Verification of the Heat-Field Concept for Sonic-Boom Alleviation

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The results of an experimental program carried out in the NASA-Langley 4×4 ft supersonic pressure tunnel to investigate the validity of the heat-field concept for sonic-boom alleviation are reported herein. The basic idea of the heat-field concept is to heat the flow about a supersonic aircraft in such a manner as to obtain an increase in effective aircraft shape (area distribution) that will result in a shock-free pressure signature on the ground. In the program, pressure signatures were measured below a series of wind-tunnel models to provide a step-by-step verification of the concept. For the key model tested, nitrogen is passed through the near of an off-axis slender fin situated below a representative SST configuration having its lift equivalence in volume. Comparisons of pressure signatures for this model with no flow, cold flow, and heated flow through the fin demonstrated substantial effects due to flow and due to heat, essentially verifying the use of mass flow and heat fields for providing equivalent area in supersonic flow. In addition, the ability of an off-axis slender fin to produce a finite-rise-time signature in the presence of lift was demonstrated by testing a wing-body model both with and without a solid fin.

I. Introduction

AN aircraft in supersonic flight generates a complex pressure field that simplifies as it propagates, giving rise to the so-called sonic boom. Generally speaking, the far-field sonic-boom pressure signature may be characterized by four parameters, namely the overpressure p_{max}, the shock pressure rise, p_s, the rise time, τ, and the impulse $_0\int^\lambda p\,dt$ (Fig. 1). For current generation aircraft of the supersonic transport (SST) class, it is estimated that the reflected overpressure will be on the order of 100-200 N/m^2 (2-4 psf) during cruise conditions.

The principal aspects of the sonic boom causing public concern are structural damage and noise. Based on extensive studies of the structural damage question, relatively few structures will be damaged by booms having overpressures below about 250 N/m^2 (5 psf).[1] Thus, it appears that adverse public reaction is associated mainly with the acoustic effects of the boom. A substantial startle effect is produced by a shock having pressure rise of the order of 100 N/m^2 or more, and this gives rise to annoyance.

Our knowledge, however, of the relative importance of the previously noted signature parameters in producing an acceptable sonic-boom signature is incomplete at this time.[2] It is known that replacement of the front and rear shock waves by pressure increments occurring over times as small as 10 msec results in a substantial reduction of acoustic power in the frequencies to which the ear is most sensitive, thereby reducing the outdoor annoyance of the boom.[1,3] Studies to determine the optimal configuration and minimum length for an aircraft having a signature without shock waves (so-called bangless boom)[4] have been undertaken by a number of investigators over the past several years[2,4,5,6] with a closed-form result evolving only recently.[2,4] These results yield a required length of 174 m (570 ft) for an SST-type aircraft with a gross weight of 272,000 kg (600,000 lbm) flying at an altitude of 18,300 m (60,000 ft) at Mach 2.7. Although this length is substantially below earlier estimates of about 305 m (100 ft),[6] it is still beyond that generally considered feasible for practical implementation. Consequently, consideration has been given to the possible use of heat or force fields distributed about the aircraft in such a manner as to create a so-called "phantom body" having the length and area distribution. This article presents the results of an experimental program in which pressure signatures were measured below a series of wind-tunnel models to provide a step-by-step verification of the heat-field concept.

Several investigators have examined heat and force-field concepts and have made estimates of the power requirements and weight penalties associated with their implementation.[1,7-9] Cheng and Goldburg,[7] upon close examination of the use of electro-aerodynamic techniques to reduce the sonic boom as first proposed by Cahn and Andrew,[10] concluded that on the order of 454,000 kg (1,000,000 lbm) of electrical equipment would be required to generate the power needed to reduce the overpressure of the leading shock wave in the sonic-boom signature by 10%. Both Batdorf[1] and Miller and Carlson,[8]

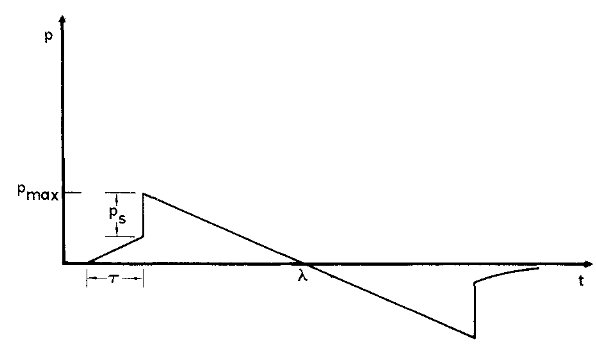

Fig. 1 Generalized sonic-boom pressure signature.

Presented as Paper 74-573 at the AIAA 7th Fluid and Plasma Dynamics Conference, Palo Alto, Calif., June 17-19; submitted June 18, 1974; revision received September 19, 1974. The author would like to acknowledge the significant contributions of his colleagues at The Aerospace Corporation to this program. S. B. Batdorf originated the basic concept that was investigated, i.e., the use of heat (e.g., as provided by direct combustion[1]) in an off-axis configuration such as a thermal fin, W. R. Grabowsky carried out the thermal analysis and preliminary design of the heating system incorporated in the thermal fin, and D. A. Durran transformed the preliminary design of all the models into detailed designs, and designed the power-supply console.

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considering a heat field generated by direct combustion of the same fuel as used by the main SST engines, concluded that the front shock wave could be eliminated through properly controlled burning of an amount of fuel corresponding to about 20% of that normally carried to meet propulsion needs. On consideration of rear shock elimination also, Miller and Carlson concluded that this penalty would increase to about 60% additional onboard fuel with no account taken of the weight and performance penalties of the system. Lipfert[9] predicted the weight penalty associated with alleviation devices in terms of additional structural weight and additional fuel weight for both shock-wave elimination and to overcome the additional drag due to the additional weight. He considered the use of hydrogen (a pollution-free-fuel) as well as hydrocarbon fuels not only for shock elimination but also to supply the propulsive power to overcome the drag penalty. The minimum weight penalty to remove both the front and rear shocks for a 272,000 kg (600,000 lbm) aircraft flying a 5000-second mission at Mach 2.7 at an altitude of 18,900 m (62,000 ft) was estimated to be 227,000 kg (150,000 lbm). These tentative conclusions indicate a questionable practicality of using free combustion to remove both front and rear shocks. However, the use of free combustion in conjunction with other ideas, such as the use of a high tail to obtain, in the area-rule sense, the required additional length needed to remove the rear shock wave,[11] could bring such concepts back into the realm of practicality. In implementation of these concepts, employment of the area rule allows off-axis combustion to be considered.[1,8] This has particular attraction in the case of the front shock, since it replaces the difficult problem of burning against the flow ahead of the aircraft with one of burning in the direction of flow beneath the aircraft.

Questions concerned with the engineering and economic feasibility of practical incorporation of the previously discussed design features in supersonic transport aircraft can be answered only after exhaustive studies by qualified design teams with representatives from all technology areas. However, there are some fundamental questions concerning the validity of the heat-field concept than can be answered by means of properly conducted, but relatively simple, wind-tunnel tests. Such was the purpose of the present investigation. To investigate the feasibility of using an off-axis heat field for removal of the leading shock wave in the sonic-boom signature, an experimental program was carried out in the NASA-Langley 4 x 4 ft supersonic pressure tunnel. Seven wind-tunnel models were tested and pressure signatures recorded at two and three body lengths in the plane directly below the aircraft. The results of these tests are reported herein.

II. Test Facility

The tests were carried out in the NASA-Langley 4 x 4 ft supersonic pressure tunnel at a freestream Mach number of 2.01 and total temperature of 317 K (570°R). All of the tests were performed at a tunnel stagnation pressure of 69 kN/m^2 (10 psia) with the exception of two signatures for the thermal-fin model that were taken at a tunnel stagnation pressure of 51.7 kN/m^2 (7.5 psia). Many tests in which sonic-boom pressure signatures have been measured for both simple and complex configurations have been carried out in this facility, and high-resolution signatures have been obtained.[12-14]

For the tests under consideration, the models were mounted on a remotely controlled actuator permitting their movement relative to a slender static-pressure probe (Fig. 2). Both the model and measuring probe were mounted on a support system that provided for remote control adjustments of longitudinal and lateral position. To obtain a signature, the probe position is held fixed in the tunnel and the model moved forward relative to it. In this manner, the flow disturbances originating at different positions on the body always traverse the same region of flow between the body and probe, thereby having any freestream flow nonuniformities in common. In this fashion, different points in the signature are referenced to a common base.

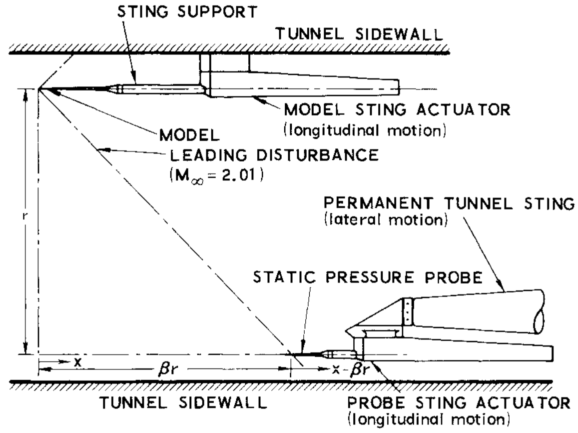

Fig. 2 Wind-tunnel test section apparatus and nomenclature.

III. Test Models, Results, and Discussion

A series of wind-tunnel models was tested in the course of the investigation. These were a basic body of revolution, a 5/2-power area distribution body (so-called frontal spike), the basic body with a solid fin below it designed to have the same Mach-plane area distribution as the frontal spike, the so-called thermal fin model in which approximately 40% of the solid fin in the previous model is replaced by a heat field, a wing-body model, and a wing-body with solid fin model.

A basic body-of-revolution (Fig. 3) having a nominal length of 25.4 cm (10 in.) was used to represent at 1/360 scale a typical SST configuration. Such a body provides a baseline signature containing front and rear shock waves. The basic-body area development was patterned after an SST area distribution taken from Ref. 6 and modified slightly so that the front shock wave could be eliminated by heat addition only (i.e., no heat removal). In this model, the lift is replaced by its equivalence in volume according to the rules of linearized theory.[15] In addition, the nose of the model is modified to have a 5/2-power area distribution such that the front shock wave will not form within a distance of about 25 cm (10 in.) from the body. The reason for this is to allow the fin of the model discussed below to interact with the basic-body disturbances before nonlinear effects become important, thereby satisfying requirements for applicability of the supersonic area rule.

The results for the pressure signature measured below this model at lateral distance r of 50.8 cm (two body lengths) is shown in Fig. 4. In this figure, the nondimensionalized overpressure $\Delta p/p_\infty$ is shown as a function of distance, $x - \beta r$, behind the Mach line through the point of initial model disturbance (Fig. 2). Comparison is shown with theoretical prediction using a computer code developed at NASA-Langley[12] based on the theory of Whitham.[16] Note that, in general, the agreement with theory is good. The magnitude of the front shock overpressure and rear expansion are underpredicted, however, and the positions of the rear expansion and rear shock are somewhat in error. A possible explanation for the latter is the fact that the theory assumes that all disturbances

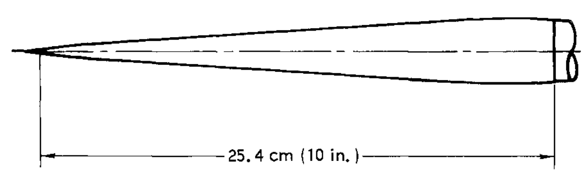

Fig. 3 Basic body of revolution representing typical SST configuration.

originate on the body axis rather than one the body surface, which would cause the theoretical disturbance to be displaced downstream.

The second model tested is a body of revolution whose area grows as the 5/2-power distance from its nose (Fig. 5). This is the minimum-length area distribution for a finite rise-time pressure signature, and is the distribution that will result in a linear pressure rise for the first compression in the signature according to the basic theory of Whitham.[16] This model was designed to represent an airplane equivalent body with a full scale length of 152 m (500 ft), [61 m (200 ft) longer than the airplane itself], and of such a contour that the area difference between the basic body and the 5/2-power frontal spike is constant aft of the nose of the basic body. This would correspond to the case of heat distributed upstream of the basic body in such a fashion as to displace streamlines in a manner identical to the 5/2-power frontal spike. The heating would end at the nose of the basic body.[1]

Figure 6 shows a comparison of theoretical and experimental pressure signatures for the frontal spike model. Note the excellent agreement with theory in this case, the measured data very nearly following the predicted linear pressure rise.

The third model tested incorporates a solid fin situated below the basic body of revolution and is designed so that the area distribution in Mach cutting planes perpendicular to the lateral symmetry plane of the fin is identical to that of the 5/2-power spike (Fig. 7). The purpose of the 5/2-power area distribution nose on the basic body (discussed previously) is to allow disturbances from the fin to interact with those from the basic body before nonlinear effects become important, thereby satisfying requirements for applicability of the supersonic area rule. The fin is designed to have a double-wedge cross section in Mach planes, with a maximum front-wedge angle of four degrees in horizontal planes. It is necessary to keep the front-wedge angle in horizontal planes small in order to remain within the bounds of linearized theory, upon which the supersonic area rule is based.[15] When these conditions are met, the pressure signature below the fin in the lateral symmetry plane of the fin is theoretically identical to that of the 5/2-power spike. This result is nearly obtained in Fig. 8, where the signature at the two-body length lateral distance is shown.

The important question, of course, is whether any local compression regions will cause a shock wave to develop at larger distances. Two such regions are seen in Fig. 8, one at the beginning and one toward the end of the linear pressure-rise region. When the signature at three body lengths in Fig. 9 is considered, however, it is seen that no significant steepening has occurred for the compressive region at the beginning, and the one toward the end of the linear region has nearly disappeared. Even if shock waves developed from these compressions, however, they would account for no more than 10% of the total pressure rise.

Based on the success of the solid fin, a so-called thermal-fin model was developed in which approximately 40% of the fin volume of Model 3 is replaced by a heat field designed to have the equivalent area distribution in Mach planes of the solid fin (Fig. 10). The heat field is generated by passing nitrogen through a 0.3175-cm (1/8-in.) diameter heating tube that continues through the model into a nozzle whose exit plane is at the rear of the fin (Fig. 11). In order to accommodate the 1/8-in. heating tube in the fin section, the scale of the thermal-fin model was increased by a factor of 1.5 over its solid counterpart. Hence, the basic-body portion of the thermal fin model has a nominal length of 38.1 cm (15 in.).

The thermal fin Mach-plane area distribution is shown in Fig. 12. In the Mach plane at the end of the heated region, better than 60% of the fin area is represented by heat. This percentage drops to 43% at 30.163 cm (11.875 in.) and remains constant thereafter. In terms of volume, about 40% of the total fin volumes is represented by the heat field.

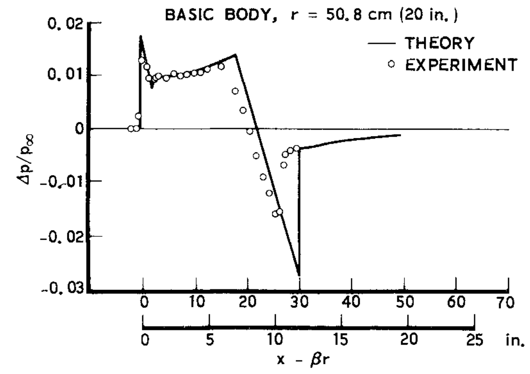

Fig. 4 Comparison of experimental and theoretical pressure distributions.

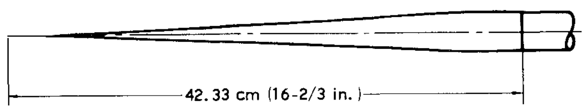

Fig. 5 5/2-power body frontal spike.

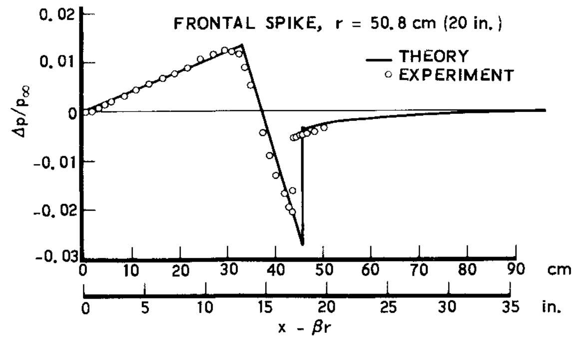

Fig. 6 Comparison of experimental and theoretical pressure distributions.

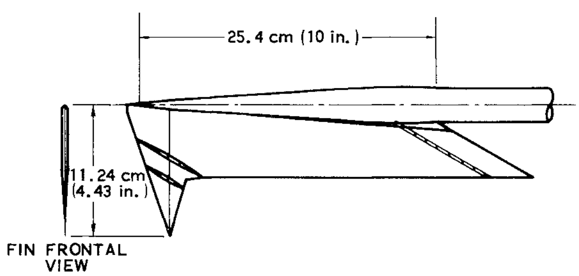

Fig. 7 Solid fin model.

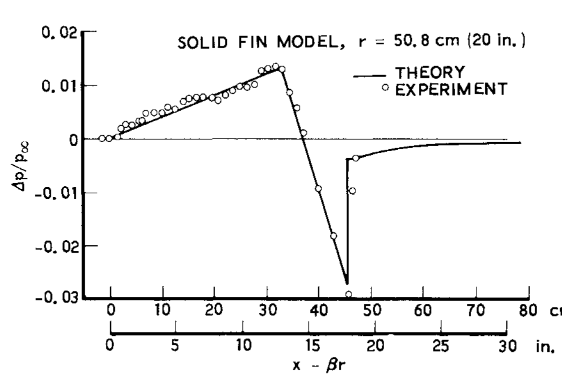

Fig. 8 Comparison of experimental and theoretical pressure distributions.

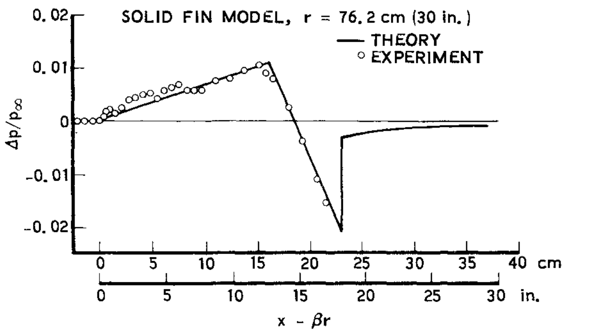

Fig. 9 Comparison of experimental and theoretical pressure distributions.

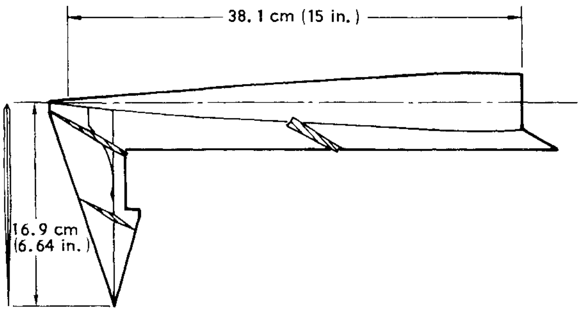

Fig. 10 Thermal fin model.

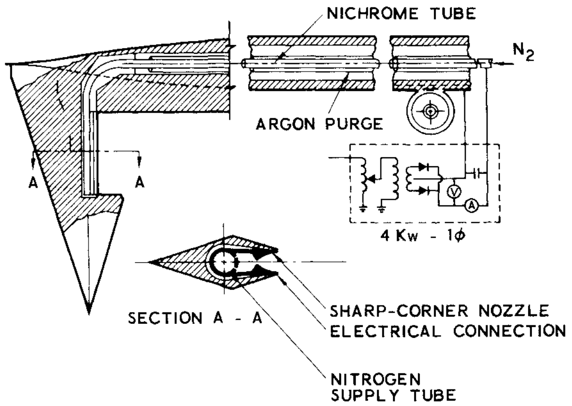

Fig. 11 Thermal-fin model cross section.

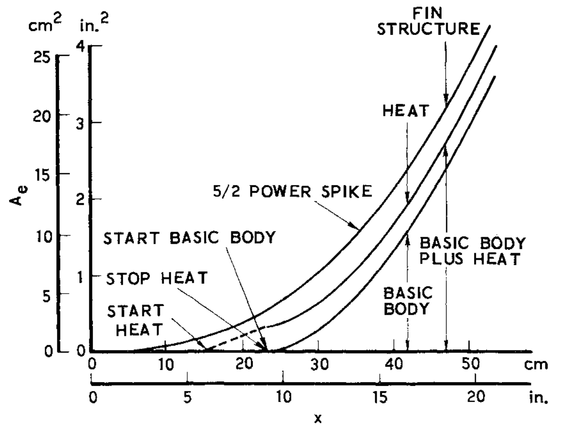

Fig. 12 Thermal-fin Mach-plane area distribution.

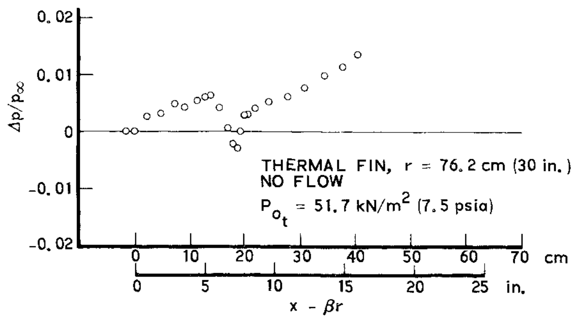

Fig. 13 Experimental pressure distributions.

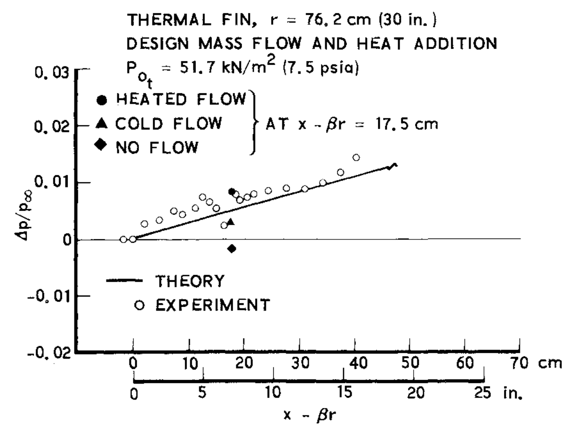

Fig. 14 Comparison of experimental and theoretical pressure distributions.

The resulting pressure signatures for the thermal-fin model are shown in Figs. 13 and 14 for a wind-tunnel total pressure of 51.7 kN/m^2 (7.5 psia) and in Figs. 15 and 16 for a total pressure of 69 kN/m^2 (10 psia). The former is the design-condition setting to obtain the proper ratio of jet-to-freestream stagnation pressures as determined by the exit static-pressure match condition. Figure 13 shows the resulting signature at two body lengths for the case of no flow through the fin nozzle. Note the large expansion and subsequent recompression. This portion of the signature corresponds to disturbances generated by the nozzle region of the fin, expansion being generated by the growing area deficit through the nozzle region, and the recompression caused by this deficit becoming constant at the end of that region. Figure 14 shows the results of passing heated nitrogen through the fin at the design mass-flow rate and temperature. Note the substantial reduction in the nozzle deficit region, although an expansion still exists at the outset. Visual examination of the model after the test indicated a greater nozzle width in the center than at the ends when the flow was turned on, giving a mass (and hence area) deficit in the lower region of the fin. This is the probable cause of the expansion-recompression region noted in the signature. Further, note the two points marked "cold flow" and "no flow." These were obtained by first turning the heater power off and then turning the nozzle flow off. In this manner, the effects of the flow and heat are separated, and it is seen that they both contribute about equally to the overpressure at this point.

Figures 15-17 show the effects of cold and hot flow on the thermal-fin pressure signature for a wind-tunnel total pressure of 69 kN/m^2. Although the nozzle is underexpanded at this pressure and the effective-area contribution is only about three-quarters that of the fully-expanded case, note that the effects on the pressure signature are still substantial.

The final models tested were the wing-body and wing-body with solid fin. The wing-body model consists of a body of revolution and a double delta wing for which the sum of the Mach cutting-plane areas due-to-volume and the equivalent

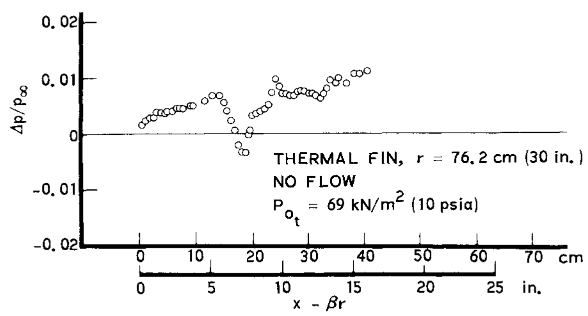

Fig. 15 Experimental pressure distribution.

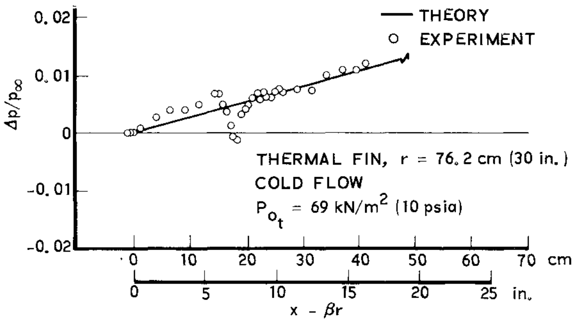

Fig. 16 Comparison of experimental and theoretical pressure distributions.

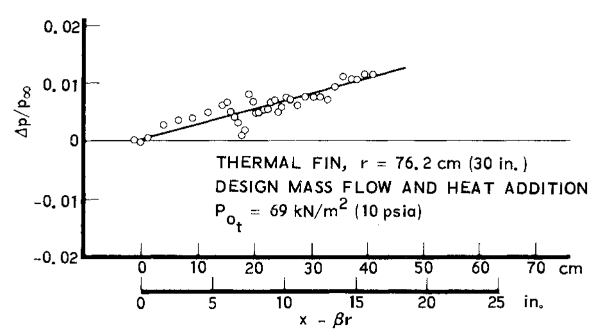

Fig. 17 Comparison of experimental and theoretical pressure distributions.

Mach-plane area due-to-wing lift is equal to the Mach-plane area distribution of the basic body of revolution (Fig. 18). Then, according to linearized theory and the basic sonic-boom signature theory of Whitham, this model should have a pressure signature below it, in the lateral symmetry plane through the fuselage centerline, equal to that of the basic body. The wing is a 67.5° sweep-angle double delta set at 40 min angle of attack. The lift for the wing was calculated using the theory of Polhamus.[17] At the end of the body, the equivalent area due-to-lift represents about 30% of the total effective area. The wing-body with solid fin model is comprised of the wing-body model with the solid fin of the third model attached below it. The reason for testing these models was to assess the effectiveness of the solid fin in producing a finite rise-time signature when lift is present in actuality as opposed to its equivalent-area form as in the basic body of revolution. Figure 19 shows the results for the wing-body alone compared with the theoretical prediction for the basic body of revolution. Recall that the wing-body was designed so that the sum of the areas due to volume of the fuselage and wing plus the equivalent area due-to-lift of the wing is equal to the area of the basic body of revolution in each Mach plane.

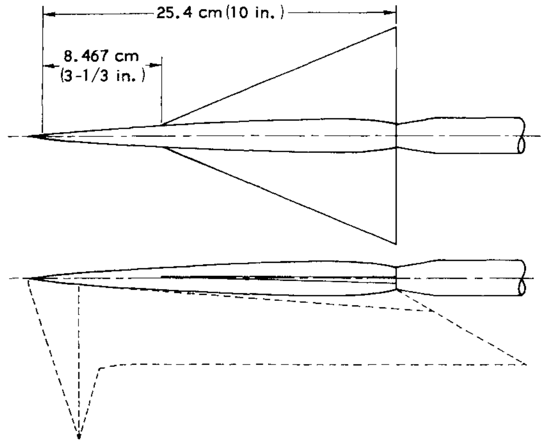

Fig. 18 —Wing-body. — —Wing-body with solid fin.

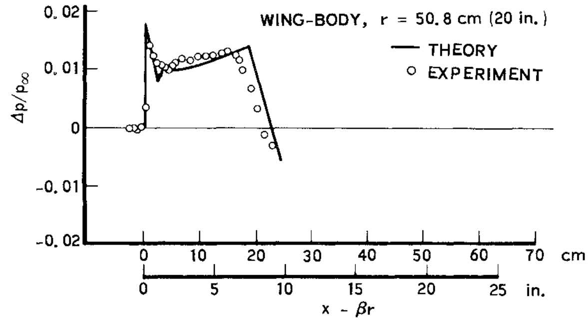

Fig. 19 Comparison of experimental and theoretical pressure distributions.

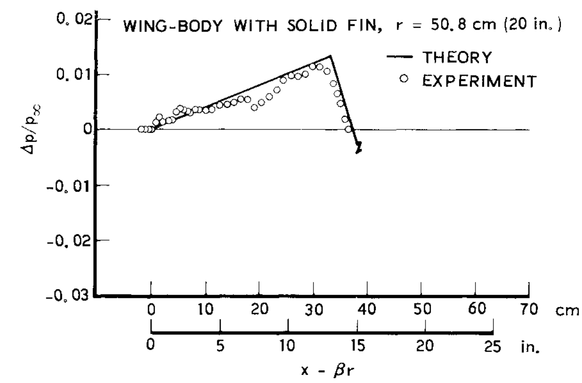

Fig. 20 Comparison of experimental and theoretical pressure distributions.

Hence, the wing-body should have the same pressure signature as the basic body of revolution. The experimental results of Fig. 19 compare well with the theoretical prediction for the basic body and with the experimental results of Fig. 4, thereby verifying the equivalence.

Figure 20 shows the results of adding the solid fin previously tested in conjunction with the basic body of revolution (Fig. 7) to the wing-body model. The resulting pressure signature compares well with the theoretical prediction except in a small region where the nose of the basic body begins to affect the signature. A sudden expansion followed by a gradual recompression is noted in this region. One probable cause of this is a small gap between the fin and basic body in that region that was noted upon detailed examination of the model after the test data were taken.

IV. Conclusions

Based on the results of the wind-tunnel tests reported herein, several conclusions may be drawn. Most importantly, the use of heat fields to provide effective areas in supersonic flow through stream-tube expansion has been verified. In addition, it has been shown that off-axis structures and/or heat fields may be used to produce finite rise-time sonic-boom signatures. Moreover, such structures are effective in producing a finite risetime signature in the ground track in the presence of a lateral lift distribution. In addition, of significance is the utilization of realistic configurations in the tests, thereby verifying the effectiveness of a heat field in eliminating the front shock wave of the sonic-boom signature for a representative SST configuration.

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