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ICAS Paper No. 70–23

PRACTICAL ASPECTS OF SONIC BOOM PROBLEMS

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The Seventh Congress of the International Council of the Aeronautical Sciences

CONSIGLIO NAZIONALE DELLE RICERCHE, ROMA, ITALY / SEPTEMBER 14-18, 1970

Price: 400 Lire

PRACTICAL ASPECTS OF SONIC BOOM PROBLEMS†

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Abstract

SST configurations selected from the point of minimizing sonic booms are investigated. It is indicated that for a total length of 300 ft and total initial weight of the same order as the present U.S. SST designs, sonic booms having shock pressure rise of the order of 0.6 lb/ft^2 can be obtained. Values as low as 0.3 are possible for airplanes designed for cross-country flights.

I. Introduction

The introduction of supersonic airplanes for commercial aviation has encountered severe criticism, and at the present time is opposed by many political and technical groups on the basis of several objections of a different nature. Some amount of the furor generated against such steps is not completely justified because it is generated by superficial and incomplete information. However, the basic objection related to the disturbances created by the sonic boom is based on solid ground, and presents one of the largest obstacles in the practical development of future supersonic aviation.

Many of the present difficulties are the result of the fact that the problem of the sonic boom has been downgraded at the beginning of the planning of the development of the first generation of supersonic transports. Therefore, programs for the first generation of airplanes have been initiated without a complete understanding of the effects of the sonic boom on the population and their reaction against it. Only when strong objections have been raised against the use of such airplanes, the responsible technical community has considered such problems of primary importance, and because of lack of an acceptable solution has been forced to limit the use of such means of transportation to overseas flights, at least for the first generation of airplanes.

The formation of sonic boom by supersonic airplanes is a physical phenomenon that cannot be eliminated when the airplane has lift and can only be reduced or modified; therefore, any technical effort in this field should be directed to answering the following basic questions:

a) What is the level of minimum practical values that can be obtained if the airplane design is optimized for minimum boom?

b) What is the minimum value acceptable if the airplane flies over a populated area?

If the answers to the first two questions give values that are of the same order of magnitude, then the third question should be answered:

c) What are the penalties of performance, and what is a possible compromise if the airplane is designed for acceptable sonic boom?

This paper reviews an effort devoted to the first task; however, some remarks related to the second and third questions are in order.

Present regulations in the U.S.A. forbid the production of sonic booms produced by airplanes of any strength over the United States. Such a limitation is related only to the airplanes; however, booms and noise of some level and similar shapes are produced by many other human activities and are acceptable everywhere. Therefore, it must be assumed that such a regulation is of a preliminary nature. The strongly restrictive limitation is presently justified because of the lack of required information raised in question b). It can be expected that better knowledge will permit changing such a restriction. It is interesting to recall indeed that similar objections (which were just as loud, but not as well organized), were leveled against the introduction of the first generation of automobiles. They were also classified as too noisy, unnecessary, uneconomical, and dangerous. Today, the population has become used to car noise in view of the practical advantages of the use of the automobile.

While all the information is not available, it is very probable that all of the objections related to possible damage to structures or buildings will be eliminated by better knowledge of the problem coupled with the reduction of the sonic boom level required to make the boom value acceptable to the people. The objection based on disturbances to

†This work was supported by the National Aeronautics Administration under NASA Grant NGL 33-016-119. The Project Monitor is Mr. Ira R. Schwartz. The work presented here is part of an investigation on sonic boom problems performed by a team at New York University under the direction of the author and Professor Lu Ting. Participating in some phases of this work are: Dr. Lu Ting, Dr. Huai-Chu Wang, Mrs. F. Kung, Messrs. A. Agnone and M. Siclari.

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humans is therefore the most serious objection to deal with at this stage of the problem. Such an objection is related to the value of the peak overpressures. Therefore, in the present discussion we will cover only this point.

It can be expected that the objection against supersonic flight will be overcome by a combination of several steps:

a) limitation of a number of sonic booms over a given populated area on the basis of selection of appropriate routes,

b) by selecting airplane configurations that minimize the sonic boom effects on the ground,

c) by adopting provisions that minimize the disturbances due to local conditions.

The sonic boom on the ground depends on the details of the airplane design and flight conditions as well as on the focusing effects due to atmospheric conditions and to terrain configurations. Therefore, all aspects of the minimization should be considered. The presence of some atmospheric conditions can increase somewhat the intensity of the peak overpressures on the ground (Refs. 1,2, and 3). The control of atmospheric conditions is impossible; however, some knowledge is already available that permits the reduction of such effects by carefully selecting time and flight paths. It is probable that for very low values of sonic boom, the sporadic increase of sonic boom overpressure due to special atmospheric conditions would not make it impossible to fly supersonic planes overland.

The problem of focusing due to terrain configurations is more serious because these conditions will exist all the time for every given flight path. In addition, such effects can produce large increases of sonic boom effects. Fortunately, such effects once detected can be reduced or eliminated by correct selection of flight paths, and by introducing special precautions in the vicinity of the focusing.

II. Local Reduction of Sonic Boom by Maneuvering of the Airplane.

The sonic boom of an airplane according to the Whitham theory[4] depends on the distribution of the lift and volume along the length. Following the approach suggested by Carlson[5], the lift and volume can be combined in a single equation, then the F function that defines the sonic boom can be expressed in terms of an integral that combines lift and volume, which is determined by transforming the lift into an equivalent cross-sectional area. The cross-sectional area due to volume, reduces to zero at the end of the airplane; therefore a cross-sectional area of the airplane at zero lift is equal to the difference in area due to the streamtubes of the engines, plus the cross section of the wake. The equivalent area due to the lift reaches the value equal to $\beta/2\ C_L S$* at the end of the wing and then remains roughly constant downstream. Then for a given distribution of equivalent cross-sectional area, which produces a given sonic boom signature, many airplane configurations can be obtained because a change in the division of equivalent cross-sectional area between volume and lift that does not change the total area distribution, does not change the sonic boom. However, when a given design is selected, then the sonic boom on the ground depends on the lift of the airplane. The signal generated by the airplane modified by the atmosphere is reflected by the ground and amplified depending on the shape of the ground. If the ground is absolutely flat and rigid, then the coefficient of amplification at high Mach numbers is 2; however, in practical conditions some decay takes place because of the nonuniformity and thickness of the ground. Therefore, usually a coefficient less than 2 is assumed for the reflection for a rough flat ground. (In the present paper a coefficient of 1.8 is used.) However, if the ground has some type of special shapes, larger amplifications are also possible (e.g., Refs. 4-5). It can be expected, for example, that larger amplifications can be obtained locally at some points at the end of a valley if the airplane flies parallel to the valley, or at the beginning of a chain of mountains.

In addition, it could be useful to reduce substantially the sonic boom when the airplanes fly in the proximity of cities. This possibility exists if the airplane reduces the lift of the vehicle by means of a maneuver. An airplane flying at high velocity can perform a pull-up maneuver of a few degrees before reaching the point where the sonic boom peak value should be reduced, and then fly a lower lift trajectory over the selected point. The airplane then can fly for several miles producing a signal which is substantially reduced.

Figure 1 gives the range obtainable for different values of C_L at a constant speed, and constant C_L trajectory as a function of the Mach number. The maneuver starts at an altitude of 40,000 ft where the airplane makes a pull-up of α degrees and then flies a trajectory at constant C_L equal to N times the C_L for cruise, with constant velocity. The airplane first increases altitude then descends and reaches 40,000 ft again at the same angle α. The maximum altitude reached is given in Figure 2. Two values of initial angle of the trajectory have been considered: one corresponding to $\alpha = 5°$, and the second to $\alpha = 10°$. The maximum value of discontinuous Δp produced by the sonic boom decreases strongly when an altitude of flight decreases and C_L decreases; therefore, the maneuver can alleviate substantially the disturbances produced by sonic boom. For example, initial Δp for a C_L corresponding to 2/3 of the horizontal value at an altitude of 40,000 ft can be as low as .4 lb/ft^2 as shown in Figure 3. The curve corresponds to an airplane having weight equal to 465,000 lbs and length on the order of 300 ft. The take-off weight of the airplane is between 650,000 and 700,000 lbs.

III. Design Criteria for Minimum Boom During Cruise.

The optimum design criteria for supersonic airplanes have been discussed in detail by several authors for the case of far field signatures. Jones[9] and Carlson[5] give expressions for minimum

*$\beta = \sqrt{M_\infty^2 - 1}$, C_L lift coefficient, S wing area, M_∞ flight Mach number.

overpressures for the conditions of far field signature. McLean[10] has shown that for the acceleration phase near field signature is possible. The author[11,12] has shown that near field signature with sonic booms having values of $(\Delta p)_{max}$ on the order of 0.8 lb/ft^2 can be obtained even for very large airplanes on the order of 500,000 lbs during cruise at 60,000 ft and M = 2.7, provided that large amount of equivalent cross-sectional area is placed near the front. Later Seabass[13] and George[14] obtained analytical expressions confirming such results. While for far field signature, it is possible to obtain simple expressions relating the length, weight, and altitude of the airplane to peak sonic boom signature, the problem is more complex for the near field because the peak overpressure is not necessarily obtained at the beginning, and even the maximum peak overpressure is not indicative of the disturbance due to the fact that both criteria are important: the value of pressure discontinuities, and the maximum overpressure. Both qualities vary with weight of the airplane, length, altitude of flight, and airplane configurations.

In the work presented here (which is a continuation of the work presented in Ref. 11), details of the signature have been related directly to the airplane configuration. The results obtained have been derived by using two different numerical programs: (1) the program generated by Carlson at NASA Langley Field (required modifications have been introduced in the original program) which assumes constant atmospheric pressure averaged between the flight altitude and ground ($p = \sqrt{p_h p_{ground}}$); and (2) the program generated by W. Hayes[15]. The latter program is more accurate; it permits analyzing maneuvers; it takes into account variable density for horizontal flight; and requires somewhat longer computing time. Some of the data have been obtained with both programs. The differences between the results of the two methods are small and not important for the conclusion reached here, and therefore are not discussed.

The parameters investigated here are weight of the airplane configuration, length, Mach number, and altitude of flight. The criteria for the selection of the range of the parameters selected are briefly outlined below:

Three different values of weights have been considered corresponding either to horizontal flight or to maneuvers; one corresponding to 460,000 lbs, the second corresponding to 320,000 lbs, and the third corresponding to 240,000 lbs. The first weight assumed is representative of the first part of cruise for an airplane take-off weight of 600,000 lbs.

A typical airplane weight distribution for such an airplane is:

total weight	600,000 lb
empty weight	240,000 lbs
fuel	310,000 lbs

Such an airplane with a payload of 50,000 lbs. should be able to fly 4,900 statute miles. In selecting these data, it has been assumed that because the airplane is a second generation airplane, it will have improved structural design and better engines. Therefore, for the same weight and range it can be more complex and have lower L/D than first generation airplanes. For such airplanes, taking into account the fuel consumption for acceleration, the weight of the airplane for cruise varies between 490,000 to 330,000 lbs. Then the 460,000 lbs is close to the worst condition. Consistent with these data, an airplane having an initial gross weight of 400,000 lbs will have a range of 2,800 statute miles, with required reserve. Such a distance is typical of cross-country flights in the U.S.A. Such an airplane at the beginning of cruise will weigh 330,000 lbs and at the end 230,000 lbs. Then the first two values assumed correspond to the initial and final phases of cruise for transatlantic airplanes of 600,000 lbs; the second and third to an airplane having smaller ranges usable for cross-country flights.

The length of the airplane is a very important parameter in order to obtain near field signature; therefore, lengths on the order of 300 ft to 400 ft have been considered. Few calculations have been performed for shorter lengths in order to emphasize the difficulty of obtaining near field signature for short airplane lengths. In addition, the available height of the airplane has also been utilized in order to increase the effective length of the airplane. Present airplanes have vertical tails about 60 ft from the ground. The height can be used advantageously to decrease sonic boom in a biplane configuration as will be discussed later. In this case the effective length can be increased roughly by h/M_∞ with respect to the physical length (h is the height of the airplane, M_∞ the flight Mach number). Therefore an effective length of 300 ft could be obtained with an airplane on the order of 200 ft actual length at M = 2.7 while 450 ft can be obtained for an airplane 300 ft long. In a biplane configuration, the wing area is distributed in two wings. However, the wing thickness required by the structure probably will not increase substantially provided that the two wings are connected rigidly near the tip to form a box structure. Then, aerodynamically the system can be designed for low drag even at transonic speed.

In the analysis the presence of a vertical tail, the fact that the engine exhausts are usually placed near the leading edge of the wings, and the presence of the wake have been taken into account. These factors tend to increase the effective length of the airplane and decrease slightly the strength of the second shock of the N wave.

The flight Mach numbers considered are 1.5, 2.7, and 4. An increase in cruise Mach number tends to decrease the fuel consumption per mile, and therefore for a given weight of the airplane, will permit a better compromise for the design from the sonic boom point of view, because it will permit some degradation of aerodynamic performances.

Several flight altitudes have been considered between 30,000 and 80,000 ft. The altitude of 60,000 ft has been assumed as typical for the present SST designed by Boeing. However, altitudes of flight below 60,000 ft are of interest. The flight altitude is a very important parameter for the utilization of near field effects. In addition, a decrease of flight altitude decreases proportionally the lateral spread of the disturbance. When far field signature is present, the increase of altitude decreases the sonic boom overpressure; therefore, high altitudes have been considered desirable in the past. (For a given weight of airplane, the $(\Delta p)_{max}$ decreases roughly proportionally to $h^{3/4}$). However, when near field effects are utilized, such a conclusion is invalid, and lower altitudes of flight appear to be advantageous. A decrease of flight altitude simplifies the engine design and reduces engine weight and will permit using a somewhat more efficient engine, thus cancelling some of the penalties due to the decrease of L/D.

The parameter that has been investigated parametrically is the shape of equivalent cross-sectional area distribution. In order to obtain realistic equivalent area distribution that could correspond to a possible airplane design, the equivalent area distribution has been divided into two regions, i.e., the front and rear regions. The important characteristic of the front and rear regions are defined mainly by two parameters: (1) the area of the equivalent cross-sectional area in the front region indicated by L_1; the length of the front region indicated by ℓ_1. The details of the distribution of the equivalent cross-sectional area in this region do not affect strongly the results provided that such a distribution is close to optimum. Therefore, in all of the analyses, the equivalent area distribution has been divided into two regions, the total value of equivalent area and their length has been changed parametrically, while the form of the distribution has been kept constant.

IV. Results of the Analysis.

The distribution of an equivalent area along the length of the airplane is a very important parameter for the shape of the sonic boom. All of the results indicate that a minimum concentration of equivalent cross-sectional area is required in front in order to obtain near field effects. Such a minimum depends on the Mach number and altitude of flight. However, the details of such a distribution are not too important provided that the distribution is not too different from the optimum shape.

As an indication of these effects, in Figure 4 the sonic boom signatures obtained for an airplane flying at 60,000 ft , M = 2.7, and having a weight of 460,000 lbs are shown. The total equivalent area in each case is constant and corresponds to a weight of 460,000 lbs. The total length of the airplane is 300 ft. For all cases, this length has been divided into two regions(ℓ_1)70 ft, and the second (ℓ_2) 230 ft. The equivalent area at the end of the front part corresponds to 15.5% of the total. This value is also kept constant for all cases. The distribution of the equivalent area as a function of the length in the front part has been changed in the different diagrams as indicated in the figure. The distribution of the rear part has been kept constant and assumed in all cases to be linear. The figure indicates that for all cases near field effects are obtained and peak values of the order of 0.9 can be obtained for values of the exponents of the expressions between 1/3 and 1/5.

A change of either the ℓ_1, or of the value of the equivalent area of the front part $L_1 = \int_0^{\ell_1} k\, x^{\,n} dx$ for a given length of the airplane changes the near field region and the value of the $(\Delta p)_{initial}$. In Figure 5, sonic booms corresponding to a given value of ℓ_1 equal to 70 ft and different values of L_1 are shown. Too small an amount of equivalent area in front, i.e., L_1 = 9% of the total gives far field signatures. The extent of the near field signature increases with the increase of L_1; however the initial (Δp) increases with the L_1, when L_1 is larger than 16%. Similar results are obtained if the value of ℓ_1 is changed, and L_1 is kept constant.

It is interesting to observe that the peak sonic boom is of the order of .9 for the conditions selected, while configurations used for the first generation of supersonic transports have for corresponding conditions values of the order of 1.9. The difference is due to the distribution of lift along the airplane. In Figures 6a and 6b, two possible configurations are shown that correspond to the equivalent area distribution selected for the case of n = 1/3 and ℓ_1 = 70 ft and L_1/L = 15.5%. For comparison, in Figure 6a a configuration used for the first generation airplane having a peak value of 1.9 is also shown in dotted lines. The configuration of Figure 6a has a larger fuselage than the original one; therefore, the changes required are obtained by means of volume changes. In Figure 6b, the wing planform has been changed and a highly sweptback wing has been used, while the fuselage has been kept similar to the fuselage of the first generation of airplanes. The second configuration permits similar and better aerodynamic performances as the original airplane for cruise conditions; however, it probably will require more wing area for low speed flight.

The other design parameter that is of extreme importance is the total length of the airplane. In Figure 7, the effect of the total length of the airplane is indicated. In this comparison, the values of ℓ_1 and L_1 are kept constant and equal to 70 ft and 15.5% of the total lift and the total length has been changed. In Figure 8 a similar comparison is shown; however, the values of L_1 and ℓ_1 are optimized for each total length. An increase of length permits decreasing somewhat the Δp_{max} and permits having a slender fuselage. A possible configuration is shown in Figure 9.

The required length of the airplane can be exchanged with the height of the airplane. This possibility suggests that a biplane having wings that do not interfere at supersonic speeds, and do not choke at transonic speeds, has some good possibilities from the point of view of reducing sonic boom. An example of such a configuration is shown in Figure 10. The wing area has been distributed on two wings, one placed on the fuselage and the other placed on top of the vertical surface. The height of the vertical surface is the same as in present SST configurations of the same size. The two wings are staggered in order to avoid choking at transonic speed, and are connected by the vertical tail and have two vertical reinforcing structures near the tip to decrease the bending stresses. The sonic boom of this configuration is shown in Figure 11. Figure 11 indicates that without increasing the length and the height of the configurations considered, it is possible to reduce the jump in Δp due to the sonic boom for an airplane 460,000 lbs flying at 60,000 ft and M=2.7 to values of the order of 0.5 to 0.6. From an annoyance point of view, the jump across the shock is the parameter to be considered.

Let us consider now the variation of the total weight at the same Mach number and altitude of flight. In Figure 12, the sonic boom of three of the airplanes having the same length and the same distribution of cross-sectional area, but different weights, are shown. A decrease in weight decreases substantially the $(\Delta p)_{max}$.

In Figure 13, the sonic boom for an airplane weight of 320,000 lbs is shown for several values of L_1. The length of the front part is constant and equal to 70 ft,and total length is also constant and equal to 300 ft. Figure 13 indicates that shocks on the order of 0.6 lb/ft^2 can be obtained for a supersonic transport useful for a cross-country flight with a weight equal to 320,000 lbs. The physical configuration of the airplane changes when the value of the lift changes even if the distribution of equivalent area is similar, because the volume of the airplane must change in proportion. For a weight of 320,000 lbs the pressure jump can be reduced to values as low as 0.4 lb/ft^2 if a biplane configuration is used.

Figure 14 gives the sonic boom configuration for a biplane flying at M = 2.70 at 40,000 and 60,000 ft altitude. The airplane is 300 ft long. A possible configuration is similar in shape to the configurations shown in Figure 16. The first and last discontinuous jumps have a value of the order of 0.38 and 0.41; after the initial jump a gradual pressure rise takes place that is not objectionable from the point of view of disturbances to the population.

The altitude of flight is also a parameter. A change of flight altitude between 60,000 and 30,000 ft does not change strongly the value of the minimum intensity of the pressure jump.Values of the order of -0.8 have been obtained in the range of altitude for optimum configurations for constant weight and length of an airplane corresponding to 460,000 lbs and 300 ft in length. However, the corresponding distribution of the equivalent cross-sectional area changes. The value of L_1 required in order to obtain low sonic boom for a given ℓ_1 decreases when the altitude decreases. In addition, it is possible to reduce substantially the value of the (Δp_{max}) in front if some higher value is accepted for the (Δp) of the trailing shocks.

The possibility of selecting the flight altitude without penalty, from the point of view of sonic boom minimization could be attractive in order to reduce the area disturbed by the sonic boom, and at the same time to obtain an acceptable aerodynamic design. The cross-section and the volume of the fuselage required for the airplane depends on the mission and aerodynamic requirements of high L/ D. When the altitude or Mach number of flight are changed for a given vehicle weight and cross-sectional area distribution, the relation between cross-sectional area contribution due to lift and due to volume changes, because the equivalent cross section due to lift is proportional to the ratio between weight and dynamic pressure (the coefficient is $\beta/2$). Then an increase in Mach number or decrease of altitude for a given total weight increases the dynamic pressure and decreases the contribution of the lift with respect to the contribution of volume, and vice versa. Then if the equivalent area distribution selected on the basis of sonic boom optimization requires too large a fuselage, two alternatives are available: either using highly sweptback wings as shown in Figure 6b; or changing the flight Mach number or altitude. If the flight Mach number increases, then the fuselage cross section corresponding to the distribution decreases; the same occurs if the altitude of flight decreases. This effect gives some flexibility in selecting acceptable configurations.

The last parameter considered is the Mach number. An increase in Mach number increases slightly the peak (Δp) for the same configuration, for the same weight and altitude of flight, but the difference is very small, Figure 15. If the Mach number and altitude of flight decreases, then the near field effects become very pronounced, and very low initial Δp can be obtained. For these conditions, initial pressure jumps as low as 0.3 are possible; however, the trailing shock does not decrease in the same proportion, and usually tends to increase, unless the equivalent airplane length is substantial. Again the biplane configuration can be used to advantage. A signature obtained for a biplane 290 ft long, flying at M = 1.5 and 40,000 ft is shown in Figure 16. The peak pressure is less than 0.3 lb/ft^2. A possible corresponding configuration is shown in Figure 17.

V. Conclusions

The results of the analysis presented here indicate that from an aerodynamic point of view it is possible to generate airplane configurations that can reduce substantially the strength of the front and tail shocks of sonic booms for airplanes designed for transatlantic operations; values as low as 0.5 lb/ft^2 are possible. Values as low as 0.4 and 0.3 are possible when the weight is reduced for cross-country operations, and the airplane is optimized for minimum sonic boom. These values are much lower than the values investigated in present flight tests. In addition, disturbances of the same order are presently accepted in normal operations in populated areas. The analysis presented here has analyzed only superficially the consequences of utilization of such concepts on airplane performances. Two steps are required in order to proceed further: (1) the acceptance of such levels of disturbances should be determined by measuring the shape and level of present disturbances currently generated in city operations, and by additional flight tests; and (2) the incorporation of such concepts in practical usable configurations for second generation supersonic transports should be investigated.

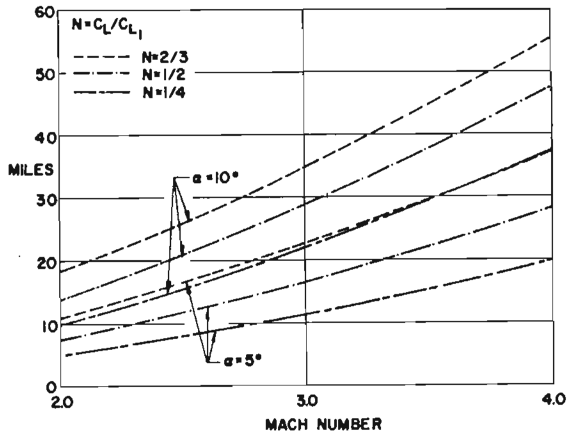

Figure 1. Range as a function of flight Mach number for a constant C_L, constant speed trajectory, starting at 40,000 ft, and angle α, and terminating at the same altitude.

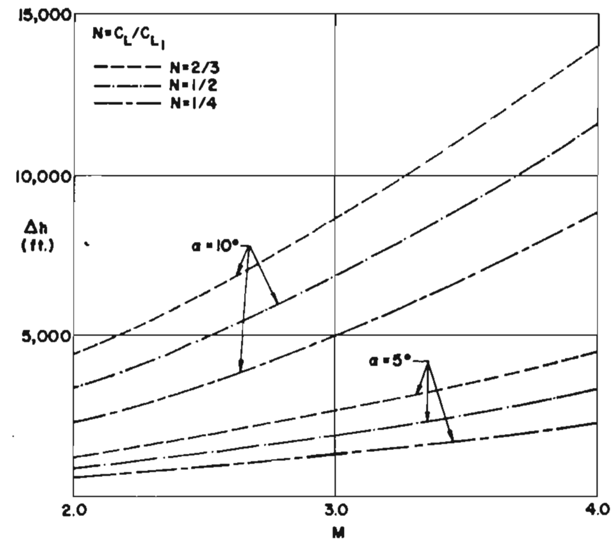

Figure 2. Maximum Δh reached for the airplane during the trajectory, corresponding to ranges given in Figure 1.

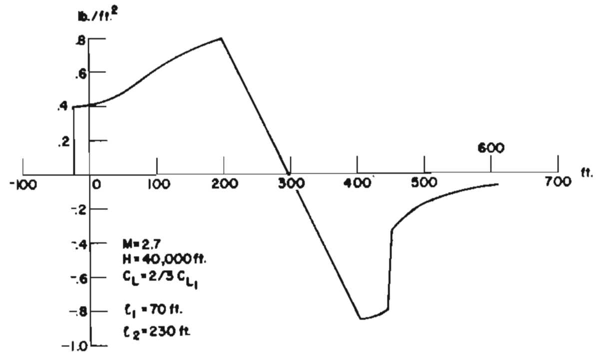

Figure 3. Sonic boom signature corresponding to an airplane 300 ft long flying at M = 2.7 and 40,000 ft altitude at a lift equal to 2/3 of the lift required for horizontal flight. Airplane weight is 460,000 lbs. The equivalent area of the front part is equal to 1/9 of the total.

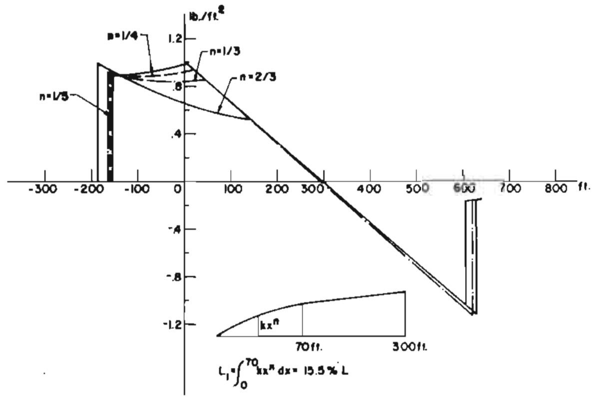

Figure 4. Effect of distribution of equivalent area in the front part. Airplane characteristics: L = 300 ft, weight 460,000 lbs, M = 2.70, h = 60,000 ft L_1 frontal area 15.5% total, ℓ_1 = 70 ft.

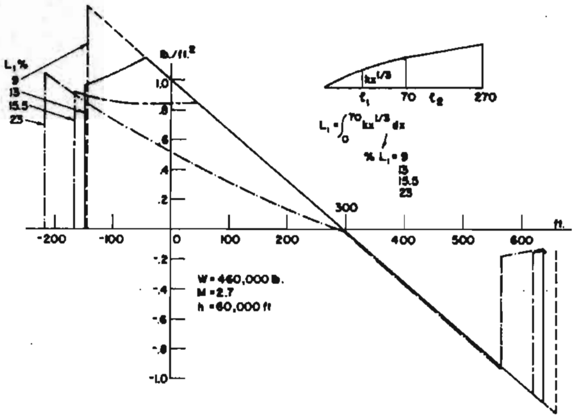

Figure 5. Sonic booms corresponding to M = 2.7, h = 60,000 ft, and weight 460,000 lbs, total length equal to 300 ft, and different equivalent area in the front part, ℓ_1 = 70 ft.

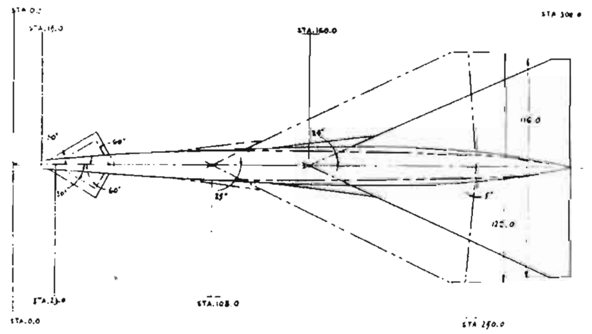

Figure 6a.

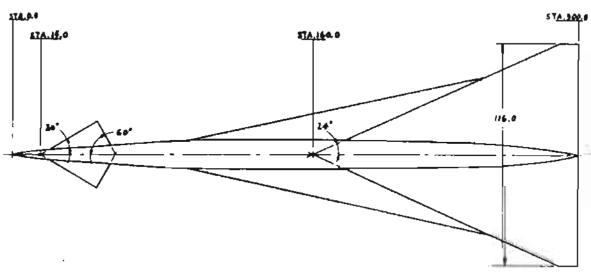

Figure 6b.

Figures 6a and 6b. Possible airplane configurations corresponding to the sonic boom shown in Figure 5 for L_1 = 15.5%L.

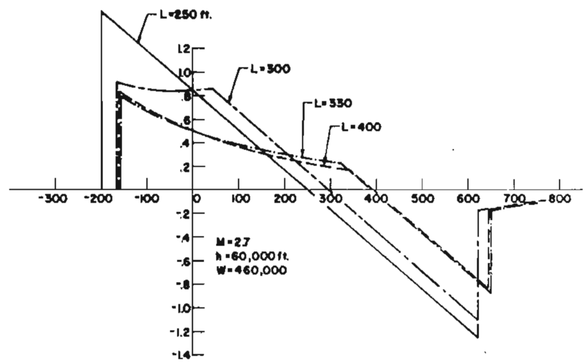

Figure 7. Effect of airplane length on sonic boom signature.

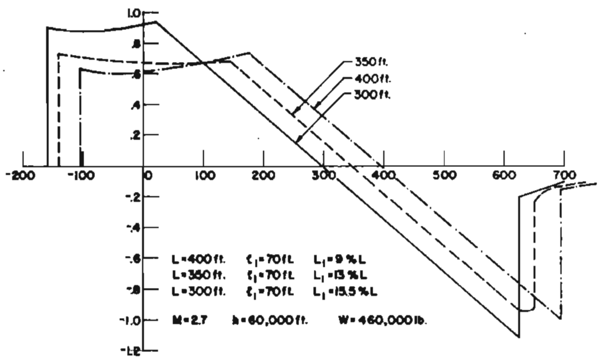

Figure 8. Effect of length on sonic boom signature.

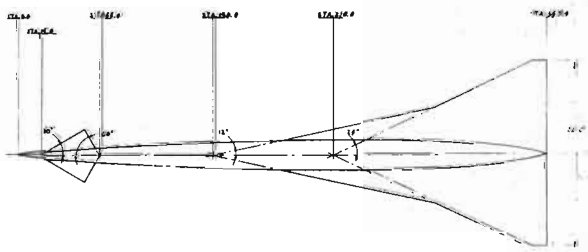

Figure 9. Possible configuration of an airplane 350 ft long having max Δp equal to 0.7 lb/ft^2.

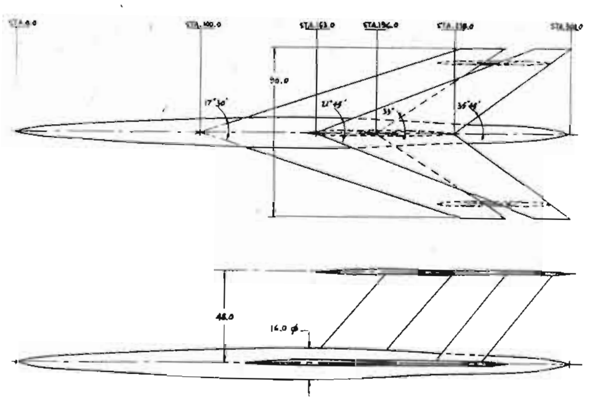

Figure 10. Schematic design of a biplane configuration 300 ft long having sonic boom signature shown in Figure 11.

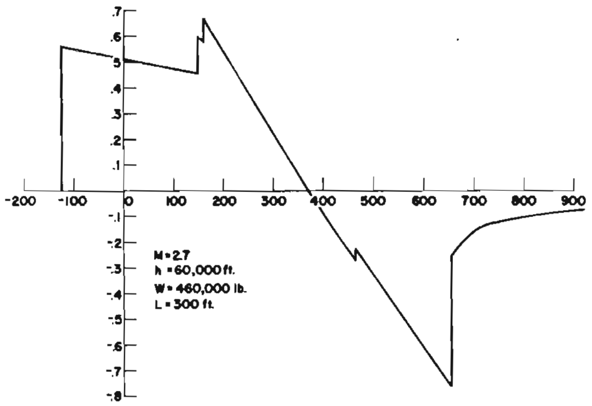

Figure 11. Sonic boom signature of the biplane shown in Figure 10.

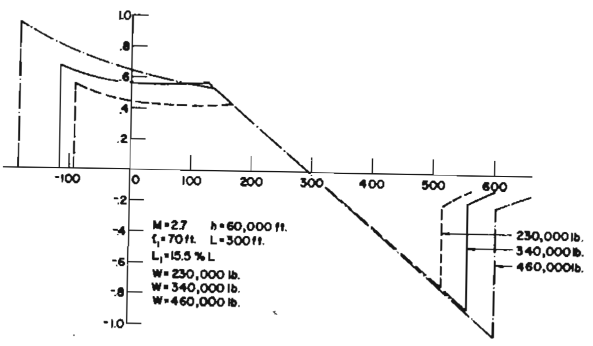

Figure 12. Sonic boom signature as a function of airplane weight.

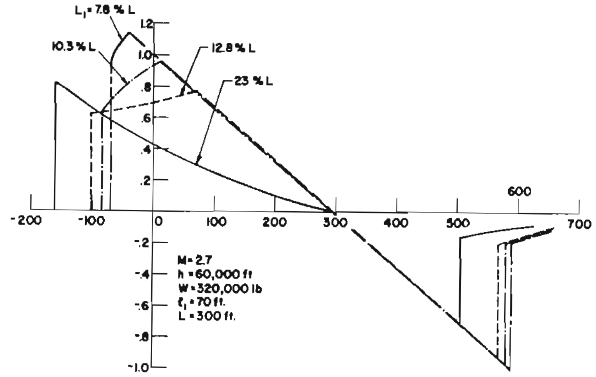

Figure 13. Sonic boom configurations for airplanes of 320,000 lbs weight, 300 ft long, and flying at M = 2.7 at 60,000 ft.

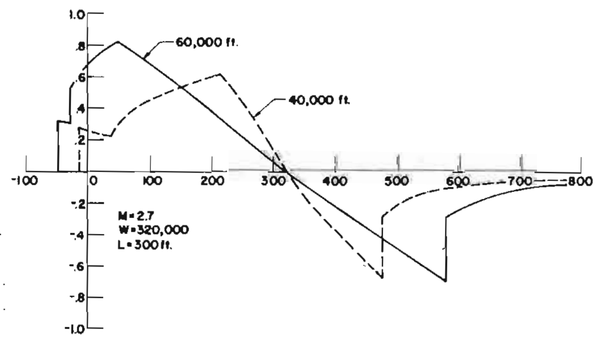

Figure 14. Sonic boom signature of a biplane configuration at M = 2.70 and h = 40,000 and 60,000 ft. Length 300 ft, height 45 ft, weight 320,000 lbs.

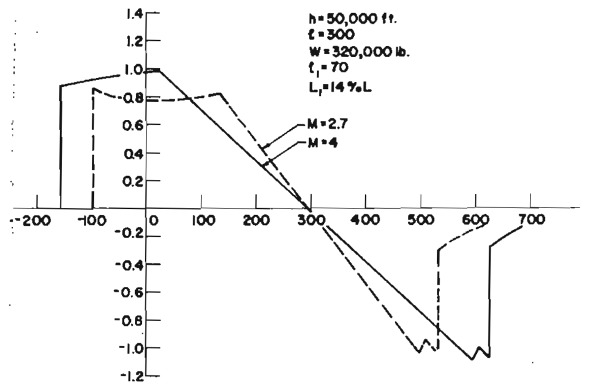

Figure 15. Sonic boom for 2 airplanes flying at M=4 and M=2.70. The airplanes have the same weight and equivalent area distribution.

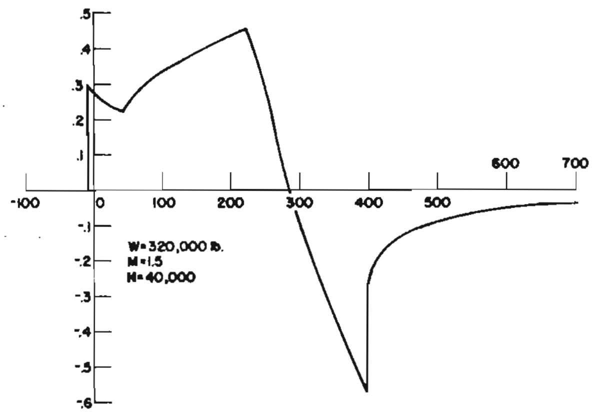

Figure 16. Sonic boom signature of the biplane shown in Figure 17 flying at M=1.50, and 40,000 ft altitude, weight 320,000 lbs, length 290 ft.

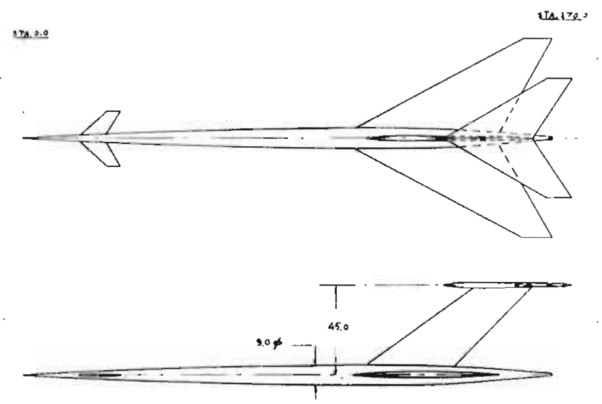

Figure 17. Possible biplane configuration corresponding to sonic boom signature shown in Figure 16.

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