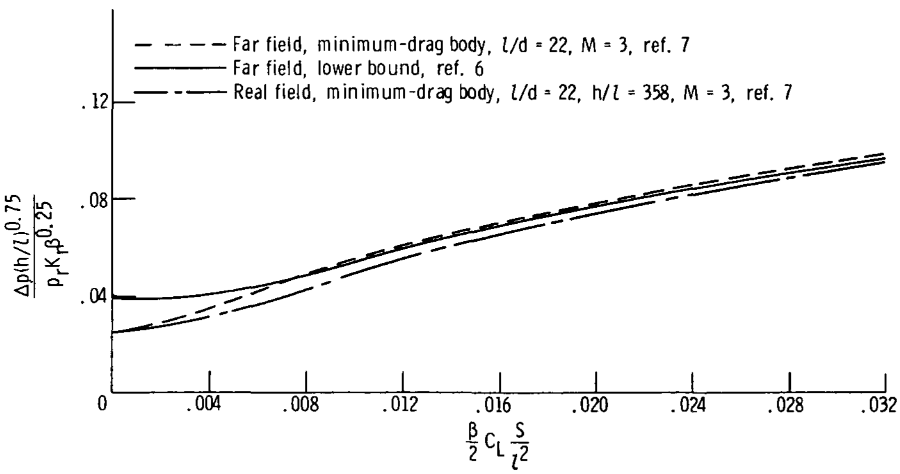
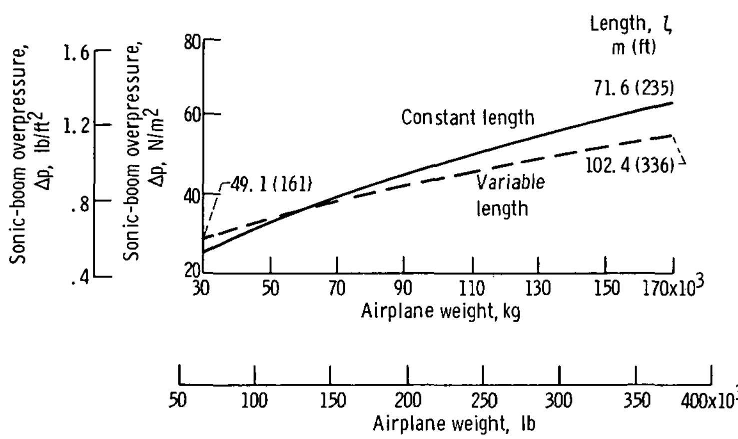
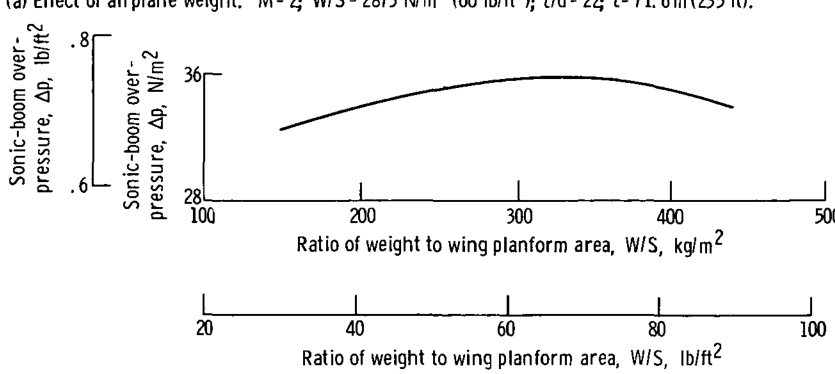


Figure 1. - Sonic-boom characteristics of minimum-drag body, and lower bound.

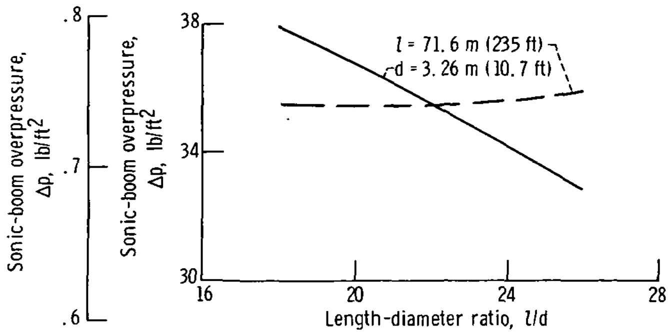


(a) Effect of airplane weight. $M = 2$; $W/S = 2873\ N/m^2$ ($60\ lb/ft^2$); $l/d = 22$; $l = 71.6$ m (235 ft).

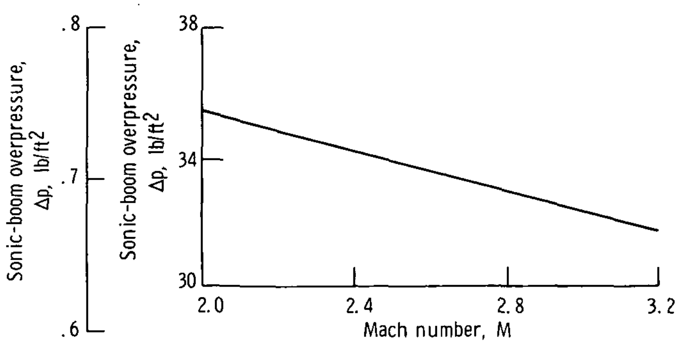


(b) Effect of wing loading. $M = 2$; $W = 56\ 700$ kg (125 000 lb); $l/d = 22$; $l = 71.8$ m (235 ft).

Figure 2. - Effect of airplane design parameters on real-field sonic boom. Mach number, M; wing loading, W/S; length-diameter ratio, l/d; length, l.



(c) Effect of fineness ratio. $M = 2$; $W = 56\ 700$ kg (125 000 lb); $W/S = 2873\ N/m^2$ (60 lb/ft²).



(d) Effect of cruise Mach number. $W = 56\ 700$ kg (125 000 lb); $W/S = 2873\ N/m^2$ (60 lb/ft²); $l/d = 22$; $l = 71.6$ m (235 ft).

Figure 2. - Concluded.

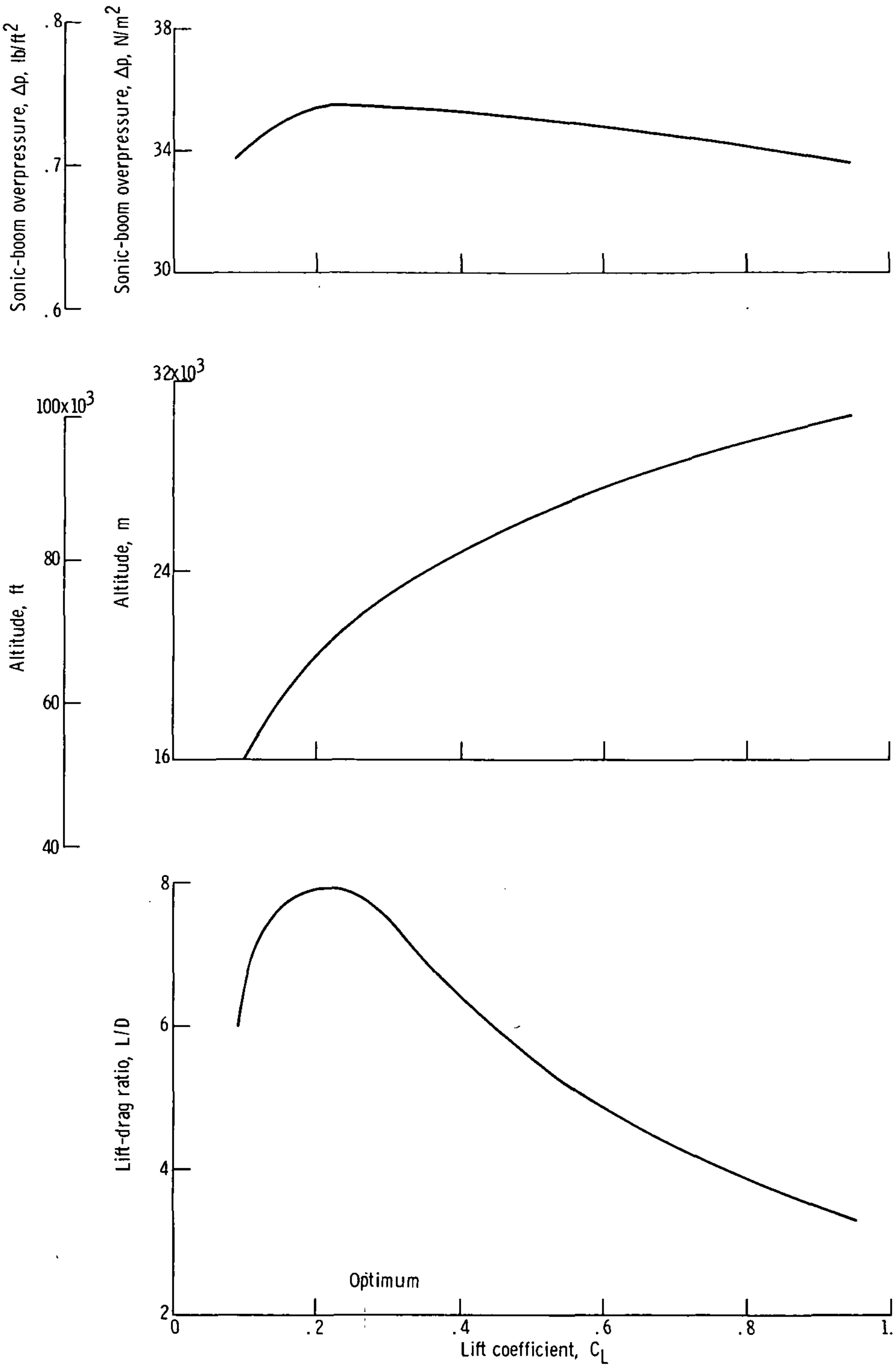
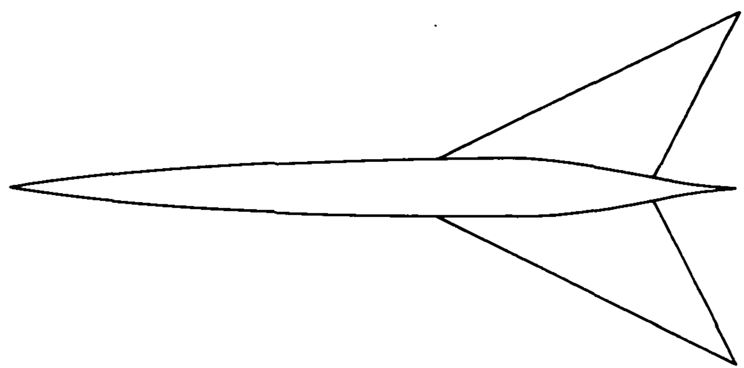
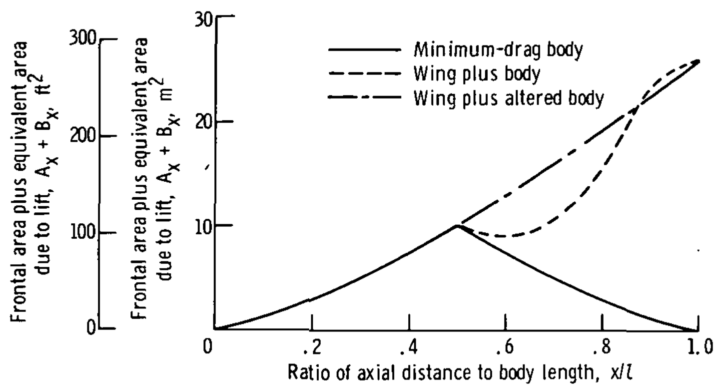


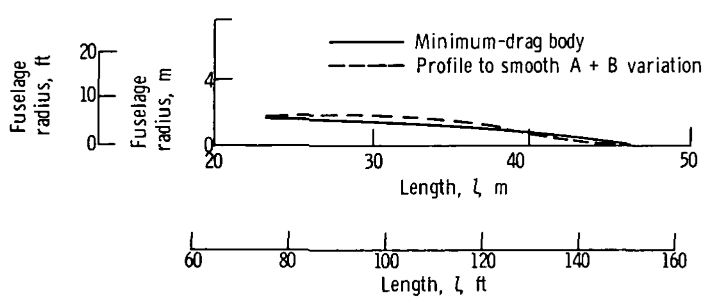
Figure 3. - Effect of operation. Minimum-drag body; arrow wing; weight, 56 700 kg (125 000 lb); Mach 2; body length, 71.6 m (235 ft); body maximum diameter, 3.26 m (10.7 ft); wing area, 193 m^2 (2083 ft^2); wing sweep, 62°; wing notch, 25 percent.



(a) Planform.



(b) Area variation. Lift coefficient, 0.155; free-stream Mach number, 2; altitude, 19 800 m (65 000 ft).



(c) Rear fuselage alteration.

Figure 4. - Nominal low-boom Mach 2 transport. Weight, 56 700 kg (125 000 lb); length, 46.0 m (150.8 ft); ratio of weight to wing planform area at takeoff, 2873 N/m^2 (60 lb/ft^2).

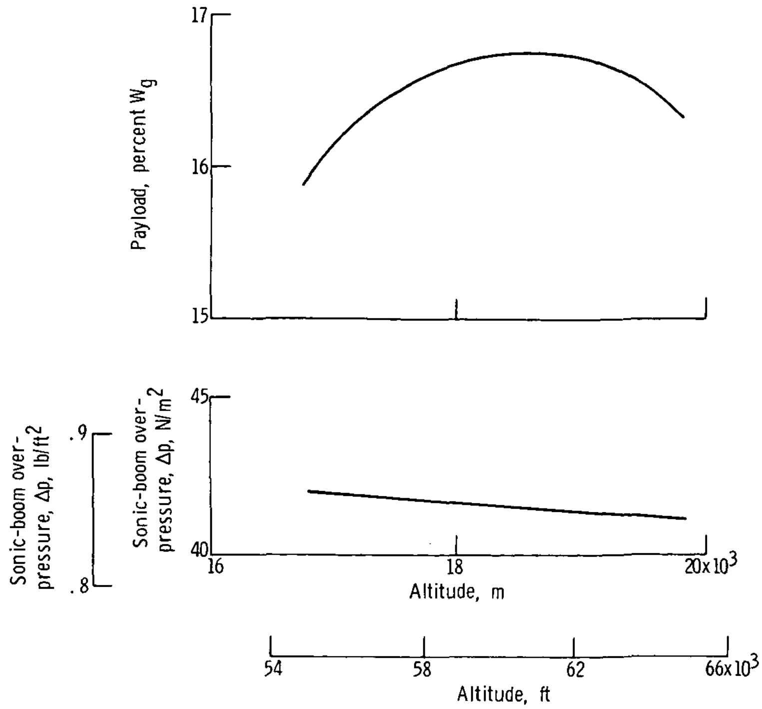


Figure 5. - Effect of cruise altitude on payload and sonic-boom overpressure. Weight, 56 700 kg (125 000 lb); Mach 2; body length, 46.0 m (150.8 ft); body maximum diameter, 3.9 m (12.8 ft); wing area, 194 m^2 (2083 ft^2); wing sweep, 64^0; wing notch, 25 percent; afterburning turbojet engines.

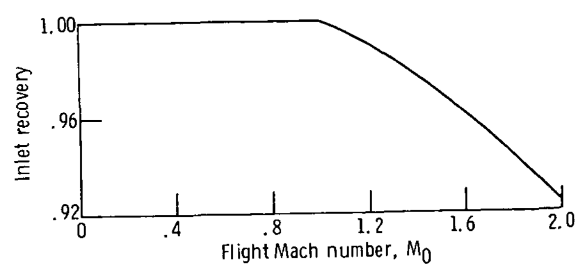


Figure 6. - Inlet pressure recovery.

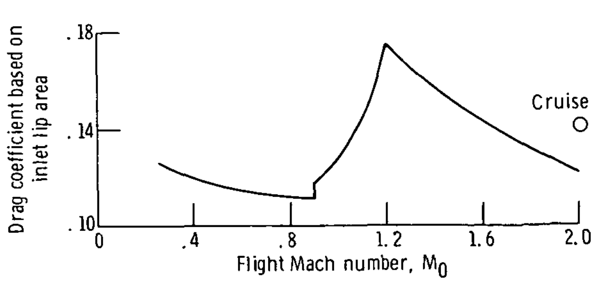


Figure 7. - Inlet and nacelle drag.

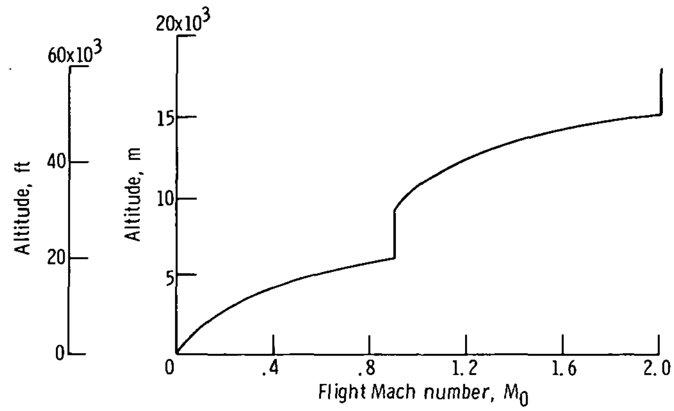


Figure 8. - Climb and acceleration path.

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