Alleviation of the Sonic Boom by Thermal Means

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Previous studies suggest that reduction of SST sonic boom noise to levels acceptable to the public requires elimination of shocks and creation of pressure rise times of 10 msec or more at the ground. Because it appears impractical to lengthen the aircraft sufficiently to achieve this objective, a feasibility investigation of the use of heat to simulate a long body was undertaken. The large power requirement dictates the use of direct combustion of cheap fuel for this purpose, and practical considerations motivate off-axis heating. This can be accomplished in principle by the use of either a thermal spike with its axis parallel to the airstream or a thermal keel that extends downward from the aircraft.

Introduction

THE basic theory of the sonic boom has been treated in a number of papers[1-5] and will not be repeated here. Stated briefly, the pressure disturbance in the near field is concentrated primarily in the region between the bow and tail shock waves (Fig. 1). Because of the fact that high pressures travel faster and low pressures travel slower than sonic velocity, the high pressures tend to move forward and low pressures fall to the rear. As a result, in the far field, we find an *N*-shaped signature characterized by two shocks. In the case of the supersonic transport (SST), it is estimated that the shock strength will be in the range 2–4 psf reflected pressure, and the time between shocks will be approximately 0.4 sec.

The main aspects of a sonic boom that cause public concern are structural damage and noise. The structural damage has been studied for at least a decade,[6,7] and a digest of the main results is presented in Table 1. Generally speaking, the data indicate that only relatively decrepit structures will be damaged below about 5 psf.

Because relatively few people are likely to experience significant property damage as a result of sonic booms of the type expected in normal operation of the SST, it appears that adverse public reaction is mainly associated with the acoustic effects. Many people are startled by a shock having an over-pressure of the order 2 psf or more, and this gives rise to annoyance. The purpose of the present paper is to report the results of a preliminary inquiry into the feasibility of effecting a substantial reduction in the noise level associated with the sonic boom.

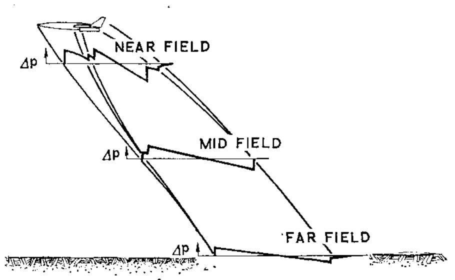

Fig. 1 Signature of supersonic aircraft.

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Table 1 Structural damage due to sonic booms

Date and place	Results
1960 Nellis AFB, Nev.	Windows damaged when $\Delta p \geq 20$ psf
1961–2 St. Louis, Mo.	Cracked plaster and broken glass (66 flights up to 3.1 psf)
1964 Oklahoma City, Okla.	Large percentage of public believed sonic boom caused structural damage; however, no structural damage in instrumented houses ($\Delta p \leq 2$ psf)
1964–5 White Sands Missile Range, N. Mex.	Test structures undamaged by 680 5-psf booms; 8-psf damaged pre-cracked plaster and glass
1965 Star Mountain, Calif.	No avalanches from 18 sonic booms $1.5 \leq \Delta p \leq 2.2$ psf
1966 Edwards AFB, Calif.	4 psf booms produced smaller wall accelerations than slamming doors or running and jumping

Noise Considerations

The spectrogram of an *N*-wave has been calculated by a number of investigators[6,8,9] and is rather complicated. The main features of the situation, however, can be described simply if the fine structure is neglected and only the envelopes are considered.

The results of such a calculation are presented in Fig. 2, which shows a comparison between zero rise time and finite rise time signatures for the nominal over-pressure 2 psf. In

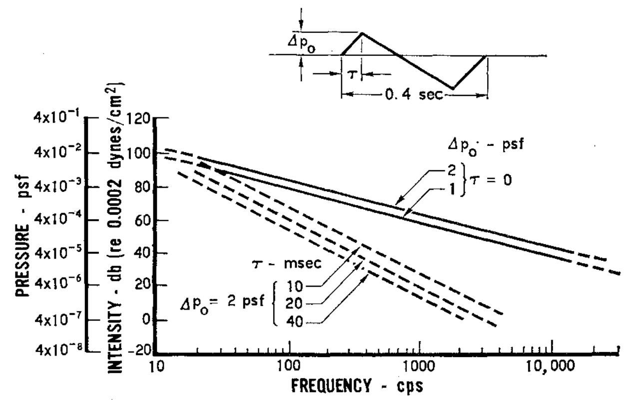

Fig. 2 Sonic boom signature spectrogram.

the case of zero rise time, the envelope is a straight line that obeys the equation

$$f_o(\nu) = \Delta p_o/\pi\nu \tag{1}$$

where Δp_o = pressure rise, T = time between shocks, ν = frequency. In the case of finite rise time τ, the envelope coincides with that for zero rise time for low frequencies; however, at high frequencies, the envelope drops to

$$f_\tau(\nu) = \Delta p_o[1/\tau + 1/(T - 2\tau)]/\pi^2\nu^2 \tag{2}$$

which reduces to

$$f_\tau(\nu) = \Delta p_o/\pi^2\tau\nu^2 \tag{3}$$

when $\tau \ll T$. The envelope for finite rise time drops below that for zero rise time at the frequency for which $f_o(\nu) = f_\tau(\nu)$, i.e., at

$$\nu_{cr} = (\pi\tau)^{-1} \tag{4}$$

Thus the slope of the zero rise time envelope is −6 db per octave, while that of the finite rise time envelope at frequencies above ν_{cr} is −12 db per octave. Accordingly, a rise time as low as even 10 msec results in a very substantial reduction of acoustic power in the frequencies to which the ear is most sensitive, i.e., $\nu > 300$ Hz. In fact, experiments with 90 subjects led Zepler and Harel[10] to conclude that such a rise time reduces the apparent noise level by about 15 db. In the case of zero rise time, a 6-db reduction in the noise level can only be achieved if the pressure is halved. In the case of the already quieter finite rise time, additional 6-db reductions in high frequency response can be achieved either by halving the pressure or doubling the rise time where rise times are small compared with the period. A comparison of these spectrograms with that of typical street noise and with the thresholds of hearing and feeling, shown in Fig. 3, suggests that a 10-msec rise time sonic boom would seem significantly louder than normal urban traffic.

It would appear from Fig. 3 that what annoys people is not an intolerable noise level but the startle effect. It has been suggested by Beranek† that being startled by sudden loud noises is a response built into our nervous systems by a process of evolution over many millennia, because it has been important to the survival of the species. As a result, we cannot easily get used to them as we can to more gradual loud noises. On the basis of the rather limited evidence available at present, Beranek estimated that the shock strength would have to be reduced below about 0.5 psf in order to reduce the startle level to that required for public acceptance. The possibility that, by clever design, the over-pressure level could be reduced by the indicated factor 4 or more is debatable. Even if such a reduction could be achieved, there is no margin for boom amplification that may be caused by maneuvering. We therefore turn our attention to means for effecting a finite rise time.

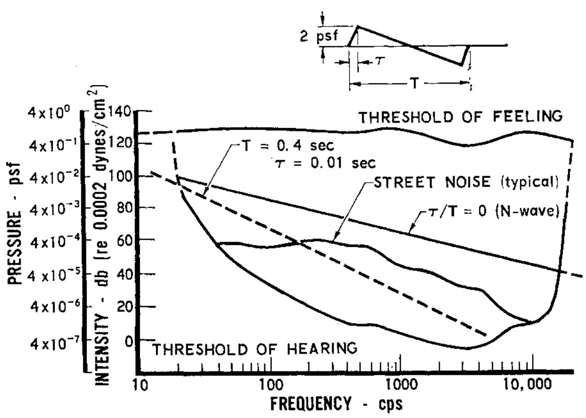

Fig. 3 Comparison of sonic boom spectrogram with auditory response.

† Beranek, L. N., Introductory remarks while serving as Chairman of Session on Aircraft Noise, 6th AIAA Annual Meeting and Technical Display, Oct. 1969, Anaheim, Calif.

Length Required for Shock-Free Signature

The fundamental tool in sonic-boom analysis is G. B. Whitham's theory for the flow about bodies of revolution traveling at supersonic speeds.[1,2,11–13] Equivalent bodies of revolution for lifting aircraft are found by the exploitation of the supersonic area rule[2,14,15] and by the introduction of an equivalent area associated with lift. If the pressure rise in the near field is made sufficiently gradual, the formation of shocks can be made to require a propagation distance greater than the operating altitude of the aircraft. It can be shown that the minimum length body for finite rise time signature is the one that causes a linear pressure rise. It follows from the Whitham theory that this is generated by a body whose area varies with spanwise location x according to the law

$$A(x) = Kx^{5/2} \tag{5}$$

Whitham's theory has been amply confirmed by wind-tunnel experiments and field tests.[16,17]

The merits of changing the far-field signature in the manner just described have been recognized for some time, and several investigators have calculated the minimum length aircraft for shock avoidance. For the SST (i.e., for an aircraft with a gross weight of 600,000 lb and traveling at an alt of 60,000 ft at Mach 2.7), a figure of 1000 ft was obtained by F. E. McLean[18] on the basis of a uniform atmosphere and inclusion of the tail wave effect. The same length was obtained by Seebass[19] for an exponential atmosphere without a tail wave effect. More recent work of Seebass[4] led to a length of 580 ft. It would appear that lengths required to achieve the desired objective are well beyond those generally considered feasible for practical implementation.

However, even if it proves impracticable to construct a vehicle of the length and shape required to eliminate shocks from the ground pressure signature, it should be possible to simulate such a body. The present paper discusses the use of heat for this purpose.

Thermal Simulation of Long Aircraft

Let us consider two supersonic stream tubes of initial area A_o (Fig. 4). One stream tube contains a mechanical spike having the desired variation of area with spanwise location x' that isentropically compresses the air and causes a radial outward flow.

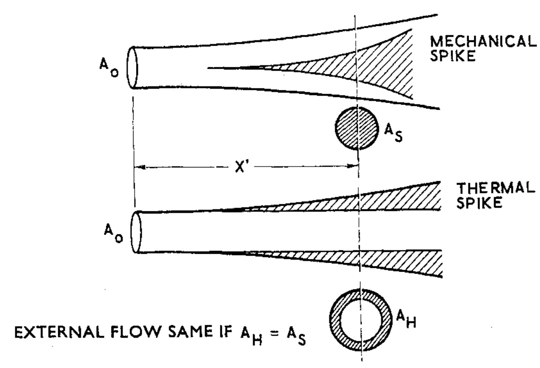

Fig. 4 Relation between thermal spike and mechanical spike.

The total area within the external circumference of the tube at point x' is

$$A(x') = A_s(x') + A_1(x') \quad (6)$$

where $A_s(x')$ is the area of the spike and $A_1(x')$ is the area occupied by air.

In the second stream tube, heat is deposited in such a way that it expands in exactly the same manner as the first stream tube, i.e., according to Eq. (6). It follows that, outside the stream tubes, the pressure distributions are identical in the two cases. We can eliminate any concern over the effects of heat transfer in the second tube by taking A_o so large that only a negligible amount of heat will flow through the walls of the tube in the time required for the air to travel the length of the very long aircraft that is to be simulated. A stream tube that is heated for the purpose of simulating the effects of a mechanical spike is referred to in the remainder of this paper as a thermal spike.

In the case of a sufficiently slender body, the pressure at x' differs only slightly from ambient; hence, $A_1(x') \cong A_o$. If it is assumed that the pressure is constant, that the disturbance velocity is entirely radial, and also that the air is a perfect gas, the fractional change in area can be equated to the fractional change in absolute temperature, i.e.,

$$[A(x') - A_o]/A_o = [T(x') - T_o]/T_o = A_s(x')/A_o \quad (7)$$

Probably the most attractive way to produce the desired changes in temperature along the axis in front of the aircraft, if it were feasible to do it in this manner, would be to use radiation. Carbon dioxide lasers are currently the most powerful and efficient, but air is virtually transparent to the 10.6 μ radiation of this laser; therefore, it could not be used for heating. Air is relatively opaque in the ultraviolet, and laser radiation in these frequencies can be produced by ion lasers; however, these are extremely inefficient. Laser-initiated radiation in the ultraviolet can also be obtained by frequency doubling, with substantially better efficiency than that of ion lasers. Alternatively, electrons at energies of several hundred kilovolts would have about the right range. Any approach, however, that requires conversion of fuel first into heat and then successively into mechanical, then electrical or radiant energy, and finally back into heat is too inefficient to look attractive when consideration is given to the total heat requirement, as will be evident shortly.

We can determine the thermal requirement for eliminating the front shock of the SST as follows: Let us assume that heat is delivered to the air within the tube (Fig. 4) in such a manner that, at each position x', the temperature is that given by Eq. (7) and is uniform over the entire cross section. This is a particularly simple model selected to permit the desired result to be calculated easily, but it will be shown that the results are independent of the model chosen. The mass of air entering this tube per second equals the mass of air leaving each second, which equals $\rho_o U_o A_o$. If the final temperature of the air is T_F, then the heat required per second is

$$\dot{Q} = \rho_o U_o A_o c_p (T_F - T_o) \quad (8)$$

where c_p is specific heat at constant pressure. From Eq. (7), however, we have

$$(A_F - A_o)/A_o = \bar{A}_H/A_o = (T_F - T_o)/T_o \quad (9)$$

where $\bar{A}_H$ is the total added area due to the heat addition. Combining Eqs. (8) and (9), we obtain

$$\dot{Q} = \rho_o U_o T_o c_p \bar{A}_H = \rho_o U_o h_o \bar{A}_H \quad (10)$$

where h_o is the specific enthalpy at cruise altitude.

The fact that the results must be independent of the details of heat distribution within a cross section can be established by showing that, when the pressure is kept constant, the increase in volume in a gas depends only on the heat added. In order to do this, let us assume that the original volume of a gas is V_o and that it is expanded when the temperature is varied from the original uniform value of T_o to a nonuniform final temperature $T(x, y, z)$. Here, x, y, and z refer to the original coordinates of a particular gas particle before the expansion takes place. The amount of heat required to produce these temperatures is given by the expression

$$Q = \int [T(x,y,z) - T_o] c_p \rho_o \, dxdydz \quad (11)$$

The volume change of the element $dxdydz$ that results from an increase in temperature from T_o to T is given by the expression

$$\Delta(dxdydz) = [T(x,y,z) - T_o]/T_o \, dxdydz \quad (12)$$

Total volume change is obtained by the integration of Eq. (12), which leads to

$$\Delta V = \int \Delta(dxdydz)$$

$$= (1/T_o) \int [T(x,y,z) - T_o] dxdydz \quad (13)$$

Combining Eqs. (11) and (12), we obtain the result

$$\Delta V = Q/\rho_o c_p T_o \quad (14)$$

which shows that the volume change depends only upon the heat added, the nature of the gas, and the initial conditions. The change is not dependent upon the initial volume of the gas to which the heat was added nor on the details of the distribution of the heat within that volume.

In order to determine the thermal power required to eliminate the front shock, we must find $\bar{A}_H$ and substitute it in Eq. (10). This can be achieved in the following manner. Figure 5 shows an effective area composed of three parts: the area due to the volume of the aircraft, the area due to lift, and the area due to added heat. (The curves for the effective area due to volume and lift were given as representative of a typical SST configuration in Ref. 3.) The length of spike is designated L_s, and heating occurs over the entire distance in front of the aircraft in such a manner that $A_H(x')$ satisfies Eq. (5). The total area, which consists of $\bar{A}_H \equiv A_H(L_s)$ added to the other two sources of effective area, must continue to satisfy Eq. (5) to the center of the aircraft if burning is stopped at the nose of the SST. It is evident from Fig. 5 that this will require some slight tailoring of the lift and volume components of effective area. The total area $A_T(x')$ at any cross section forward of the center of the aircraft is given by

$$A_T(x') = \bar{A}_H (x'/L_s)^{5/2} \quad (15)$$

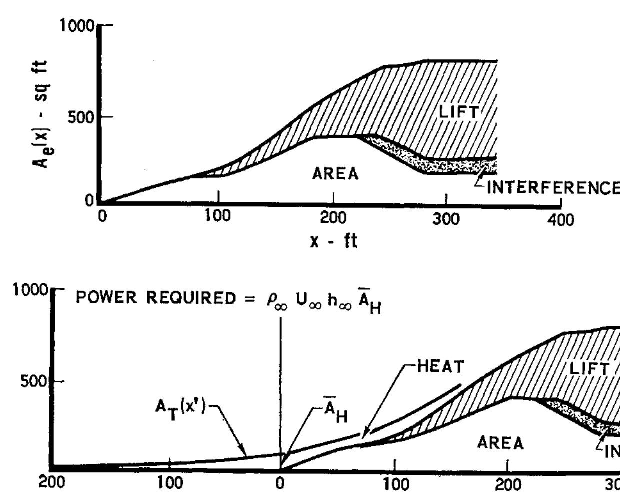

Fig. 5 Determination of power required for thermal simulation of front spike.

Equation (15) is written in such a form as to guarantee that the total area will be equal to $\bar{A}_H$ at $x' = L_s$, but one more condition is required to fix A_H. If we choose the condition that, at the center of the aircraft (where $x' = L_c$), Eq. (15) continues to apply

$$A_T(L_c) = A_V(L_c) + A_L(L_c) + \bar{A}_H = \bar{A}_H(L_c/L_s)^{5/2} \tag{16}$$

Solving Eq. (16) for $\bar{A}_H$, we obtain

$$\bar{A}_H = (A_V + A_L)[(L_c/L_s)^{5/2} - 1]^{-1} \tag{17}$$

According to unpublished calculations by Busemann, Swigart, and Murdock, a thermal spike 200 ft long is required to achieve a rise time of 10 msec in the case of an aircraft 300 ft long, with the effective area distribution due to volume and lift as shown in Fig. 5. By the use of these values, the result $\bar{A}_H = 120$ ft^2 is obtained from Eq. (17). If we substitute this area into Eq. (10) and use the following values for aircraft velocity and environmental conditions during cruise: $U_o = 2700$ fps, $\rho_o =$ density at 60,000 ft $= 0.000224$ slugs/ft^3; $c_p =$ specific heat at constant pressure $= 0.27$ Btu/lb/°F, $T_o = 390$°R, the required power of approximately 2.1×10^5 Btu/sec is obtained.

The significance of 2.1×10^5 Btu/sec can be assessed if it is compared with the power needed to operate the SST. If the weight is 600,000 lb and the lift-drag ratio is about 6, the output power is approximately 3.5×10^5 Btu/sec; i.e., we need about 60% of the SST cruise power. This result makes unattractive any approach that uses the mechanical power generated by the aircraft as its starting point.

If the heat can be produced in the desired regions in a properly tailored fashion by direct combustion of the same fuel as is used to propel the SST, the price of eliminating the front shock may not be too severe. If the over all efficiency of the SST under cruise conditions is 30%, the fuel used for propulsion purposes is consumed at a rate equivalent to $3.5 \times 10^5 \div 0.3 \cong 1.2 \times 10^6$ Btu/sec. Thus, the thermal cost previously discussed amounts to a little less than 20% of the SST's cruise-rate fuel consumption.

The possibility of using heat for sonic boom alleviation was first brought to public attention by Carlson and Miller of NASA's Langley Research Center.[20] Their studies covered both a heat-field and a force-field approach, and were primarily concerned with the magnitude and distribution of energy required to be delivered to the air rather than the mechanism of energy generation and application. In the example they used to illustrate the application of the heat field method to a representative SST configuration, there was an estimated power requirement equivalent to about twice the power necessary to sustain the airplane in steady, level cruise flight.

The large reduction in power in the present investigation, as compared with that obtained by Carlson and Miller, is primarily due to three factors. First, and most important, the fuel requirement need not be twice the nominal unless the heat field power is to be extracted from engine output power. With direct burning of fuel, as proposed in the present study, the heat field power estimated by Carlson and Miller could be supplied by a rate of fuel consumption equal to about 70% of that at the cruise level. Second, their calculation of the required length was based on a uniform atmosphere, whereas the present analysis takes advantage of the reduction due to the approximately exponential variation of air density. Third, their estimate was for a ground pressure signature with a rise time of almost 200 msec, while the present analysis uses a much shorter rise time in order to minimize the cost in power. When these factors are properly taken into account, no sizable discrepancy remains in the results of the two studies that were obtained by somewhat different analytic techniques.

Although the thermal spike can be made completely equivalent to the mechanical spike so far as the pressure rise in front of the aircraft is concerned, an important difference occurs at the nose of the SST. In the case of a mechanical spike, the nose of the aircraft is shrouded and does not give rise to any shock. In the thermal case, the heated airstream passes over the nose, and the effective area directly behind the nose is the sum of an effective area due to heat addition and an effective area due to the fuselage. For purposes of discussion, two cases may be distinguished. In the first, the nose creates a bow shock; in the second, the nose is tapered so as to produce a gradual pressure rise. If the total effective area obeys Eq. (5), one might expect that there would be a bow shock in the first case that weakens with distance from the axis and disappears at the edge of the heated region. In the second case, the creation of the front shock should be avoided entirely.

At the January 1969 AIAA meeting in New York, Cheng and Goldberg[21] presented a paper that analyzed a sonic boom alleviation technique based on the use of a force field.[22] This scheme proposed the use of an electrostatic field to ionize the oncoming air and gradually turn it so as to result in a gradual pressure rise, thus, hopefully, avoiding shock formation. Cheng and Goldberg pointed out that a blunt body is in effect created in this approach which gives rise to a detached shock, thus aggravating the problem instead of solving it. Their results have been widely interpreted as invalidating any nonmechanical approach, including the use of heat, for avoid ance of the shock. Actually, of course, their results apply only under conditions giving rise to strong interaction and not under the weak interaction conditions proposed here.

Off-Axis Heating

There are essentially two types of difficulty associated with on-axis heating. One type is connected with finding satisfactory means for deployment of the heat, concerning which more will be said later. The second has to do with the fact that the heated air must then sweep over the SST, thus further heating the skin of the vehicle, which is already a serious problem. In the case analyzed by Carlson and Miller,[20] the stagnation temperature at the nose was increased above its already high value by about 400°F as a result of using heated air. Because of these difficulties, off-axis heating appears more attractive.

Off-axis heating can, in principle, still be axisymmetric. For instance, the heat can be deployed in the surface of a cylinder that is coaxial with the SST. While such an approach would alleviate the skin heating problem, at least over the fuselage, it is difficult to see how heat could be deployed in a practical manner in this configuration. Also, the portion of heat that is delivered in the region preceding the plane of the aircraft appears not to be contributing its full share to the solution of the problem. Accordingly, we will restrict our attention here to asymmetric off-axis heating in which the heat is deployed below the aircraft in or near the vertical plane.

From a conceptual point of view, the simplest way of accomplishing this objective is to employ a thermal spike identical to that just discussed, but with its location shifted down from the original axisymmetric position to some point below (Fig. 6). In order to obtain the same signature, the thermal spike in a displaced position must have its front located on the Mach line going through the front of the original axisymmetric thermal spike location as shown in Fig. 6. The idea here is simply that, within the framework of linearized theory, a point on the ground experiences a pressure disturbance that is independent of the location of the source along the Mach line.

Instead of generating heat in a distributed and properly tailored fashion along the horizontal axis as shown in Fig. 6,

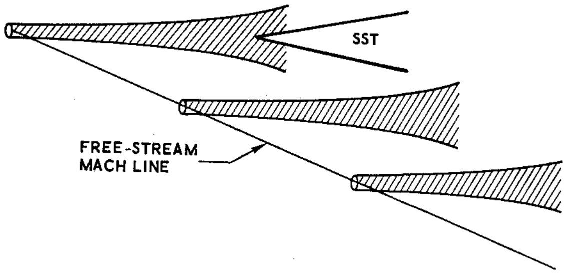

Fig. 6 Thermal spikes creating same signature at ground locations in vertical plane.

one can alternatively produce the heat in a distributed and properly tailored fashion along the vertical axis as shown in Fig. 7. In this approach, a thermal keel of heated air would be created below the SST. This can be done in a manner that would not heat the aircraft itself, and thus avoid one of the difficulties to which the on-axis approach is subject. According to the supersonic area rule, if the Mach plane passing through each point on the axis intersects the same total stream tube area increment in the thermal keel (including any solid structures required to deploy the heat) as it would have in the case of the axisymmetric thermal spike as shown in Fig. 7, the disturbance in the far field will be the same for the keel as for the spike.

The equality of the areas formed by the intersection of the Mach plane with the thermal spike and the thermal keel implies the equality of these two areas as viewed from in front. Because the front projections are the actual effective areas for purposes of computing the equivalent body of revolution, it is apparent that the increase in area created by the thermal fin should be the same as that created by the thermal spike in the previous considerations. Consequently, the heat consumption would be the same in both cases if, in the process of creating the heat, there were no losses and no net drag or thrust.

In the case of the thermal keel, it is readily seen that the effective area corresponding to a given point on the axis is the total area between the bottom of the keel, i.e., $z = 0$, and the height at which $z = x \tan\mu$. Accordingly, the thickness $t(z)$ should vary with height in such a way as to satisfy the relation

$$\int_0^{x\tan\mu} t(z)dz = K(x)^{5/2} \tag{18}$$

Differentiating this expression, we obtain

$$t(z)\tan\mu = \tfrac{5}{2}K(x)^{3/2} \tag{19}$$

This implies that

$$t(z) = (\tfrac{5}{2}K\cot^{5/2}\mu)z^{3/2} \tag{20}$$

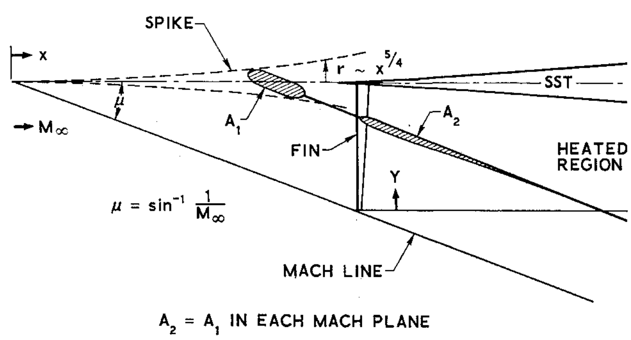

Fig. 7 Thermal keel.

Combustion processes take place over a distance that is inversely proportional to the pressure and directly proportional to the velocity of the combusting fluid, and there is a lower limit to the pressure at which combustion can be sustained. Because of this, at the ambient conditions and velocity of the SST, external burning of hydrocarbons such as JP-4, if feasible at all, can be expected to occur over distances of a hundred feet or more. As a result, external burning is impractical for the thermal keel approach, where we want the air to be fully expanded (and thus combustion completed) as far upstream as possible. Conversely, it is ideally suited to the thermal spike, where the burning must continue for approximately 200 ft behind the point of its initiation. It therefore appears that horizontally distributed combustion (thermal spike) should employ external burning, and vertically distributed combustion (thermal keel) should employ internal burning as in a ramjet of some type.

Discussion

In order to complete this study, it is necessary to further discuss several subjects that were either omitted in the preceding sections or were touched on only very lightly. The first of these is the question of how to deal with the rear shock in the N-wave signature.

This matter has recently been investigated by Carlson and Miller.[23] Their calculations indicate that, if heat from a rear-mounted vertical fin is used, the rate of fuel consumption for elimination of the rear shock is of order 40% of that for the propulsion during cruise. A rate this large is somewhat discouraging, but it should be pointed out that there are alternatives. For instance, an engine can be used to reduce the stream tube cross section.[24] With optimum shaping of the effective area by means of cross section reduction and appropriate modifications in the effective area due to volume, lift, and heat addition, the fuel cost of eliminating the rear shock might be substantially reduced.

The second question is the tolerance in the area distribution required for a finite rise time. No experimental data have come to the attention of the author, but it is likely that the shorter the rise time, the more sensitive the ground signature will be to deviations from the ideal $\frac{5}{2}$ power body. For instance, if the rise time at the ground is only ϵ% of the rise time in the the near field, one might expect that a local excess of order ϵ% in dA/dx on the body would cause a local shock in the far field. Allowance should be made for this sensitivity in the choice of rise time.

The third point that should be mentioned is that the preceding discussion was primarily concerned with the boom in the vertical plane below the aircraft. Actually, of course, the sonic boom covers a zone extending about 30 miles on each side of the ground track. When the pressure disturbance is propagated at an angle to the vertical, the effective scale height is larger and the path length is longer; consequently, the rise time observed at the ground is reduced. Thus, in order to achieve a rise time exceeding some specified lower limit over the whole zone, the length of the thermal spike or thermal fin must be increased over the value required in the vertical plane only.

A lengthening of the spike or fin will entail a still higher rate of fuel consumption. In order to put this matter in proper perspective, however, it should be borne in mind that the fractional increase in takeoff fuel weight required for boom alleviations is less than the fractional increase in the rate of fuel consumption during cruise. This is due to several factors: the subsonic portions of the flight do not require boom alleviation; under conditions of maximum thrust, the fraction devoted to boom alleviation is greatly reduced; and part of the takeoff weight is fuel reserve. The economic impact of devoting a large amount of fuel to boom alleviation is mitigated by the fact that fuel is estimated to be less than 20% of the total operating cost of the SST.[25]

The discussion up to this point has been concerned primarily with the signature observed in the field. During a sonic boom, the noise indoors may actually be greater than that outdoors if windows and doors rattle. Thus, in the presence of rattle, there is probably little or no advantage in a finite rise time. When induced building vibrations are suppressed, however, the indoor waveform is generally similar to the external one except for a substantial reduction in peak pressure and an increase in rise time.[6] Accordingly, one would expect that in a rattle-free, reasonably solid house, the total attenuation would be the sum of that due to the gradual rise time and that due to the reduced peak pressure. It seems likely that neither the time nor the expense required to rattle-proof a typical house would be very great, since the nominal 2 psf is less than the pressure generated by a 30-mph gust.

Finally, there is the question of whether the fuel required for debooming will eliminate or drastically reduce the payload. Here the important thing to note is that overland flights requiring boom alleviation are generally shorter than overseas flights dictating the basic design. Thus, by the use of Breguet's theory, if the weight of ancillary equipment for boom alleviation is neglected, an airplane that is basically designed for flights from New York to Paris (approximately 3200 naut miles) and that uses half its takeoff weight in fuel can fly from New York to Los Angeles (2200 naut miles) with the same cargo, initial fuel, and the like, with a fuel allocation for boom alleviation as high as 45% of that required for propulsion. While the Breguet theory is, of course, a vast oversimplification, the foregoing result suggests that payloads higher than those for transoceanic missions are compatible with a substantial allocation of fuel for boom alleviation in domestic operations.

Conclusions

Present indications are that, to be acceptable to the public, the noise level of a supersonic transport must be reduced below that associated with the nominal 2 psf shock by at least 12 db. There appears to be little prospect of doing this by a reduction of the shock strength without a major change in aircraft weight or operating characteristics. The only approach that appears capable of yielding the desired noise alleviation is to eliminate shocks from the pressure signature and substitute rise times of 10 msec or more.

A properly shaped aircraft several times as long as any SST currently under development could reduce the boom to acceptable levels but is believed to be structurally infeasible. Accordingly to theory, however, an airplane of the length desired can be simulated by the use of heat. The basic concept of using off-axis heating instead of a solid structure to produce the desired pressure disturbance is under investigation in the NASA Langley 4- by 4-ft tunnel.

Rough calculations indicate that the heat needed to prevent formation of the front shock (the only one considered in the present analysis) is unacceptably large if it is derived from the output power of the aircraft. If it is obtained by direct combustion of cheap fuel such as JP-4, however, penalties in increased fuel consumption and reduced payload may not be prohibitive. If this be the case and if the wind-tunnel experiments validate the underlying theory, the utility of the approach will depend primarily on the cost of eliminating the rear shock, the practical engineering aspects of designing an appropriately shaped aircraft, and accomplishing the necessary heat deployment.

A very cursory examination has been made of means for deploying the heat in the desired fashion. Two possible modes for doing this are the thermal spike approach, preferably employing external burning, and the thermal keel approach, preferably employing internal burning.

While it is by no means clear that either approach can be developed into a practical design, the converse is also true, i.e., it has not been shown that they cannot. All that can be said at the present time is that there are reasonable grounds for believing that, if it develops that the advantages of overland supersonic flight outweigh the associated penalties in weight, complexity, operating costs, etc., suitable means will be found for reducing the noise of an SST sonic boom to an acceptable level. While such a prospect is too tenuous to justify delay in the schedule of any SST currently under development, it encourages the hope that it may be possible to cruise supersonically over the United States with second or third generation SST's. Such a prospect should bring about a more favorable attitude toward the development of the SST on the part of the public and also stimulate efforts to overcome this latest barrier in the march toward ever-increasing domestic flight speed.

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