JOURNAL OF PROPULSION AND POWER
Vol. 9, No. 4, July–Aug. 1993

Preliminary Assessment of Exhaust Systems for High-Mach (4–6) Fighter Aircraft

A. P. Kuchar* and J. P. Wolff†
GE Aircraft Engines, Cincinnati, Ohio 45215

Exhaust systems designed for high-Mach fighter aircraft will operate at nozzle pressure ratios near 100 for a Mach 4 aircraft to possibly over 600, depending on inlet recovery, for a Mach 6 aircraft. Also, the nozzle pressure loading will be two to four times higher than current Mach 1.5–2.2 exhaust systems, and the exhaust gas temperatures will be significantly higher. These new operating conditions for high Mach nozzles will result in substantial increases in both nozzle size and weight relative to current exhaust systems. Analytical studies have been conducted to assess the internal performance of the conventional axisymmetric nozzle, the two-dimensional, convergent-divergent (2DCD) nozzle, and the single expansion ramp nozzle (SERN) operating at high Mach conditions. Weight estimates have also been completed for the 2DCD nozzle matrix. As expected, results of the studies show that the 2DCD and SERN concepts can achieve desired levels of performance at the expense of size and weight. In general, the exhaust systems for Mach 4–6 aircraft will have to be three to four times as large and four to five times as heavy as those currently employed on Mach 1.5–2.2 aircraft. The axisymmetric nozzle has geometric constraints that limit its performance capability.

Nomenclature

Ai	= local flow area
A_8	= nozzle throat area
A_9	= nozzle exit area
A_{9e}	= SERN external projected area
A_{9i}	= SERN internal projected area
C_A	= angularity coefficient
C_{FG}	= thrust coefficient
C_{FGR}	= resultant thrust coefficient
C_{FGX}	= axial thrust coefficient
C_V	= velocity coefficient
D	= tailpipe diameter
f/a	= nozzle gas stream fuel-air ratio
L	= 2DCD or axisymmetric nozzle secondary flap length
LC	= SERN cowl length
LR	= SERN ramp length
M_0	= Mach number
mi	= local mass flow
P_{T8}	= nozzle gas stream total pressure
$P_{T8}//P_0$	= nozzle pressure ratio
$P1$	= engine inlet pressure
P_0	= ambient pressure
Q	= freestream dynamic pressure
T_{T8}	= nozzle gas stream total temperature
Vi	= local velocity
Vs	= fully expanded isentropic velocity
W	= nozzle width
α	= secondary nozzle half-angle
α_i	= local flow angle
γ	= ratio of specific heats
θ	= SERN cowl angle

Presented as Paper 89-2356 at the AIAA/ASME/ASEE/SAE 25th Joint Propulsion Conference, Monterey, CA, July 10–12, 1989; received July 21, 1991; revision received Dec. 1, 1992; accepted for publication Jan. 25, 1993.

*Manager, Nacelle/Exhaust System Aero Technology, Advanced Technology Operation, Aircraft Engine Engineering Division, 1 Neumann Way, M/D A330. Member AIAA.

†Staff Engineer (retired), Preliminary Exhaust System Design, Advanced Technology Operation, Aircraft Engine Engineering Division, 1 Neumann Way.

Introduction

AIRCRAFT engine technology advancements over the past decade in materials, structures, and analytical methods, along with anticipated advancements for the next decade, have renewed interest in the Mach 3–6 operating regime for both commercial transports and military weapons systems. The type of propulsion system cycle required for these aircraft will depend on the flight regime selected for a particular application. For example, a turbofan engine cycle can be used for Mach numbers less than approximately 3.5, however, a dual-mode turboramjet cycle will probably be required for Mach 4–6 aircraft. Reference 1 provides a brief discussion of potential powerplants for these systems.

The exhaust systems for these high-Mach applications will operate in regimes significantly different from current Mach 2.2 systems. These operating regimes will require advanced technology development in exhaust system aerodynamics, structures, materials, and cooling. Reference 2 discusses some of the new problems encountered in the high-Mach operating regime and the resulting impact on the exhaust system.

Analytical studies of high-Mach exhaust system candidates have been conducted to assess their performance and to identify the key geometric parameters which affect performance. Empirical design criteria (established for Mach 1.5–2.2 operating conditions) extrapolated to high-Mach operating conditions along with computational fluid dynamic (CFD) techniques have been used to conduct these studies. Corresponding preliminary studies have been conducted to assess the weights of one of the nozzle concepts. This article discusses the results of these initial aerodynamic performance and weight estimate studies.

Aerodynamic Studies

Approach

For the straight flap axisymmetric and two-dimensional, convergent-divergent (2DCD) nozzles, current technology aerodesign criteria was extrapolated to anticipated high-Mach operating conditions (nozzle pressure ratio and area ratio), and one-dimensional calculations were conducted to predict nozzle performance for a range of nozzle geometries. Performance assessments for the single expansion ramp nozzle (SERN) concept were made using a CFD Euler code.[3]

Representative high-Mach turboramjet cycle data covering a wide range of operating conditions from sea level static takeoff through transonic accel, and up to Mach 6 at 100,000 ft was used to assess the axisymmetric and 2DCD nozzles. This was done because performance predictions could be made rather quickly and inexpensively using extrapolated design criteria and computerized performance prediction techniques. For the CFD analyses performed on the SERN, only a Mach 5 design point condition was assessed due to the significantly higher cost and longer analysis time associated with CFD techniques.

Two-Dimensional, Convergent-Divergent Nozzle Study

The analysis procedure for the 2DCD and axisymmetric nozzles was conducted in the following manner. For any internal flow nozzle which has symmetry about its centerline such that the thrust vector always acts parallel to and along the nozzle centerline, internal performance can be estimated by first determining the peak thrust coefficient of the nozzle and then calculating the expansion thrust loss. Design curves established for axisymmetric and 2DCD nozzles define the peak thrust coefficient constituents as a function of nozzle geometric parameters. The expansion thrust losses can then be calculated from fundamental thermodynamic expansion process calculations.[4] These expansion losses are dependent on gas stream properties (temperature, pressure, and fuel/air ratio), nozzle pressure ratio, and nozzle area ratio. For the analysis of axisymmetric and 2DCD nozzles at high-Mach conditions, the design curves that define the peak thrust coefficient had to be significantly extrapolated to nozzle geometries representative of high-Mach operation.

For current technology, 2DCD and axisymmetric nozzles characterized by straight flaps, the peak thrust coefficient has been previously determined[5] to be the product of the angularity (C_A) and velocity (C_V) coefficients:

$$C_{FG\,\mathrm{PEAK}} = C_A C_V \tag{1}$$

The angularity coefficient accounts for the nonaxial flow and nonconstant static pressure at the nozzle exit plane. The velocity coefficient accounts for viscous friction losses along the nozzle wall surfaces. Previous analytical studies combined with scale model tests of these nozzles have established and verified characteristic curves of these coefficients as functions of nozzle area ratio and secondary nozzle half-angle as discussed in Ref. 5. These curves, however, were established for ranges of geometries typical of exhaust systems used for applications up to Mach 2.2. These curves were extrapolated to very high area ratios and half-angles typical of those which could be expected for high-Mach operation.

For each of a number of flight conditions ranging from sea level static to high-Mach cruise, the nozzle area ratio for maximum internal performance was determined by trading off angularity vs expansion losses. At a constant nozzle flow condition (P_{T8}/P_0, T_{t8}, f/a) with a fixed length L and throat area (A_8), the three nozzle losses (friction, angularity, and expansion) will vary with area ratio as shown typically in Fig. 1. Since the friction loss is relatively constant, the area ratio at which the maximum internal performance occurs is determined largely by a tradeoff of the angularity and expansion losses. The resulting area ratios for maximum internal performance and the corresponding thrust coefficients will correlate for any given nozzle design vs nozzle pressure ratio. This implies an independent control system for nozzle throat and exit areas. This was done for four 2DCD nozzle lengths to provide a range of achievable performance levels vs nozzle size. The schematic in Fig. 2 shows the engine and nozzle outlines (to scale) for each nozzle length. Three nozzle widths and corresponding sets of length ratios (L/D) are shown. The performance analysis was conducted for the family of nozzles whose width equals the tail pipe diameter. From an aerodynamic standpoint, if the nozzle width is doubled and the flap

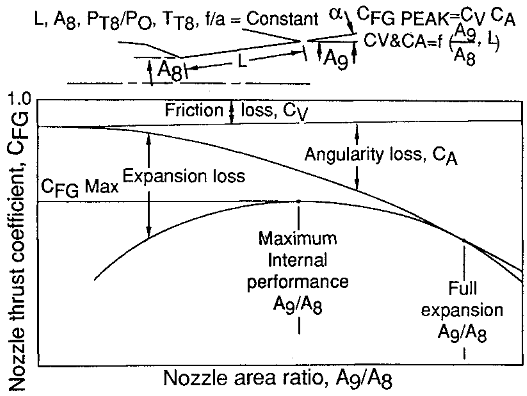

Fig. 1 Internal performance optimization for a converging-diverging nozzle.

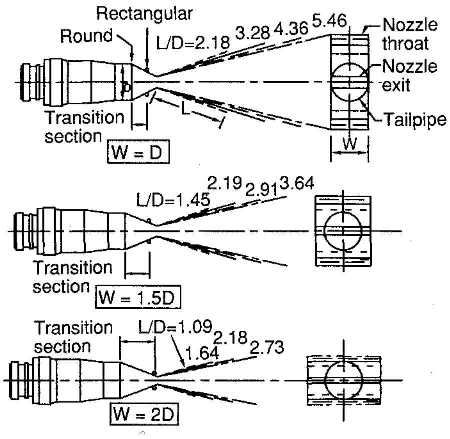

Fig. 2 2DCD nozzles studied.

length is halved to maintain a given throat area, the nozzle geometric cross section is proportionally constant, and the key nozzle aerodynamic parameters (area ratio and secondary flap angle), are therefore, unchanged. Thus, the performance curves derived for the nozzle width equal to the tail pipe diameter can apply to all the sets of nozzle widths and length ratios shown. This assumes, as was done for this study, that aspect ratio, within the range of nozzles studied, has a second-order effect on performance. The reason this point is made is that it became apparent that long 2DCD nozzles (for the width equal to the tail pipe diameter) at high-Mach conditions would have an unusual shape when viewed from the rear (Fig. 2) and would probably not install well in an aircraft. The wider, shorter family of nozzles were thus devised.

Results of the forementioned aerodynamic trade study for the 2DCD nozzle concept are shown in Fig. 3. Included in Fig. 3 are the area ratios at which maximum internal performance occurs in the upper part of the figure and the corresponding performance in the lower part. To achieve nozzle goal performance levels shown in the lower part of Fig. 3 needed for a viable Mach 5 propulsion system, exhaust system length must be in the L/D range of curves 1 and 2. The large size of the exhaust system relative to the engine becomes apparent. From an installed standpoint, nozzle length and width can be traded at a fixed level of internal performance to obtain the best installation in the aircraft. Based on the scaled schematics in Fig. 2, it seems apparent that a nozzle that is as wide as the tail pipe would not install as well as a wider nozzle. It is noted that, because of the angularity vs expansion loss tradeoff shown schematically in Fig. 1 for a given flap length, the area ratios in Fig. 3 are substantially less than what would be needed for full expansion. The full expansion area ratio at Mach 5, for example, for a representative high exhaust gas temperature level is over 20, and the corresponding expansion thrust loss is 1–2% ΔC_{FG} depending on nozzle length.

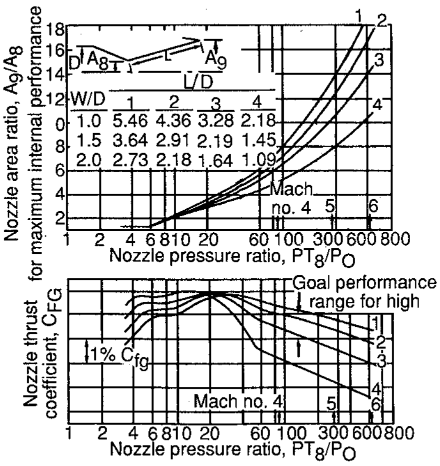

Fig. 3 Relative performance estimates for straight flap, 2DCD nozzles.

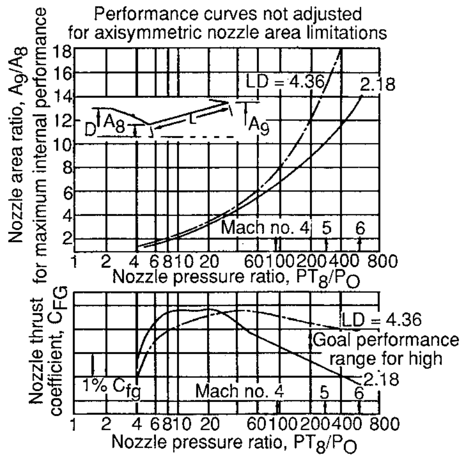

Fig. 5 Relative performance estimates for axisymmetric nozzles.

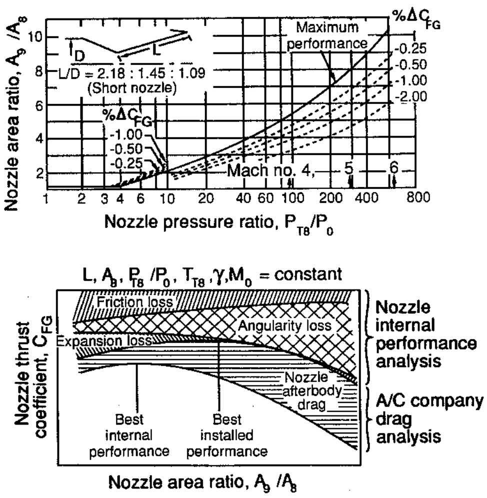

Fig. 4 Installed performance optimization for a 2DCD nozzle.

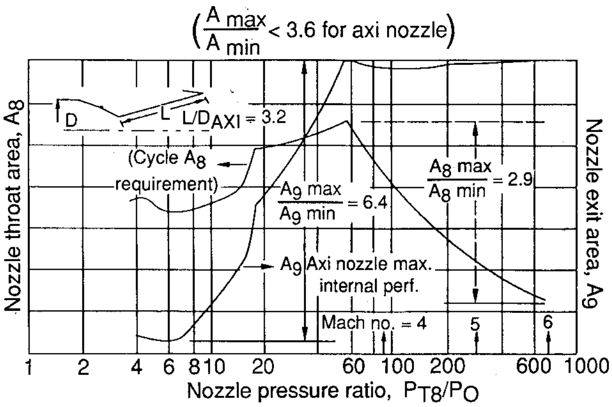

Fig. 6 Nozzle area requirements.

The nozzle performance analysis also included an assessment of performance sensitivity to changes in area ratio. The area ratio schedules shown in Fig. 3 provide the maximum internal performance for a given nozzle length by optimizing the angularity and expansion losses with area ratio as previously discussed (Fig. 1). From an installed standpoint, however, the nozzle exit area, A_9, may be too large at high-Mach conditions to provide an attractive, low-drag installation. Conversely, at low-Mach (subsonic/transonic) operation, the exit area may be too small for low afterbody drag. Thus, on an installed basis, it may be beneficial to deviate from the Fig. 3 area ratio schedules to provide the best installed performance. Typical results from the performance sensitivity study are shown in the upper part of Fig. 4 for the 2DCD short nozzle length (L/D curve 4). For example, at Mach 5, a reduction in area ratio from the maximum performance value of 7.8–6.6 is a 15% A_9 reduction, but results in only a 0.25% reduction in internal performance. If the external drag is reduced a significant amount with this area change, it would be better to operate the nozzle at the lower area ratio. The lower part of Fig. 4 schematically displays this analysis process at a specific design point and exemplifies the aircraft and engine company integration required to assess the exhaust system, select the best installed configuration, and establish or optimize the nozzle area ratio schedule.

Axisymmetric Nozzle Study

Similar aerodynamic studies were performed for axisymmetric nozzles, and results for two flap lengths are shown in Fig. 5. The axisymmetric nozzle performance is at approximately the same levels as the 2DCD nozzle. However, a characteristic of current technology axisymmetric nozzles makes achievement of the area ratio schedule and, therefore, the performance in Fig. 5 at all conditions impossible. Axisymmetric nozzles consist of overlapping flaps and seals, and the variation in physical diameter ratio (either at A_8 or at A_9) from maximum to minimum area cannot exceed a value of approximately 1.9 in practice (2.0 in theory) without mechanical interference of the flaps and seals at low areas or disengagement at high areas. This results in a practical maximum-to-minimum area ratio limit of approximately 3.6 in either A_8 or A_9. As an example, for an intermediate length-to-diameter ratio of 3.2, the maximum-to-minimum A_9 required to satisfy an optimum performance area ratio schedule is about 6.4, as shown in Fig. 6. Although the nozzle throat area variation requirements established by the engine cycle could be achieved, the exit area variation required for maximum internal performance is nearly double the mechanical limit. Thus, significant performance losses relative to the levels in Fig. 5 would occur either at the high Mach end or low Mach end (or both) depending on which would be least detrimental to the aircraft mission, because the area ratio schedule could not be achieved. For this reason and the initial judgment that the axisymmetric nozzle would probably not integrate with or install well in a high-Mach aircraft, it is not considered to be an attractive candidate.

Single Expansion Ramp Nozzle Study

The SERN may be an attractive high-Mach aircraft nozzle candidate because it has an inherent ability to be highly integrated with the aircraft. For example, a significant portion of the nozzle expansion ramp surface can become part of the aircraft surface, thus offering the potential of reduced aircraft/propulsion system overall weight at the expense of perhaps a

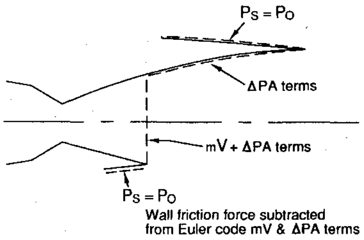

Fig. 7 Control volume for SERN performance analysis.

more complicated engine nozzle/aircraft interface and thrust vector directional control.

Analysis of the SERN concept was performed using a two-dimensional Euler CFD code. The code ouput provides relevant flowfield properties such as Mach number, static pressure and temperature, velocity, flow angle, and flow per unit area. Flowfield results were integrated across the exit plane at the cowl trailing edge and along the ramp surface as shown schematically in Fig. 7 to calculate the inviscid thrust coefficient as follows:

$$C_{FG\text{inviscid}} = \frac{\int_{\substack{\text{nozzle}\\ \text{exit}}} \mathrm{d}miVi \cos \alpha i + \int_{\substack{\text{nozzle}\\ \text{exit}}} (Psi - P_0)\, \mathrm{d}Ai + \int_{\text{ramp}} (Psi - P_0)\, \mathrm{d}Axi}{Vs \sum \mathrm{d}mi} \tag{2}$$

where $Vs = f(P_{T8}/P_0, T_{T8}, \gamma)$.

Wall static pressure distributions from the Euler code results were input into a boundary-layer correlation analysis to assess the viscous wall friction losses, ΔC_{FGV}.

The calculated thrust coefficient from the combined CFD and boundary-layer analysis was thus

$$C_{FG} = C_{FG\,\text{inviscid}} - \Delta C_{FGV}$$

Because of the asymmetry of the SERN, a thrust vector angle will be inherent in its performance characteristics. Thrust vector angle (TVA) for this study was defined and calculated as

$$\text{TVA} \equiv \tan^{-1}(Fy/Fx) \tag{3}$$

where

$$Fy = \int_{\substack{\text{cowl}\\ \text{exit}}} \mathrm{d}miVi \sin \alpha i + \int_{\text{ramp}} (Psi - P_0)\, \mathrm{d}Ayi \tag{4}$$

$$Fx = \int_{\substack{\text{cowl}\\ \text{exit}}} \mathrm{d}miVi \cos \alpha i + \int_{\substack{\text{cowl}\\ \text{exit}}} (Psi - P_0)\, \mathrm{d}Ai + \int_{\text{ramp}} (Psi - P_0)\, \mathrm{d}Axi \tag{5}$$

Both the axial and resultant thrust coefficients as defined below are presented:

$$C_{FGX} = \frac{Fx}{Vs \sum \mathrm{d}mi} \Delta C_{FGV} \tag{6}$$

$$C_{FGR} = \left[C_{FGX}^2 + \left(\frac{Fx}{Vs \sum \mathrm{d}mi} \right)^2 \right]^{1/2} \tag{7}$$

For the SERN study, analyses were limited to a Mach 5 operating condition with a nozzle pressure ratio of 292. Based on the 2DCD nozzle studies, a nozzle width-to-tailpipe diameter ratio of 1.5 appeared attractive and was selected for the SERN analysis.

The SERN has many more flowpath geometric variables to consider than the straight flap axisymmetric and 2DCD nozzles. Some of these variables, shown in Fig. 8, include ramp shape and length (LR), ramp trailing-edge position (A_{9e}), cowl length (LC) and shape, internal exit area (A_{9i}), and cowl angle (θ). This excludes any flow path variables up to the nozzle throat, shown to be symmetric in Fig. 7. For this study, the nozzle up to the throat was assumed to be two-dimensional and symmetric, the cowl shape was a straight line, and the ramp surface and shapes and cowl length were defined concurrently in the following manner. Using the Mach 5 design point pressure ratio of 292, the lower cowl angle (θ) was set at 0 deg, and an isentropic ramp contour was defined using the method of characteristics. A design point ramp length was then established by locating the point on the ramp where the static pressure reached ambient. The ramp was subsequently held in this position, and length variations were achieved by cutting it back. Cowl length, for a given ramp design, was established and held constant by locating the point on the cowl where the last characteristic of the first reflection of series of characteristics impinged on the cowl. This is shown schematically in Fig. 9. The cowl angle was changed (i.e., 5 deg) and the process was repeated to define a different ramp shape/orientation.

SERNs with fixed ramps tend to exhibit a characteristic of providing high performance at the design point consistent with the ramp definition, and lower performance at other "off-design" conditions (lower pressure ratios). For this study, it was perceived that, if the isentropic ramp were defined for a Mach 5 pressure ratio, it may have very low performance at much lower Mach numbers (i.e., subsonic and transonic operation). Thus, a family of isentropic ramp shapes and design point lengths for different cowl angles were defined for nozzle pressure ratios consistent with Mach 4, 4.5, and 5. Figure 10 defines the geometric parameters for the matrix of SERNs evaluated and presents the nozzle cross sections to scale.

Results for the Mach 4.5 design point family of nozzles analyzed at the Mach 5 nozzle pressure ratio are shown in Figs. 11, 12, and 13 vs ramp length. Several trends and characteristics are evident from these plots. The TVA curves in Fig. 11 are nearly identical in shape. If the curves were displaced vertically by an amount equal to the cowl angle, they would all lie on one curve within plus or minus one-quarter of a degree. Thus, for a given design point pressure ratio, the ramp shapes defined using the method of characteristics for each of the cowl angles were nearly identical even though the

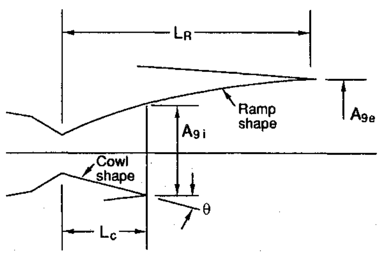

Fig. 8 SERN geometric variables.

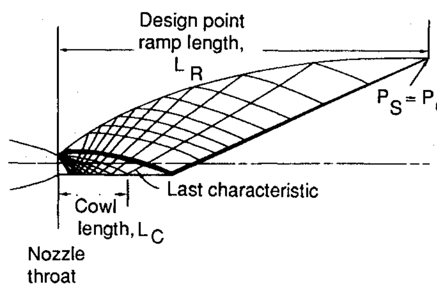

Fig. 9 SERN design approach using method of characteristics.

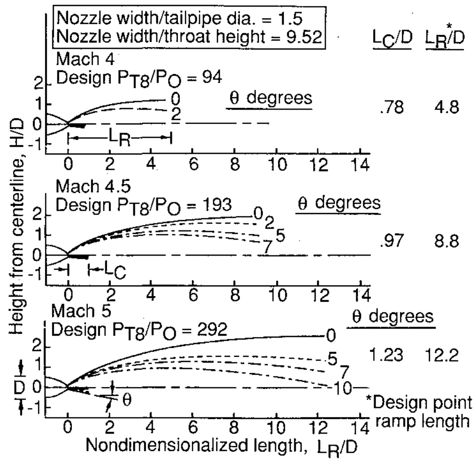

Fig. 10 SERNs studied.

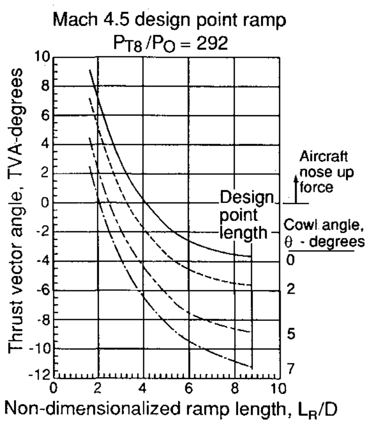

Fig. 11 Cowl angle and ramp length effects on thrust vector angle.

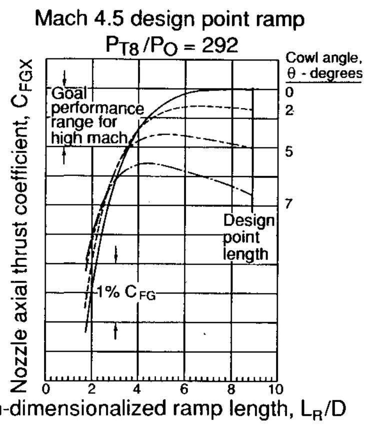

Fig. 12 Cowl angle and ramp length effects on axial thrust coefficient.

nozzle throat orientation relative to the cowl changed with each cowl angle. The implication from these results is that, for a preliminary study, the ramp contour need only be defined once for a selected cowl angle (i.e., 0 deg), and the thrust vector angle characteristic can be defined/varied by simply rotating the ramp and cowl through a selected angle and then translating the TVA curve by the same amount. The thrust coefficient characteristics in Figs. 12 and 13 were consistent with the TVA trend. The fact that the resultant coefficients all lie on one curve within plus or minus 0.1% indicate the basic efficiency of the four nozzles are essentially the same at any given length. Thus, with just one isentropic ramp definition and the corresponding TVA and resultant thrust coefficient curve, axial thrust coefficient can be traded off against TVA for a given ramp length, assuming that it is probably desirable to achieve a specific TVA for the aircraft at a super-

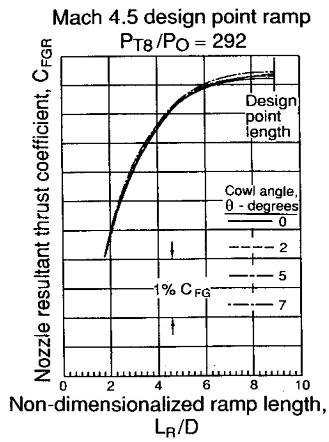

Fig. 13 Cowl angle and ramp length effects on resultant thrust coefficient.

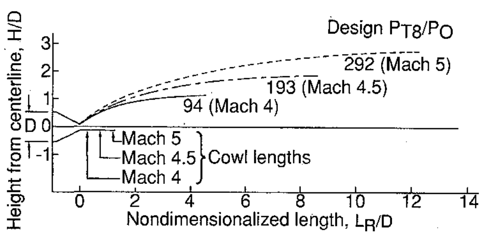

Fig. 14 Design point ramp length comparisons at 0-deg cowl angle.

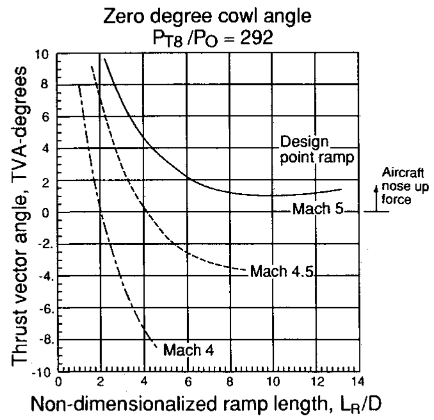

Fig. 15 Design point ramp and ramp length effects on thrust vector angle.

sonic cruise design point. Figure 13 also displays an obvious tradeoff of ramp length vs performance. For example, the ramp could be cut back to an LR/D value of about 6 to reduce weight with no significant performance penalty. Results for each of the Mach 4 and 5 design point family of nozzles were similar; the TVA curves were very close in shape, and the resultant thrust coefficient curves were similar although not quite as close (within plus or minus one quarter percent) as the Mach 4.5 nozzle results.

Of significant interest would be to compare the performance characteristics for each design point ramp shape at one cowl angle. The ramps for each of the three design point Mach numbers are shown in Fig. 14 for a 0-deg cowl angle, and performance results are shown in Figs. 15–17. The TVA trends are as expected, considering the relative orientation of the ramps shown in Fig. 14. Contrary to Fig. 11, however, the TVA curve shapes are not the same. A somewhat unexpected result was that the resultant thrust coefficients were nearly the same at a given length, particularly as the thrust coefficient approached within a percent of the maximum. It was anticipated that the larger internal area ratio (A_{9i}/A_8) for the Mach

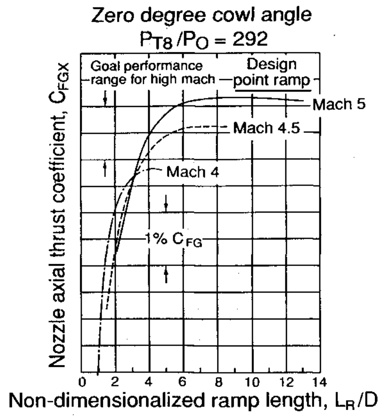

Fig. 16 Design point ramp and ramp length effects on axial thrust coefficent.

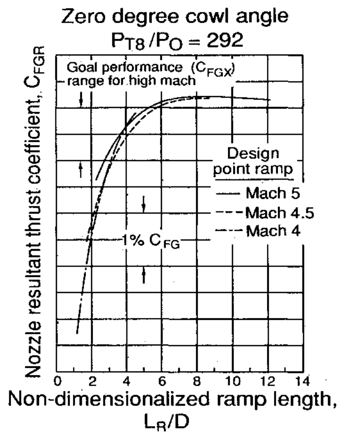

Fig. 17 Design point ramp and ramp length effects on resultant thrust coefficient.

5 design point ramp would be more efficient than the other nozzles, but no clear trend was evident.

Several conclusions were reached from these comparisons. First, extending the ramp beyond an LR/D greater than approximately 6 would result in little internal performance improvement. And second, the ramp design point pressure ratio at which the shape is defined is not a significant parameter which affects resultant thrust coefficient efficiency at a typical Mach 5 nozzle pressure ratio. Thus, ramp length and TVA are the key parameters which can be traded off to establish an axial thrust coefficient level at a high Mach design point.

Cowl length effects were also analyzed for the Mach 5 design point ramp at a 0-deg cowl angle. The cowl length was increased and decreased by one-third. Results indicated that a negligible performance gain is achieved (less than 0.1%) by a one-third incease in cowl length, suggesting that the cowl length selection criteria discussed earlier provided near maximum performance. A weight vs performance tradeoff could be made, however, by reducing the cowl length.

The SERN analyses and results were for one high-Mach operating condition (Mach 5) only. Because of the inherent TVA variation of SERNs with geometry as well as nozzle pressure ratio, it would be required to assess the performance and TVA for this nozzle concept over the complete operating range including subsonic and transonic conditions to eventually select a ramp length and orientation. These off-design conditions could bring ramp shape (design point pressure ratio) back into the picture as a design variable. While it has the potential advantage of being highly integrated with the aircraft, the inherent vector angles associated with the concept would make it more difficult to install considering the entire operating flight envelope.

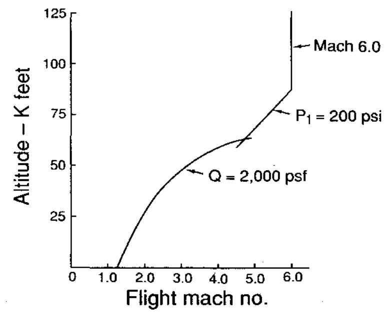

Fig. 18 Flight envelope limits for exhaust system weight study.

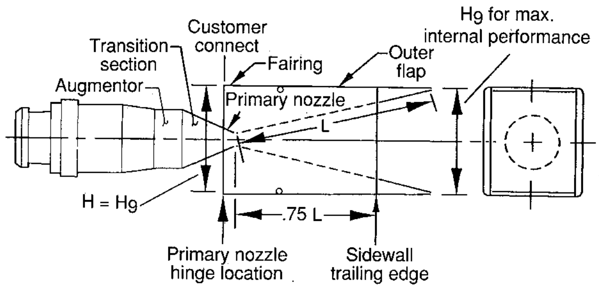

Fig. 19 Geometry assumptions for 2DCD exhaust system weight study.

Weight Studies

Approach

Weight studies were performed for the family of straight flap, 2DCD nozzles consistent with the aerodynamic study. Nozzle operating conditions were based on the same variable cycle engine used for the aerodynamic studies. A factor that strongly affects the exhaust system weight is the right side of the high-Mach flight envelope. This study was based on a right-side envelope limit shown in Fig. 18 with a maximum dynamic pressure (Q) of 2000 psf or a maximum engine inlet pressure ($P1$) of 200 psi, whichever yielded the lowest $P1$. In the ramjet mode of operation, the differential pressure which establishes loading across the nozzle structural components is essentially established by $P1$ since the ambient pressure is very low and the bypass duct and afterburner pressure losses are relatively small. For a Mach 4 design point, this flight envelope limit yielded a maximum nozzle total pressure of about 110 psi, and for the Mach 5 and 6 conditions, the maximum total pressure was 200 psi. Limiting the flight envelope to higher $P1$ and Q values would increase the nozzle weight.

The materials assumed for this weight study were advanced (year 2000) high-temperature, high strength-to-density materials such as carbon/carbon, intermetallic matrix composites, and advanced titanium aluminides. The exhaust system weights presented include the augmentor components such as flameholders and spraybars in addition to the exhaust transition duct and liner, primary and secondary flaps, sidewalls, external flaps and fairing, and nozzle actuators and associated piping. The sidewall was assumed to be three-quarters of the length of the secondary flap. The forward edge of the external fairing (aircraft customer connect) was assumed to begin at an axial station slightly forward of the primary nozzle hinge station. Correspondingly, the fairing height at this point was assumed to be equal to the A_9 height required for maximum internal performance for any given secondary flap length (L/D) and design point Mach number. These geometric assumptions are shown in Fig. 19.

Results

Typical results of the weight study are shown in Fig. 20 for an exhaust system width ratio (W/D) equal to 1.5. The weights are expressed as the fraction (exhaust system weight)/(exhaust

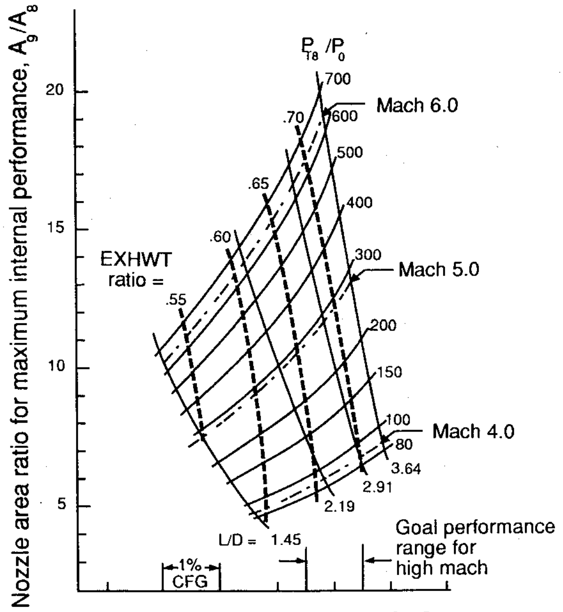

Fig. 20 2DCD exhaust system weight estimates for W/D = 1.5.

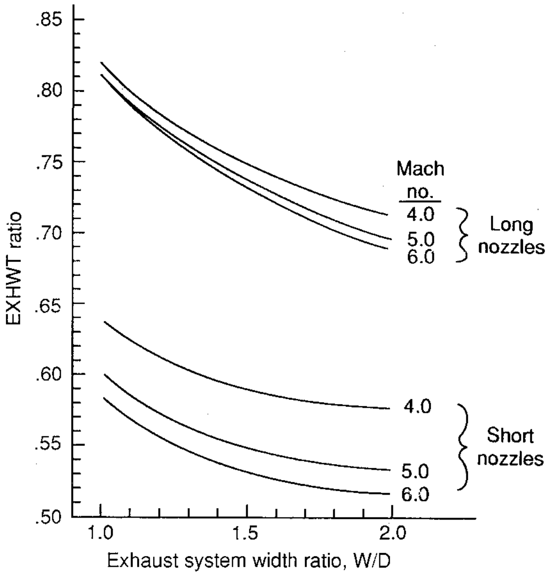

Fig. 21 2DCD exhaust system width effects on weight trends.

system weight plus engine weight) which is called EXHWT ratio. Consistent with the geometric assumptions previously discussed to estimate the weight, the independent parameters for a given W/D are the design point pressure ratio (or Mach number) and the nozzle length ratio, L/D. For example, at a Mach 5 design point with an L/D of 2.91, the EXHWT ratio is slightly less than 0.69. Thus, over two-thirds of the total engine and exhaust system weight is attributed to the exhaust system. Put another way, the exhaust system is 1.5 times as heavy as the engine. As would be expected, the lines of constant EXHWT ratio run somewhat parallel to the lines of constant L/D. Thus, for a given design point pressure ratio, the exhaust system weight is roughly proportional to L/D. The carpet plots shapes were defined using the area ratio and thrust coefficient for maximum internal performance (consistent with Fig. 3) shown on the vertical and horizontal scales. To achieve the goal range of internal performance, the EXHWT ratio would have to be anywhere from about 0.65 to 0.75 for this W/D depending on the design point Mach number and the nozzle length (actual level of thrust coefficient). This is substantially higher than current axisymmetric exhaust systems designed for Mach 1.5–2.2 applications which have EXHWT ratios of approximately 0.15.

By cross-plotting these results, including results for W/D values of 1.0 and 2.0, Fig. 21 shows the effect of nozzle width-on-weight ratio for the long and short family of nozzles. Increasing the width ratio from 1.0 to near 1.5 results in a significant decrease in the exhaust system weight. Further widening yields a more moderate weight decrease. This trend is in a favorable direction when considering the installation configuration (Fig. 2). The weight reduction with increasing width is primarily due to the rate of sidewall weight decrease being more than the rate of transition duct weight increase. It is expected that further increases in width would eventually increase the exhaust system weight.

Weight/Performance Tradeoffs

Results from the 2DCD nozzle aerodynamic and weight studies were combined to investigate the weight/performance tradeoff obtained by designing to nozzle area ratios that were less than the maximum internal performance values. Figure 22 shows a cross-plot of the ΔC_{FG} vs A_9/A_8 shown in Fig. 4 for the short nozzles at the Mach 4 nozzle pressure ratio. Similarly, Fig. 23 shows the EXWHT ratio vs A_9/A_8 for the same family of short nozzles. These two figures were com-

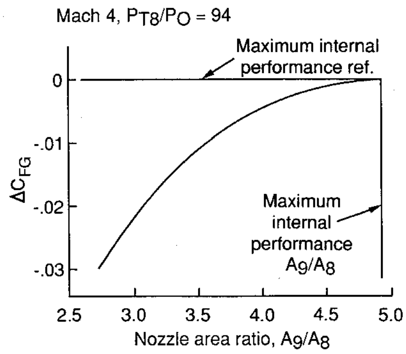

Fig. 22 2DCD thrust coefficient sensitivity to area ratio for the Mach 4 design point short nozzle.

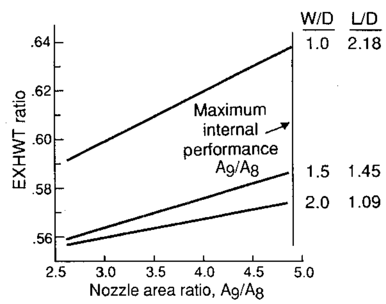

Fig. 23 2DCD exhaust system weight sensitivity to area ratio for the Mach 4 design point short nozzle.

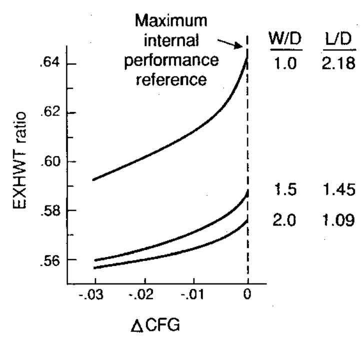

Fig. 24 2DCD exhaust system weight vs performance tradeoff for the Mach 4 design point short nozzle.

bined to produce curves of EXWHT ratio vs ΔC_{FG} in Fig. 24. The weight ratio increases rapidly as ΔC_{FG} approaches zero reflecting the point discussed earlier that A_9 can be reduced significantly from the maximum internal high-Mach performance value with little change in thrust coefficient. These curves again indicate that the wider flow paths yield the lowest weight nozzles and that the designs should probably be sized for a maximum area ratio below that required to meet maximum internal high-Mach performance. How much below will depend on the specific aircraft sizing studies which relate the change in EXHWT ratio vs the change in both ΔC_{FG} and external drag.

Conclusions

The major conclusions derived as a result of the preliminary studies discussed in this article include the following:

1) Exhaust systems for high-Mach (4–6) fighter aircraft will have to be substantially larger and heavier, as expected, than current systems to achieve performance levels needed to make a viable propulsion system. At a Mach 5 condition, the 2DCD nozzle may be three to four times as large and four to five times as heavy (percent of total weight) as current systems. Weight is essentially proportional to nozzle length as expected, and a width-to-tailpipe diameter ratio between 1.5–2.0 appears to be desirable from a standpoint of both minimum weight and installation in an aircraft. Although no weight estimates were made for the SERN, it is concluded that this concept should be lighter than the 2DCD nozzle on a total aircraft/propulsion system basis because of its inherent ability to be more easily integrated with the aircraft.

2) Axisymmetric nozzles do not appear to be viable candidates for Mach 4–6 applications because of inherent maximum to minimum area limitations and corresponding performance degradations.

3) The straight flap, 2DCD nozzle can achieve the desired level of high-Mach performance. Nozzle length for a given width is the key parameter which establishes performance capability, but both width and length are the key drivers which affect weight and the installation in the aircraft.

4) The SERN can also achieve the desired level of high-Mach performance. From these studies, ramp length and orientation appear to be the major parameters which affect performance level and thrust vector angle. Design point pressure ratio for ramp shape definition appears to have little effect on performance.

These studies provided a better understanding of the size and weight that will be required for several exhaust system concepts to achieve desired levels of internal performance. High-Mach operation presents new and challenging operating conditions which cause these significant exhaust system changes, and the study presented herein barely scratches the surface of this new technology arena. New, highly integrated exhaust system concepts need inventing to reduce the overall aircraft/propulsion system weight. Ejector exhaust systems capable of handling large quantities of secondary air will likely be required to offset the large inlet/engine airflow mismatch at transonic conditions. The technology data base for these and other challenges is very limited and identifies a need for significant future analytical and experimental development.

References

[1]Ward, B. P., and Hewitt, F. A., "High Speed Airbreathing Propulsion," AIAA Paper 88-3069.

[2]Dusa, D. J., "High Mach Propulsion System Installation and Exhaust System Design Considerations," AIAA Paper 87-2941, Sept. 1987.

[3]Rowe, R. K., Collopy, G. B., and Rout, R. K., "Flowfield Performance Prediction of a 2D-CD Vectored Nozzle Using Self Adaptive Grid Solution," AIAA Paper 91-2367, July 1991.

[4]Oates, G. C., "Aircraft Propulsion Systems Technology and Design," AIAA Education Series, AIAA, Washington, DC, 1989, Chap. 5.

[5]Speir, D. W., and Blozy, J. T., "Development of Exhaust Nozzle Internal Performance Prediction Techniques for Advanced Aircraft," AIAA Paper 81-1490, July 1981.