NASA CR-135157
PWA-5489

ANALYTICAL SCREENING OF LOW EMISSIONS, HIGH PERFORMANCE DUCT BURNERS FOR SUPERSONIC CRUISE AIRCRAFT ENGINES

FINAL REPORT

by

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March 1977

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA Lewis Research Center
NAS3-19781

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-135157		
4. Title and Subtitle Analytical Screening of Low Emissions, High Performance Duct Burners for Supersonic Cruise Aircraft Engines		5. Report Date March 1977 6. Performing Organization Code
7. Author(s) R. P. Lohmann and G. T. Riecke		8. Performing Organization Report No. PWA-5489 10. Work Unit No.
9. Performing Organization Name and Address Commercial Products Division Pratt & Whitney Aircraft Group United Technologies Corporation East Hartford, Connecticut 06108		11. Contract or Grant No. NAS3-19781 13. Type of Report and Period Covered Contractor Report
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

Project Manager, Donald Schultz, Combustion and Pollution Research Branch
NASA Lewis Research Center, Cleveland, Ohio 44135

16. Abstract

An analytical screening study was conducted to identify duct burner concepts capable of providing low emissions and high performance in advanced supersonic engines. The study involved the definition of duct-burner configurations ranging from current augmenter technology to advanced concepts such as premix-prevaporized burners. Aerothermal and mechanical design studies provided the basis for screening these configurations using the criteria of emissions, performance, engine compatability, cost, weight and relative risk. The results indicated that technology levels derived from recently defined experimental low emissions main burners are required to achieve both low emissions and high performance goals. A configuration based on the Vorbix (Vortex burning and mixing) combustor concept was analytically determined to meet the performance goals and is consistent with the fan duct envelope of a Variable Cycle Engine. It has the potential for low emissions levels that are substantially below those of more conventional augmentor concepts, although they do not fully satisfy the goals set by NASA for this study. This duct burner configuration has a moderate risk level compatable with the schedule of anticipated experimental programs. More advanced concepts, employing premix-prevaporized combustor technology have been defined that may provide lower NO_x emissions, particularly at supersonic cruise, but have higher risk and are therefore not compatible with the projected schedule of follow-on experimental programs.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Duct Burner Augmentor Supersonic Engines Emissions Duct Burner Performance Variable Cycle Engines	

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
UNCLASSIFIED	UNCLASSIFIED	93	

* For sale by the National Technical Information Service, Springfield, Virginia 22151

NASA-C-168 (Rev. 6-71)

FOREWORD

This report summarizes a contracted study of duct burners for advanced supersonic propulsion systems conducted for NASA by Pratt & Whitney Aircraft. This study was conducted in the period from April 1976 to December 1976 under Contract NAS3-19781.

The NASA project manager for this study contract was Donald F. Schultz, Combustion and Pollution Research Branch, Lewis Research Center, Cleveland, Ohio. Key P&WA personnel were Dr. Robert P. Lohmann, Study Program Manager, and George T. Riecke who directed the study efforts.

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SUMMARY

This report presents the results of an analytical screening study of low emissions, high performance duct burner concepts that are applicable to advanced supersonic engines. Duct burners provide thrust augmentation at critical operating conditions for Variable Cycle Engines (VCE) being evaluated for advanced supersonic aircraft.

To begin the study, several pilot and high power stage concepts were defined, ranging from current technology combustion systems to advanced concepts such as premixed-prevaporized configurations. These concepts were screened on the basis of their potential emissions and performance characteristics and the best pilot and high power stage concepts were combined to provide eight promising duct burner configurations. Further analysis of these configurations involved aerothermal sizing, engine and aircraft performance estimates, and refined emissions projections. This provided definition to evaluate these concepts with respect to the following criteria: emissions, chemical efficiency, thrust efficiency, compatibility with a selected VCE configuration, effect on engine cost, weight and dimensions, and relative risk. Based on these criteria, four concepts were selected for further evaluation. The final phase of this study program involved preliminary design studies to identify potential mechanical, durability and operational problems for each selected configuration.

Based on the results of these analytical screening studies, the duct burner concepts evaluated may be grouped into three categories as follow:

- The evaluation of a piloted radial V-gutter configuration, which is based on current augmentor technology, indicated this concept was severely deficient in both chemical combustion and thrust efficiency at the critical supersonic cruise operating condition. The NO_x emissions of this configuration were low, satisfying the program goal at sea level takeoff at which the chemical combustion efficiency was also projected to be adequate. However, the thrust efficiency decrement at supersonic cruise is unacceptable because it leads to increases in aircraft takeoff gross weight of more than 6 percent.

- Staged combustor concepts, based on the technology evolved from experimental low emissions main burner programs, meet the duct burner performance goals. One of these, a three-stage duct burner, employing the Vorbix (Vortex Burning and Mixing) combustor principle was identified by analysis as the best configuration in this category. This duct burner concept has the potential to meet the overall engine requirements including pressure loss, thrust efficiency and ignition margin, and is compatable with the VCE geometry and dimensions. While this analytical study indicates this duct burner does not meet the program emissions goal specified by NASA, it is projected to provide emissions levels substantially below those of configurations based on current technology. These projected levels are consistent with those of main burners employing this technology. The Vorbix concept has undergone extensive evaluation and optimization during the NASA/P&WA Experimental Clean Combustor Program for application to the main burner systems of advanced engines. Consequently it represents only a moderate level of risk that is compatable with the projected schedule of the follow-on experimental programs.

- Advanced technology combustors based on premixed-prevaporized concepts may be capable of meeting the performance and size requirements while also having the potential for lower NO_x emissions than the moderate risk concepts. However, achieving the projected emissions and performance levels is dependent on accomplishing the levels of combustion homogeneity associated with idealized premix combustion systems. Extensive research and development programs followed by substantiation testing of the duct burner would be necessary with these systems. They are considered to be high risk concepts and are therefore not compatible with the schedule of the testbed program which requires a duct burner for evaluation of the coannular noise effect.

The variable geometry concepts, because of their severe effects on engine dimensions, are not considered suitable for duct burner systems.

1.0 INTRODUCTION

Pratt & Whitney Aircraft (P&WA) has been conducting advanced supersonic propulsion studies under the National Aeronautics and Space Administration (NASA) sponsored Supersonic Cruise Aircraft Research (SCAR) program. These studies have identified the Variable Stream Control Engine (VSCE) as a promising Variable Cycle Engine (VCE) concept for an advanced supersonic commercial transport aircraft. This engine shows the potential for significant environmental and economic improvements relative to first-generation supersonic turbojet engines. Reference 1 contains a detailed description of the Variable Stream Control Engine.

1.1 THE VARIABLE STREAM CONTROL ENGINE

The Variable Stream Control Engine (VSCE) is an advanced technology version of a duct burning turbofan engine that employs variable geometry components and a unique throttle schedule for independent control of two coannular flow streams to provide reduced jet noise at takeoff and high performance at cruise. Figure 1-1 shows the basic arrangement of the major components of the VSCE. The low spool of this twin spool configuration consists of an advanced technology, multi-stage, variable geometry fan and low pressure turbine. The high spool consists of a variable geometry compressor driven by an advanced single-stage high temperature turbine. The primary burner and duct burner require low emissions, high efficiency combustor concepts. The nozzle is a two stream, coannular design with variable throat areas in both streams and an ejector/reverser exhaust system.

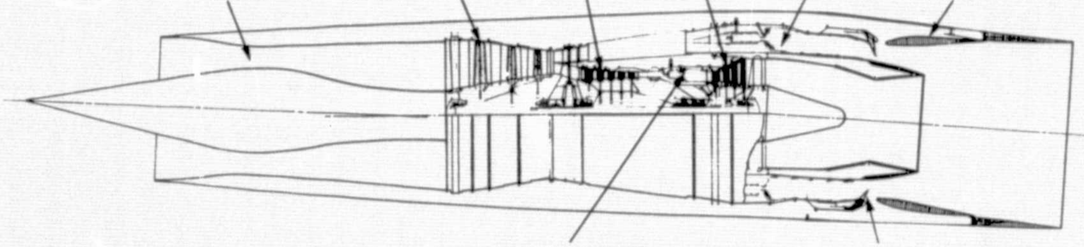


Figure 1-1 Variable Stream Control Engine Concept

The capability to independently control the temperature and velocity in both primary (high spool) and fan streams through the use of a duct burner and the variable area coannular exhaust nozzle provides an inherent reduction in jet noise during takeoff. This noise reduction characteristic is based on an inverse velocity profile, where the fan stream jet velocity is 60 to 70 percent higher than the primary stream velocity. Results from a P&WA model nozzle test program sponsored by NASA indicate that noise levels measured for coannular nozzles with this inverted velocity profile are approximately 8 EPNdB (effective perceived noise level in dB) lower than a single-stream nozzle operating at the same airflow and thrust levels (Ref.

2, 3 and 4). These results are based on both static tests and wind-tunnel tests simulating take-off flight conditions. Based on these model tests, the coannular noise benefit represents a breakthrough in jet noise control.

Figure 1-2a depicts the engine operating mode used to generate this unique inverted velocity profile for takeoff operation. As indicated, the primary stream is throttled to an intermediate power setting so that the jet noise associated with the primary stream is low. To provide both the required takeoff thrust, and the inverse velocity profile, the duct burner is operated at its maximum exit temperature of approximately 1700K (2600°F). This condition sets the cooling requirements for the duct burner and nozzle system. Relative to military engine augmentor systems, which approach stoichiometric combustion, the peak duct burner temperatures for the VSCE are relatively low, and will not compromise the life capability of this commercial engine.

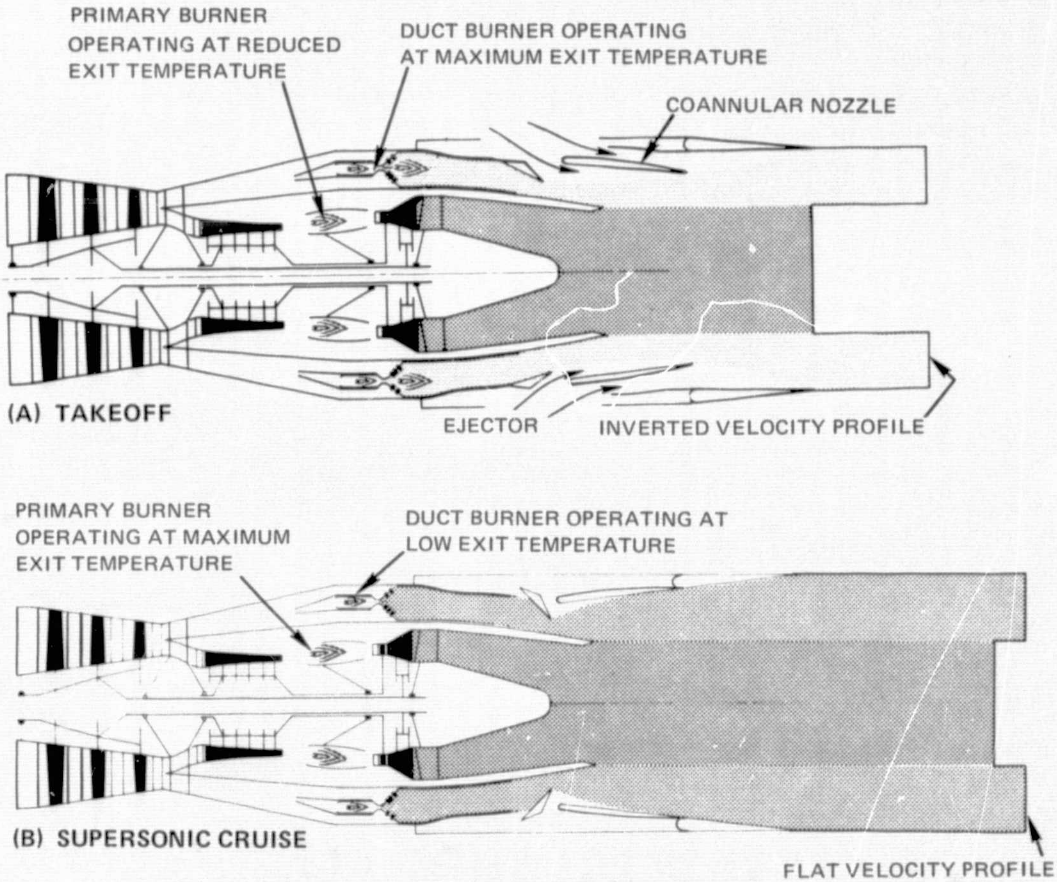


Figure 1-2 Operating Modes of the Variable Stream Control Engine

For supersonic operation, the main or primary burner exit temperature is increased (relative to takeoff), and the high spool speed and flow rate are matched to the higher primary burner exit temperature. This matching technique is referred to as the inverse throttle schedule –

inverse relative to conventional subsonic engines which cruise at much lower temperatures and spool speeds than they require for takeoff conditions. This feature enables matching the high spool to a higher flow rate at supersonic conditions relative to a conventional turbofan. In effect, this high-flow condition reduces the cycle bypass ratio. The level of thrust augmentation required in the duct-burner during supersonic operation can therefore be reduced. At this condition, the exhaust temperature from the coannular streams are almost equal and, as shown in Figure 1-2b, the velocity profile is flat, to provide optimum propulsive efficiency. As a result, the cruise fuel consumption characteristics of the VSCE approach those of a turbojet cycle designed exclusively for supersonic operation. The inverse throttle schedule feature enables sizing the VSCE propulsion system for optimum supersonic cruise performance, while also meeting FAR Part 36 noise levels at the other end of the operating spectrum, by means of the coannular noise benefit.

1.2 RELATION OF DUCT BURNER TO THE OVERALL VARIABLE CYCLE ENGINE PROGRAM

The coannular nozzle as well as the duct burner have been identified as critical technologies for the Variable Stream Control Engine. A Variable Cycle Engine (VCE) testbed program, using the F100 engine, is planned to demonstrate these two technologies in an engine that simulates the VCE conditions and environment. The arrangement of this testbed engine is illustrated in Figure 1-3. The F100 will be modified by replacing the afterburner and single-stream nozzle with the selected duct burner and the coannular nozzle and ejector system. As shown in Figure 1-3, the duct burner is located behind rather than around the engine, to minimize changes to the F100 engine, and a TF30 iris nozzle is used for the variable nozzle of the bypass stream. Following successful completion of the testbed engine program, a Variable Cycle Engine Experimental program is envisioned. While currently in the preliminary planning stage, this program could involve incorporating the variable geometry fan of the VSCE into a new low spool which would be combined with existing high spool engine components, a duct burner and a coannular nozzle/ejector system to provide a more advanced test vehicle. The duct burner in this engine could be an improved version of that used in the testbed, incorporating more advanced technology. The VCE Experimental demonstration program could involve, in addition to substantiation of the low spool technology, an opportunity to study duct burner - fan interactions and other component integration aspects in a more realistic environment than that of the testbed engine.

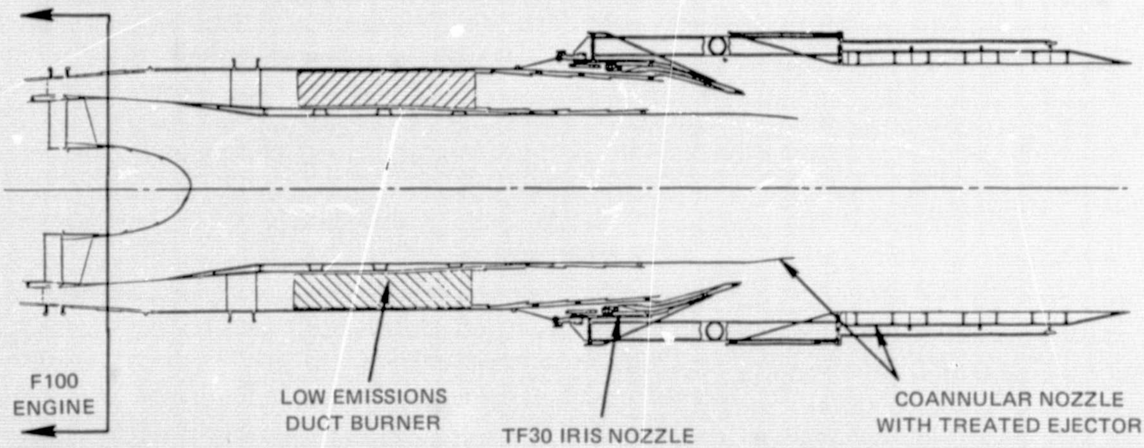


Figure 1-3 Critical Technology Test-Bed Engine Arrangement

Figure 1-4 shows the relationship of the duct burner analytical screening study to the overall Variable Cycle Engine program. This study program provided the opportunity for comprehensive evaluation of the problems and risks associated with various duct burner concepts and led to the selection of a configuration compatible with the schedule and objectives of the VCE testbed program. Following this study, a rig test program will be conducted to provide experimental substantiation and optimization of this configuration prior to its incorporation in the testbed engine.

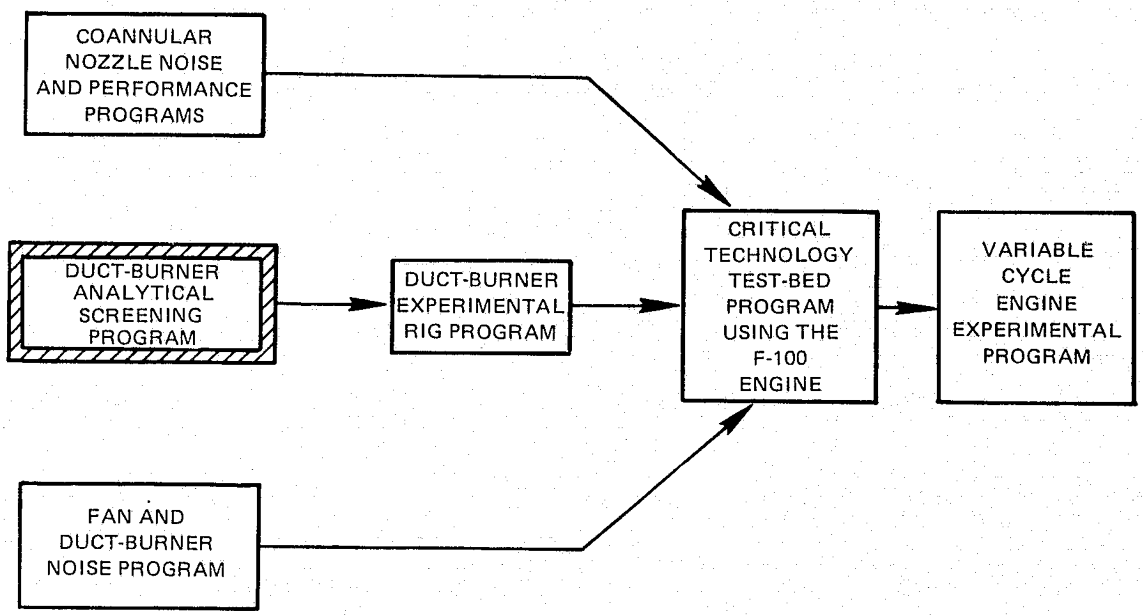


Figure 1-4 Critical Technology Programs for Advanced Supersonic Variable Cycle Engines

1.3 DUCT BURNER CONSTRAINTS AND REQUIREMENTS

The goal in establishing the conceptual design of the duct burners was to satisfy the aerodynamic constraints of the VSCE-502B engine while also obtaining low emissions. These included size (frontal area and length), pressure loss, chemical and thrust efficiency. The operating conditions for the VSCE-502B duct burner are listed in Table 1-I for takeoff, transonic climb and supersonic cruise. The fuel-air ratio and temperature rise requirements at sea level takeoff are those required to meet FAR 36 noise limits. The duct reference Mach numbers are consistent with the radial duct height of 33 cm (13 inches).

The performance goals for the duct burner are listed in Table 1-II. The supersonic cruise operating condition is critical to the economic operation of the aircraft and the fan duct pressure loss and thrust efficiency are specified at this condition. The low ignition fuel-air ratio is dictated by operational constraints. Although the duct nozzle area is varied to match the duct burner exit temperature, the initial pressure pulse produced by the initiation of combustion, if severe enough, could stall the fan. Experience with conventional duct burners has indicated that if ignition occurs at a fuel-air ratio of 0.002 or lower, this pressure pulse will be sufficiently weak to avoid perturbing the fan.

TABLE 1-I

OPERATING CONDITIONS FOR ADVANCED SUPERSONIC ENGINE DUCT BURNER

Variable Stream Control Engine, VSCE-502B

	Takeoff	Transonic Climb	Cruise
Flight Mach Number	0.3	1.3	2.4
Altitude - m (ft)	0	11,110 (36,500)	16,130 (53,000)
Duct P_T – N/m^2 (psia)	260,800 (37.8)	182,200 (26.4)	253,800 (36.8)
Duct T_T – K (°F)	438 (330)	445 (342)	604 (628)
Duct Reference Mach No.	0.161	0.119	0.120
Duct Airflow – kg/sec (lb/sec)	247 (543)	127 (279)	154 (338)
Fuel/Air Ratio	0.0385	0.030	0.013
Duct Exit T_T - K (°F)	1603 (2430)	1500 (2053)	998 (1340)
Fuel Flow - kg/sec (lb/sec)	9.5 (20.91)	3.81 (8.37)	1.99 (4.39)

TABLE 1-II

PROGRAM PERFORMANCE GOALS

Thrust efficiency at supersonic cruise	94.5%
Fan duct total pressure loss at supersonic cruise	6.5%
Ignition fuel-air ratio	0.002

Table 1-III shows the emissions goals established by NASA for this program. The goal levels of carbon monoxide and unburned hydrocarbons emission indices are representative in that they are typical of those necessary to achieve the more general combustion efficiency goal. These goals are intended only as a standard for comparison and are not related to any proposed or established regulations for advanced supersonic aircraft.

TABLE 1-III

PROGRAM EMISSIONS GOALS

Flight Condition	Pollutant	Emission Index (g pollutant/kg fuel)
Supersonic cruise and sea level takeoff	NO_x	1.0
	CO	30.0
	THC	2.5
	Smoke (SAE Smoke No.)	15.0
All Operating Conditions	Combustion Efficiency	99%

1.4 PROGRAM ORGANIZATION

The objective of this study was, through systematic analytical screening of combustor concepts, to identify those concepts with the potential for low emissions and high performance in advanced supersonic engines. To permit evaluation of a maximum number of concepts while conducting the study in sufficient depth, this program was organized to include three levels of screening with the accepted concepts being analyzed in progressively more detail.

In the initial phase, combustion concepts were considered that ranged from improved versions of current state-of-the-art duct burners through the technology levels demonstrated in the NASA sponsored Experimental Clean Combustor and pollution Reduction Technology Programs, to such advanced concepts as variable geometry premixed-prevaporized combustors. These candidate concepts were screened on the basis of their emissions reduction potential when applied to the pilot and/or high power stage of a duct burner in the VSCE engine. The data base used in predicting the characteristics of these combustor concepts included the NASA Experimental Clean Combustor and Pollution Reduction Technology Programs (Ref. 5 through 9), Pratt & Whitney Aircraft's related experience with main burners and augmentors (Ref. 10 through 13) and the published results of investigations by NASA (Ref. 14, 15 and 16), other engine manufacturers and research laboratories (Ref. 17 and 18).

Following the first level of screening, the selected concepts were used to define a number of duct burner configurations which were subsequently studied in greater detail. This phase involved the aerothermal definition of each concept, estimating the performance of the duct burner and its impact on the overall engine cycle performance over the total mission, refined estimates of the emissions characteristics and qualitative evaluation of such factors as cost, weight, potential for acoustic instabilities and development risk. All of these factors were considered in the second screening process which resulted in the selection of four configurations for more detailed analysis and mechanical design studies in the final phase of this program. Two of the selected configurations were considered to require moderate levels of risk consistent with the goals and schedule of the VCE testbed engine.

The final phase of the program involved mechanical design studies of the four duct burner configurations selected in the second phase. Since the two concepts with moderate development risk were considered candidates for the VCE testbed program, the studies on these configurations were conducted in the airflow size of the F100 engine that is expected to be used in that program. These studies provided an opportunity to identify potential mechanical, and durability and operational problems for each configuration. The final step was to identify one concept in each development risk category that appeared to offer the greatest potential for satisfying the performance requirements and future emissions restrictions on supersonic aircraft.

2.0 STUDY PROCEDURES

This section describes the analytical procedures used in this study. Where these procedures led to the definition of specific design criteria, these constraints are identified.

2.1 EMISSIONS

The emissions projections of this study are based on the results of numerous experimental programs including primarily those directed at the evaluation of low emissions main burners. These programs have been conducted on a variety of combustor concepts with the majority being evaluated in the range of inlet pressures, inlet temperatures and fuel air ratios encountered in engine main burners. In order to evaluate the emissions of these concepts in the duct burner environment, the emission data were scaled from the inlet conditions of temperature and pressure used in the test programs to those of the duct burner.

The procedure used in the scaling involved normalizing the experimental NO_x and CO emission data against the appropriate scaling factor for the inlet conditions. Figure 2-1 shows representative data from Reference 5 and 14 prepared in this manner. The emissions are projected from this data using the appropriate values of fuel-air ratio and inlet temperature and pressure in the duct burner. The development and empirical verification of the scaling factors shown are discussed in detail in Reference 19. The scaling factor for the effect of inlet temperature on NO_x emissions has been recently refined using data reported in Reference 14 which involved testing an experimental burner over a range of inlet temperature with all other parameters maintained constant.

When multiple stages or combinations of stages other than those for which a data base exists are to be examined, it was necessary to establish the influence of each stage on the overall emissions. If the concept in each stage had been tested independently, the contribution of each stage could be superimposed to construct a composite estimate. Assuming the emissions contribution of each stage is relatively independent, the total emissions from a combined concept can be obtained by fuel flow weighing the emissions of each component, as shown below.

$$\text{Composite EI} = \frac{W_{\text{fuel stage A}} \times EI_{\text{stage A}} + W_{\text{fuel stage B}} \times EI_{\text{stage B}}}{W_{\text{fuel stage A}} + W_{\text{fuel stage B}}}$$

When the emissions characteristics of the first stage of a staged burner are adequately defined, the contribution from the second stage can be computed by manipulation of the above equation.

Direct scaling of the carbon monoxide emissions characteristics of the reference combustors to the duct inlet conditions generally indicated that they would be deficient in combustion efficiency. However, as shown by the data of Figure 2-1, the high CO emissions data obtained at low fuel-air ratios diminished rapidly when the average combustion gas temperature approached 1600K (2400 F) which, in main burners, occured at a fuel-air ratio of about 0.022. Consequently, it was assumed that if the stage air and fuel schedule could be adjusted to produce this environment in the latter regions of the stage, the excess carbon monoxide could be consumed by increasing the residence time in the stage above that in the reference burner. Since the NO_x production rates are low at this temperature level it was assumed that no additional NO_x emissions would be generated during the additional residence time.

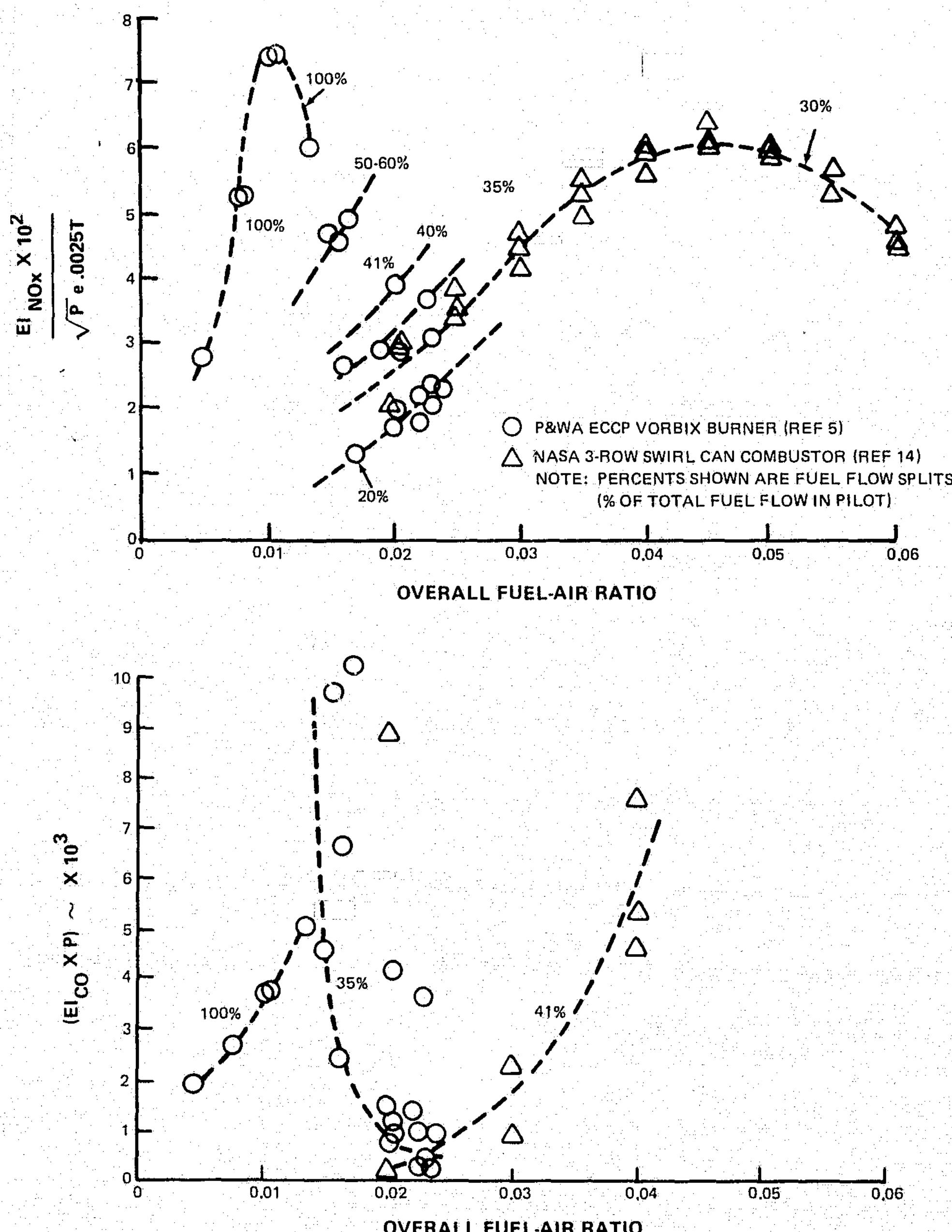


Figure 2-1 Representative Experimental Burner Emissions Characteristics

No direct scaling or computation of unburned hydrocarbon emissions was attempted. Experience with low emissions combustors generally demonstrates that unburned hydrocarbon emissions tend to parallel those of carbon monoxide. Based on this experience, a ratio of CO to unburned hydrocarbon emissions indices of ten was assumed in estimating the combustion efficiency.

Similarly, the smoke characteristics of the duct burner concepts were not evaluated directly. However, experience with main burners has indicated that excessive smoke may be produced in very rich combustion zones. Using this experience as a guideline, an upper limit on primary zone equivalence ratio of 2.0 was imposed in the study.

The application of the above procedures to the projection of the emissions characteristics of a particular combustor concept involved selection of a representative data base for the particular concept. The data was examined to determine the optimum operating condition – i.e., primary zone equivalence ratio, and in the case of staged concepts, the stage fuel flow split. This condition became the reference point for projecting the emissions charactersitics of the concept. In subsequent aerothermal definition studies, it was imperative that the airflow scheduling, geometric proportions, and other critical aerodynamic parameters be selected so as to duplicate the operating conditions and configuration of the reference burner to assure the validity of the emissions projections.

2.2 COMBUSTION ZONE SIZE DEFINITION

The ignition capability of the duct burner dictated the volume requirements of the first stage of many of the duct burner concepts. Extensive studies of experimental combustors by Odgers and Carrier (Ref. 20) has demonstrated that the lean stability limit of swirl stabilized combustion correlates well with the primary zone volumetric loading corresponding to 40 percent combustion efficiency. Since ignition must occur at a fuel-air ratio above this stability limit, the loading level corresponding to 80 percent combustion efficiency shown on Figure 2-2 was used as the criteria to establish the volume requirements of the primary zone of the first stage of the duct burners.

The premixed combustor concepts rely on bluff body flame stabilization with the blocked area of the perforated plate flameholder providing the bluff body area. The correlations of Reference 21 were used in establishing the size of these flameholders and their stability characteristics.

The volumetric loading (heat release per unit volume and pressure) or the residence time in the reference burner were used as bases for defining the size requirements for the remaining duct burner stages. However, as mentioned in Section 2.1, in many instances the carbon monoxide emissions characteristics projected from the data from the reference combustor indicated that the combustion efficiency would be below the goal levels. Consequently, the air and fuel scheduling to these stages were established so as to provide a gas temperature level of about 1600K (2400 F) in these stages and additional residence time was allowed in this region to produce further CO oxidation. The overall residence time in the stage was established using a stirred reactor kinetic analysis. Figure 2-3 shows the carbon monoxide-residence time relation defined by this analysis at pressure and temperature levels corresponding to supersonic cruise. Similar curves were generated for the inlet conditions corresponding to sea level takeoff and transonic climb conditions and were used to establish the combustion efficiency-residence time characteristics at these conditions.

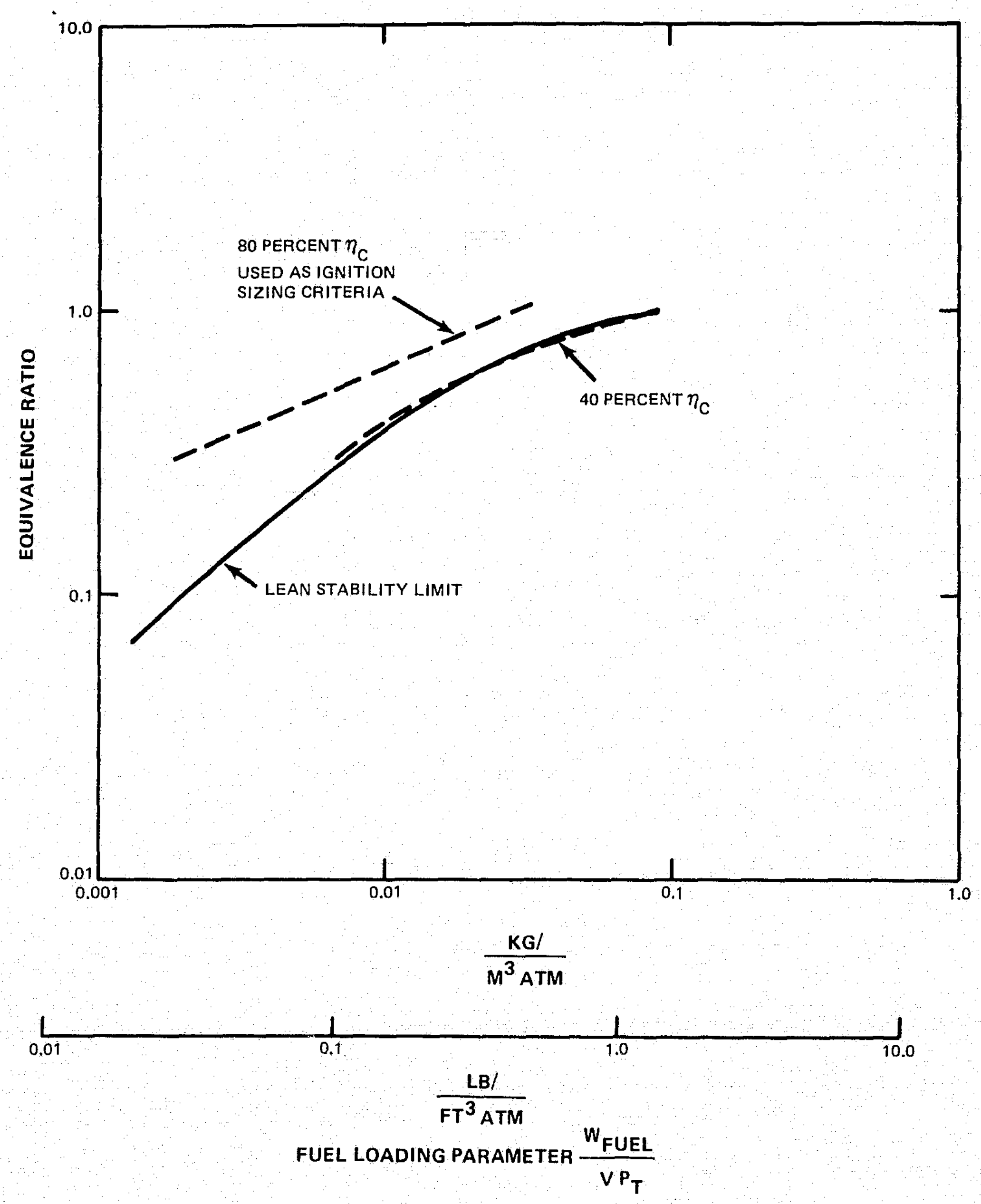


Figure 2-2 First Stage Volume Criteria (Based On Data of Reference 20)

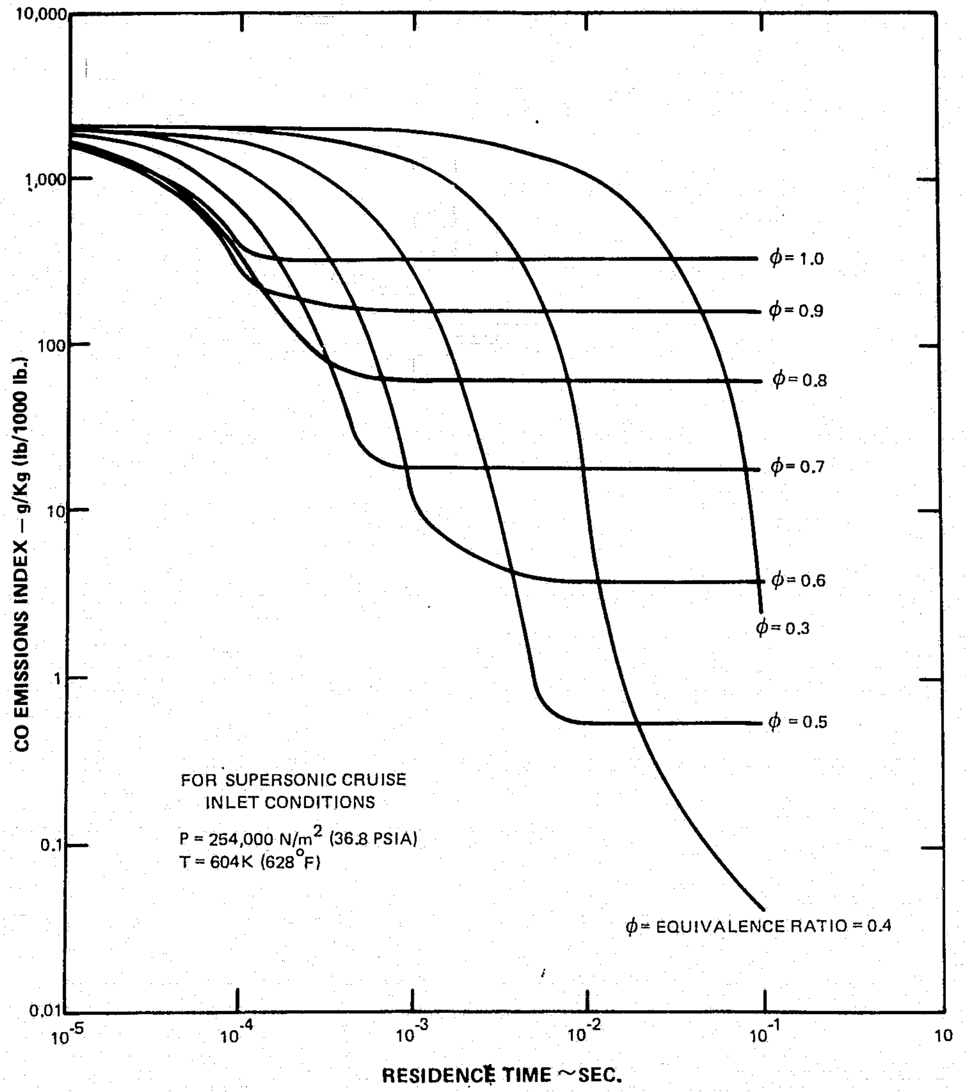


Figure 2-3 *Carbon Monoxide-Residence Time Relation From Kinetic Analysis*

The kinetic analysis assumes a homogeneous combustion mixture and this assumption is most accurate for the premixed-prevaporized concepts studied in this program. In a non-homogeneous reaction, the rate of mixing may alter the residence time requirements substantially. As shown in Figure 2-3, with a lean overall mixture, locally rich regions require substantially less residence time and this analysis may be quite conservative in application to concepts that are not mixing limited.

2.3 AEROTHERMAL DEFINITION

The procedure of Sections 2.1 and 2.2 established the gross airflow distribution in the duct burner, the conceptual definition of the burner (i.e., type of fuel injection, mixing devices and staging requirements) and parameters such as reaction zone volume or residence time consistent with stability or combustion efficiency requirements. The reference velocity – the velocity at the appropriate cross section of the burner in the absence of combustion– was the parameter used to establish the radial height-length proportions of the combustion zones from the volume or residence time requirements. The selection of the reference velocity was generally based on compromises between the need for low frontal area and the desire to duplicate the velocity levels in the reference combustor used for emissions projections. In the case of the first stage of the burners, other parameters, such as the dome height necessary for stabilization of recirculation zones, also influenced the selection of reference velocity levels.

In many of the burner concepts, jets of combustion air or fuel-air mixtures are introduced into the front end of the stages and the ensuing combustion process is assumed to occur after these jets have been mixed with the internal flow. Consequently, definition of the overall length of the stage requires consideration of, not only the residence time required to allow the combustion process to achieve the desired degree of completion, but also the length required for this initial mixing process. The length of these mixing zones were assumed to be related to the axial distance required for penetration of the combustion air jets. Data on jet penetration from References 22 and 23 were incorporated into an analytical model to predict the length of the mixing zone. This model reflected such factors as jet to combustion gas momentum ratio, length scales and, in the case of swirling jets, the effect of the tangential component of velocity on the mixing process.

The apertures for admitting air to the combustor were sized to be compatable with the design pressure drops across the combustor liner. Where available, data from the experimental airflow calibration of similar parts was used to define the effective flow area (AC_D) and establish the overall size of the component. In defining the density, location and orientation of the burner components, the configuration of the reference combustor used as a data base for the emissions projections was duplicated as much as possible to assure that these projections would be realistic.

The Mach number of the flow approaching and that bypassing the pilot stages of the burner is relatively high. Many of the designs incorporate ram air chutes and hoods to recover the dynamic pressure of the high Mach number flow and maintain high liner pressure drops consistent with strong penetration and mixing of combustion air jets. Where these pressure recovery devices were not employed it was assumed that apertures in the burner were fed by the local static pressure in the shroud. To prevent aspiration of combustion gases from the burner, it was stipulated that an aspiration margin – the ratio of the static pressure drop across the liner to the dynamic pressure in the burner shroud – be at least 1.8. This criteria, in effect, dictated the maximum allowable shroud Mach number in the absence of pressure recovery devices on the liner cooling air inlets.

2.4 DUCT BURNER PERFORMANCE

The total pressure loss across the duct burner was assumed to consist of two components – the pressure drop across the metering elements of the burner and the momentum pressure losses associated with the mixing and combustion processes in the burner proper.

For the initial aerothermal definition studies of Section 3.3, the pressure drop across the liner was assumed prior to sizing the combustion air apertures and the heat addition momentum pressure loss was computed from the Rayleigh analysis of constant area flow (Ref. 24) using the Mach number associated with the reference velocity as an initial condition. In the axially staged combustors successive mixing and combustion processes, involving additional combustion air being introduced into the burner, the losses were computed on the assumption that the dynamic pressure associated with the entering combustion air was disipated in the mixing process. A mixed state of the gas was defined that served as the initial condition for the computation of the heat addition loss in the subsequent combustion process in that stage. This stepwise procedure was continued through all of the stages to define the total pressure at the exit.

During the Mechanical Design studies of Section 3.4, a more detailed analysis was conducted on one duct burner configuration. A computer program, developed during previous augmentor design efforts, was used for this analysis. The program includes shroud pressure loss effects and the varying static pressure in the gas stream is used to compute the local pressure drop across the liner. Flow balancing routines are employed to calculate the airflow distribution through the liner. Stepwise control volume calculations along the length of the augmentor are employed to calculate the total pressure losses associated with the introduction of cooling air, combustion air, as well as those incurred in the incremental heat release process.

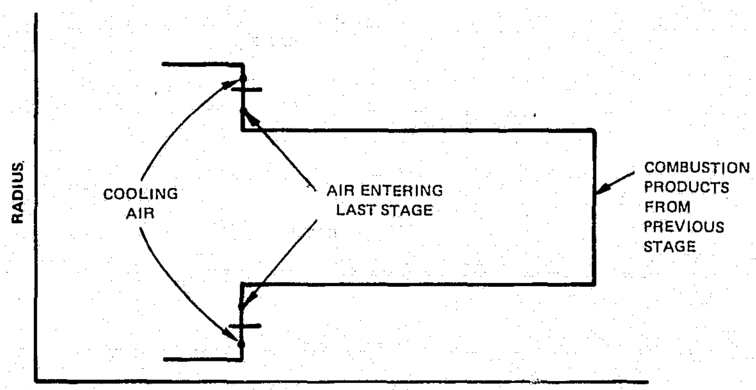
This program also incorporates subroutines to compute liner metal temperature distributions based on the local cooling flow rates predicted by the flow analysis. The details of these computations are discussed in Section 2.6.

The definition of thrust efficiency combines the concept of chemical combustion efficiency with the influence of temperature variations at the exit nozzle throat on the thrust:

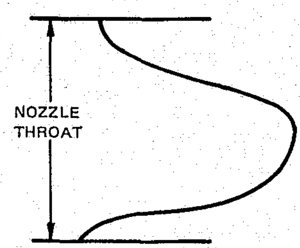
$$\begin{pmatrix}\text{Thrust}\\ \\ \text{Efficiency}\end{pmatrix} = \begin{pmatrix}\text{Chemical}\\ \text{Combustion}\\ \text{Efficiency}\end{pmatrix} - \begin{pmatrix}\text{Temperature}\\ \text{Profile}\\ \text{Inefficiency}\end{pmatrix}$$

The temperature profile at the exit of a main burner must have a precise radial distribution and controlled variation from the average temperature (pattern factor) to provide reasonable turbine life. In contrast, the temperature distribution at the fan duct nozzle should be as uniform as possible to produce the maximum thrust. Chemical combustion efficiency, as related to CO and THC emissions, pertains to the degree of completion of the chemical reaction. However, the chemical reaction can be essentially completed with a significant variation in exit temperature and, hence, velocity profile. Consequently, the thrust efficiency is directly related to the mixing accomplished in the burner. However, even with perfect mixing, the uniformity of the exit temperature profile is limited by the necessity of cooling the exit nozzle. Some of the cooling air must be at or near the temperature limit of the nozzle material.

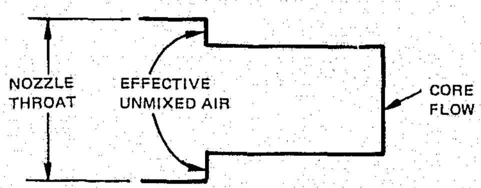
Mixing parameters, including pressure drop and the ratio of duct length to height, were incorporated in a thrust efficiency model in which the degree of mixing achieved is related to the thrust efficiency. The representation of a multi-staged duct burner in this simplified analysis is accomplished by using a composite system of flows. The critical stage of the duct burner for mixing is the last axial stage prior to the nozzle throat. The flow entering this stage can be described by Figure 2-4a. Mixing occurs primarily between the flows from the previous stage and the air entering the last stage, with the cooling flow incrementally added along the length of the mixing region. The mixing of these flows result in an acutal temperature profile similar to that shown in Figure 2-4b. This profile can be represented by a simple stepped profile shown in Figure 2-4c, with convenient definitions that produces the identical thrust. The portion of the air at the inlet temperature is defined as effectively unmixed air and the fraction of the air in the unmixed mode is related to the thrust efficiency decriment by the relationship of Figure 2-5.



A. FLOW AT INLET TO LAST STAGE



B. ACTUAL TEMPERATURE PROFILE AT NOZZLE THROAT



C. IDEALIZED TEMPERATURE PROFILE PRODUCING SAME THRUST

Figure 2-4 Simplified Model of Mixing in Duct Burners

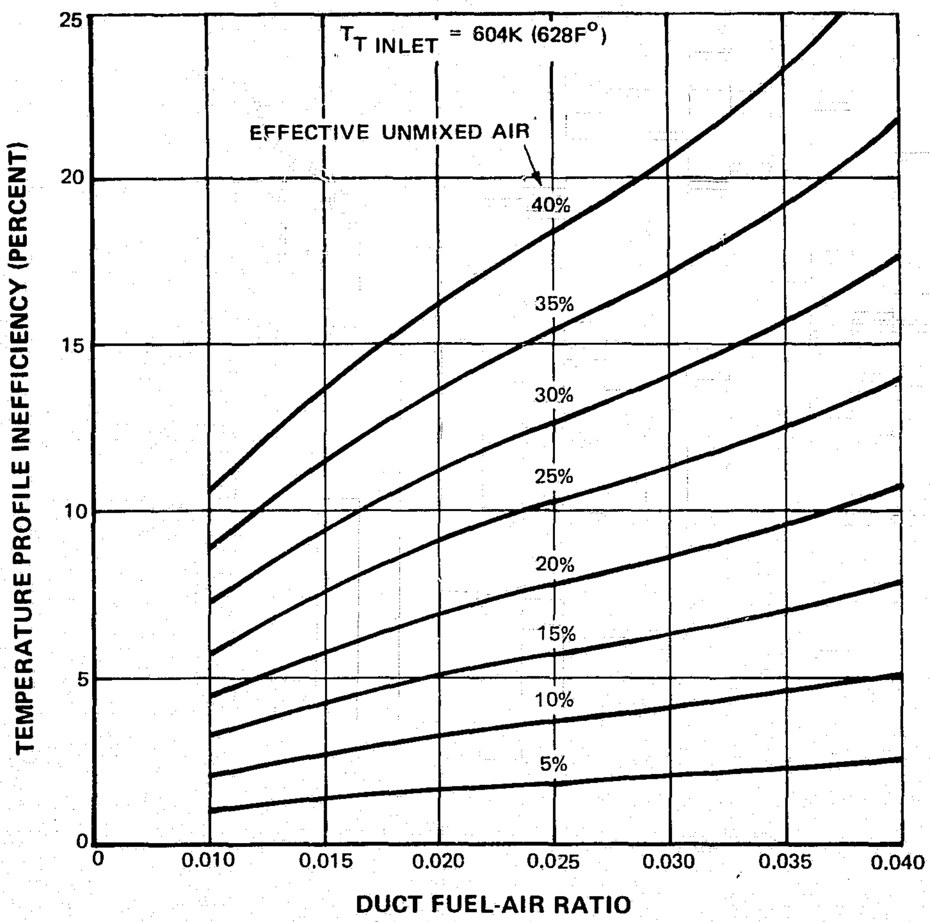


Figure 2-5 Relation Between Temperature Profile Inefficiency and Effective Unmixed Air

The definition of the unmixed air fraction is dependent on parameters such as the mixing mode in the last stage, the length and pressure drop across the stage and liner and nozzle flap cooling airflows. A qualitative correlation between these parameters was established on the basis of limited data available on the performance of augmentors (Ref. 10, 11, and 15). The lack of sufficient data to precisely define the influence of these parameters on the mixing process results in concern for the validity of the absolute levels of predicted thrust efficiency. However, the analysis is internally consistent and when applied consistently to all of the configurations evaluated is expected to predict realistic trends and relative performance increments.

2.5 ENGINE AND AIRCRAFT PERFORMANCE

Engine performance, weight, and nacelle diameter were defined based on the conceptual design of the duct burner. Variations in these propulsion system parameters result in changes to the base aircraft which provides a screening criterion for the burner study. The baseline airplane used in the screening study is the NASA Langley Reference Aircraft described in Reference 25. This is basically a modified arrow wing aircraft carrying 292 passengers with a design range of 7408km (4000nm). The VSCE-502B engine was "installed" in the baseline aircraft for this screening study. This aircraft has a gross weight of 345,600kg (762,000 lbs), however, for purposes of this study, the aircraft takeoff gross weight was scaled to maintain constant range as the propulsion system characteristics varied.

The gross weight scaling is accomplished through the use of an aircraft trade factor approach that has been developed and tested through the process of complete preliminary design. Introducing incremental changes to propulsion system performance, weight, and diameter and then evaluating their effects on the aircraft over the entire mission profile provides the capability to determine the impact of any system change in terms of an increment of takeoff gross weight. This method of mission evaluation has proven to be quite accurate for preliminary studies, and provides an efficient means of evaluating small cycle variations.

The effect of the burner on the engine performance is evaluated in a manner similar to the mission analysis. Small perturbations to burner performance are introduced to the VSCE-502B engine performance simulation and the resultant effects are normalized and used to evaluate variations in duct burner performance. Specifically, if the burner pressure loss or thrust efficiency had been compromised in a particular configuration, through the engine sensitivity factors these deviations are translated to a change in cycle fuel economy (thrust specific fuel consumption). Then, aircraft trade factors are employed to provide the proper gross weight increment.

The weight of the base engine was defined using established techniques and increments to this base were calculated for the duct burner configurations based on changes in radial height, length, and burner weight, as estimated from its relative complexity. The radial height, in addition to its impact on case weight, causes variations in the maximum nacelle diameter to accommodate nozzle actuation requirements. Those concepts that required diameter increases resulted in larger drag characteristics for the engine installation. The installed nacelle drag is influenced by several factors, but pod wave drag was considered dominant and was the only aspect considered in evaluating the drag increase in the screening study. The impacts of the other drag components were either insignificant or too uncertain to include in a preliminary study of this nature. Installed propulsion system weight and nacelle wave drag impact the aircraft as does the engine performance. This impact is analyzed in the same manner, using aircraft trade factors.

The following relations were used to determine the net change in aircraft gross weight from the individual increments:

$$\Delta\,\mathrm{TOGW} = \frac{\partial \mathrm{TOGW}}{\partial \mathrm{TSFC}}\;\Delta\,\mathrm{TSFC} + \frac{\partial \mathrm{TOGW}}{\partial \mathrm{W}}\;\Delta\,\mathrm{W} + \frac{\partial \mathrm{TOGW}}{\partial \mathrm{D}}\;\Delta \mathrm{D}$$

where:

$$\Delta\,\mathrm{TSFC} = \frac{\partial \mathrm{TSFC}}{\partial \eta_T}\;\Delta\,\eta_T + \frac{\partial \mathrm{TSFC}}{\partial\,\Delta P_T/P_T}\;\Delta\;\frac{\Delta P_T}{P_T}$$

2.6 LINER COOLING

The use of louver cooling liners was assumed throughout this study. During the aerothermal definition studies of Section 3.3, liner cooling flow requirements were estimated by scaling on a cooling flow per unit surface area basis from an existing duct burner design (Ref. 10). Similar techniques were used to estimate the cooling flow requirements for the duct stream nozzle flaps of the VSCE-502B engine. These scaling procedures recognized differences in gas path temperatures and velocity levels between these configurations.

The computer program discussed in Section 2.4 provided a detailed thermal analysis of the liner of one of the duct burner configurations during the mechanical design studies reported in Section 3.4. The airflow and pressure loss routines described in Section 2.4 provided definition of the cooling flows through individual louvers and empirical film cooling correlations incorporated in the program establish the scrubbing gas temperatures. Local metal temperatures are computed through a heat balance that includes gas side heating and shroud side cooling effects through both convective and radiant heat transfer modes.

2.7 FUEL/AIR HEATING SYSTEMS

The evaluation of premixed-prevaporized combustor concepts involved the definition of heat transfer systems for the preheating of fuel and air prior to their introduction into the burner. The heat transfer and friction characteristics of the components involved were determined from the experimental data of Reference 26 for surfaces of representative geometries. The sizing of heat exchangers was accomplished using the effectiveness - NTU analysis also presented in Reference 26.

3.0 DISCUSSION OF RESULTS

3.1 INTRODUCTION

The results of this study are presented in three parts that parallel the three step progressive screening approach outlined in Section 1.4. Section 3.2 presents the results of the initial phase in which various combustion concepts were evaluated with respect to their emissions characteristics and combustion stability. This survey culminated in the conceptual definition of eight candidate duct burner configurations that appeared to offer the lowest levels of emissions. These configurations were studied in further detail and Section 3.3 discusses the aerothermal definition of these duct burners and their impact on the overall VSCE-502B engine cycle. The discussion is concluded with a division of the configurations into moderate and high development risk categories and a ranking of the configurations in terms of their emissions and performance characteristics, cost, weight and overall compatability with the remainder of the powerplant. Mechanical design studies were conducted on four of these configurations — two of each from the moderate and high development risk categories and the results of these studies are discussed in Section 3.4.

Throughout the study, the emphasis has been placed on establishing the feasibility of various combustion concepts solely from the point of view of aerothermal definition and on the identification of potential problem areas. Resolution of these problems, involving in many instances the filling of technology voids, was clearly beyond the scope of this study. An effort has been made to indicate the uncertainties that occurred in this study and their implications are discussed in the following sections. In general, these uncertainties appear to affect the absolute level of predicted performance rather than the relative comparison of the candidate duct burner configurations.

3.2 PRELIMINARY SCREENING

The intent of the preliminary screening phase of the study was to evaluate a large number of combustor concepts to assess their potential for incorporation into a duct burner. The level of screening in this phase was limited primarily to consideration of the emissions characteristics of the candidate concepts.

Even without the constraint of low emissions, the requirement that a duct burner be capable of lighting at very low fuel-air ratios and providing smooth modulation to high fuel-air ratio levels leads to the need for a piloted or multistage combustor system. The process of screening the large number of duct-burner concepts that can be synthesized from combinations of combustor concepts was simplified by noting that some design constraints and goals are unique to only one of the stages. The overall stability of the combustion process including lighting at low fuel-air ratios and adequate lean blowout margin are constraints unique to the pilot stage. Furthermore, considering the fuel-air ratio turndown between the maximum augmentation and supersonic cruise conditions shown in Table 1-I, stability constraints as well as emissions optimization will dictate that only the pilot stage be operational at the cruise condition. While some coupling is recognized in the quench rate of the pilot exhaust gases by the mixing of bypass air, the cruise emissions characteristics are essentially independent of the type of high power stage employed. Similarly, experience with staged combustors indicates that, to a first approximation, the emissions at high fuel-air ratios are relatively independent of the pilot stage configurations. Consequently, the potential of a particular combustion concept incorporated in the high power stage of a duct burner could be assessed from the emissions levels projected at the high fuel-air ratio sea level takeoff operating condition.

The remaining parts of this Section include a description of the various combustor concepts evaluated in this phase of the study and the potential benefits and difficulties associated with each. Projected emissions, in terms of combustion efficiency and the emission index for oxides of nitrogen, are presented for each concept. These characteristics were determined by scaling experimental data from the particular concept to the inlet pressure, inlet temperature and operating fuel-air ratio of the duct burner in the VSCE-502B engine using the techniques of Section 2.1. Candidates for the pilot stage were evaluated at the supersonic cruise condition of Table 1-I while the emissions characteristics of the data base were scaled to the corresponding sea level takeoff condition for the high power stage concepts. In the majority of the cases, the data base for the combustor concepts consisted of emissions measurements obtained in rig tests of experimental main burners while the remainder originated in laboratory research programs.

Combustion stability was also introduced into the screening of the pilot stage concepts during this phase of the study. The duct burner must be capable of lighting without producing a pronounced pressure pulse that might propogate upstream and stall the fan. Experience has indicated that an adequately soft light may be obtained if the duct burner is capable of ignition at an overall fan duct fuel-air ratio of 0.002, and this level was set as the goal. Lighting fuel-air ratios were established for each pilot stage concept using the procedures of Section 2.2 which essentially establishes the stability characteristics of the pilot stage in terms of the capability to accommodate fuel-air ratio turn-down relative to the supersonic cruise point at which it must produce the minimum emissions levels.

3.2.1 Single Stage Pilots

A survey was conducted of available experimental single stage main burners that had been designed primarily for low emissions to identify the concepts that could be most advantageously employed as a pilot stage in the duct burner. These combustors had been constructed with various fuel injection devices including pressure atomizing nozzles, aerating or air assist nozzles and carburetor tubes with both rich and lean primary zone stoichiometry. As shown on Figure 3-1, the configurations employing pressure atomizing or aerating fuel nozzles resemble conventional main burners. The fuel nozzle includes or is surrounded by a swirler to establish a stable recirculating flow in the primary zone. The carburetor tube concept, also shown on Figure 3-1, employs a fuel injector and swirlers or other turbulence generating devices at the inlet of a mixing tube. Combustion is stabilized in the recirculating flow formed at the sudden enlargement at the end of the tube.

Experimental data obtained from these low emissions main burners were used as the basis for projecting the emissions characteristics of single stage pilots for the duct burner. Aerating fuel nozzles were evaluated in the NASA/P&WA Pollution Reduction Technology Program (Ref. 7) while both aerating and pressure atomizing nozzles have been evaluated in several experimental JT9D burners (Ref. 12) with both rich and lean primary zone stoichiometry. Lean carburetor tube combustors were tested under the Pollution Reduction Technology Program cited above while similar configurations with rich primary zone stoichiometry were evaluated in experimental JT8D and JT9D main burners.

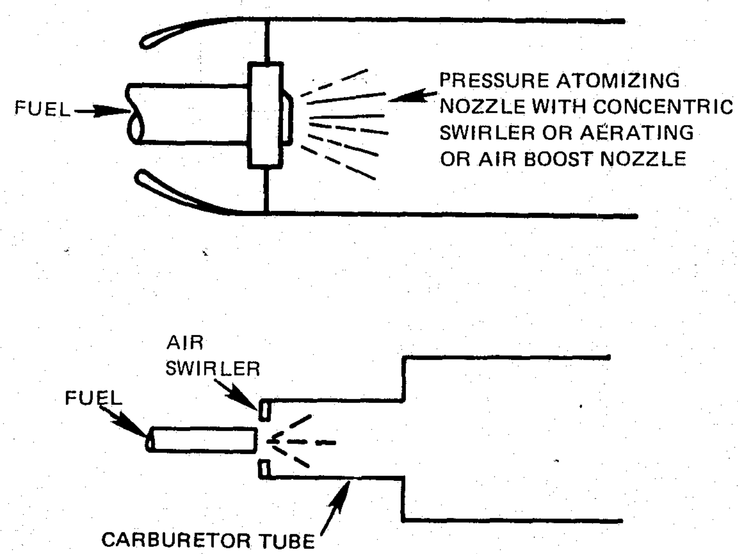


Figure 3-1 Single Stage Pilot Concepts

The procedures of Section 2.1 were used to project the emissions levels of these single stage combustors at the inlet conditions of the duct burner at supersonic cruise and the results are shown on Table 3-I. The specified primary zone equivalence ratios correspond to those at which optimum emissions characteristics were observed in the reference combustor. Direct scaling of the carbon monoxide emissions indicated that all of these concepts would have more than adequate combustion efficiency at the duct burner pressure levels. This table also shows the projected ignition fuel-air ratio for each of these burner concepts. Where data on ignition or lean blowout fuel-air ratio was available, these projections were based on the available turndown of fuel-air ratio from the level that produced optimum emissions. For concepts that lacked this experimental data, which included primarily the rich primary zone configurations, the minimum ignition fuel-air ratio was assumed to occur at a primary zone equivalence ratio of 0.5.

These results indicate that, in general, the adaptation of main burner low emissions technology to a single stage duct burner pilot yields relatively high levels of NO_X and a limited ability to meet the 0.002 fuel-air ratio requirement at ignition, particularly with the lean primary zones. A rich front end provides significantly lower NO_X emissions and lower ignition fuel-air ratios than the lean front end configurations without compromising combustion efficiency. Although the carburetor tube concept with a lean stoichiometry produces low NO_X emissions, it has a very limited turndown capability. There does not appear to be any substantial difference in the projected emissions or ignition characteristics of pilot stages operating with pressure atomizing or aerating/air blast fuel nozzles.

TABLE 3-I

PERFORMANCE OF SINGLE STAGE PILOTS AT SUPERSONIC CRUISE

	LEAN PRIMARY ZONE			RICH PRIMARY ZONE		
	Pressure Atomizing Nozzle	Aerating or Air Assist Nozzles	Carburetor Tube	Pressure Atomizing Nozzle	Aerating or Air Assist Nozzles	Carburetor Tube
Cruise NO_x EI	4.6	4.3	3.5	3.1	3.3	5.8
Cruise CO EI	13.	6.0	4.5	26.	11.4	7.3
Combustion Efficiency	>99.	>99.	>99.	>99.	>99.	>99.
Lighting Fuel Air Ratio	.0054	.0054	.0054	.0028	.0028	.0028
Primary Zone Equivalence Ratio	1.0	1.0	1.0	1.9	1.9	1.9

The ignition fuel-air ratio of the rich primary zone configurations might be reduced by increasing the primary zone equivalence ratio at the supersonic cruise design point. However, further enrichment of this zone beyond the present equivalence ratio of 1.9 could lead to the formation of smoke. Development of the pilot to balance lean ignition and smoke, with the possible incorporation of variable geometry, could eventually produce a configuration that satisfies the ignition requirement.

3.2.2 Dual Stage Pilots

Dual stage pilots offer two advantages over the single stage pilots of the preceeding section. Addition of a prechamber sized for stable combustion at a duct fuel-air ratio of 0.002 eliminates the lighting problem Use of the prechamber also provides axial staging of both the fuel and air at the supersonic cruise condition, permitting the use of staged low emissions main burner technology to reduce the cruise emissions. Several different dual stage pilot concepts were evaluated and the results are summarized on Table 3-II.

TABLE 3-II

PERFORMANCE OF DUAL STAGE PILOTS AT SUPERSONIC CRUISE

	Single Stage With Rich Primary Zone	Piloted V-Gutter	Premixed Prevaporized With Prechamber	Prechamber Vorbix	
				Lean Prechamber	Rich Prechamber
NO_x EI	3.1	1.2	0.6	2.6	2.6
CO EI	26.0	>250	<13.0	<13.0	<13.0
Combustion Efficiency %	>99.0	<91.0	>99.0	>99.0	>99.0
Lighting Fuel Air Ratio	.0028	0.002	0.002	0.0031	0.0016
Prechamber Equivalence Ratio At Cruise	–	1.0	0.48	0.45	0.95

The vorbix combustor shown on Figure 3-2 has been evaluated extensively under the NASA/P&WA Experimental Clean Combustor Program (Ref. 5 and 6) and other programs. The principles of its operation and design features are discussed in detail in Reference 27. The projected emissions of a vorbix pilot at supersonic cruise were based on data from Reference 5. The particular condition selected for the projection was a simulated main burner sea level takeoff point at which both stages were operational and the pilot to secondary fuel flow split was adjusted to produce the most favorable tradeoff between NOx and CO emissions. The equivalence ratio in the prechamber stage was low and this configuration was designated the lean prechamber concept on Table 3-II. The results indicated that the use of a dual stage vorbix pilot produced lower NOx emissions than the rich single stage pilot at supersonic cruise, but that the ignition fuel-air ratio was not improved because of the lean prechamber design. Examination of additional data on vorbix main burners indicated that a rich prechamber configuration could be defined that offered cruise NOx emissions comparable to that achieved with a lean prechamber. Being designed for a high equivalence ratio at cruise, the prechamber in this configuration had more than adequate fuel-air ratio turndown capability to satisfy the soft ignition requirement.

Direct scaling of the carbon monoxide emissions from these experimental burners indicated that the combustion efficiency would be below the 99 percent goal at supersonic cruise. To achieve this combustion efficiency goal, the residence time in the secondary stage of the pilot was increased beyond that of the reference experimental burner. By maintaining a gas temperature of about 1625K (2500°F) in the downstream part of this stage, the excess carbon monoxide could be consumed without generating additional NOx. To provide this environment at the 0.013 fuel-air ratio at supersonic cruise the pilot requires about 46 percent of the duct flow.

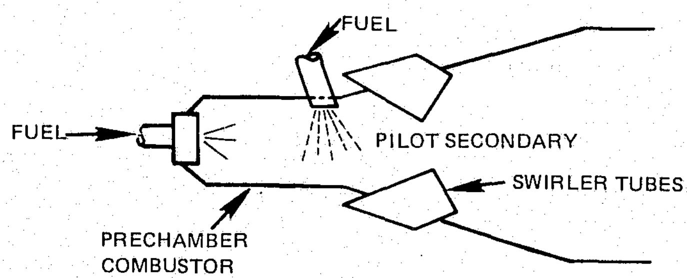


Figure 3-2 Prechamber Vorbix Pilot

A dual stage premixed pilot burner, shown schematically on Figure 3-3 was also evaluated. The prechamber is isolated from the secondary stage of the pilot and, with an airflow of about 6 percent of the duct flow, has adequate stability for ignition at a duct fuel-air ratio of 0.002. At supersonic cruise both the prechamber and pilot secondary stages are operational at an equivalence ratio of about 0.48 which represents an optimum with respect to stability and the lean combustion conducive to low NOx generation. While experimental premixed main burners have been tested extensively under the NASA/P&WA Experimental Clean Combustor Program (Ref. 5) and other programs, these combustor were subject to more stringent design constraints than encountered in the duct burner application. Because of the high inlet pressure and temperature levels encountered in main burners, these combustors were designed with low residence time in the premixing passages to avoid potential autoignition. The environment in the fan duct is much less severe and ignition delay times are more than ten times longer than in the main burner. Consequently, mixing is not compromised by inadequate residence time and more extensive efforts may be made to create the truly homogeneous fuel-air mixtures characteristic of those achieved in idealized premixed combustors. The research combustor of Reference 13 is typical of these ideal premixed systems and uses prevaporized fuel injection into a 0.92 meter (3 foot) long premixing duct, that contains several mixing baffles and turbulence generating screens, before the mixture is discharged through the flameholder. Assuming that the degree of premixing achieved in these research burners could be attained in the duct burner, data from these configurations were selected as a data base for this analysis.

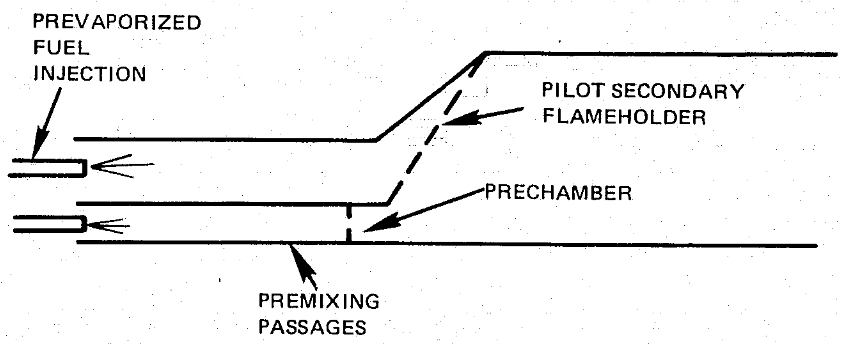


Figure 3-3 Two Stage Premixed Pilot

The data of Reference 13 was obtained over a limited range of inlet temperature consistent with high power operation of a main burner. The lower inlet air temperatures at which the duct burner operates could have a considerable impact on the rate of fuel vaporization. This inlet temperature effect was deducted from data obtained on the Vortex Air Blast combustor of Reference 17. This burner appeared to achieve a reasonably high degree of premixing at high inlet temperatures but as shown on Figure 3-4 exhibited an increase in NOx emissions as the inlet temperature was reduced. Assuming that this trend was attributable to fuel vaporization limitations, it was used as a basis for extrapolating the data from the research burner of Reference 13 to project the NOx emissions of the dual stage premixed pilot at supersonic cruise.

Like the prechamber vorbix pilot, the data from the research rig indicated that the combustion efficiency would be below the 99 percent goal at supersonic cruise, but it was assumed that, by increasing the residence time in the pilot stage, the goal could be achieved without compromising the NOx emissions.

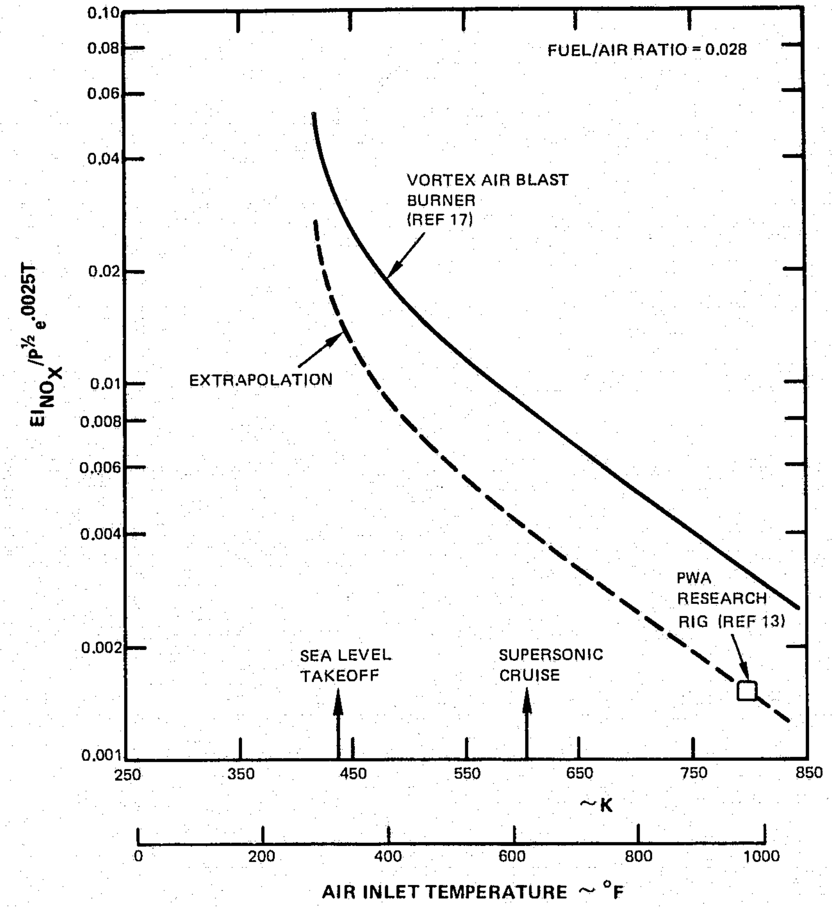


Figure 3-4 *Extrapolation of Data Obtained from a Research Premixed-Prevaporized Burner to Low Inlet Temperatures*

The final dual stage pilot evaluated was the piloted radial V-gutter concept shown in Figure 3-5. It combined a small pilot burner with V-gutter flameholders arranged on both sides of the pilot in a radial array. The pilot is sized for ignition at a duct fuel-air ratio of 0.002 by introducing 6 percent of the duct flow in the low velocity combustion zone. At duct fuel-air ratios above 0.006, the pilot operates at stoichiometric proportions and the remaining fuel is injected upstream of the radial V-gutters. The hot gases from the pilot mix with the flow in the wakes of the V-gutters to maintain continuous ignition and stability in the high velocity stream. Several radial stages of fuel injection would be required for stable operation and smooth modulation over the entire range of fuel-air ratio but only those immediately adjacent to the pilot would be operational at supersonic cruise.

The emissions characteristics of this configuration were based on data from the high temperature rise combustor of Reference 14 for the pilot and on the analysis of data from the radial-axial staged burner of Reference 8 for the V-gutters. The results indicated that the high temperatures in the pilot produce a significant NOx contribution from this component. However, partial vitiation and short residence times in the recirculation zones behind the radial V-gutters produced a low NOx contribution. By sizing the pilot as small as possible, its high NOx signature is reduced and a low overall NOx level can be maintained.

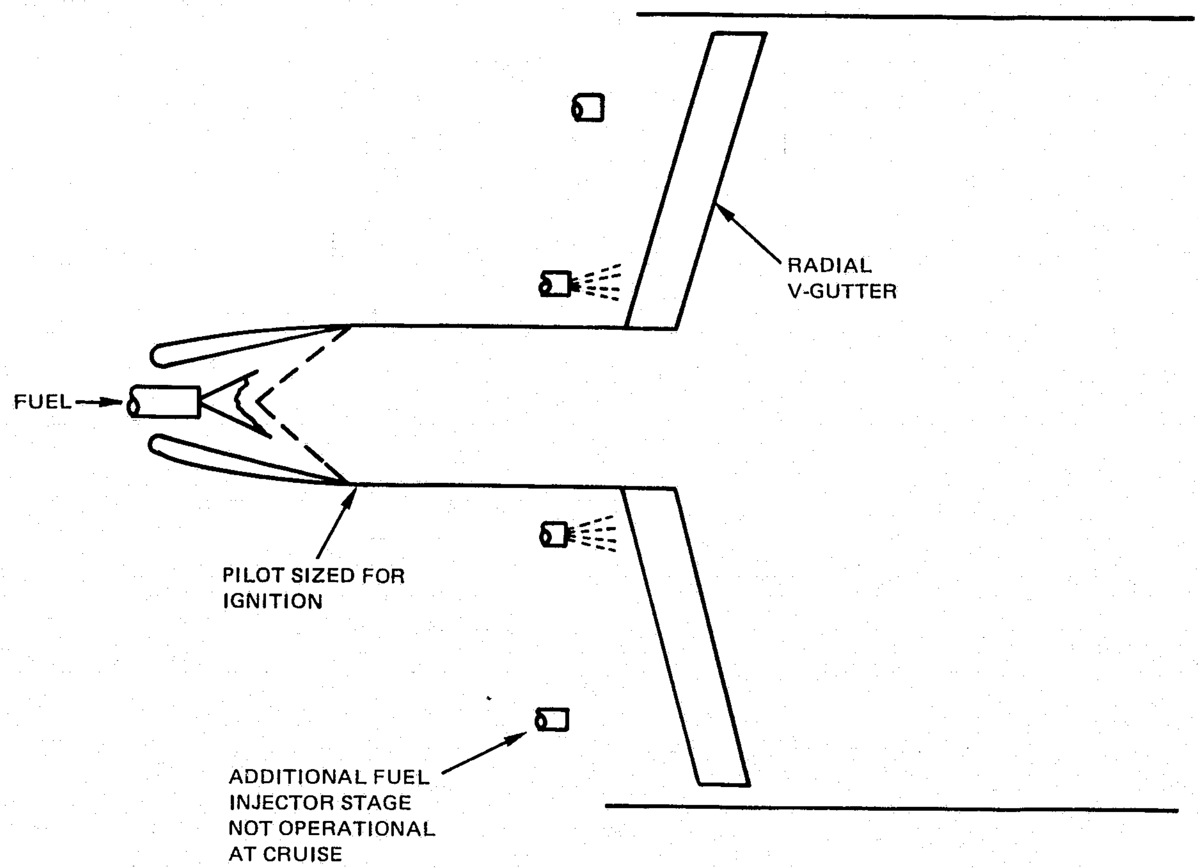


Figure 3-5 Piloted Radial V-Gutter

Part power operation, such as at the cruise fuel-air ratio of 0.013, requires burning over only about 15 percent of the radial length of the V-gutter. Under these conditions the reaction zones behind the V-gutters are in direct contact with the cold bypass air. This causes considerable quenching and high levels of CO. Data obtained from duct burners substantiate combustion efficiencies comparable with the projected CO EI levels of about 250 or more at this operating condition.

In general, the results of the study of dual stage pilots indicate that the use of these concepts eliminate the ignition fuel-air ratio limitations of the single stage configurations. The prechamber vorbix pilot is shown to offer some cruise NOx emissions reduction over single stage pilots. Low Nox levels might also be attainable with a piloted radial V-gutter concept if the low combustion efficiencies produced by peripheral quenching can be eliminated. The dual stage premixed pilot is the only concept projected to meet the cruise NOx emissions goal, and the realization of this goal will require extensive efforts to achieve a true homogeneous fuel-air mixture in this configuration.

3.2.3 High Power Stage

Six different combustor concepts were evaluated as candidates for the high power stage of the duct burner. Emissions data from representative combustion devices were scaled to the sea level takeoff operating condition of the duct burner to evaluate the relative performance of these concepts and the results are presented on Table 3-III. Since these concepts represent the second stage of various experimental combustors, the overall emissions projections could be biased by the performance of the pilot stage. To eliminate this effect, the procedures of Section 2.1 were used to differentiate between the emissions contribution from the pilot and secondary stage of the reference burner. The projections of the overall emissions at sea level takeoff were made on the assumption of the appropriate, and in most cases, a common pilot configuration as shown on this table.

TABLE 3-III

PERFORMANCE OF HIGH POWER STAGES AT SEA LEVEL TAKEOFF

Configuration	Radial V-Gutter	Vorbix	Ram Induction	Swirl Can	Premixed Prevaporized	Previtiated Mixer
NO_X EI	0.81	1.5	1.4*	1.4	1.2	1.6
CO EI	150 - 30	13.2	13.2	13.2	< 13	13.2
Combustion Efficiency %	95 - 99	> 99	> 99	> 99	> 99	> 99
Pilot Configuration Assumed	Low Airflow Pilot	Vorbix	Vorbix	Vorbix	Premixed Prevaporized	Vorbix

*high uncertainty

A description of each of the high power stage concepts is provided in the following paragraphs. The data base used for the emissions projections is indicated and unique features and problems associated with the particular concept are identified.

Radial V-Gutter Concept

As shown on Figure 3-5, this is the same concept evaluated as a dual stage pilot, and the same data base was employed to project its emissions characteristics at takeoff. The major difference from cruise operation is that combustion is occurring over the entire radial span of the V-gutters at the higher augmentation levels. This eliminates the peripheral quenching that occurs at cruise and permits achieving higher combustion efficiency. Projections based on emissions data indicate a combustion efficiency of about 95 percent; however, assuming the terminal stages of combustion are not mixing limited, the efficiency could be increased by increasing the residence time. Since a larger fraction of the fuel is burned at the V-gutters where the projected NOx production is low, the overall NOx emissions of this configuration improve as the fuel-air ratio is increased.

Vorbix Concept

The vorbix high power stage could be either of the two types shown in Figure 3-6. The intent of this concept is to produce discrete vortices that increase the rate of mixing and reduce the residence time at high temperatures. Both swirler tubes and delta wings operating at an angle of attack can be employed to generate the vortices. The delta wings have the additional advantage of providing pressure loss control and variable flow splits by varying the incidence angle of the wings. In both concepts, fuel is injected into the pilot exhaust to be vaporized before mixing with the swirling combustion air. While the swirler tube concept is preferred because it offers more positive aerodynamic control, limited testing in a simulated mixed flow turbofan augmentor (Ref. 11), indicates that with proper design, comparable levels of mixing are possible with the delta wing configuration. The emissions characteristics of this configuration were projected from the data of Reference 5 and imply an optimum fuel flow split between the pilot and high power stages.

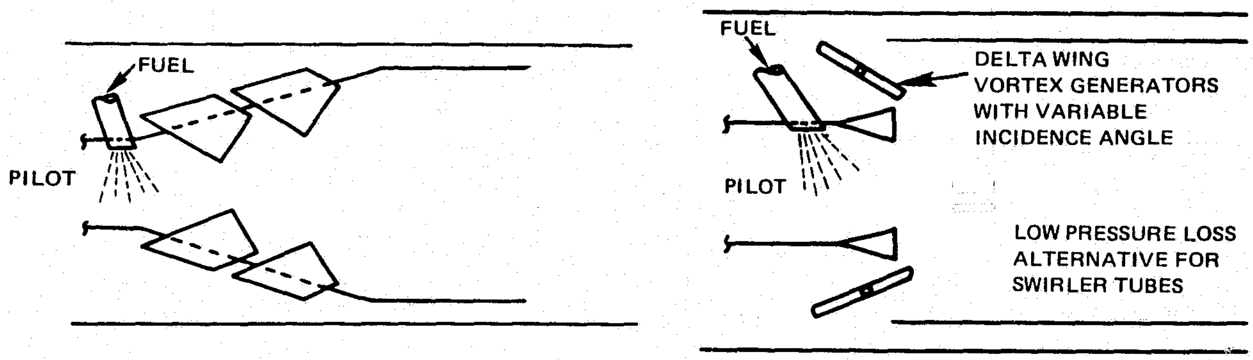


Figure 3-6 Vorbix High Power Stage Concepts

Ram Induction Concept

The ram induction concept shown on Figure 3-7 induces mixing with ram air chutes, using a high pressure drop and flow deflection vanes to enhance the mixing process. Fuel is injected into the air as it enters the chute. Some difficulty was encountered in establishing a valid data base for this concept. While ram induction has been employed to enhance mixing in experimental combustors (Ref. 16) and in a duct burner (Ref. 10), it was not used in conjunction with staged fuel injection. Reference 9 reports the results of tests conducted on a single stage burner that was modified to permit the introduction of partially premixed air and fuel through tubes protruding through the walls of the burner. While the premixing accomplished in the tubes was comparable to that anticipated in a ram induction duct burner, it was suspected that the tubes were not located sufficiently far downstream to accomplish staged burning because the NOx emissions projected from this data was more than double that predicted for the other concepts. The projections shown on Table 3-III are based on data from the configuration of Reference 18. This burner employed splash plates over combustion air holes in the liner to deflect externally atomized fuel through the aperture with the combustion air. However, this combustor was tested over a limited range of fuel-air ratios and was found very sensitive to inlet temperature. Consequently, the confidence in the projected emissions characteristics of this concept is low.

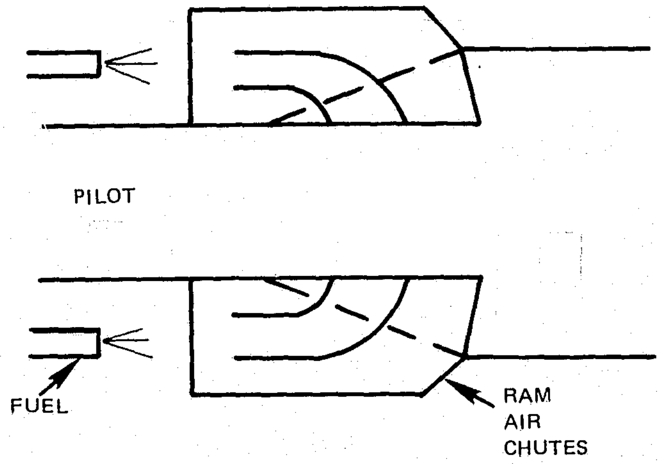


Figure 3-7 Ram Induction High Power Stage

Swirl Can Concept

The swirl can concept, shown on Figure 3-8 employs an array of tubes containing swirl vanes. Fuel is injected into the upstream end of the tube and an external baffle is fit around the downstream end to serve as a flameholder. The fuel is aerated from the swirler vane and tube surfaces and the swirling motion stabilizes the combustion zone downstream of the module. Experimental main burners, employing staged arrays of these swirl tubes, have been evaluated under the Experimental Clean Combustor Program (Ref. 5 and 8) and in other programs (Ref. 14). The data of Reference 14 was obtained at fuel-air ratios comparable to sea level takeoff operation of the duct burner and was used as the reference for projecting the emissions characteristics of this concept.

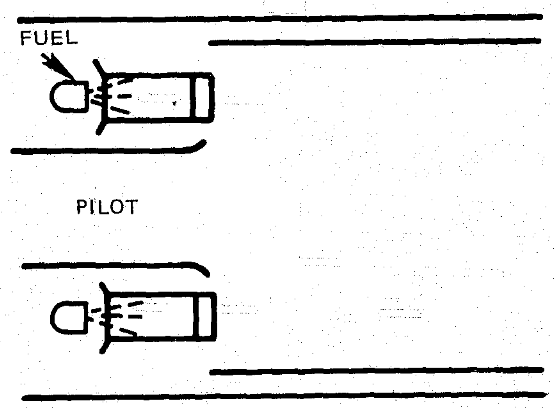


Figure 3-8 Swirl Can High Power Stage

Premixed Concept

As shown on Figure 3-9, when this concept is employed in the high power stage, the premixing passages extend axially upstream adjacent to the combustion zone of the pilot stage. In projecting the emissions characteristics at sea level takeoff it was assumed that a premixed pilot stage was employed. Minimum NOx emissions would be achieved when both stages are operated at the same equivalence ratio. The tabulated results are based on the same extrapolation of research combustor rig data used to evaluate the premixed pilot burner concept and, consequently, reflect the same assumptions with regard to the homogeneity of the fuel-air mixture prepared in the premixing passages. However, the low inlet air temperature at sea level takeoff, leads to concern over the accuracy of this extrapolation of the research burner data. At supersonic cruise, the 604K(628°F) duct air temperature was above the 560K (550°F) nominal vaporization temperature of Jet A fuel, whereas the air temperature is only 439K(330°F) at takeoff. Even if the fuel were vaporized prior to being introduced into the premixing passage, it would be liable to recondense on contact with the cold air to produce a droplet rather than vapor phase combustion process with deteriorated emissions. While the extent of fuel condensation and the associated emissions penalties is unknown, this potential limitation introduces some uncertainty regarding the projected emissions levels.

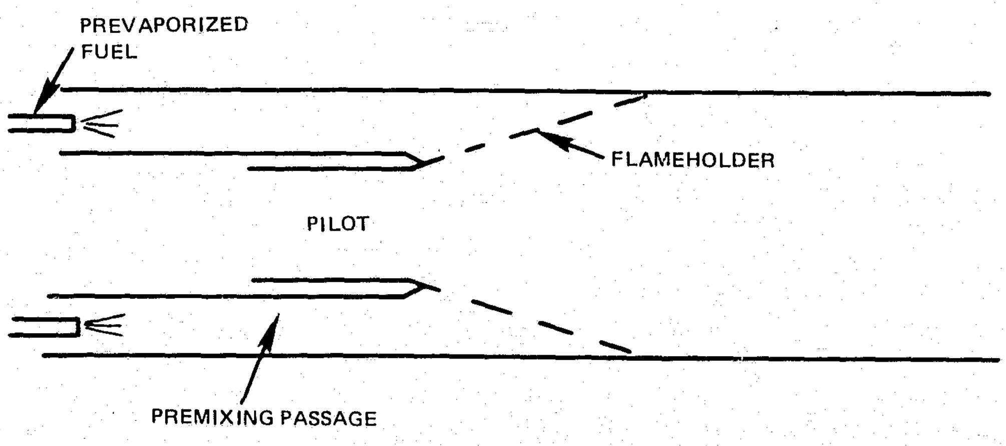


Figure 3-9 Premixed High Power Stage

Previtiated Concept

It is recognized that previtiation of the inlet air to reduce its oxygen content also reduces flame temperature and inhibits the formation of oxides of nitrogen. A dual combustor system, such as that shown on Figure 3-10, could be employed to exploit this concept. The upstream burner could serve as the sole heat source at low duct fuel-air ratios and as the previtiator for the downstream burner at the high augmentation levels.

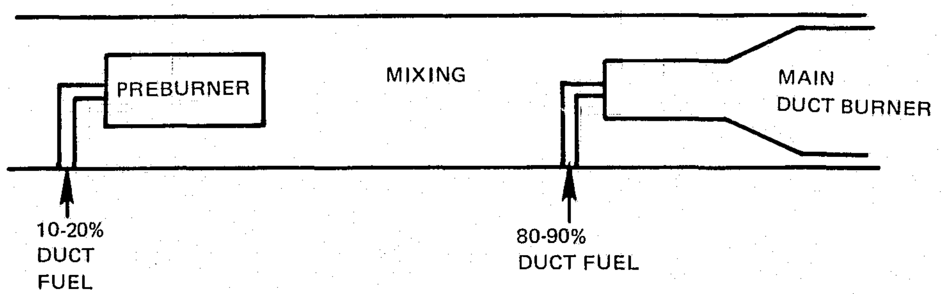


Figure 3-10 Previtiated Duct Burner Concept

This concept was analyzed from data obtained on an experimental burner that was tested with both an indirect fired heat exchanger and with a heater burner in the inlet air duct. Comparison of the emissions characteristics of the experimental burner with each inlet air heating source provided the increments in emissions produced by the previtiation. A comparison of the projected emissions with those of the vorbix high power stage with the same assumed pilot stage indicates that the NO_X emissions actually deteriorated when the high power "stage" was uncoupled from the pilot stage in this manner because of the higher effective inlet temperature to that stage.

In general, the results of the screening of the candidate high power stages reveal very little difference in the projected emissions characteristics despite the use of widely varying fuel injection, preparation and mixing techniques. Only the radial V-gutter concept, which has perhaps the least control over fuel preparation and dispersion and the weakest mixing modes, shows any substantial advantage in terms of NO_X emissions. Perhaps the reason behind this apparent insensitivity to parameters that are generally associated with emissions control – such as mixture homogeneity and mixing – is the relative high equivalence ratio of about 0.65 at which the duct burner operates at takeoff.

3.2.4 CONCEPTUAL DEFINITION OF DUCT BURNERS

Based on the results of the screening of the candidate pilot and high power stages, a number of conceptual duct burners were defined for further evaluation in the aerothermal design phase of the study. These concepts were selected with the objective of synthesizing duct burners that offered the best low emissions and performance potential while also providing comparisons that would reveal the general problems associated with low emissions duct burners.

Following are the selected concepts and the rationale for their continued evaluation:

- Premixed-Prevaporized

  The premixed-prevaporized concept has potential for the lowest emissions, particularly at supersonic cruise. Fuel and air heating concepts are investigated to determine the potential for maintaining vaporized fuel in the cold duct air.

- Piloted Radial V-Gutter

  The piloted radial V-gutter system is the simplest mechanical arrangement for a duct burner. It has potential for meeting the emissions goals at high power, although both combustion and thrust efficiency are severely limited at part power operation. It serves as a low cost and weight reference for comparison to the other concepts.

- Prechamber Vorbix Pilot with Vorbix High Power Stage

  The vorbix combustor concept has been evaluated extensively under the NASA/P&WA Experimental Clean Combustor Program and other programs. This concept offers the best substantiated technology and appears capable of meeting all operational requirements while providing moderate emissions levels.

- Single Stage Pilot with Vorbix High Power Stage

  This configuration would be simpler than the above combustor and would allow studying the effect of prechamber and ignition problems with otherwise well substantiated concepts.

- Piloted Ram Induction Burner

  Evaluation of this concept and comparison with those above would reveal the effect of high power stage configuration on the overall duct burner geometry and aerothermal definition.

- Variable Geometry Duct Burners

  Variable geometry duct burners could offer advantages in emissions or performance or provide more compact or simpler burner configurations. Evaluation of variable geometry premixed and vorbix concepts and comparison with their fixed geometry counter parts would provide an assessment of the merits of variable geometry.

3.3 AEROTHERMAL DESIGN STUDIES

3.3.1 Introduction

The objective of the aerothermal design studies was to define the geometry and major details of the configuration of duct burners based on the combustor concepts evaluated and selected in Section 3.2. Following this definition phase, estimates were made of the performance and emissions characteristics of the resultant configurations. This data was used to assess the impact of each duct burner candidate on the overall engine weight and size as well as on the engine performance over the entire mission.

Figure 3-11 shows the configuration of the fan duct in the VSCE-502B engine used as a base for this study. Preliminary estimates of the performance and size of a low emissions duct burner, reported in Reference 1, had been used to establish the geometry of the fan duct and the goal for sizing the candidate duct burners in this study was to be compatible with this envelope. The fan duct incorporates a diffuser, having a net area ratio of 2.25, immediately downstream of the intermediate case. With a length to inlet radial height ratio of 7.1, the data of Reference 28 indicate this diffuser will operate in an unstalled mode if the flow is reasonably uniform at the fan discharge. Two sets of struts span the diffuser, with the larger set carrying fuel, oil, bearing compartment scavenge and actuator service lines to the core engine. The second set, located near the diffuser exit, support the rear of the core engine from an engine mount on the outer fan case. For the present study, it was assumed that the combustion zones of the duct burner would be located downstream of the diffuser exit plane and that any components that extended into the diffuser, such as flow splitters, burner hoods, or premixing passages would be segmented around or integrated with these struts.

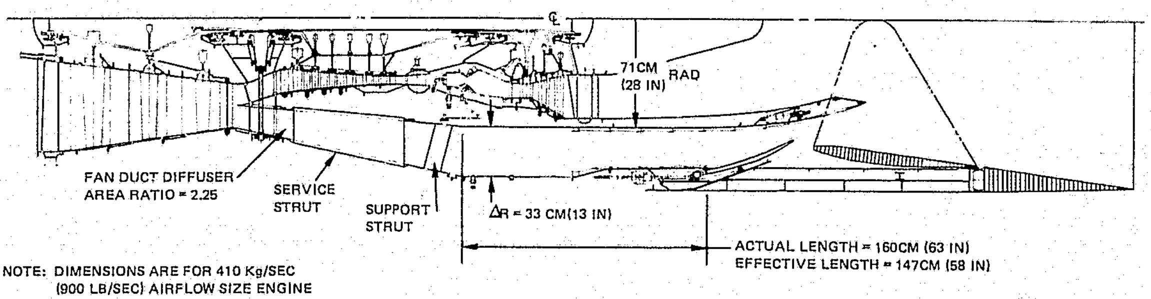


Figure 3-11 Cross Section of the VSCE-502B Engine Showing Fan Duct Geometry

At the front end of the duct burner, the fan duct has a radial height of 33 cm (13 inches) and inner and outer radii of 71 cm and 104 cm (28 and 41 inches), respectively. The reference Mach numbers of Table 1-I are based on the unblocked flow in this part of the fan duct. Further downstream the duct height is reduced to 26.4 cm (10.4 inches) to permit integration of the exhaust nozzle supports and flap actuators without increasing the diameter of the nacelle. The above cited preliminary studies indicated that the duct burner would require a length of 160 cm (63 inches) beyond the diffuser exit plane. Since the fan duct length dictates the overall length of the engine, there was a strong incentive in this study to minimize the length of the duct burner. Because of the core engine support requirements, there is little benefit to be derived in reducing the length of the diffuser and advanced technology concepts, such as boundary layer bleed, would find application primarily in improving the performance rather than the length of this component.

Since the region of constricted duct height for nozzle support and actuator integration is located at the rear of the high power stage, downstream of the more critical initial mixing and reaction zones in this stage, the length in this region is dictated primarily by residence time requirements to achieve the combustion and thrust efficiency goals. To simplify the aerothermal design studies, the fan duct was assumed to have a constant radial height over the entire length of the duct burner. Duct burner lengths defined on this basis are referred to as "effective lengths". As will be shown in Section 3.4, where the problems of nozzle integration are discussed in greater detail, the adjustment from effective to actual length is accomplished by maintaining the desired increment of residence time in the region of constricted duct height. In the VSCE-502B configuration of Figure 3-11, the difference between effective and actual duct burner length amounts to 12.7 cm (5 inches).

With regard to the emissions characteristics of the candidate duct burners, the major effort in the aerothermal design phase involved establishing configurations that were compatible with the data base for the particular concept as defined in Section 3.2. Parameters such as local equivalence ratios and zone residence times were maintained consistent with the analysis of that section. Deviations in other factors, such as combustor geometric proportions, fuel source density, liner pressure drops and reference velocities, from those of the reference combustor employed in the data base for projecting the emissions characteristics in Section 3.2 were to be minimal to maintain confidence in those emissions projections.

An additional consideration in the aerothermal design beyond the geometric sizing and emissions constraints was the performance goals for the duct burner. As shown on Table 1-II these included an overall fan duct total pressure loss of 6.5 percent and a thrust efficiency of 94.5 percent at the supersonic cruise operating condition. While no specific goal levels were established for these parameters at other operating conditions, it was expected that the performance at these conditions would be consistent with the goals established at supersonic cruise when the differences in duct burner fuel air ratio and duct Mach number are recognized.

The high supersonic cruise thrust efficiency goal presents a coupling of performance and emissions goals. Decrements in thrust efficiency arise from two causes: inadequate chemical combustion efficiency and nonuniformity of the exit temperature distribution caused by inadequate mixing or the accumulation of liner cooling air at the peripheries of the exhaust flow. Even with the mixing and liner/nozzle flap cooling influences minimized by aggressive design concepts, experience indicates that the emission goal level of a chemical combustion efficiency of 99 percent must be met at supersonic cruise conditions if the thrust efficiency goal is to be achieved.

The general approach employed in most of the configurations to achieve low emissions and high thrust efficiency at supersonic cruise involves sizing the pilot stage to have adequate residence time to achieve the 99 percent chemical combustion efficiency goal within the pilot stage proper. A high power stage that produces rapid and intense mixing of the pilot exhaust flow and the bypass air is also required. This accomplishes the dual purposes of generating the mixing necessary to achieve the uniform exhaust flow consistent with high thrust efficiency while it also quenches the pilot exhaust gases to suppress residual NOx production.

In assessing the total pressure loss across the fan duct it was assumed that a loss of 2 percent of the fan discharge total pressure was incurred in the diffuser at supersonic cruise and consequently the overall duct total pressure loss goal of 6.5 percent could be satisfied with a 4.5 percent loss across the duct burner at this flight condition. The constraint of minimal engine frontal area generally leads to the need for combustor reference velocities and shroud Mach numbers considerably higher than those encountered in main burners, and particularly at the high fuel-air ratios associated with transonic climb and sea level takeoff operation, can lead to substantial momentum total pressure losses associated with mixing and heat release processes. This leads to strong interdependence between pressure loss, duct radial height and reference velocity in the duct burner. This effect is demonstrated on Figure 3-12 which shows the variation of the momentum total pressure loss and the required fan duct height with the reference velocity in the pilot and high power stage of a representative low emissions duct burner. These characteristics were generated with the following assumptions; the pilot stage airflow was 45 percent of the duct flow which is consistent with the equivalence ratios necessary to produce the 1600 to 1700K (2400 to 2600°F) temperature environment desired in the rear of the pilot stage; the pilot burner shroud Mach number was 0.20; 90 percent of the duct flow passed through the high power stage at sea level takeoff and the liners of this stage were located 2.8 cm (1.5 inches) from the fan duct wall. The results of this analysis indicate that the pilot stage of the burner is compatible with the 33 cm (13 inch) radial duct height goal and a relatively small, 0.5 percent, heat addition momentum loss will be incurred if this stage is designed with a reference velocity of about 46m/sec (150 ft/sec). Since this is only somewhat higher than the 30 to 37 m/sec (100 to 120 ft/sec) reference velocity at which most of the main burner emissions experience has been obtained, a reasonable level of confidence in the projected emissions characteristics of these pilot stages is expected. With a momentum total pressure loss of only 0.5 percent, the supersonic cruise pressure loss goal can be satisfied with a 4.0 percent pressure drop across the liners of the pilot stage - assuming effective recovery of the dynamic pressure of the diffuser exit flow. Since the liner pressure drop establishes the available turbulent energy for mixing the reactants inside the combustor (Ref. 29 and 30), the high liner loss in the pilot stage also implies the capability of achieving mixing comparable to that accomplished in the data base combustors.

The situation is different in the high power stage, in that a duct height of the order of 50 cm (20 inches) would be required to maintain reference velocities comparable to experimental low emissions main burners. This would cause intolerable increases in engine diameter, and if the 33 cm (13 inch) duct height goal is to be achieved, the high power stages must be designed to reference velocities about double those of main burner experience. Confidence in the projected emissions at high power levels is compromised to some extent by this deviation.

At the high power stage reference velocity levels necessary to achieve the duct height goal, the curve of Figure 3-12 indicate the momentum total pressure loss across the duct burner due to heat addition is about 7.1 percent at sea level takeoff. The 4.0 percent liner pressure drop previously established for the pilot stage at supersonic cruise increases to about 7.0 percent at sea level takeoff because of the 33 percent increase in duct reference Mach number at this operating condition. Consequently, overall duct burner total pressure loss of the magnitude of 14 percent at sea level takeoff can be anticipated with duct burners that reflect this sizing approach.

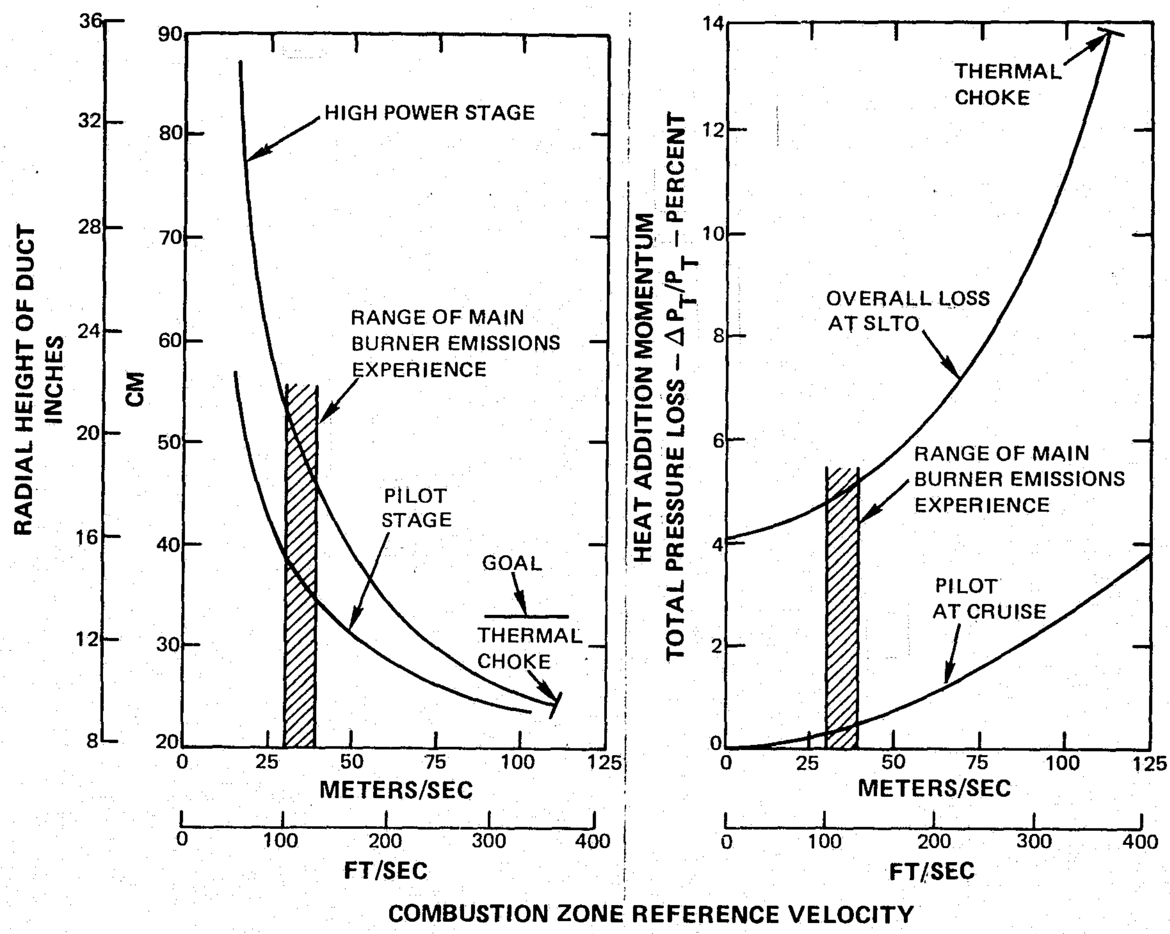


Figure 3-12 Effect of Reference Velocity on Momentum Pressure Loss and Radial Duct Height

The aerothermal definition of the duct burner concepts required some compromising between the size, emissions and performance goals. The general approach followed in these studies involved designing to the emissions and performance goals while accepting compromises in the size and complexity of the duct burner. (The singular exception was the above cited acceptance of reference velocities in the high power stage that were about double those of low emissions main burner experience to avoid excessive radial duct heights.) Following the completion of the aerothermal design, studies were conducted to assess the impact of the various compromises that had been incorporated in these definitions on the overall performance of the engine. The flight profile used to evaluate the study concepts was a nominal all supersonic mission which allocates the largest amount of its fuel to the cruise leg and in so doing, emphasizes the cruise performance of the engine. Because of this performance emphasis, small penalties to engine length and diameter can be acceptable if they lead to improvements in pressure loss or thrust efficiency levels.

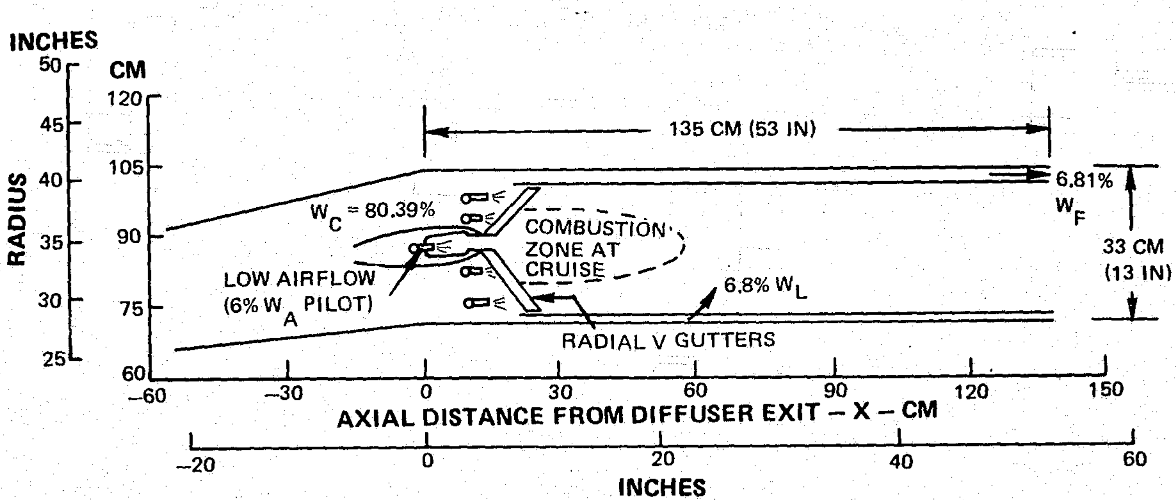
Because of the approach followed in the aerothermal design, most of the duct burner concepts were found to meet or exceed the cruise performance goals of 94.5 percent thrust efficiency and 4.5 percent total pressure loss and the size and complexity of the systems became the more significant screening parameters. An evaluation of the concepts was conducted on the basis of overall engine weight which was impacted by variations in duct heater length, radial height, and complexity. Specifically, increased length requires a longer engine case and burner liner. Radial height increases caused larger case weights, and when the diameter increase extended back to the nozzle interface, the nozzle size and weight also increased. Complex burners generally had heavier fuel systems. Since the weight of the duct burner is a small fraction of the overall engine weight, small changes in burner geometry had only minor effects on the total system weight.

Configurations which required an increase in the radial height of the fan duct also produced an increase in nacelle drag. The combined effect of drag, engine weight increments and increments in duct burner performance were established in terms of the net change in takeoff gross weight of the aircraft which reflected these deficiencies in terms of excess fuel requirements over the nominal mission.

The following parts of this section present the results of the aerothermal definition of each of the duct burner concepts identified in Section 3.2. The unique features and impact of each concept on the overall performance of the engine are discussed. Several fixed geometry concepts are reviewed first to demonstrate the effect of combustor concept and the use of prechamber pilot stages on the aerothermal definition. Premixed-prevaporized combustor concepts are discussed in a separate sub-section that is prefaced with an evaluation of preheating systems required with this type of combustor. Variable geometry concepts are also discussed as a group and the conclusions regarding the adaptability of variable geometry are generalized to include aspects other than low emissions capabilities. This section is completed with a summary of the results of this phase of the study and the screening of the configurations to select those that were evaluated in further detail in the subsequent mechanical design study phase.

3.3.2 Piloted Radial V-Gutter Concept

Figure 3-13 shows the aerothermal definition of a duct burner employing the piloted radial V-gutter combustor concept. The airflow distribution shown on this and subsequent definitions of other concepts is that computed for the supersonic cruise operating condition. The pilot is sized to accommodate stoichiometric combustion in six percent of the fan duct airflow and achieves 97 percent combustion efficiency within the pilot combustion chamber proper at the sea level takeoff operating condition. The low airflow pilot is employed to permit lighting at a 0.002 duct stream fuel-air ratio and to sustain combustion in the wake of the radial V-gutter flameholders. A hood is employed over the pilot for effective recovery of the residual dynamic pressure at the diffuser exit, permitting the pilot to function with the intended 4 percent liner pressure drop at cruise. Feeding the pilot combustion and cooling air apertures from the hood also permits higher Mach numbers in the bypass passages. In this particular configuration, the flameholder approach Mach numbers are 0.15 and 0.20 at supersonic cruise and sea level takeoff respectively. An array of canted narrow V-gutters are employed, with the blockage being established to produce the desired 4 percent cold flow pressure loss at supersonic cruise. In defining the flameholder dimensions several factors favor a large number of narrow flameholders. These include: combustion efficiency - burner length trades bias the selection toward a greater number of flameholders to minimize the circumferential distance over which the reaction zones must spread before merging; it is also suspected that the majority of the NO_X generation in the V-gutter combustion regions occurs in the recirculation zones behind the flameholder and narrower flameholders would reduce the size and consequently, the residence time in these zones. However, stability requirements dictate the minimum flameholder width and satisfying this criteria at a minimum pressure condition during the transonic acceleration became the dominant factor in establishing the geometry shown.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	1.18	0.40	>0.81
η_C - %	91	99	99
η_T - %	85	–	91.5
$\Delta P_{T\ HOT}$ - %	4.5	–	13.2
SIZE PENALTY	NONE		
WEIGHT PENALTY	– 118 KG (–260 LB)		
Δ TOGW	6.35%		

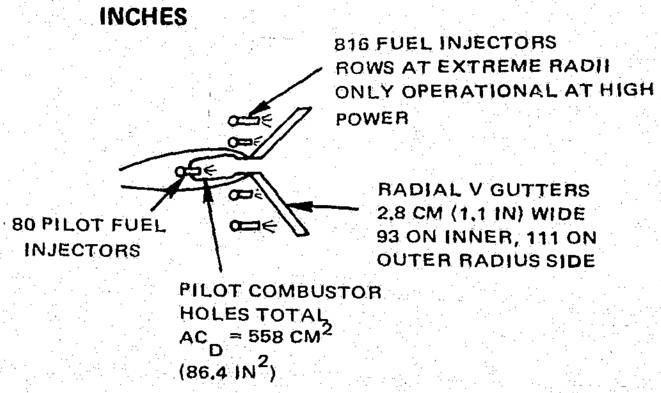


Figure 3-13 Aerothermal Definition of the Piloted Radial V-Gutter Duct Burner

A three stage fuel injection system is required with one stage supplying the pilot burner. The second stage having spray rings immediately inboard and outboard of the pilot, injects fuel into the bypass air adjacent to the pilot and is operational at low duct fuel-air ratio conditions, including supersonic cruise. At high duct fuel-air ratio points, such a transonic climb and sea level takeoff, the third stage feeding spray rings near the extreme radii of the V-gutters are activated.

The combustion zone downstream of the V-gutters has a constant radial height and is enclosed by louvered liners. The liner and nozzle flap cooling air quantities shown on Figure 3-13 and on the remaining configurations in this section were defined on the basis of scaling cooling flow per unit surface area from a previous duct burner design (Ref. 10) according to the procedures of Section 2.6. These techniques recognize differences in gas temperatures and reference velocities between these configurations. While the liner cooling flows shown occur at supersonic cruise, these flow rates will increase at higher augmentation levels because the static pressure in the downstream regions of the duct burner are depressed. At sea level takeoff augmentation levels the cooling flow in the furthest downstream louvers is expected to be about double that established at supersonic cruise. In this respect, duct burners have a self compensating cooling system with more cooling air being provided to the liner when the scrubbing gas temperatures and velocities are highest.

The projected emissions and performance characteristics, tabulated on Figure 3-13 indicate that this concept is capable of meeting the emissions goal of 99 percent chemical efficiency with NO_x emissions indices below the goal level of 1.0 at transonic climb and sea level takeoff conditions. The low NO_x emissions is attributed to the low residence time in the recirculation zones in the wakes of the V-gutters despite the high nominal equivalence ratio of 0.65 at sea level takeoff. At high duct fuel-air ratios, this configuration is operating with a relatively homogeneous initial fuel-air mixture across the entire depth of the duct and reasonably uniform exit temperature profiles consistent with high thrust efficiency levels can be anticipated.

At supersonic cruise the performance deteriorates with both the combustion and thrust efficiency substantially below the goal levels. At this condition, combustion occurs in the pilot and over part of the radial span of the V-gutters. Considerable quenching of the reactions occur near the interface between the combustion zone and the cold bypass air at the extremes of the V-gutters and causes the projected reduction in combustion efficiency. The thrust efficiency is low at cruise, not only as a consequence of the low combustion efficiency but also because of the absence of forced radial mixing modes that are necessary to mix the combustion products in the center of the duct with the cold nonvitiated air at the radial peripheries. The NO_x emissions at supersonic cruise are low, only slightly above the goal level, but the low chemical efficiency must be considered a major factor in the low NO_x levels projected.

From the point of view of compatibility of the duct burner with the engine, this concept fits readily into the prescribed duct envelope and, being lighter than the estimated low emissions duct burner, has a negative weight penalty. However, this beneficial aspect is negated by the deficiency in thrust efficiency at supersonic cruise that requires a 6.35 percent increase in aircraft takeoff gross weight because of the additional fuel that must be consumed in the cruise leg to maintain the desired thrust level.

Figure 3-14 shows a schematic illustration of a variation in the piloted radial V-gutter configuration that could enhance the combustion efficiency at cruise by isolating that part of the V-gutter airflow being vitiated from the colder bypass air. As in the configurations shown in subsequent parts of this section, the cruise combustion zone could be sized for high combustion efficiency before the combustion products are mixed with the bypass air. However, this improvement in combustion efficiency is expected to increase the cruise NO_x emissions substantially. Furthermore, this concept is still deficient in the capability to generate radial mixing modes and, even with the improvement in combustion efficiency, it is doubtful that the cruise thrust efficiency goal can be achieved. In conclusion, despite the fact that low residence times in the reaction zones behind the flameholders can have beneficial effects on NO_x emissions, the mixing limitations of radial V-gutter concepts preclude the achieving of the goal thrust efficiency levels.

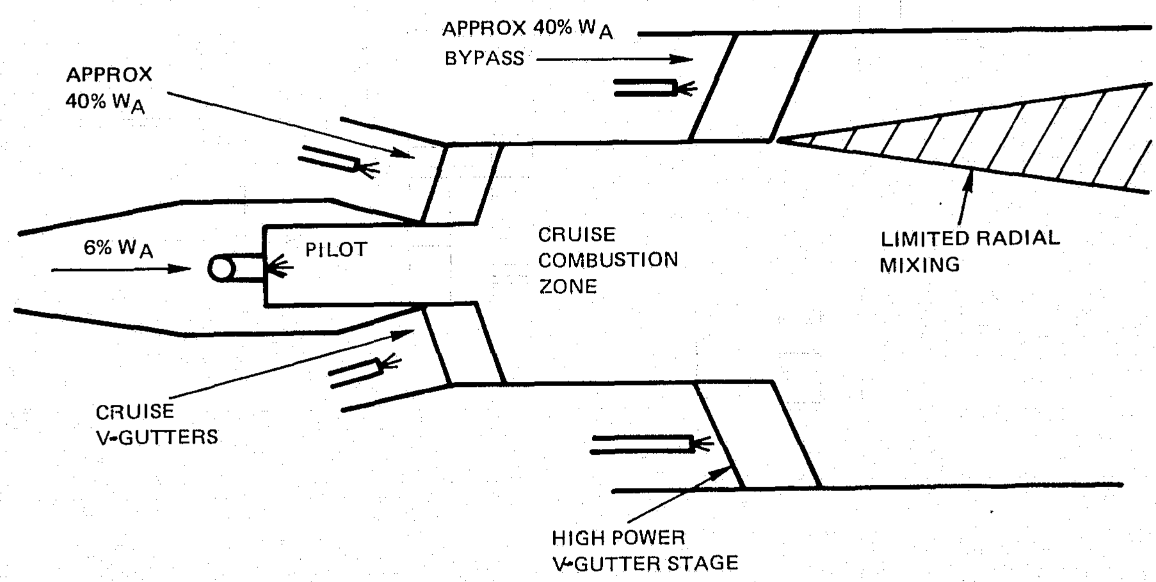
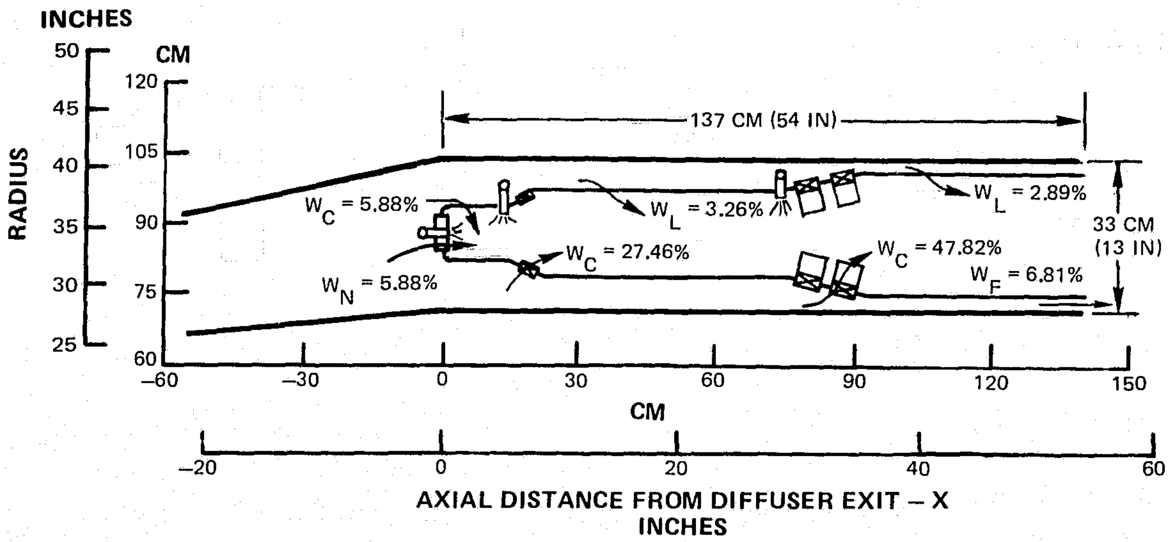


Figure 3-14 Alternative Configuration of the Radial V-Gutter Duct Burner

3.3.3 Prechamber Vorbix Concept

Figure 3-15 shows the aerothermal definition of this duct burner which employs three stages of combustion with each individual stage having an independent fuel system. A prechamber, flowing 11.76 percent of the duct air, is employed to provide the capability to light at the goal fuel-air ratio of 0.002. The combustion products from the prechamber enter a pilot secondary stage where they are mixed with additional air entering through swirler tubes. Additional fuel is injected into the prechamber combustion products and the combined prechamber and secondary stage function as a two-stage vorbix combustor as described in Section 3.2.2. The third stage incorporates similar swirler tubes for air induction and a fuel injection system at the downstream end of the pilot secondary zone. This fuel system is operational at the high fuel-air ratios associated with transonic climb and sea level takeoff and, at these conditions, the third stage also functions as a vorbix burner with the combustion products from the two preceeding stages serving as a pilot heat source.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	2.75	1.22	1.78
η_C - %	99	92.5	99
η_T - %	94.5	–	88
$\Delta P_{T\ HOT}$ - %	4.25	–	14.0
SIZE PENALTY	NONE		
WEIGHT PENALTY	105 KG (230 LB)		
Δ TOGW	0.35%		

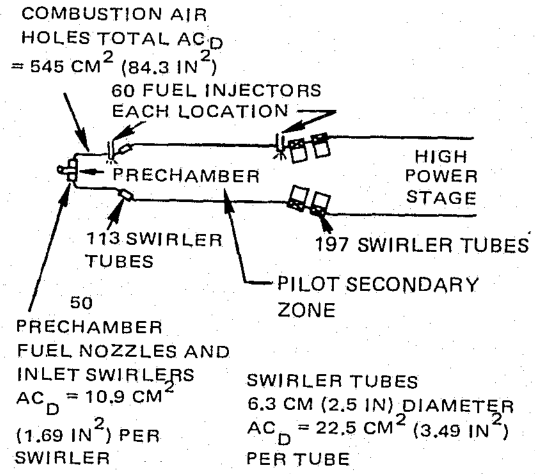


Figure 3-15 Aerothermal Definition of the Prechamber Vorbix Duct Burner

The combustion zones were sized using the procedures and requirements of Section 2.2. The prechamber has a relatively low reference velocity of 22.2 m/sec (73 ft/sec) at sea level take-off and its volume was established by the ignition requirements. The pilot secondary stage was designed with a reference velocity of 43 m/sec (140 ft/sec) at supersonic cruise to provide a configuration compatible with the 33 cm (13 inch) duct height goal. The air scheduling to this stage was established to produce gas temperatures of 1625K (2500°F) at the rear of this stage to permit continuing oxidation of carbon monoxide without encountering high NO_X generation rates. The length of this stage is dictated by the combination of the swirler tube jet penetration and mixing and the residence time required to achieve 99 percent combusion efficiency at supersonic cruise. The high power stage operates at a reference velocity of 67 m/sec (220 ft/sec) at sea level takeoff, with this level also being selected to satisfy the duct height goal. Like the pilot secondary stage, the length of this stage is dictated by jet penetration and mixing criteria and the residence time required to achieve the combustion efficiency goal at sea level takeoff.

The swirler tube and fuel source density were established on the basis of experience derived in the evaluation of vorbix combustors in the programs of References 5, 6 and 27. However, the vane angle in the swirler tubes has been reduced to increase the airflow capacity of the tubes and enhance jet penetration, at the expense of some centrifugal mixing action. While not shown on Figure 3-15, a hood would probably be required over the prechamber and the swirler tubes of the pilot secondary stage to assure adequate airflow to these components. The Mach number in the shroud surrounding the pilot secondary stage is 0.18 at supersonic cruise and 0.24 at sea level takeoff, which provides static to static pressure drops of 1.8 and 3.0 percent, respectively, across the cooling air louvers at these conditions. While these pressure drops might be adequate for the metering of cooling air, they represent only about 80 percent of the dynamic pressure of the shroud flow which violates the aspiration criteria of Section 2.3. This problem is eliminated by extending the hood enclosing the prechamber downstream over these louvers to provide a higher pressure cooling air source that is isolated from the high Mach number bypass flow. Because the cooling air requirements are low, the additional radial height of the hood does not create excessive blockage in the shrouds. While the Mach numbers will be elevated above the cited levels at the entrance to the shroud, the hood may be tapered to attach to the liner at the downstream end of the pilot. This forms a diffuser of very moderate proportions in the shroud that will minimize the losses associated with the higher entrance Mach number.

The projected emissions characteristics of this configuration at sea level takeoff and supersonic cruise are consistent with the estimates of Section 3.2 with the goal levels of combustion efficiency being attained. However, the efficiency of transonic climb is substantially below this goal. This problem is common to all of the axially staged concepts evaluated in this study and is a consequence of the kinetic analysis of the CO oxidation process that was used to predict the combustion efficiency. The high power stage was sized to provide the residence time necessary to achieve 99 percent combustion efficiency at the sea level takeoff design point. When this stage is operated at lower equivalence ratios at transonic climb (typically 0.46 as opposed to 0.61 at takeoff), this analysis indicates that the residence time is inadequate and that an additional 43 cm (17 inches) of stage length would be required to satisfy the 99 percent combustion efficiency goal. However, experience in the application of these kinetic analyses to combustion data indicates that the predictions are increasingly conservative at lower equivalence ratios because inevitable nonhomogeneity in the mixtures create rich regions with substantially higher CO oxidation rates. Consequently, while it is recognized that the combustion efficiency at transonic climb may present a problem, it does not appear that length increases of the magnitude predicted by this analysis are necessary to resolve it. For this reason, the aerothermal definitions shown in this section are all based on the sizing of the high power stage at sea level takeoff but the lower combustion efficiencies are shown at transonic climb to provide an indication of the magnitude of this problem.

The thrust efficiency of this concept is projected to be high, satisfying the performance goal, at supersonic cruise. Relative to the piloted radial V-gutter concept of Section 3.3.2, this improvement is accomplished by the isolation of the combustion regions from the bypass air and through the radial mixing accomplished in the high power stage by directing the swirler tube air jets toward the center of the duct. However, at high duct fuel-air ratios the majority of the heat release occurs in the combustion process in the high power stage. When inevitable non-uniformities in fuel distribution are superimposed on the existing mixing process the uniformity of the exit temperature distribution, and hence the thrust efficiency, is expected to deteriorate with increasing duct fuel-air ratio. This thrust efficiency – augmentation level characteristic is representative of all of the axially staged duct burners evaluated (although it is modified somewhat in the premixed-prevaporized concepts of Section 3.3.6), and is the inverse of the performance characteristic of the piloted radial V-gutter configuration of Section 3.3.2. From the point of view of compatability with the engine performance requirements, if deficiencies in thrust efficiency must be accepted in certain ranges of fuel-air ratio, the axially staged concepts offer a thrust efficiency-augmentation level characteristic superior to the V-gutter configuration.

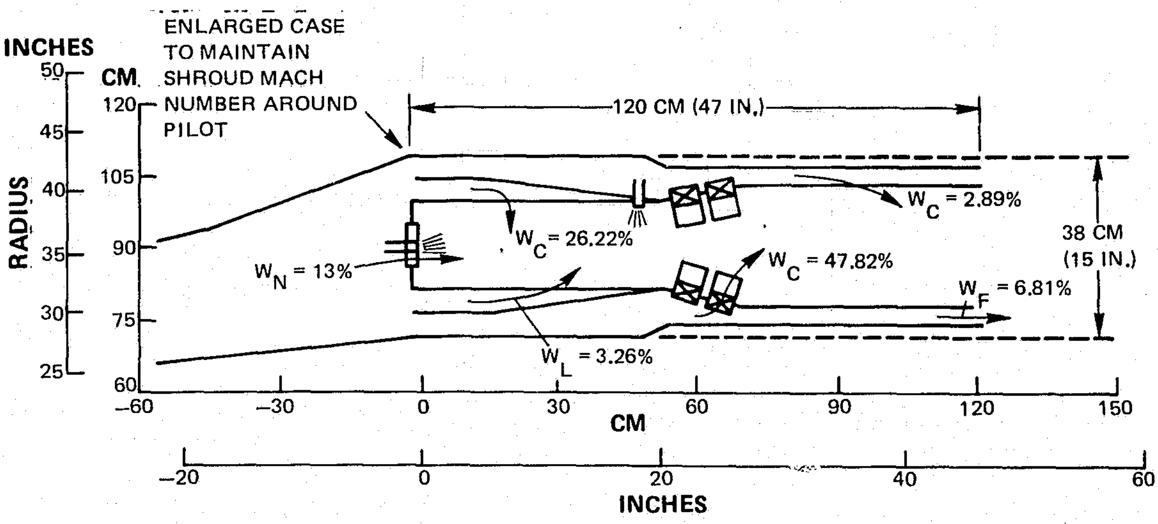
With the prechamber vorbix concept satisfying the cruise thrust efficiency and pressure loss goals, the net impact of this configuration on overall powerplant and aircraft performance is small. The duct burner fits into the proposed duct envelope. The small weight penalty associated with this configuration, relative to the preliminary design used to establish the baseline, represents more realistic definition of the duct burner in this study, particularly with regard to the number of fuel injectors and associated components, rather than an actual deficiency of this configuration relative to other concepts.

3.3.4 Single Stage Pilot Vorbix Concept

Figure 3-16 shows the aerothermal definition of this duct burner which differs from the prechamber vorbix configuration of Section 3.3.3 in the use of a single stage pilot rather than the prechamber – secondary stage pilot arrangement. The major advantage of this concept over the prechamber vorbix configuration is the use of two rather than three stages of fuel injection. The pilot is sized to produce 99 percent combustion efficiency within the pilot proper at supersonic cruise and its air scheduling and equivalence ratio – residence time history duplicates that of the experimental JT9D low emissions burner employed as a data base in the evaluation of Section 3.2. The pilot is designed with a rich (θ = 1.9) primary zone followed by a rapid quenching with dilution air jets to reduce the equivalence ratio to about 0.40 after which carbon monoxide consumption may be continued without generating excessive NO_x. The studies of Section 3.2 indicated that the emissions characteristics were insensitive to the type of fuel injector employed – aerating or pressure atomizing. However, in view of the large fuel flow turndown required on this stage, aerating injectors would be preferred. The introduction of pressure atomizing fuel nozzles would require the use of a duplex fuel system that would, in effect, negate the advantage of this concept.

Relative to the secondary zone of the pilot in the prechamber vorbix configuration, the single stage pilot is sized for a seven percent higher reference velocity; 46 m/sec (150 ft/sec) as opposed to 43 m/sec (140 ft/sec) at supersonic cruise. While a hood would also be employed over the pilot to prevent aspiration through the cooling air louvers, this concept also requires the introduction of dilution air through the pilot liner. To achieve effective penetration, these jets should be generated by the maximum available liner pressure drop. Consequently, the hood was sized to allow the combined dilution and pilot liner cooling flow to pass around the front end of the pilot. With a Mach number of 0.12 at this location at supersonic cruise, a static to static pressure drop of 3.0 percent exists across the liner and with appropriate contouring of the hood even higher liner pressure drops could be achieved at the dilution holes further downstream. However, the larger hood increases the blockage height of the pilot from 19 cm

(7.5 inches) to 28 cm (11 inches). This required an increased fan duct height to avoid excessively high shroud Mach numbers, which would produce high losses and compromised performance in the high power stage. Maintaining the shroud entrance Mach number at the levels encountered in the prechamber vorbix configuration at supersonic cruise required that the duct height be increased 5.08 cm (2 inches) to 37.8 cm (15 inches) in the region around the front end of the pilot.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	3.21	1.2	1.75
η_C-%	99	92.5	99
η_T – %	94.5	–	88
$\Delta P_{T\ HOT}$ - %	4.5	–	14.7
SIZE PENALTY	10.2 CM (4.0 IN) Δ DIA.		
WEIGHT PENALTY	123 KG (270 LB)		
Δ TOGW	0.45%		

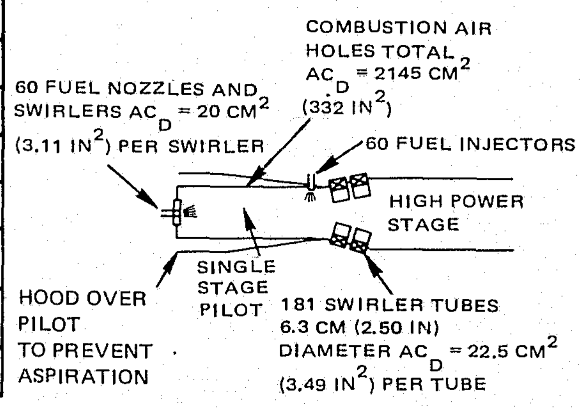


Figure 3-16 Aerothermal Definition of the Single Stage Pilot Vorbix Duct Burner

While this increase in duct height implies an increase in the net area ratio of the diffuser, there is no change in diffusion requirements and the increased exit area merely reflects increased blockage of the exit plane. The pilot hood could readily be extended into the diffuser to maintain the flow area distribution while the cases diverge to produce the required exit height, and there would be no need for a longer or a more aggressive diffuser design concept.

With a common high power stage there is little difference between the performance or combustion efficiency characteristics of this concept and the prechamber vorbix duct burner. The total pressure loss is shown to be slightly lower with the single stage pilot configuration despite both configurations being designed for the same liner pressure drop. In the context of the Rayleigh line analysis employed to define the heat addition momentum pressure loss, this result implies the losses are reduced with fewer stages of heat addition.

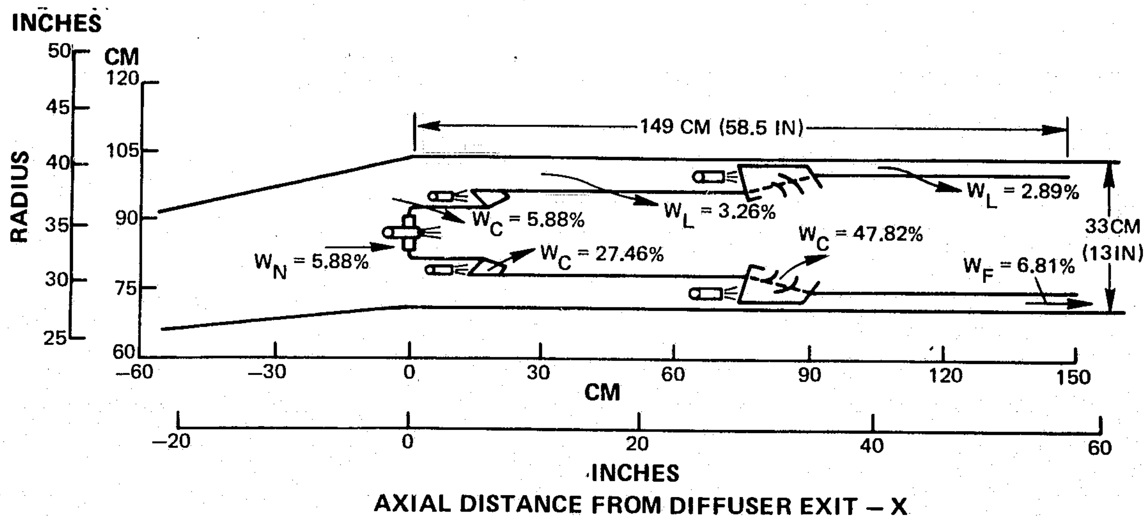
While the NO_x emissions are essentially the same as those projected for the prechamber vorbix duct burner at high fuel-air ratios, the supersonic cruise NO_x emissions are 0.46 gm/kg (lb/1000 lb) higher. This difference obviously reflects the advantage of the two stage vorbix pilot over the single stage pilot. The ignition fuel-air ratio of the single stage pilot is projected at 0.0028 as opposed to the goal of 0.002 to avoid creating excessive back pressure pulses at lightoff. While improved ignition characteristics might be achieved with development, it might also be necessary to employ sequential lighting or variable geometry air swirlers on the pilot to achieve this goal without compromising performance or emissions.

In terms of overall engine/aircraft performance this configuration is also comparable to the prechamber vorbix duct burner. The 10.2 cm (4 inch) increase in flow path diameter is restricted to the region immediately adjacent to the pilot stage and could be accommodated in front of the nozzle support structure (see Figure 3-11) to avoid an increase in overall nacelle diameter. While the number of fuel injectors and manifolds, the most significant factor in establishing the weight of the duct burner proper, are reduced in this configuration, the overall weight penalty is slightly higher because of the larger diameter of the fan duct cases.

3.3.5 Ram Induction Duct Burner Concept

Figure 3-17 shows the aerothermal definition of a three stage duct burner incorporating ram induction of the combustion air in the pilot secondary and high power stages. The configuration is similar to the prechamber vorbix concept of Section 3.3.3. Since it is designed to the same stochiometry and reference velocities as that configuration, the overall gaspath layout and dimensions are nearly identical to those of Figure 3-15. The only significant aerodynamic difference is in the use of ram air chutes rather than swirler tubes to introduce combustion air. The chutes are effective devices to recover the dynamic pressure of the shroud flow and employ turning vanes to divert the captured flow into the radial direction. With the entire total to static liner pressure drop (4 percent at supersonic cruise) available as a radially directed velocity vector, better penetration of the air jets into the combustion gas is anticipated. However, lacking the tangential component of velocity generated in the vorbix swirler tubes and the associated centrifugal mixing aspects, the small scale mixing may not be as intense. This factor is recognized in the aerothermal definition of Figure 3-17 with an additional 11.4 cm (4.5 inches) of axial length being required to accomplish mixing in the front end of the pilot secondary and the high power stages.

This configuration also employs spray rings upstream of each set of ram chutes to inject the fuel into the combustion air as it is introduced into the burner. While this leads to the need for a greater number of fuel injectors than required in the prechamber vorbix configuration, the injectors are of simpler construction and do not produce any substantial additional weight penalties.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	—	—	—
η_C - %	99	—	99
η_T - T	94.5	—	88
$\Delta P_{T\ HOT}$ - %	4.25	—	14.0
SIZE PENALTY	NONE		
WEIGHT PENALTY	105 KG (230 LB)		
Δ TOGW	0.35%		

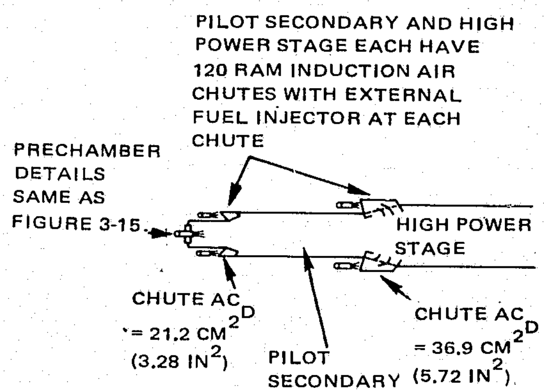


Figure 3-17 Aerothermal Definition of the Ram Induction Duct Burner

While the data base was inadequate for realistic projection of the NO_X emissions characteristics of this concept, the design is based on the same residence time criteria for CO oxidation as the vorbix configuration, and comparable combustion and thrust efficiency levels are anticipated if there are no fundamental mixing limitations. Based on studies of the capability of premixing of the fuel in the premixed-prevaporized duct burner concepts of Section 3.3.6, the fuel preparation in the air chutes is expected to be poor. Consequently, the ram induction duct burner would be expected to produce considerably higher NO_X emissions than a premixed system. If the fuel injection system were modified to permit flash vaporization of the fuel in the combustion products of the upstream stage, the ram induction configuration would differ from the prechamber vorbix concept only in the mode of mixing. In this case the NO_X emissions might be expected to approach those of the latter.

The analysis of Section 3.2 indicated that, with the exception of the radial V-gutter and premixed – prevaporized burner concepts, the candidate high power stage configurations differed primarily in the fuel preparation and mixing techniques employed, and no substantial advantages in sea level takeoff emissions were established. The aerothermal design study of the ram induction burner has demonstrated that, once a three stage configuration is established to satisfy ignition, supersonic cruise and takeoff design points and reference velocities are selected to be compatable with the goal duct height, the general layout of the duct burner is well defined. Consequently, should future technology programs define concepts that offer superior fuel preparation or mixing characteristics, their incorporation into the duct burner at that time would be relatively straightforward and have minimal impact on the overall configuration.

3.3.6 Premix-Prevaporized Duct Burner Concepts

The premixed-prevaporized combustor concepts identified in Section 3.2 differ from premixed burner technology of Reference 5 in that the lower pressures and temperatures in the fan duct produce longer ignition delay times, allowing more time to prevaporize the fuel and produce a more homogeneous fuel-air mixture prior to the initiation of combustion. At the pressure levels at which the duct burner operates, the nominal boiling temperatures of Jet A fuel is about 560K (550°F), whereas the duct air temperature at the supersonic cruise operating condition is 604K (628°F). Theoretically, the duct air has more than adequate thermal energy to vaporize cold liquid fuel producing a mixture having a bulk temperature above the boiling point. However, considering the problems involved in dispersing a liquid fuel uniformly, it appears that a more homogeneous and completely vaporized fuel distribution would be achieved if the fuel were vaporized prior to being injected into the premixing passages. At the lower flight Mach numbers encountered at sea level takeoff and transonic climb conditions, an additional problem occurs because the duct air temperature is only 438 to 450K (330 to 350°F). Even with fuel preheating to temperatures in excess of 560K (550°F), the prevaporized fuel is liable to condense when injected into the colder air. The extent of recondensation and the associated increments in emissions performance are uncertain, but if they are large, it would be desirable to preheat the duct air prior to combustion at these operating conditions. The additional fuel vapor fraction achieved with air preheating could offset the effect of the increased inlet temperature on NO_x emissions.

Preheating System Definition

The design studies of premixed-prevaporized duct burners were initiated with an evaluation of various fuel and air preheating concepts to determine those feasible from the point of view of heat transfer capability, system size and weight, and engine performance increments. These concepts were then incorporated into the subsequent definition of the duct burners proper. For the purpose of these feasibility studies, the fuel supply temperature was assumed to be 294K (70°F), i.e., there was negligible aerodynamic heating of the fuel tanks and the duct burner fuel was not used as a sink for airframe or engine heat rejection.

Figure 3-18 shows a schematic view of a regenerative fuel heating system in which the fuel is circulated through tubes that form the walls of the duct burner combustion zone. The tubes are exposed directly to the combustion gases, but a heatshield is employed on the rear side to prevent cooling of partially heated fuel by the shroud flow. To avoid fuel boiling during the preheating process, the fuel is supplied at a pressure level above the critical point pressure of 22 atmospheres and after heating to 560K (550°F) is flash vaporized when it is throttled to the duct pressure level in an expansion valve or the injector orifice. Systems of this type have been used extensively to cool rocket motor thrust chambers and the program of Reference 7 demonstrated their effectiveness during short duration tests in a simulated main burner environment.

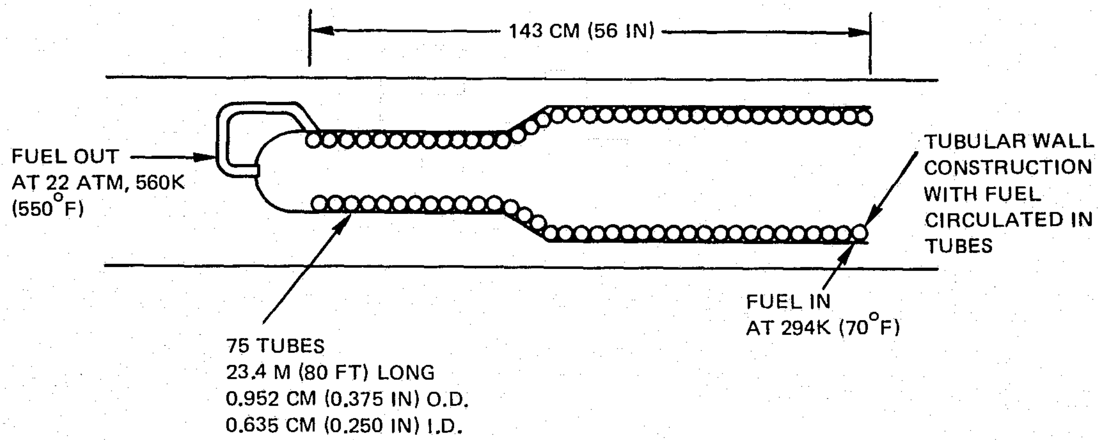


Figure 3-18 Regenerative Fuel Preheating System

The system consists of seventy-five tubes with the duct burner fuel flow distributed uniformly between tubes. Each tube is 23.4 meters (80 feet) long, wrapping around the circumference of the engine about four times, with the total regenerative surface having an axial length of 143 cm (56 inches). The system is sized for heating the sea level takeoff flow, but the flow may be reduced to the supersonic cruise level while obtaining adequate fuel preheating and structural thermal protection without the need to shut down part of the heating system. Relative to a louvered liner of the same axial length, incorporating the regenerative fuel heating system of Figure 3-18 would impose an engine weight penalty of 440kg (970 lb) exclusive of fuel headers and additional control systems.

In principle, a similar regenerative system could be employed to preheat the duct air in the event that recondensation of the fuel causes severe deterioration of the emissions characteristics at the low Mach number operating conditions. The utility of this concept would be fundamentally limited by the ability to generate high heat transfer rates across the wall of the combustion zone while maintaining low pressure drops in the air passages. A preliminary evaluation indicated that, with representative combustion zone wall areas and air passage pressure drops, this type of system could provide less than 10 percent of the desired 122K (220°F) air temperature rise in the duct flow at sea level takeoff. Augmenting the heat transfer to the duct air by passing it through radial tubes spanning the duct burner combustion zone – in effect making the combustion zones crossflow tube bundle heat exchangers – was found to be impractical because of the high total pressure losses involved. To moderate these losses to tolerable levels would require two to threefold increases in the radial height of the fan duct.

A partial regenerative heating concept, that permits significant preheating of the combustion air in the high power stage without compromising duct geometry or performance, is shown schematically in Figure 3-19. The high power stage premixing tubes are extended axially through the pilot stage combustion zone, and air heating is accomplished while fuel-air mixing is occurring in the tubes. Preliminary evaluation of this concept, using representative airflow splits between the pilot and secondary stage, indicates that this system could be incorporated into the 33 cm (13 inch) duct height without exceeding the goal pilot stage reference velocity or total pressure loss. The tubes would have to be 1.7 meters (5.6 feet) long to provide sufficient heat transfer area to produce an exit temperature of 560K (550°F), whereas the pilot stage is not expected to be more than .61 to .76 meters (2.0 to 2.5 feet) long. Consequently, the preheating capability would be limited to about 50 percent of that desired and the air temperature at the discharge into the secondary combustion zone would be about 500K (440°F) at sea level takeoff. The tube wall temperature would be about 840K (1100°F) so that, if liquid fuel were to contact the wall, it would be rapidly evaporated. At the low pressure levels at which the duct burner operates, the autoignition time for fuel in contact with the tube wall is more than four times the residence time in the tube so the risk of premature combustion would be minimal.

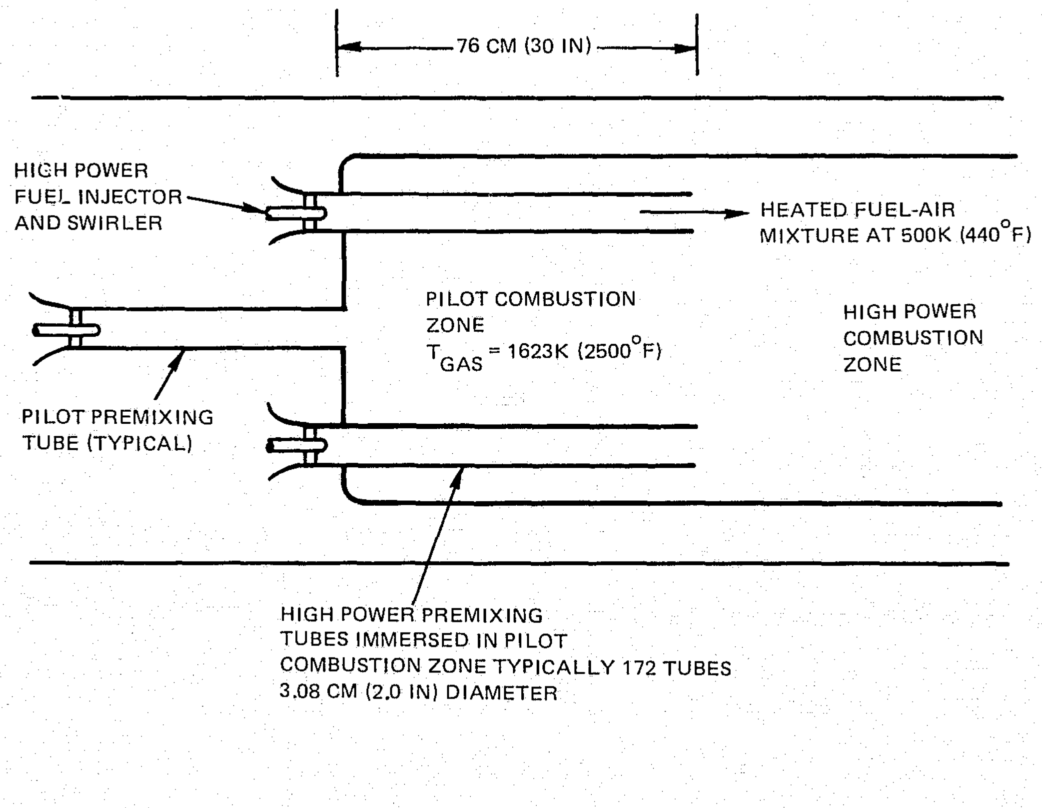


Figure 3-19 High Power Stage Air Preheating Concept

This approach could also be used to preheat the pilot combustion air by using the U-shaped premixing tubes that return the mixture to the front end of the combustion zone. The increased length of the tubes would permit preheating the combustion air completely to the 560K (550°F) temperature level desired, but the need for the return leg on the tube would introduce more than double the blockage in the pilot combustion zone. This would require either an increase of 30 percent in the frontal area of the duct burner or a 40 percent increase in the reference velocity in the pilot stage with associated increases in the momentum loss and the length of the pilot. Combining these problems with the fact that the preheating of the pilot air would also occur at supersonic cruise which would be unnecessary and have an adverse effect on NO_x emissions at that flight condition, suggests that the use of this approach be restricted to the high power stage of the duct burner.

The duct burner fuel and/or combustion air could also be preheated by bleeding hot gases from the core engine. Table 3-IV shows the conditions in the core engine at the exit from the high-pressure compressor and the high-pressure turbine, and indicates that either of these sources has sufficient pressure drop to the fan duct to assure adequate flow through the ducts or heat exchangers required in the preheating system and that the bleed gases are at high enough temperatures for effective heat transfer.

TABLE 3-IV

SOURCES OF CORE ENGINE BLEED FOR PREHEATING

	Bleed Source	
	Compressor Exit	Turbine Exit
Supersonic Cruise		
Bleed Pressure - $10^6 N/m^2$ (psia)	1.43 (207.6)	0.44 (64.4)
Temperature - K (°F)	985 (1313)	1375 (2019)
Pressure Drop to Fan Duct - $10^6 N/m^2$ (psia)	1.18 (170.8)	0.19 (27.6)
Sea Level Takeoff		
Bleed Pressure - $10^6 N/m^2$ (psia)	1.95 (282.6)	0.605 (87.7)
Temperature - K (°F)	787 (957)	1239 (1773)
Pressure Drop to Fan Duct - $10^6 N/m^2$ (psia)	1.69 (244.8)	0.344 (49.9)

The use of core engine bleed for preheating implies obvious performance penalties for the engine cycle, with the penalty increments being a strong function of the bleed flow rate. Table 3-V presents a summary of the core engine bleed rate required to preheat various combinations of the duct burner fuel and air with the bleed flows having been computed on the basis of an enthalpy balance to a mixed state at the 560K (550°F) fuel vaporization temperature. These results indicate that the hotter turbine bleed gas is more than three times as effective a preheating medium than air bled from the compressor exit, and that the quantities of bleed gas required to accomplish any significant duct air preheating at the sea level takeoff condition are excessive. Consequently, core engine bleed was only considered feasible for fuel preheating.

TABLE 3-V

CORE ENGINE BLEED FLOW REQUIRED TO PREHEAT DUCT BURNER FUEL AND AIR TO 560K (550F)

	Compressor Exit % WAE	Turbine Bleed % WAE
For Supersonic Cruise Operation		
Pilot Fuel	3.21	1.62
For SLTO Operation		
Pilot Fuel	6.84	2.20
Pilot Air	36.81	11.78
Pilot Air and Fuel	43.65	13.98
Total Fuel	20.26	6.58
Total Air	79.28	25.37
Total Air and Fuel	99.54	31.95

An analysis was conducted to establish the effects of the flow extraction on engine thrust and fuel economy at takeoff and cruise. It was assumed that the bleeds were variable with flight condition, including total shut-off when the duct burner was not in operation. As shown for the cruise condition on Figure 3-20, the results indicated that the turbine was not only the best bleed source from the point of view of heat transfer capability, but that smaller performance penalties were incurred with turbine as opposed to compressor exit bleed.

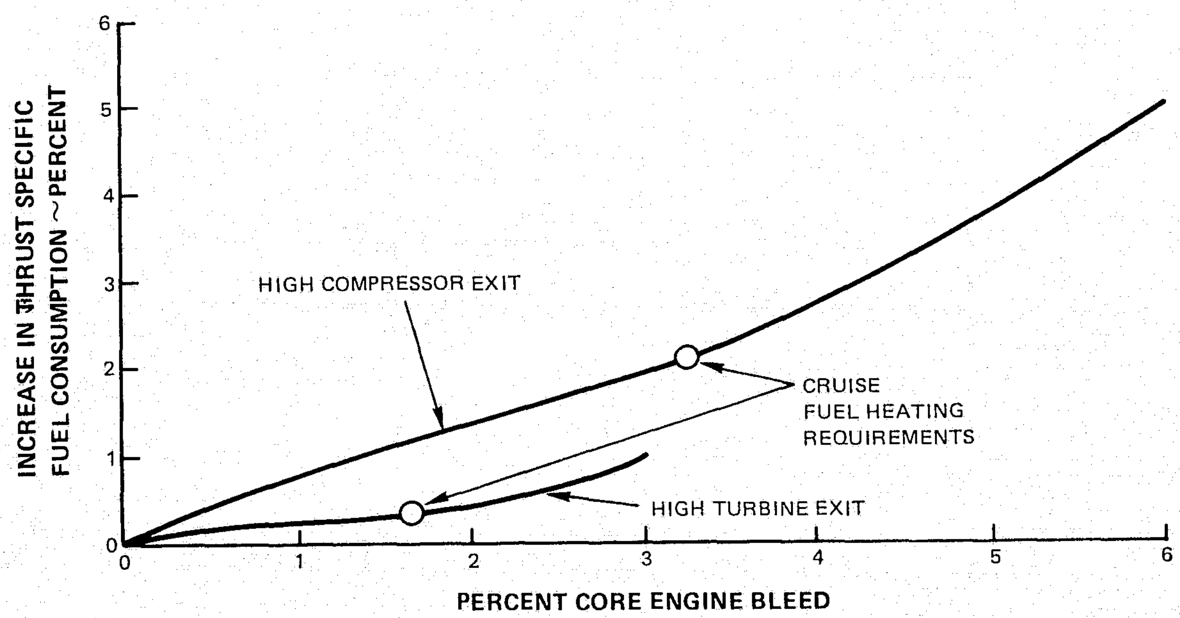


Figure 3-20 Effect of Core Engine Bleed on Specific Fuel Consumption at Supersonic Cruise

The core engine performance deterioration varies with the flight condition being evaluated. The takeoff condition for the VSCE engine is very sensitive to engine cycle changes because of the nature of its coannular noise benefit. Studies indicate that the full effect of coannular noise reduction is obtained only with a certain jet stream velocity profile. Bleeding of core flow while maintaining the proper exhaust profile requires minor rematching, but the amount of mission fuel affected is small and has little impact on the aircraft. At the supersonic cruise condition, thrust deficiencies caused by the engine bleed are overcome by increasing the level of duct augmentation. This causes a 0.4 percent penalty in cruise fuel economy, and produces a net increase of one percent in aircraft gross takeoff weight.

Examination of the conceptual definition of a fuel preheater operating with core engine turbine bleed indicate that, while the fuel could be vaporized readily by injecting it into the bleed gases and routing this mixture to the premixing passages in the duct burner, spontaneous ignition would occur almost immediately because of the high bleed gas temperatures. Figure 3-21 shows a schematic diagram of a fuel preheating system which circumvents this problem by employing a heat exchanger to partially preheat the fuel. The fuel is then injected into a chamber where it is mixed with the partially cooled bleed gases and the remaining heating and vaporization process occurs. The figure shows the variation in heat exchanger volume, or proportionately weight, with the exit temperature of the bleed gases as different degrees of fuel preheating are achieved in the heat exchanger. An auxiliary scale shows the ignition delay time before spontaneous combustion will occur in the mixed bleed gases and fuel in the mixing chamber. The available time for mixing diminishes rapidly at bleed gas temperatures above 700 K (800°F) and a heat exhanger having more than 70 percent of the volume required for full fuel preheating would be necessary. Based on these results, it was concluded that unless the required heat exchanger was very large, the additional weight and complexity of fuel-gas mixing chambers would not offset the 30 percent reduction in heat exchanger volume and it would be more attractive to accomplish all of the fuel preheating in the heat exchanger.

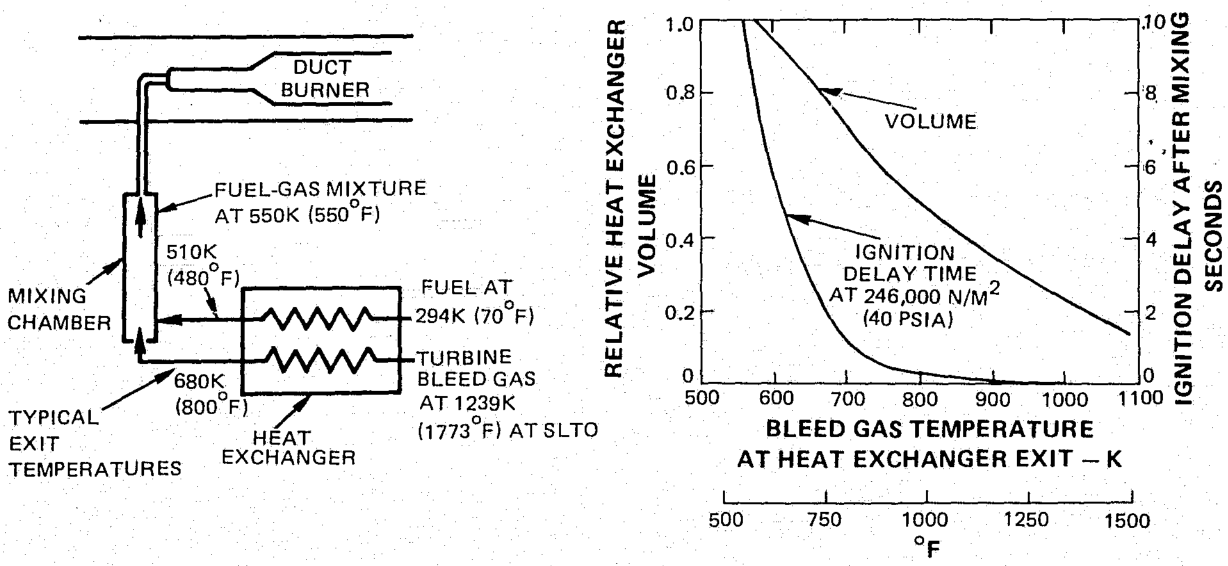


Figure 3-21 Fuel Preheating System Using Turbine Bleed

An analysis was conducted to establish the size and performance characteristics of the required heat exchanger. The allowable gas side pressure loss in the heat exchanger proper was restricted to 55 percent of the core engine turbine exit to fan duct pressure differential to allow for pressure losses in the bleed ports and ducts, and the heat exchanger was assumed to be a multipass cross counterflow type having a compact finned heat transfer surface. A heat exchanger module having a heat transfer volume of $0.071 m^3$ ($0.765 ft^3$) and approximate dimensions of 7.7 cm x 30.5 cm x 91.5 cm (3.06 in x 12 in x 36 in) was found adequate to heat the fuel at supersonic cruise conditions. Five of these modules would be required for complete fuel preheating at sea level takeoff.

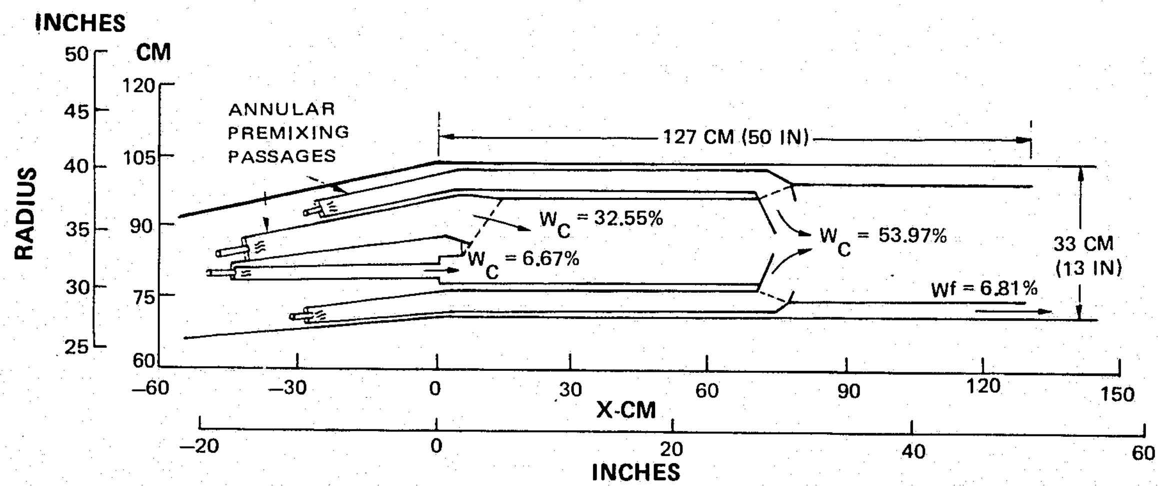
These heat exchangers could be installed around the outside of the engine behind the fan case without compromising the frontal area of the nacelle installation around the engine (see Figure 1-1), but this would require that the hot turbine bleed gases be ducted across the fan duct. The bleed ducting problem would be simplified if the heat exchangers were installed in the region between the core engine cases and the inner wall of the fan duct, but interference with main burner fuel manifolds, compressor vane actuators, and other accessories could complicate this installation. The individual heat exchanger modules weigh 64 kg (140 lb), including cases but exclusive of bleed gas ducts or additional control systems, and would produce a net engine weight penalty of 320 kg (700 lb) if the five modules required for fuel preheating at sea level takeoff were incorporated. The system requires turbine bleed rates of about 1.65 and 6.90 percent of core engine airflow at supersonic cruise and sea level takeoff, respectively.

In summary, these studies have indicated that from the point of view of heat transfer capability, size, and engine performance impact, two means of preheating the duct burner fuel are feasible. These involve regenerative heating in the liner of the duct burner and preheating with hot gases bled from the turbine of the core engine. Both of these approaches involve engine weight penalties of the order of 400 kg (880 lb) and the latter imposes additional TSFC penalties. The only feasible means of preheating the duct air, if it is required to inhibit recondensation of the vaporized fuel, involves immersing the premixing passages of the high power stage in the pilot combustion zone and even this approach is length limited, producing only about half the desired air temperature rise. It should be noted that this study has not addressed problems such as fuel deposition on the heat transfer surface or structural integrity and these areas could require extensive research and development efforts and further design evaluation to produce viable components.

Aerothermal Definition

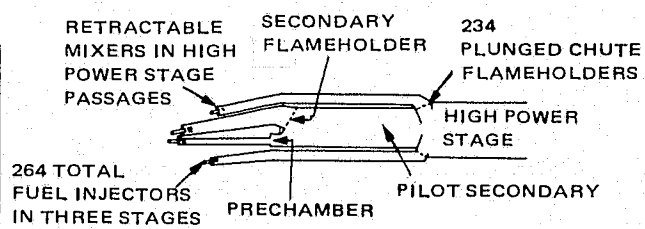
Figures 3-22 and 3-23 show the aerothermal definition of two duct burners employing premixed-prevaporized combustion concepts. Both of these configurations are three stage burners incorporating a small prechamber stage for ignition, a secondary stage operational at supersonic cruise, and a high power stage. The configuration of Figure 3-22 was assumed to incorporate a regenerative fuel heating system with all of the fuel being circulated through the burner liner before injection. In the burner of Figure 3-23, fuel preheating was assumed to be accomplished with turbine bleed through a heat exchanger and a louvered liner was employed on the burner. This configuration also incorporated air preheating in the high power stage.

The airflow to the prechamber stage is established by the stability characteristics of premixed systems, as defined in Section 2.2, and provides for ignition at a duct fuel-air ratio of 0.002. At supersonic cruise the equivalence ratio in the combined prechamber and pilot secondary stages is 0.48. This produces temperatures of about 1600K (2500°F) in this stage to achieve CO oxidation without encountering high NOx generation rates. The flameholder in the pilot secondary stage is sized for adequate stability at this flight condition and equivalence ratio. At high augmentation levels the high power stage is operational and fuel scheduling is established to provide a uniform equivalence ratio of 0.603 (0.645 in the configuration of Figure 3-23 where richer mixtures are required because of the need for liner cooling air) at sea level takeoff. This provides the leanest overall mixture strength with the lowest possible NOx production. Flameholder surface area is not a critical factor in the high power stage because the combustion process is sustained by the hot combustion products from the pilot stages.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	0.52	0.6	1.12
η_C - %	99	87.9	99
η_T - %	94.5	–	95.5
$\Delta P_{T\ HOT}$ - %	4.25	–	14.0
SIZE PENALTY	NONE		
WEIGHT PENALTY	735 KG (1620 LB)*		
Δ TOGW	2.35%		

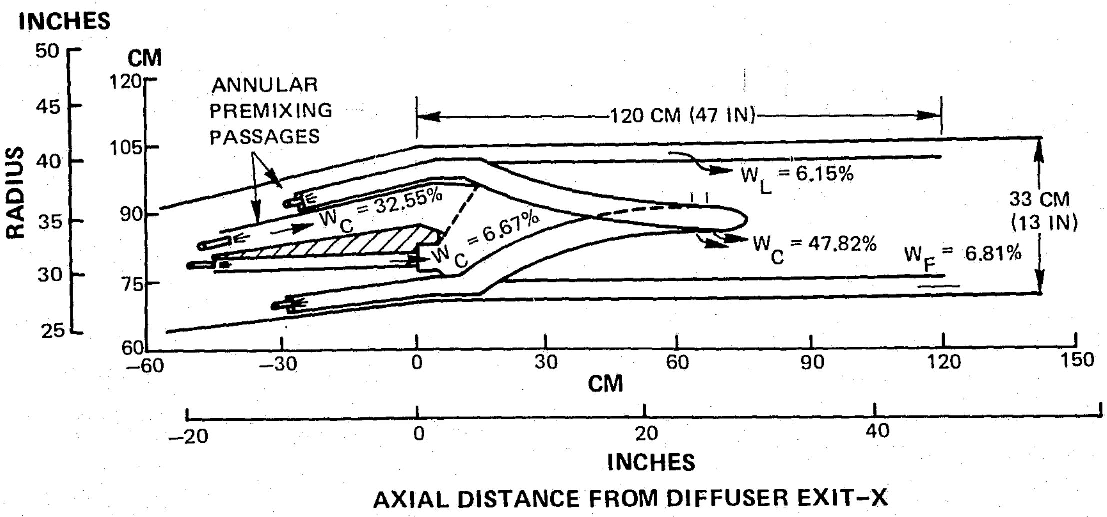
*INCLUDES FUEL PREHEATER



FLAMEHOLDER AREAS

STAGE	AC_D CM^2	AC_D IN^2	SURFACE AREA CM^2	SURFACE AREA IN^2
PRECHAMBER	1190	185	3400	528
PILOT SEC.	5810	902	16600	2577
HIGH POWER	8550	1325	0	0

Figure 3-22 Aerothermal Definition of Premixed-Prevaporized Duct Burner with Regenerative Fuel Preheating



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	0.52	0.62	1.29
η_C	99	87.9	99
	94.5	—	95.5
$\Delta P_{T\ HOT}$	4.25	—	14.0
SIZE PENALTY	NONE		
WEIGHT PENALTY	770 KG (1690 LB)•		
Δ TOGW	3.45%†		

•INCLUDING FUEL PREHEATER
†INCLUDING CORE ENGINE BLEED PENALTY

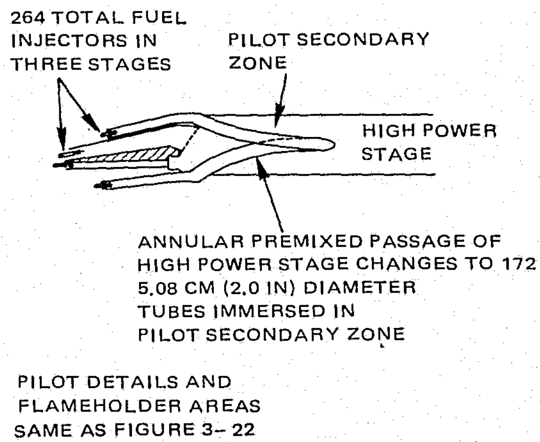


Figure 3-23 Aerothermal Definition of Premixed-Prevaporized Duct Burner With Air Preheating in the High Power Stage and Fuel Preheating With Turbine Bleed

The emissions characteristics of these duct burners have been projected on the basis of data from laboratory tests of ideal premixed-prevaporized burners (Ref. 13) and, if these levels are to be achieved, comparable degrees of premixing must be accomplished. Reference 31 presents data from the testing of premixing systems and was used as a basis for the design of the premixing passages on these configurations. The cited data indicated that fuel-air mixtures having nonhomogeneities of 8 percent or less were generated in tubes with swirlers at the entrance. A one percent pressure drop was maintained across the tube and the residence time in the premixing tube was 17.7 msec. These values were used as design goals but when excessive passage lengths were found to be required to produce this residence time, it was assumed that by increasing the pressure loss across the passages to two percent, comparable mixture uniformity could be acheived in one half this residence time. The premixing passages shown in these two configurations are sized to satisfy these criteria. The passages are annular and since they extend into the diffuser, they must be segmented around the struts. Mixers, consisting of vortex generators, turbulence screens or counterotating vanes, are installed at the entrance to these passages to reproduce the effect of the swirlers in the configuration of Reference 31.

The premixing passages are sized for a passage Mach number of 0.137 at supersonic cruise. With a passage pressure loss of 2 percent at this condition the overall liner loss goal of 4.0 percent is satisfied with a 2 percent flameholder pressure drop. Based on the data of Reference 32, the autoignition safety factor (ignition delay time/residence time in the passage) in these passages is more than 800 at supersonic cruise, and even higher at sea level takeoff because of the long ignition delay associated with the low pressure and temperature levels.

The thrust efficiency characteristics of the staged premixed prevaporized duct burners differ from those of the previously discussed axially staged burners in that very high efficiency is achieved at sea level takeoff. At this condition, there is no need for strong penetration of the fuel-air jets entering the high power stage because all of the stages are operating at the same equivalence ratio and a uniform exit temperature distribution consistent with the high thrust efficiency is anticipated regardless of the radial position of the reaction zones in the high power stage. However, rapid and vigorous mixing is required in the high power stage at supersonic cruise to quench residual NOx production and produce the gas temperature uniformity necessary to achieve the 94.5 percent thrust efficiency goal. To enhance the penetration and mixing of the bypass air entering through the high power stage flameholder at this condition, the mixers on the inlet of the premixing passages of this stage are stipulated to be retractable or variable. When this stage is operational; the mixers are active and produce a 50-50 split of the available pressure loss between the premixing passage and the flameholder. At the lower duct fuel-air ratio conditions, such as supersonic cruise, where this stage functions only as a mixing zone the inlet mixers are inactive and a greater fraction of the overall pressure drop is available at the flameholder to produce higher jet velocities. Jet penetration is also augmented in the configuration of Figure 3-22 by the use of chute flameholders. The downstream end of the immersed premixing tubes of the burner on Figure 3-23 can also be arranged in an array that provides the optimum coverage of the combustion zone crossection for this purpose.

The combustion efficiency of these configurations at the transonic climb condition is lower than that projected for the staged vorbix concepts of Sections 3.3.3 and 3.3.4. The low inlet pressure and temperature dictate that a larger fraction of the fuel must be introduced in the prechamber and pilot secondary stages to sustain stable combustion in the extremely homogeneous fuel-air mixtures entering these stages. Consequently, the high power stage must operate at a very lean equivalence ratio (0.38 as opposed to 0.46 in the staged vorbix concepts) at transonic climb and greater deficiencies in combustion efficiency are projected by the kinetic analyses. While the previous cited uncertainties regarding this analysis are valid in the case of premixed systems, the fact that the projected deficits are larger suggests that the problem may be more acute with these configurations.

The presence of the premixing tubes, either surrounding or immersed in the pilot combustion zone does not compromise the sizing of these duct burners. Both of these configurations fit the 33 cm (13 inch) duct height while retaining a 43m/sec (140 ft/sec) reference velocity in the pilot zone at supersonic cruise. The high power stage is sized for the same 67 m/sec (220 ft/sec) sea level takeoff reference velocity used in the preceding configurations.

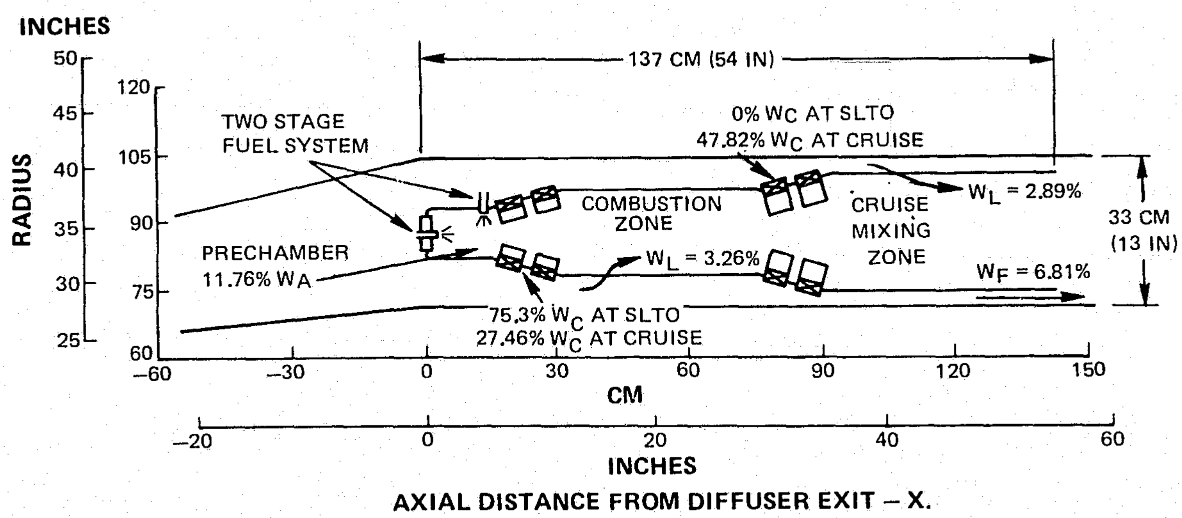
Both of the premixed-prevaporized duct burners produce weight penalties of about 640 kg (1400 lb) above those of the prechamber vorbix and ram induction configurations leading to increases of 2 percent in aircraft takeoff gross weight over those concepts. Approximately half this increment is due to the weight of the fuel preheater, i.e., the regenerative liner or the heat exchanger, while the remainder is attributed to the premixing passages and their supporting structure. An additional one percent penalty in aircraft takeoff gross weight is accrued with the turbine bleed fuel preheating system because of the deteriorated performance of the core engine.

3.3.7 Variable Geometry Concepts

The staged duct burner concepts described in the previous sections employ a fixed airflow distribution and multiple fuel systems, with injectors located at several radial and axial positions. As the overall fuel-air ratio of the duct burner is increased from the ignition level, the multiple fuel systems are sequentially activated and the fuel flow split between the systems adjusted to maintain the desired equivalence ratio levels in each stage of the burner. An alternative approach to equivalence ratio control involves variation of the airflow distribution in the combustor. Analytical studies and operational combustor tests, such as those of Reference 33, have demonstrated that the emissions from single stage main burners may be reduced over wide ranges of combustor inlet conditions and fuel-air ratio through the use of variable geometry airflow components. This approach offers the advantage of a simpler fuel system but introduces the additional complexity of actuation of the variable geometry mechanisms that produce the airflow distribution changes. When combined with the potential for continuous control of local equivalence ratios with overall fuel-air ratio, as opposed to the stepwide modulation capability of staged fixed geometry systems, the variable geometry concept appears attractive. Studies were conducted to assess the potential advantages of incorporating variable geometry features in the duct burner to enhance the performance capabilities, achieve lower emissions levels or reducd the size and complexity of the burner. In the remainder of this section, the application of variable geometry to the prechamber vorbix duct burner of Section 3.3.3 and the definition of a variable geometry single stage premixing-prevaporized duct burner are discussed and some general observations regarding the utility of variable geometry in duct burners are presented.

Variable Geometry Vorbix Duct Burner

Figure 3-24 shows the aerothermal definition of a variable geometry version of the prechamber vorbix duct burner of Section 3.3.3. The burner incorporates the pilot prechamber stage of its fixed geometry counterpart to retain the desirable lighting characteristics, to provide a heat source for fuel vaporization and to sustain combustion in the zone immediately downstream. This configuration differs in the use of multiple rows of air induction swirlers at the discharge from the prechamber and in the use of variable geometry to divert airflow through either these swirlers or those at the downstream end of the combustion zone. At supersonic cruise, 36 percent of the swirlers at the prechamber exit and all of those at the end of the combustion zone are open. This produces an airflow distribution identical to that of the fixed geometry configuration and, with all other geometric parameters identical to that configuration, the same emissions and performance characteristics. At sea level takeoff, the swirler tubes at the downstream end of the combustion zone are closed while all of those at the prechamber exit are opened admitting all of the combustion air not used in the prechamber at this location. This results in a simplification of the three stage fuel system employed in the fixed geometry configuration because the need for fuel injection in the high power stage has been eliminated. While nearly all combustion occurs in the previously designated pilot secondary zone, the overall length cannot be reduced because the volume of the high power stage is still naeded for mixing the combustion products and the bypass air to achieve high thrust efficiency at supersonic cruise. The sea level takeoff NO_x emissions of this configuration are expected to be higher than those of the fixed geometry burner because combustion occurs in two rather than three stages and at nominally higher equivalence ratios.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_x	2.75	1.54	>2.17
η_C - %	99	99	99
η_T - %	94.5	–	92
$\Delta P_{T\ HOT}$	4.25	–	18.5
SIZE PENALTY	NONE		
WEIGHT PENALTY	186 KG (410 LB)		
Δ TOGW	0.6%		

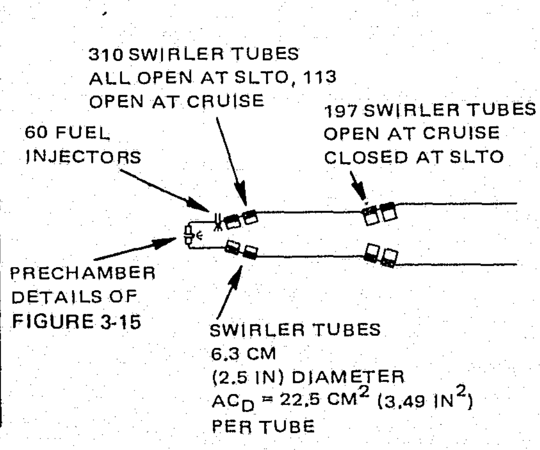


Figure 3-24 Aerothermal Definition of the Variable Geometry Vorbix Duct Burner

Uncertainty regarding the takeoff emissions characteristics arises because the ratio of the mass of fuel that must be vaporized in the prechamber combustion products to the mass flow in that stream is large. The reference velocity of 103 m/sec (339 ft/sec) in the combustion zone at sea level takeoff is more than three times the level of experience in low emissions main burners. This causes further uncertainty in the projection of the emission characteristics and also leads to serious pressure loss problems at this operating condition with the analysis indicating thermal choking in the combustion zone. Increasing the radial height of the combustion zone would relieve this problem, but if the shroud Mach number at supersonic cruise is to be maintained, it is estimated that a 12.7 cm (5 inch) increase in engine diameter would be required to reduce the momentum total pressure loss to the 14 percent levels incurred in the fixed geometry configurations at sea level takeoff.

This pressure loss problem demonstrates the unique difficulty in attempting to incorporate variable air scheduling in a duct burner. Regardless of the application – duct burner or main burner – variable geometry systems incorporate parallel flow paths consisting of the combustion zone and the bypass passage or burner shroud. Each of these components must be sized for the maximum airflow situation in the individual branch. In main burners, the reference velocities are nominally low and shroud heights are not subject to geometric constraints other than the capability of adequate pressure recovery. However, in the duct burners, a premium is placed on duct height and both reference velocities and shroud Mach numbers are as high as possible within overall pressure loss constraints. The inherent increase in shroud and/or burner frontal area that must be introduced to accommodate varying flow rates in the parallel flow system becomes a significant penalty not present in the main burner.

Variable Geometry Premixed-Prevaporized Duct Burner

Figure 3-25 shows the conceptual definition of a single stage fully variable geometry premixed-prevaporized duct burner and shows the airflow distribution in the burner at various operating conditions. This configuration features an axially translating cowl to accomplish the flow shifting. A single combustion zone with a perforated plate flameholder is employed and, as the cowl is translated forward it blocks off progressively more of the holes in the flameholder while uncovering apertures for the admission of dilution air further downstream in the combustion zone. The upstream lip of the cowl interacts with the walls of the fan duct diffuser to vary the relative inlet areas of the premixing and bypass air passages in accord with the desired airflow split. At sea level takeoff the cowl is located in the full aft position and all of the combustion air is forced through the premixing passage and the flameholder to provide the lowest possible equivalence ratio in the combustion zone. As the fuel-air ratio is decreased, the cowl moves forward reducing the flameholder airflow to maintain a combustion zone equivalence ratio consistent with optimum emissions and stability, while the remaining duct air flows through the bypass passage to the dilution air ports. The cowl is in the full forward position for ignition and only 6.67 percent of the duct flow enters the premixing passage. The radial front wall of the combustion zone has adequate flameholder area for stable combustion at the ignition fuel air ratio of 0.002. This concept accomplishes the objectives of reducing the duct burner to a single stage burner with optimum equivalence ratio capability at all operating conditions and the use of a single stage fuel injection system. Fuel preheating could be accomplished by either the regenerative liner or the core engine bleed approaches shown on the fixed geometry staged premixed prevaporized duct burners of the preceding section.

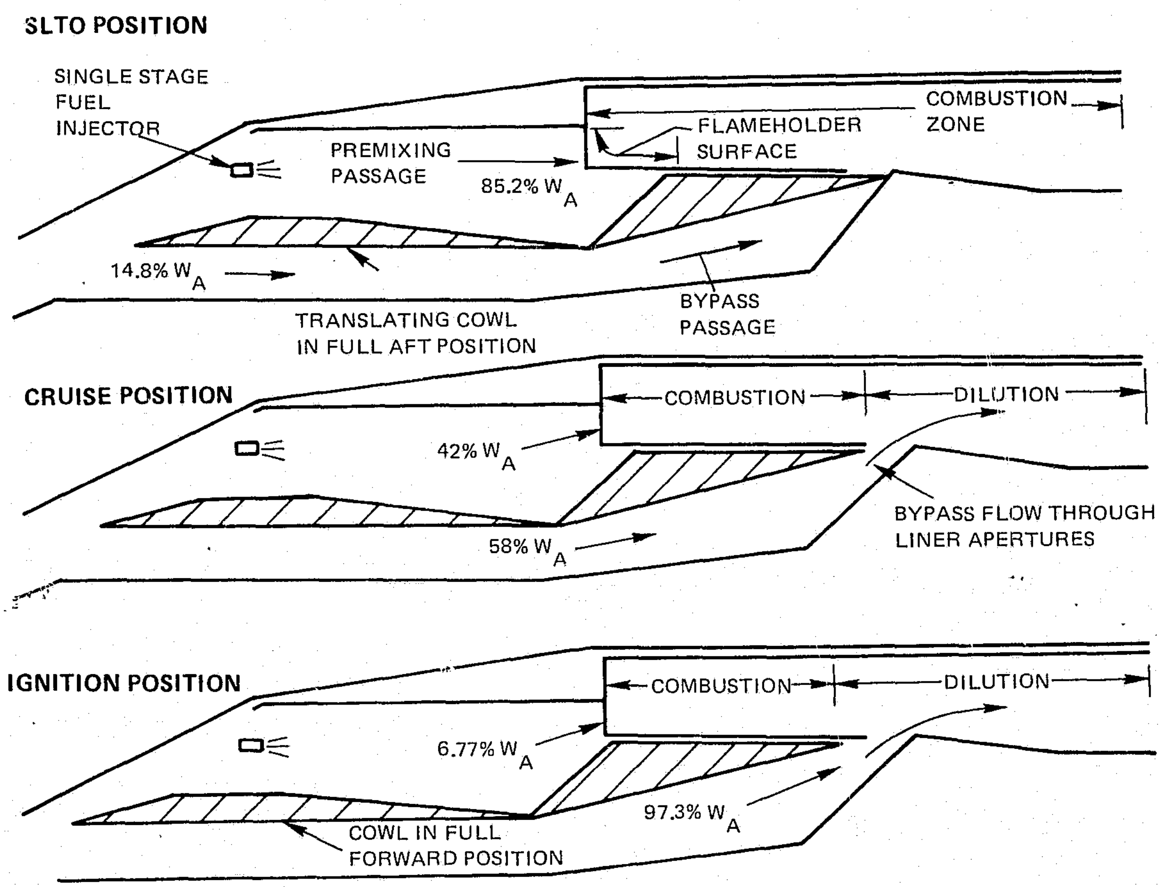
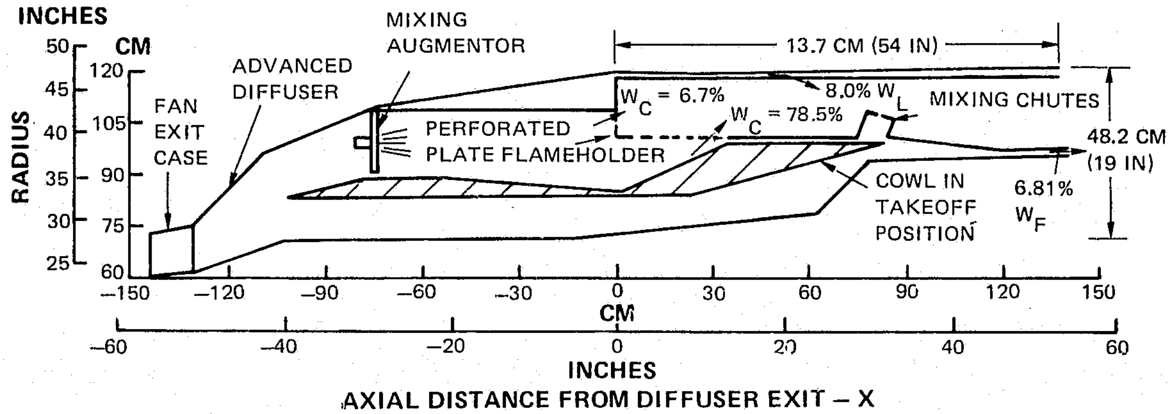
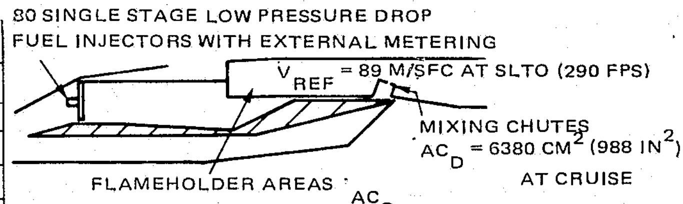


Figure 3-25 Conceptual Definition of a Single Stage Variable Geometry Premixed-Prevaporized Duct Burner

Figure 3-26 shows more of the details of the aerothermal definition of this duct burner and its projected performance. Obviously, the problem of duct height and total pressure loss encountered with the variable geometry vorbix combustor is also present in this configuration. The radial height of the duct is dictated by three flow area constraints at the axial location of the front of the combustion zone (axial location X = 0). These include: sufficiently low Mach number in the bypass passage at supersonic cruise to avoid high pressure losses; low premixing passage velocity to prevent aspiration of combustion gases from the first row of holes in the cylindrical flameholder surface and; sufficient combustion zone height to maintain reference velocities consistent with reasonable levels of momentum total pressure loss. Even the 48 cm (19 inch) duct height shown on this figure is optimistic in this regard because the bypass passage is nearly choked at ignition and the momentum total pressure loss in the combustion zone at takeoff will, most likely, produce a thermal choking situation in that component.



	SUPERSONIC CRUISE	TRANSONIC CLIMB	SEA LEVEL TAKEOFF
EI NO_X	0.52	0.82	1.12
η_C - %	99	99	99.9
η_T - %	94.5	—	95.5
$\Delta P_{T\,HOT}$ - %	6.6	—	24.5
SIZE PENALTY	30.5 CM (12 IN) Δ DIA.		
WEIGHT PENALTY	770 KG (1700 LB)		
Δ TOGW	4.15%		



	CM^2	IN^2
IGNITION	788	122
SUPERSONIC CRUISE	5880	910
SEA LEVEL TAKEOFF	11600	1640

SURFACE, BLOCKAGE = 50%

Figure 3-26 Aerothermal Definition of the Variable Geometry Premixed-Prevaporized Duct Burner

The variable geometry premixed-prevaporized concept does not appear to offer any substantial performance or emissions advantage over its staged counterpart at the takeoff or cruise design points. In addition to the frontal area problems, it is noted that the flow dividing lip of the cowl and the premixing passage overlap into the diffuser section. This creates the need for advanced diffuser concepts; causes potentially severe interaction effects between the diffuser and the cowl and leads to complex integration of these components with the service and structural struts spanning the fan duct in this area.

General Aspects of Variable Geometry Duct Burners

The two configurations discussed above demonstrate the problems associated with attempting to incorporate variable geometry concepts into duct burners and indicate that, from the point of view of reducing emissions, they do not offer any advantages over comparable staged burners and can present significant additional complexities and performance penalties. However, limited application of variable geometry might be of value in resolving the problem of low combustion efficiency at the transonic climb operating condition. This situation arises because of low equivalence ratios in the high power stage of fixed geometry configurations and could be offset by delaying the introduction of some of the air in this stage. Dual air induction aperatures could be located at the front end and at an intermediate axial position in the high power stage, and would be varied to admit all of the stage combustion air at the front end at takeoff and only part of this air at climb. However, with the additional problem of mixing the air introduced at the downstream aperatures into the combustion products to obtain high thrust efficiency at climb, it is doubtful that this approach would offer any advantage over the previously advanced solution of extending the length of the duct burner to provide adequate residence time for carbon monoxide oxidation. Furthermore, because of the higher equivalence ratio at the front end of the high power stage, the NO_x production would also be higher.

Other variable geometry concepts could be envisioned that would offer the capability of reducing the total pressure loss across the duct burner. Variable geometry apertures on all major components that would open to accommodate the increase in corrected flow in the duct between the supersonic cruise and the sea level takeoff could reduce the burner pressure loss by 3 to 4 percent at the later operating condition. However, since the liner pressure drop is related to jet penetration and the turbulent energy available for mixing (Ref. 30) this relatively small gain could compromise the emissions characteristics and the thrust efficiency at takeoff. Likewise, a low emissions pilot could be combined with collapsing or retractable flameholders in the high power stage and the duct pressure loss at supersonic cruise reduced because of the low blockage in the stream bypassing the pilot burner. However, the pressure drop across the air inlets on the high power stage is necessary to create the high velocities that enhance jet penetration and mixing which are essential to attaining high thrust efficiency at the supersonic cruise condition.

In general, it appears that the introduction of variable geometry into duct burners, regardless of the intended purpose, creates performance, emissions or size penalties that more than offset any advantages derived from its use. These consequences are a result of the strong interaction between geometric-pressure loss and mixing-pressure loss constraints in the duct burner.

3.3.8 Overall Evaluation

Following the completion of the aerothermal design studies, the eight duct burner configurations were screened with the objective of selecting the four most attractive concepts for further evaluation in a mechanical design study. The screening was initiated with a division of the concepts into categories representing "moderate" and "high" development risk. It was further stipulated that at least two of the four selected concepts would be in the moderate development risk category. In this manner the final phase of the study would be directed at concepts that could be readily incorporated into the VCE Critical Technology Test Bed Engine with relatively low risk as well as those that offered the greatest potential for low emissions following extensive development.

Table 3-VI presents a summary of the emissions and performance characterisitcs of the duct burners, their critical dimensions and overall impact on aircraft performance. The table reflects the above cited division into risk categories with the staged and variable geometry premixed-prevaporized combustor concepts identified as being in the high development risk category. For emphasis, the parameters that are deficient relative to the goals are boxed. The table also includes a ranking of the concepts on the basis of fabrication cost and projected combustion noise level at sea level takeoff.

TABLE 3-VI

AERORTHERMAL DESIGN SUMMARY

	EMISSIONS						GEOMETRY				PERFORMANCE					OVERALL PERFORMANCE		
	Cruise		SLTO		Transonic Climb		Length		Duct Height		Cruise		SLTO		Lighting		Noise Ranking	Cost Ranking
Configuration	EI_{NO_x}	η_c %	EI_{NO_x}	η_c %	EI_{NO_x}	η_c %	cm	in	cm	in	η_T %	ΔP_T %	η_T %	ΔP_T %	F/A	ΔTOGW		
Goal	1.0	99	1.0	99	–	99	147	58	33.2	13	94.5	4.5	–	–	.002	0	–	–
MODERATE DEVELOPMENT RISK																		
Prechamber Vorbix	$\boxed{2.75}$	99	$\boxed{1.78}$	99	1.22	$\boxed{92.5}$	137.2	54	33.02	13	94.5	4.25	88	14.0	.002	+.35%	4	3
Vorbix-Single Stage Pilot	$\boxed{3.21}$	99	$\boxed{1.75}$	99	1.2	$\boxed{92.5}$	119.4	47	$\boxed{38.1}$	$\boxed{15}$	94.5	4.5	88	14.7	$\boxed{.0028}$	+.45%	5	2
Piloted V Gutter	$\boxed{1.2}$	$\boxed{91}$	> .81	99	.4	99	134.6	53	33.02	13	$\boxed{85}$	4.5	91.5	13.2	.002	+ 6.35%	Not Ranked	1
Prechamber Ram Induction	$\boxed{–}$	99	$\boxed{–}$	99	–	$\boxed{–}$	148.6	58.5	33.02	13	94.5	4.25	88	14.0	.002	+.35%	3	4
Variable Geometry Vorbix	$\boxed{2.75}$	99	$\boxed{>2.17}$	>99.9	1.54	99	137.2	54	33.02	13	94.5	4.25	92	18.5	.002	+.6%	6	5
HIGH DEVELOPMENT RISK																		
Three Stage Premixed	.52	99	$\boxed{1.12}$	99	.64	$\boxed{87.9}$	127	50	33.02	13	94.5	4.25	95.5	14.0	.002	+ 2.35%*	1	6
Three Stage Premix-Air Preheat	.52	99	$\boxed{1.3}$	99	.62	$\boxed{87.9}$	119.4	47	33.02	13	94.5	4.25	95.5	14.0	.002	+2.45%*	1	7
Variable Geometry Premixed	.52	99	$\boxed{>1.2}$	>99.9	.82	99	127	50	$\boxed{48.25}$	$\boxed{19}$	94.5	4.4	95.5	21.42	.002	+ 4.15%	2	8

*Increase 1% with Turbine Bleed Fuel Preheating System

The cost ranking is based on complexity of the configuration with primary emphasis on the number of fuel injectors and manifolds. The piloted radial V-gutter configuration is shown to be the least expensive while the premixed-prevaporized concepts, with their associated fuel preheaters and premixing systems, are the most complex and costly.

The combustion noise ranking was based on an analytical model of the noise generation mechanisms, supplemented by measurements obtained during testing of several experimental low emissions main burners (Ref. 34). While this model relates noise production with such fixed parameters as inlet pressure and temperature level it also identifies fluctuations in the local heat release rate as the dominant acoustic energy production source and correlates combustion noise inversely with the number of fuel sources. Since a homogeneous fuel-air mixture is assumed to be generated in the premixing passages of the premixed-prevaporized combustors, each aperture on the flameholder is considered to be a distinct fuel source and these concepts are estimated to have the lowest combustion noise. The data of Reference 34 indicate lower combustion noise levels with multi rather than single stage combustors and the ranking reflects this by favoring the three stage concepts over the two stage and variable geometry configurations. No ranking is assigned to the piloted radial V-gutter because it is beyond the range of experience of acoustic evaluations. With the possible exception of the radial V-gutter concept, combustion noise ranking is not a significant factor in the screening because the difference in noise level between the extremes of the remaining concepts is not expected to be more than 4 dB.

In general, the tabulated data indicate that the most serious deficiencies relative to the study goals are in the NOx emissions. The premixed-prevaporized burners are the only concepts with the potential for meeting the goal at supersonic cruise. While the piloted radial V-gutter configuration is projected to produce NOx emissions only slightly in excess of the goal, it is accomplished at the expense of combustion and thrust efficiency. At sea level takeoff, only the radial V-gutter configuration is projected as being capable of satisfying the NOx emissions goal and the advantage of premixed-prevaporized combustors over the other axially staged concepts is considerably less than it is at supersonic cruise.

The overall aircraft performance, in terms of increments in takeoff gross weight, generally follows the division in development risk. Increments of 2 to 4 percent in takeoff gross weight are associated with the additional weight of the premixed-prevaporized combustors. Most of the concepts in the moderate development risk category have takeoff gross weight penalties of less than 0.5 percent and these reflect deficiencies in the preliminary duct burner definition used as a baseline for these studies. The singular exception is the piloted radial V gutter which incurs a large penalty because of the inadequate thrust efficiency at supersonic cruise. The overwhelming significance of cruise thrust efficiency is apparent on considering that the variable geometry premixed-prevaporized configuration has a smaller gross weight penalty than the piloted radial V-gutter concept despite a heavier burner, fuel preheater weight and a 30.5 cm (12 inch) increase in nacelle diameter.

The selection of the four concepts for evaluation in the mechanical design studies was based on the data of Table 3-VI. The prechamber vorbix configurations was clearly the best concept in the moderate development risk category. The single stage piloted vorbix configuration was selected as the second candidate in this category with emphasis in the design phase to be concentrated on the duct height and the inadequate ignition characteristics. In the high development risk category, the staged premixed-prevaporized configurations were selected over their variable geometry counterpart.

3.4 MECHANICAL DESIGN STUDIES

The intent of the mechanical design studies was to further refine the aerothermal design and to establish preliminary mechanical definition of the four selected duct burner concepts. The expansion of the aerothermal definition included the identification of fuel injector constraints and more precise definition of liner cooling requirements.

The preparation of preliminary mechanical designs of the duct burner configurations provided the opportunity to study numerous problems such as complexity associated with fuel manifolds and the burner support structure, fabrication and assembly problems as well as the structural integrity of the burner components. A number of mechanical problems were identified in these studies. While their resolution was beyond the scope of this program, their impact on engine performance has been indicated.

The two vorbix duct burner concepts selected from the configurations in the moderate development risk category are candidates for incorporation in the VCE Component Technology Testbed Engine. Since it is anticipated that this program will employ an F-100 engine as a test vehicle, the aerothermal definition of these duct burners has been re-established in a smaller airflow size consistent with this engine. The problems associated with this redefinition are discussed and changes in projected emissions and performance characteristics because of the different airflow size are identified.

The mechanical design studies of the advanced premixed-prevaporized duct burner concept were conducted in the airflow size of the VSCE-502B engine and, as such, were consistent with the aerothermal definitions of Section 3.3. Fuel preheating is considered to be essential to create a near homogeneous mixture of fuel and air in the premixed system of these burners. To investigate the mechanical problems related to fuel preheating, one of the concepts studied was assumed to employ regenerative fuel heating in the duct burner liner while the other used bleed gas from the turbine of the core engine to preheat the fuel in an external heat exchanger. The later also incorporates air preheating in the high power stage.

3.4.1 Fuel System Requirements

In the nominal supersonic cruise mission, the duct burner is operated at the maximum augmentation level at takeoff, but is shut down immediately afterward for the early stages of climbout. At an altitude of about 7600 meters (25,000 feet) and a flight Mach number of 0.8, the duct burner is restarted and, as the aircraft proceeds through transonic acceleration, the augmentation level is progressively increased until a duct fuel-air ratio of 0.030 is achieved at the 1.3 Mach number, 11,000 meter (36,000 foot) condition. Beyond this point, the augmentation level is reduced and the duct fuel-air ratio is eventually maintained at 0.013 at supersonic cruise. This mission leads to the need for continuous modulation of the duct fuel-air ratio over the range from 0.002 at ignition through 0.030 or higher during transonic acceleration in combination with a fixed point operating capability at a fuel-air ratio 0.0385 for takeoff operation. Consideration of only the fixed point operation at ignition, supersonic cruise, transonic climb and takeoff, established the need for a three stage fuel system in most of the duct burner concepts evaluated in Section 3.3. Figure 3-27 shows a representative scheduling of fuel in the prechamber vorbix configuration, which is also typical of that required in all of the three stage duct burners. As the augmentation level is increased after ignition, the fuel-air ratio in the prechamber stage is increased until a duct fuel-air ratio of 0.008 is achieved, after which further augmentation demand is satisfied by activating the pilot secondary fuel system. The schedule in this stage is established so as to satisfy the optimum prechamber – pilot secondary fuel flow split at the supersonic cruise fuel-air ratio while the pilot secondary stage is operated progressively richer at high fuel-air ratios. When a fuel-air ratio of about 0.017 is reached, the high power stage is activated and the fuel flow split between this stage and the pilot secondary system is adjusted to the optimum levels at transonic climb and takeoff.

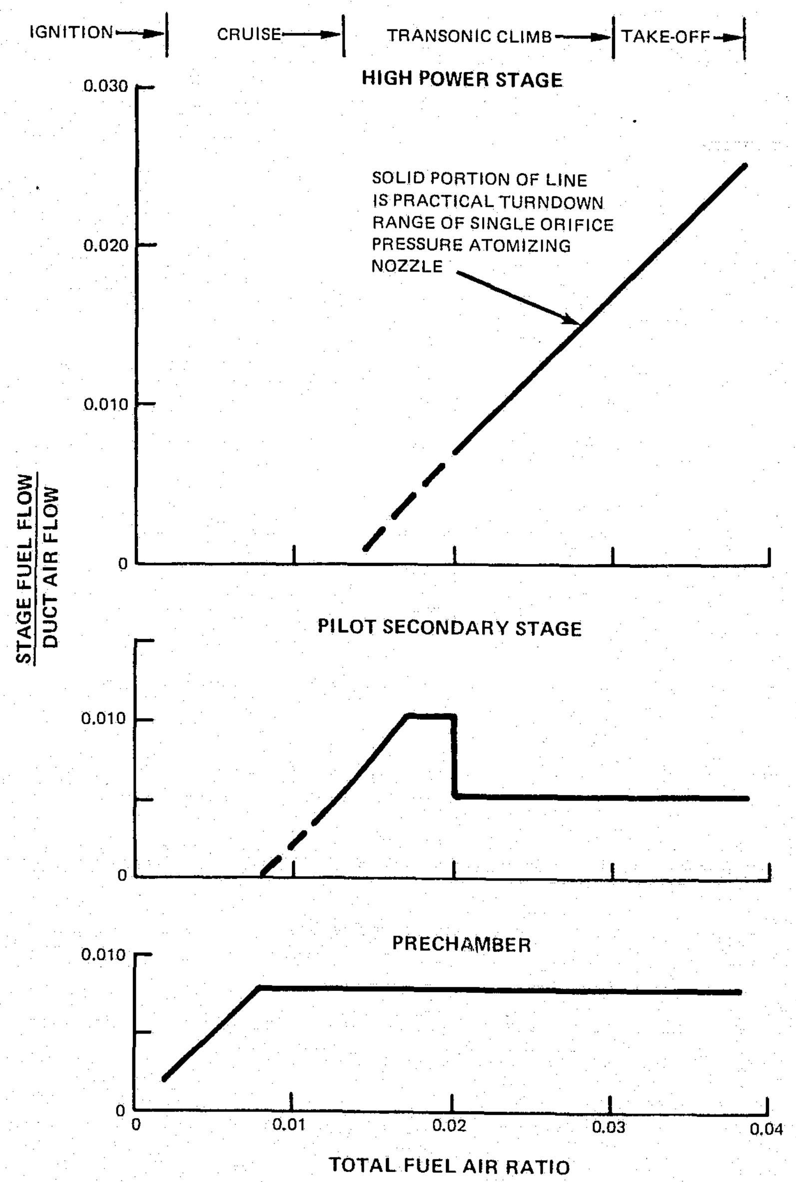


Figure 3-27 Typical Fuel Schedule For a Three Stage Duct Burner

This sequence would satisfy the continuous modulation requirements but would require essentially infinite fuel flow turndown capability in the pilot secondary and high power stage fuel systems. The simplest fuel system would rely on the pressure drop in the fuel injectors to accomplish flow metering. If single orifice pressure atomizing fuel injectors were incorporated in these stages, realistic fuel flow turndown limits would be dictated by the need for a good spray quality at low pressure drops and by the fuel pump pressure limit at high flows. Assuming values of 173,000 N/m^2 (25 psi) and 8,300,000 N/m^2 (1200 psi) for these limits, and including the effects of varying duct airflow, the range of operation of these fuel systems would be restricted to those shown by the solid portions of the curves of Figure 3-27. This causes gaps in the operating envelope in the range of fuel-air ratios just above the cited staging points. While the engine could accelerate through these gaps, steady state operation at inadequate pressure drop on the fuel injectors could cause deterioration in performance because of inadequate spray quality, and the risk of fuel deposition in the nozzle if it is exposed to hot gases. The extent of these gaps could be reduced or even eliminated by various revisions to this simple fuel scheduling approach. Optimization of the system with respect to the turndown capability of each stage, including overshoots in the loading of an individual stage, as shown on Figure 3-27 in the case of the pilot secondary stage, could narrow these operating range gaps. Circumferential sequencing of the fuel injectors in individual stages accomplished with valves on every other fuel injector that open when a preselected fuel manifold pressure is achieved, could also be employed for this purpose. More elaborate approaches could involve the use of more complex fuel controls in combination with variable area pressure atomizing fuel nozzles.

The fuel flow turndown of the prechamber system dictates that, with the above fuel pump pressure limit, the injector pressure drop would be about 103,000 N/m^2 (15 psi) at ignition. Assuming that an aerating or air boost fuel injector were employed to perform the actual fuel atomization, this pressure drop would be used solely for flow metering and the turndown capability of this stage would be more than adequate.

If a two-stage duct burner were employed, the fuel schedule of the prechamber and the pilot secondary stage on Figure 3-27 would be combined to provide that of the pilot stage. The fuel system in this stage would need a fuel flow turndown ratio in excess of 18. With the assumed fuel supply pressure limit of 8,300,000 N/m^2 (1200 psi), the pilot fuel injectors would operate at a pressure drop of only 24,000 N/m^2 (3.5 psi) at ignition. Assuming that this pressure drop would be used solely for metering and that spray quality would be achieved by aerating, the head differential in a fuel manifold on the VSCE-502B would produce a top to bottom injector fuel flow variation of ± 12 percent at ignition. These variations would diminish rapidly as the fuel flow is increased and, in the case of the single stage pilot vorbix duct burner, could provide some improvement in the ignition capability by locally enriching the pilot at the bottom of the engine.

In general, either of the staged burner configurations in the moderate development risk category could be operated in a test rig or in the VCE Critical Technology Testbed Engine with a simple fuel control system relying on injector pressure drop for metering. However, in a flight engine, requiring continuous fuel-air ratio modulation, it may become necessary to employ more complex fuel control systems.

Fuel flow turndown is a more severe limitation in the premixed-prevaporized duct burners because the fuel is at pressure levels above the critical point to prevent boiling in the preheater. With a critical point pressure of about 2,230,000 N/m^2 (325 psia) the fuel pump pressure limit restricts the fuel flow turndown capability of fixed orifice fuel injectors to less than 2:1. Consequently, variable area metering, either in each injector or in a separate distribution valve, is necessary with these concepts.

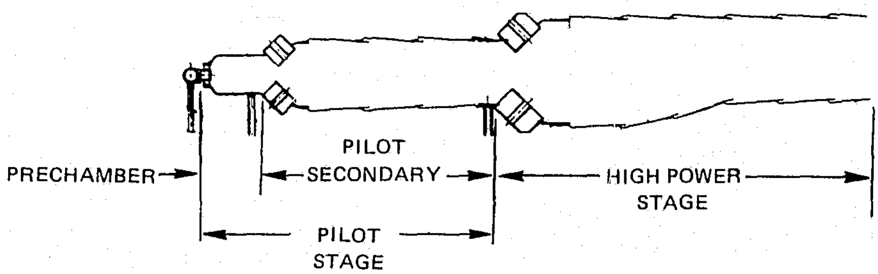
3.4.2 Liner Cooling Requirements

The liner cooling flows employed in the aerothermal definitions of Section 3.3 were based on estimates derived from a previous duct burner design (Ref. 10). A comprehensive analysis of the liner cooling requirements was conducted during the mechanical design study. A computerized augmentor liner cooling analysis developed from experience in the design and development of augmentors was used to determine the duct burner liner cooling flows. The analysis was conducted for the most severe thermal design point, sea level takeoff, where duct pressure, Mach number, and fuel-air ratios are simultaneously at the highest level encountered in the flight envelope.

One configuration, the prechamber vorbix concept, was selected for detailed analysis. Assuming the heat load distribution of all four configurations is the same, i.e. local gas temperatures and gas velocities are comparable, the liner cooling flow distribution determined for this concept was scaled to the other configurations on the basis of surface area. For the premixed-prevaporized concepts, this approach will tend to be conservative as perfect homogeneity at a lean fuel-air ratio should eliminate local hot spots in the regions of initial mixing that were assumed to exist in the vorbix configuration. Since the liner cooling levels are low relative to a main burner, less than 10 percent of the duct flow, this difference is not expected to be highly significant.

As shown on the schematic diagram of Figure 3-28, the inner and outer liner of the duct burner were assumed to consist of louver cooled panels. The nominal panel length was 18 cm (7 inches) but shorter length louvers were employed in the front end of the stages and in the prechamber where nonhomogeneity of the mixture was expected to produce locally hot regions. The nominal height of the louver lips was assumed to be 0.64 cm (0.25 in). The cooling airflow requirements were based on achieving a maximum liner metal temperature of 1115K (1550°F) so as to be consistent with the long term durability desired in possible commercial applications.

The table on Figure 3-28 shows the cooling air requirements of the three duct burners in the VSCE-502B engine size. Because of the nearly identical size, there is little variation between configurations. However, the levels are substantially higher than the supersonic cruise levels shown on the aerothermal definitions of Sections 3.3. A major part of this difference is due to the higher liner pressure drop at the sea level takeoff design condition of this analysis. This effect is most pronounced in the high power stage where the combination of a momentum total pressure loss of about 7 percent and a high combustion gas Mach number produce a nearly threefold increase in the static to static pressure drop across the liner relative to that encountered at supersonic cruise. An additional discrepancy of about two percent of the duct flow between this analysis and the previous estimates is due to the stipulation of a more conservative metal temperature limit than that used in the design of Reference 10.



LINER COOLING AIRFLOW IN PERCENT OF TOTAL DUCT FLOW

COMPONENTS	PRECHAMBER VORBIX	SINGLE STAGE PILOT VORBIX	PREMIXED-PREVAPORIZED
PRECHAMBER	1.62	0	1.47
PILOT SECONDARY	2.99	3.03	2.85
HIGH POWER STAGE	4.91	4.91	5.54
TOTAL	9.52	7.94	9.86

Figure 3-28 Liner Cooling Air Requirements in VSCE-502B Engine at Sea Level Takeoff

3.4.3 Aerothermal Redefinition for the VCE Critical Technology Testbed Engine

Studies are currently being conducted to define and plan the VCE Critical Technology Testbed Program. At this time, it appears that an F-100 engine will be used as the gas generator for this test. Figure 1-2 shows the conceptual arrangement of this engine. Since the two duct burners employing the vorbix concept are the candidates for use in the test bed program, the mechanical design studies of these configurations were conducted in a size compatible with this engine.

Table 3-VII shows the design point fan stream conditions in the F-100 engine. Comparison with the conditions in the VSCE – 502B of Table 1-I indicates that with the exception of the airflow rate, they are essentially identical to the VSCE-502B. When difference in the mean diameter of the fan ducts are recognized, the duct height in the F-100 is about half that in the VSCE-502B at comparable duct Mach numbers. Consequently, resizing of these combustors for the F-100 test bed program involved primarily reducing the radial height dimensions of the burner and the shroud passages. More specifically, recognizing that scaling down the size of burner components is not linear in the airflow size, these duct burners have been redesigned so as to operate at the same pressure drop, combustion zone reference velocities and shroud Mach numbers incorporated in the initial definition in the larger airflow size of the VSCE-502B.

TABLE 3-VII

F-100 ENGINE DESIGN CONDITION

Duct P_T – N/m^2 (psia)	283,000 (41)
Duct T_T – K (°F)	441 (335)
Duct Airflow – kg/sec (lb/sec)	48.6 (107)
Fuel-Air Ratio	0.0385
Duct Exit T_T – K (°F)	1065 (2435)
Approx. Mean Radius of Fan Duct – cm (inches)	43.6 (17.2)

While the local velocities in the combustion zones of the F-100 size duct burners are maintained equal to those of the full size VSCE-502B engine design, the annular shrouds outside the combustor were modified in some locations because of mechanical limitations. Scaling the shroud annulii in the high power stage and those under the pilot hood on the basis of equal velocities resulted in wall gaps of about 0.38 to 0.65 cm (0.15 to 0.25 inches). The sensitivity of such heights to fabrication tolerances and thermal distortion could result in maldistribution or restriction of the cooling airflow. To avoid these potential problems, the radial gaps between these walls were limited to a minimum of 1.27 cm (0.50 inches). The change in pressures due to reduced shroud velocities only affects the pressure drop across the cooling liners and can be compensated for by reducing the size of the cooling air apertures. This deviation is not considered significant to the demonstration of the duct burner emissions, performance or operational capability.

The shrouds for the bypass airflow around the pilot stage were designed to reproduce the axial Mach number distribution anticipated in the VSCE-502B configuration. This assured equal pressure losses to the entrance to the swirler tubes of the high power stage and, when combined with equal velocity levels inside the burner, produced the same pressure drop across these components.

Similarity of the aerodynamic aspects critical to the emissions characters was also maintained in the aerothermal redefinition of these duct burners. In scaling from the VSCE size to the test bed size, mixing in the vicinity of the swirler tubes and total residence time in the particular stage were the major considerations. Mixing of the vortex jets is controlled by jet penetration and swirl strength. By maintaining the pressure drop across the swirler tube and the velocity of the gas into which the air is injected constant, and the geometric proportions of circumferential spacing and diameter of the jets relative to the duct height invariant, the similarity is maintained. The success achieved in maintaining simultaneous similarity in all of these geometric parameters is largely due to the unique combination of radial height and mean gaspath radius that is preserved between the VSCE-502B and the testbed engine.

Despite the similarity established in the aerothermal revisions of the selected duct burner concepts for the VCE Component Technology Test Bed Engine, two major aspects of the design are affected. The reduction in size from the VSCE engine requires proportionally more airflow for liner cooling and a significant change in the duct burner aspect ratio, i.e., the ratio of length to radial hieght of the combustion zones.

Since the liner cooling flow per unit surface area is essentially invariant with burner size, the smaller size of the F-100 engine requires an increase in the fraction of duct air used for this purpose. Table 3-VIII shows the projected liner cooling air requirements for both duct burner concepts in the F-100 airflow size. These estimates are based on the analysis used in Section 3.4.2 with the same liner construction and metal temperature limitations.

TABLE 3-VIII

LINER COOLING AIR REQUIREMENTS FOR F-100 SIZE DUCT BURNERS

	Cooling Flow at Sea Level Takeoff in Percent of Duct Airflow	
Component	**Prechamber Vorbix**	**Single Stage Pilot Vorbix**
Prechamber	1.87	0
Pilot Secondary Stage	7.58	7.68
High Power Stage	9.22	9.22
TOTAL	18.67	16.90

Comparison of these cooling flow levels with those of the VSCE-502B on Figure 3-28 indicates they are near double those of the larger size engine – a consequence of the fivefold reduction in duct airflow in combination with only about a 50 percent reduction in liner surface area.

This increase in fractional cooling flow level influences both the emissions characteristics and the thrust efficiency of the duct burner. The diversion of more air for liner cooling increases the nominal equivalence ratio in the combustion air in the high power stage by about five percent which is expected to increase the sea level takeoff NOx emissions by three percent over that of the equivalent burner sized for the VSCE-502B. The increased liner cooling flow is also expected to have an adverse effect on the thrust efficiency because the concentration of excess cooling air at the peripheries of the gaspath will accentuate the temperature profile.

Scaling the duct burner for the same velocity environment while maintaining equal residence times results in a larger aspect ratio of the burner. Although similarity is maintained in the initial mixing zone in the proximity of the swirler tubes, the increased aspect ratio is expected to increase the net mixing accomplished at the end of the burner. This improves the temperature profile at the nozzle and results in an improved thrust efficiency. Analysis indicates this improvement will offset the detrimental effect of the high liner cooling flow and that the combined influence of these effects will cause an increase in the thrust efficiency of about 2.0 percent.

3.4.4 Mechanical Design Evaluation

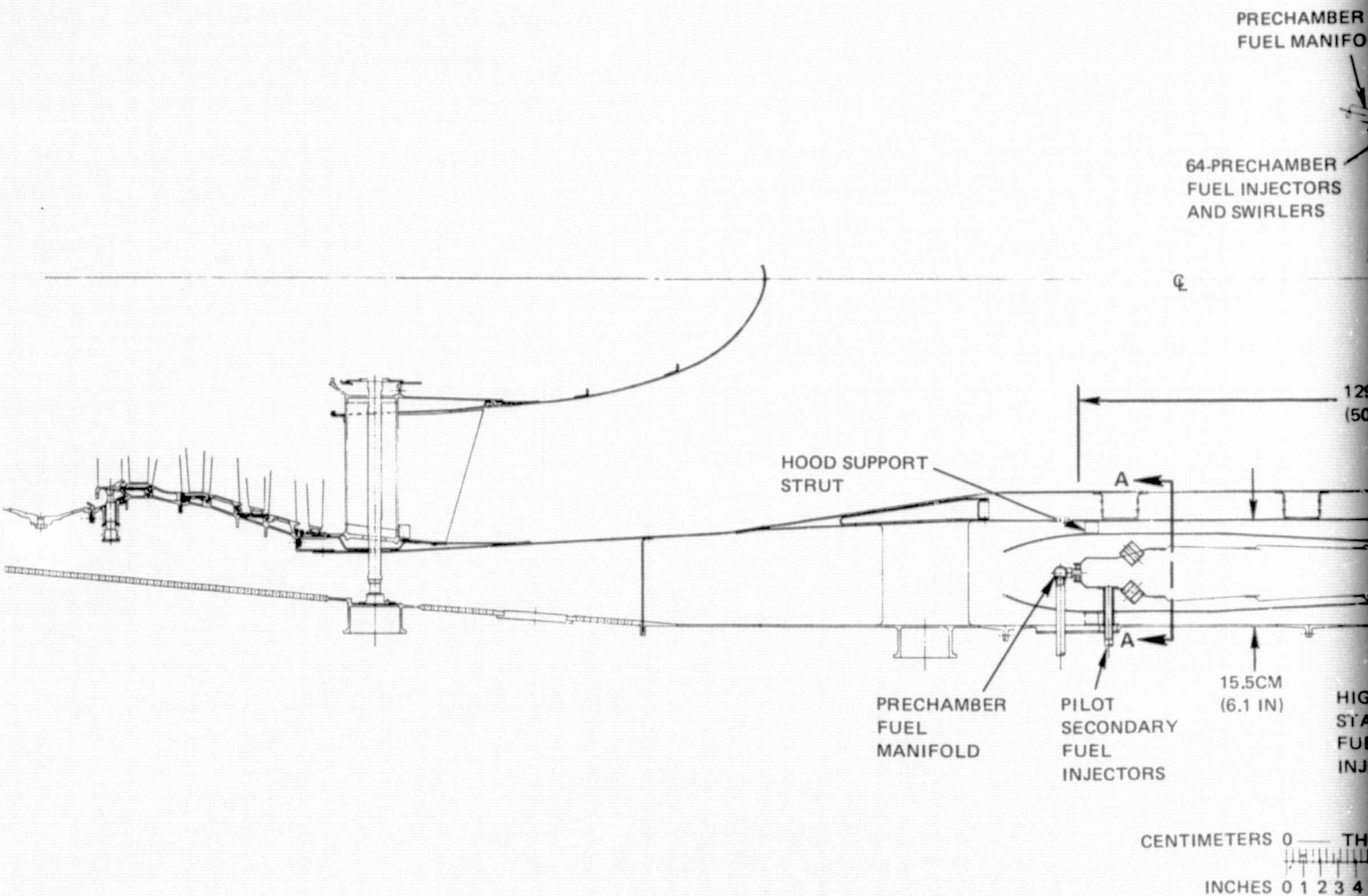
Figures 3-29 through 3-32 show preliminary designs of the four concepts selected for further evaluation in the mechanical design study phase of the program. The intent of this study was to execute a preliminary mechanical design in sufficient depth to identify problems of a structural, mechanical or maintenance nature as well as mechanical constraints that might affect the emissions or performance characteristics. Attempts to solve the identified problems were generally beyond the scope of this program.

Many of the problem areas identified are related to structural and maintainability aspects. Their significance is established with respect to the long life commercial application requirement of a supersonic transport aircraft. As the design studies progressed, it became apparent that many of the problems encountered were common to all of the configurations being evaluated, and must be associated with the duct burner in general. These problem areas are summarized in Table 3-IX and the areas of operational concern associated with each are identified. In general, the problems divide into two categories – liner durability concerns and fuel system complexity. Additional design studies should address the liner durability area and future developments in the cooling and structural aspects of main burner liners may be found applicable to the duct burner. The problem of fuel system complexity is inherent in the design of duct burners with high fuel-air ratio turndown requirements dictating the need for multi-stage fuel systems in even the simplest of configurations. Inovative design approaches may be pursued to enhance the reliability and maintainability of these components, but the operation and performance constraints will continue to dictate the complexity of the system.

The following parts of this section describe the mechanical design features of each of the four duct burners evaluated. Problem area identification is restricted to those aspects that are unique to the particular configuration.

Prechamber Vorbix Concept

Figure 3-29 shows the conceptual mechanical design of this configuration as it would be incorporated in the F-100 sized VCE Critical Technology Testbed Engine. The aerothermal definition procedures of Section 3.4.3 led to the specification of a 15.5 cm (6.1 inch) radial duct height for this configuration. This height is about 10 percent larger than required if all of the velocity levels in the initial aerothermal definition were duplicated. The increase is required to satisfy the minimum annular passage height criteria of the previous section in the gaps between the liners of the pilot secondary zone and the hood over these components as well as in the shrouds around the liners in the high power stage. However, the velocities at all other locations in and adjacent to the duct burner are identical to those in the initial definition in the VSCE-502B airflow size. The overall length of the burner is consistent with the initial definition when it is recognized that the axial length of the mixing regions in the front end of the stages are reduced because of the smaller radial height. Nozzle integration does not present any problem in the testbed engine configuration. The crosssectional area of the high power stage is constant over its entire length and this stage is merely canted inboard slightly to match the nozzle gaspath. The swirler tube density is also consistent with the previous aerothermal definition, but the double row of swirler tubes on the high power stage has been replaced with a single row of larger tubes inclined at a 45 degree angle because experience indicates superior jet penetration and mixing with such an array. Slip joints are employed in the liners upstream and downstream of the high power stage fuel injectors to accommodate the axial thermal expansion of the liners.



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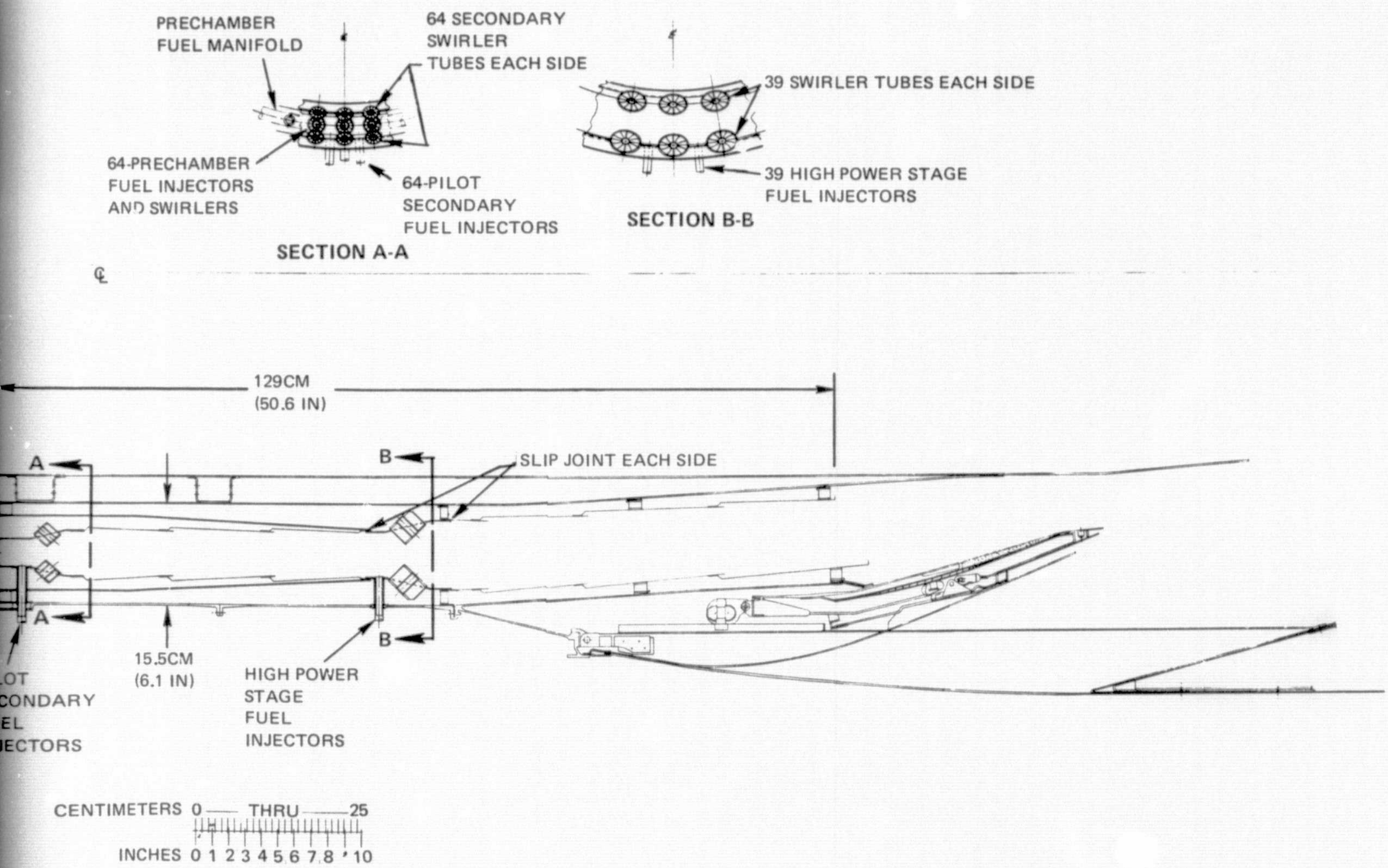


Figure 3-29 Mechanical Configuration of the Prechamber Vorbix Duct Burner in the Testbed Engine

TABLE 3-IX

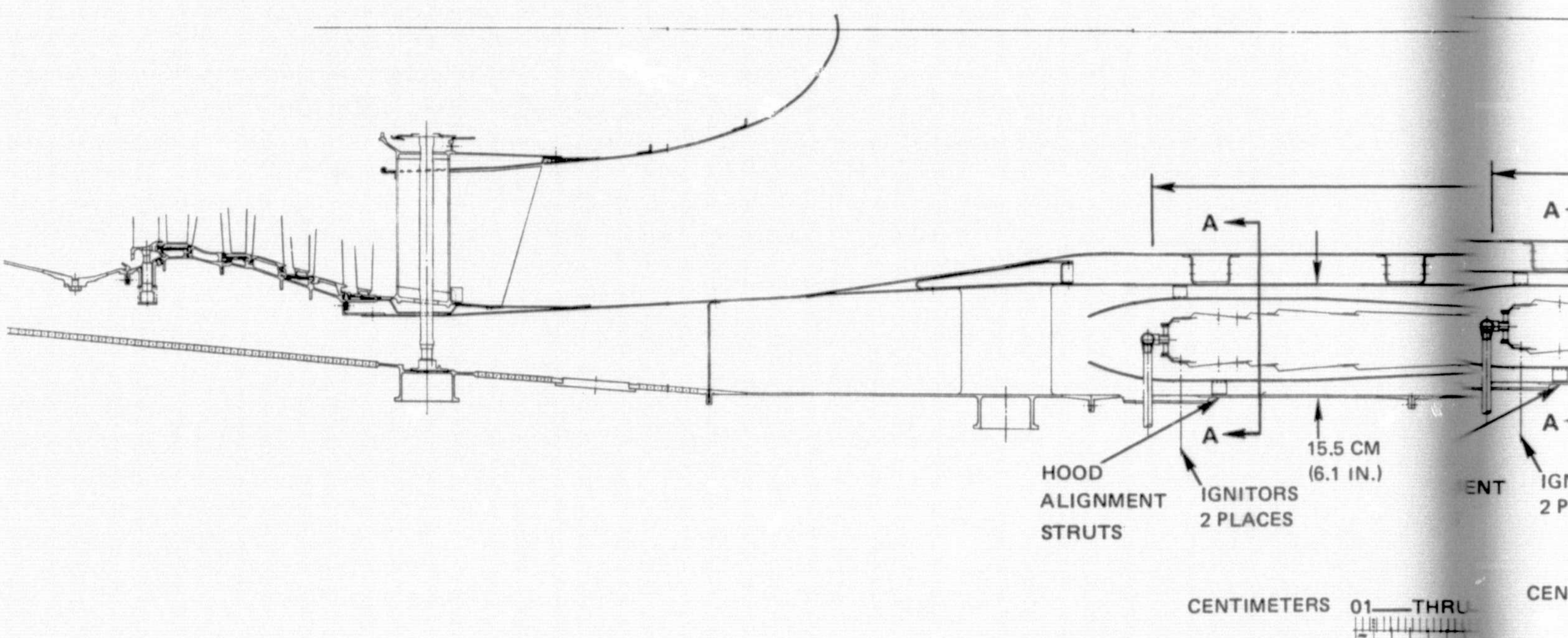
MECHANICAL DESIGN PROBLEMS COMMON TO ALL CONFIGURATIONS

Problem	Operational Concern	Comments
Internal fuel nozzle manifolds	• Maintenance cost • Reliability	• Limited access to nozzles for cleaning and repair. • Ring manifold structure prone to vibratory failure.
Numerous fuel nozzles and case penetrations for fuel lines	• Maintenance cost • Weight • Reliability • Initial fabrication cost	
Complex fuel management and manifolding	• Maintenance cost • Weight • Reliability	• Duct burners normally require more complex controls than main burners. However, emissions will require more precise control. • Axial staging prevents clustering of fuel manifolds at a convenient location of the engine.
Large diameter flat louver panels are prone to high frequency cyclic fatigue	• Maintenance cost • Burner life	
Severe temperature gradients on louvered cooling liners make them prone to low frequency cyclic fatigue	• Maintenance cost • Burner life	• Duct burner liners are typical of louvered liner construction for all burners. However, lower cooling air temperature in duct burner tends to aggravate the temperature gradients.

In addition to the general mechanical design problems associated with all of the duct burners, the installation of the swirler tubes in this configuration causes concern. The high air velocities in the tubes keep the metal temperatures in the tubes low while the adjacent areas of the liner are scrubbed by hot combustion gases. The resultant thermal gradients could cause high stresses and cyclic fatigue failure in the vicinity of the tube to liner weldments.

Single Stage Pilot Vorbix Concept

As shown on Figure 3-30, the mechanical configuration of this duct burner is nearly identical to that of the prechamber vorbix configuration of Figure 3-29. The mechanical problem areas identified with that concept are also present in this configuration. The potential severity of some of these problems are reduced in that there are fewer swirler tube-liner weldments and fewer fuel manifolds.



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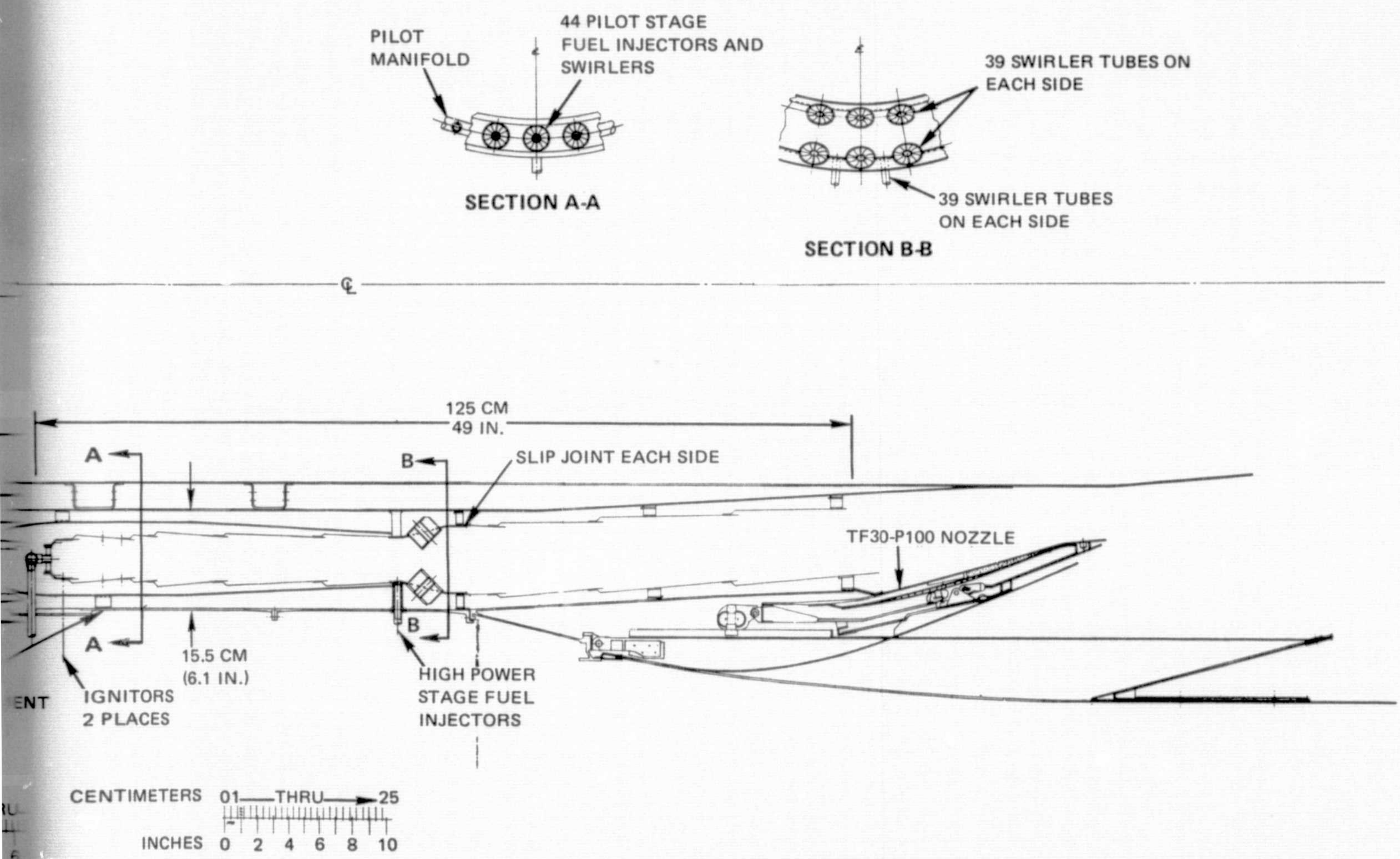


Figure 3-30 Mechanical Configuration of the Single Stage Pilot Vorbix Duct Burner in the Testbed Engine

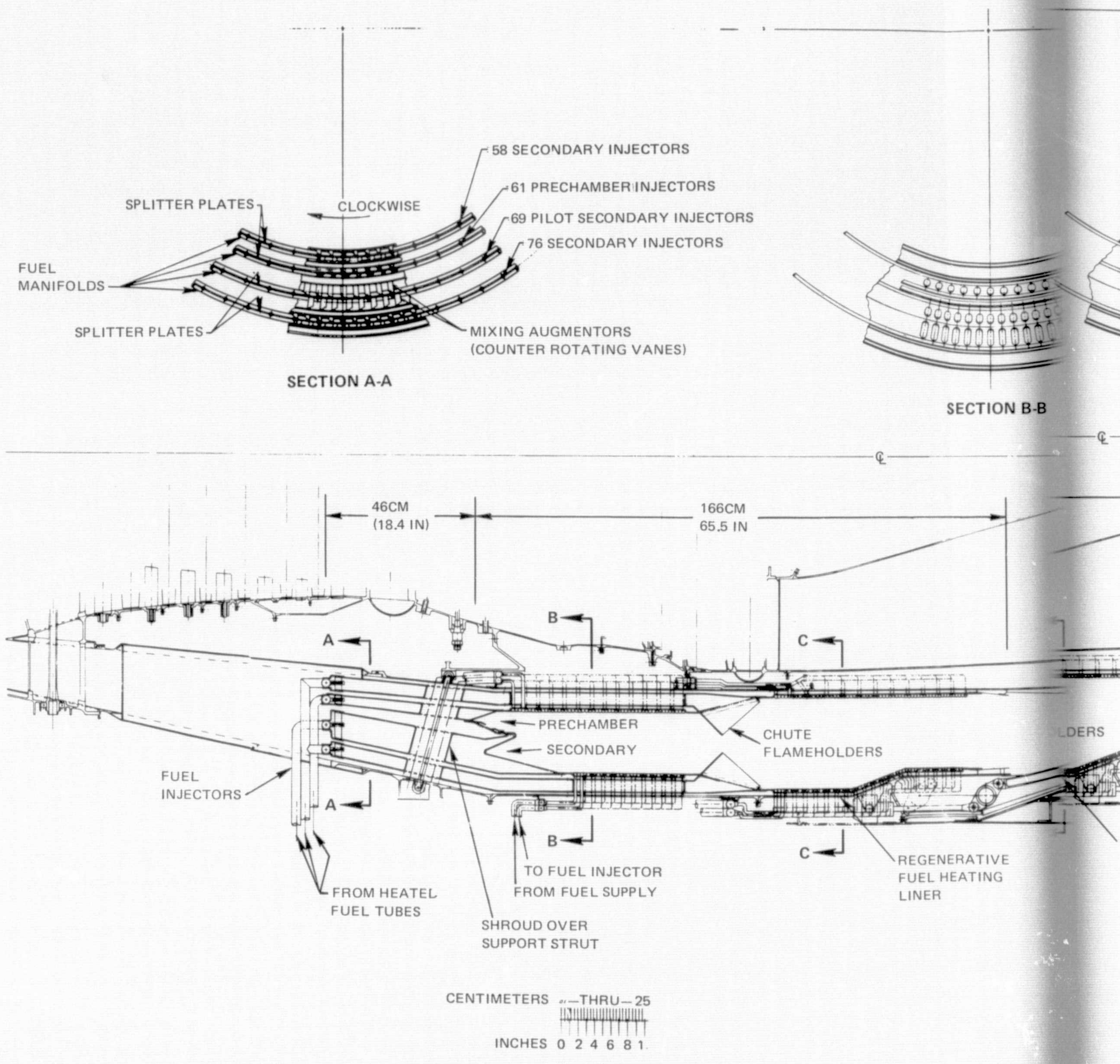
In the initial aerothermal definition of this configuration for the VSCE-502B, the duct height was increased about 16 percent over the goal height in the vicinity of the front of the pilot. This additional duct height was required because a large hood was employed over the pilot to maintain high pressure levels in the dilution air as it passed around the front end of the pilot. When redefined in the F-100 airflow size, this configuration was designed to fit the same radial duct height as the prechamber vorbix concept. Since the hood over the pilot secondary stage cooling louvers had been oversized to satisfy mechanical constraints in that configuration, a substantial fraction of the required hood flow area became available. The additional hood flow area required for effective pressure recovery was obtained by reducing the radial height of the pilot upstream of the dilution air apertures. However, this constriction increases the reference velocity in the primary combustion zone of the pilot by 20 percent, which could compromise the emissions characteristics. Furthermore, higher reference velocities tend to make ignition more difficult, and with this concept being projected as incapable of ignition at fuel-air ratios below 0.0028 prior to this revision, this approach to reducing the duct height does not appear to be desirable.

While the problem of inadequate ignition capability might be resolved with development, it may also be necessary to pursue more elaborate approaches. Circumferential sequencing of the fuel flow to every other fuel nozzle in the pilot stage during ignition would be difficult because the combustion might not propagate across the unfired modules. Variable airflow devices that restrict the flow through the pilot swirlers during ignition could be of value. A relatively simple variable blockage swirler, activated by a unison ring, is described in Reference 35.

Premixed Prevaporized Concepts

Figures 3-31 and 3-32 show the mechanical configurations of the premixed-prevaporized duct burners when these concepts are incorporated in the VSCE-502B engine. These configurations are both three stage burners and differ primarily in the fuel preheating mode. The configuration shown on Figure 3-31 employs regenerative fuel heating in the duct burner liner while that of Figure 3-32 assumes the use of turbine bleed gases to preheat the fuel in a heat exchanger. The later also employs immersed premixing tubes in the high power stage to accomplish some degree of air preheating in this stage.

The premixing passages on both of these configurations extend into the diffuser and must be segmented around the trailing edge of the service strut. The core engine support struts also cross the premixing passages and are enclosed in streamlined fairings to maintain the isolation of the passages. Counterotating vanes are employed as mixing augmentors in the entrance to the premixing passages. The vanes in the passages leading to the high power stages are variable – being rotated into the axial direction when the duct burner is operated at low augmentation levels to reduce the pressure loss in these passages. This produces a higher pressure drop at the flameholders with more effective penetration and mixing of the bypass air to achieve high thrust efficiency.



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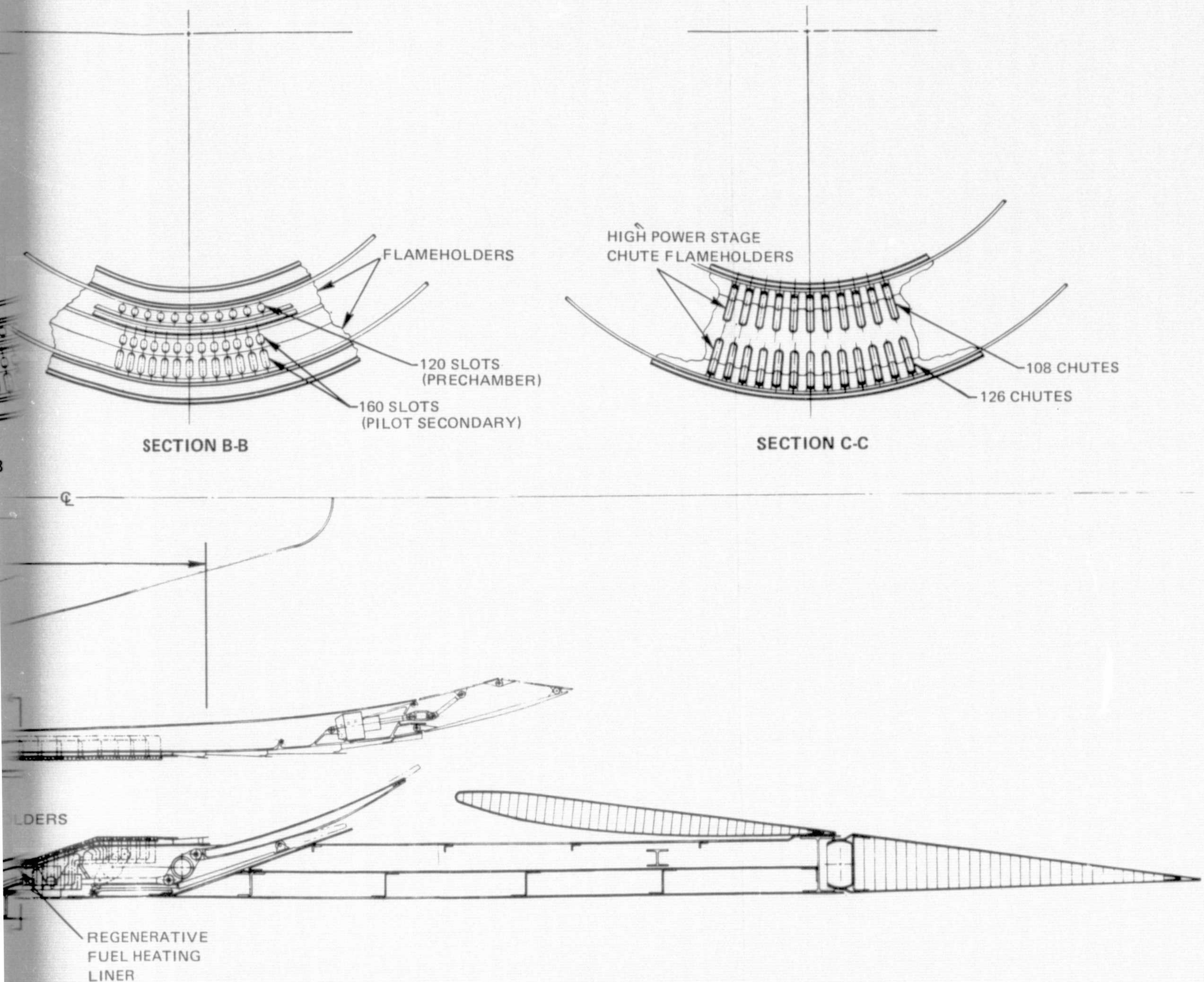


Figure 3-31 Mechanical Configuration of the Premixed-Prevaporized Duct Burner with Regenerative Fuel Preheating in the VSCE-502B Engine

FOLDOUT FRAME 2

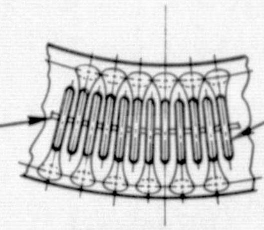
Integration of the fan duct exhaust nozzle with the duct burner leads to a compromise on the overall length of the fan duct. The initial aerothermal definition of Figure 3-21 established a 61 cm (24 inch) effective length for the high power stage. When this stage was integrated with the nozzle actuation mechanisms it was found that the constriction in duct height would begin almost immediately downstream of the flameholder chutes. Since the bulk of the heat release at sea level takeoff occurs in the region immediately downstream of these flameholders, a constriction at this location would increase the total pressure loss substantially. Consequently, it was stipulated that the full duct height be maintained for at least one liner depth downstream of the high power stage flameholder to permit mixing and initial reaction prior to the constriction. This constraint leads to an increase of about 20 cm (7.8 inches) in the effective length of the duct burner. It should be noted that this problem would be encountered in the integration of any of the axially staged concepts into the VSCE-502B engine. Before considering this length increment a penalty, it should be noted that this additional length might be necessary to provide the residence time needed to satisfy the combustion efficiency goal at transonic climb.

A number of mechanical problems, in addition to those listed on Table 3-IX, arise when the advanced technology premixed prevaporized combustor concepts are introduced into the duct burner. These areas are listed on Table 3-X and may be generally categorized into concerns over durability and the strucutral complexity of the premixing passages.

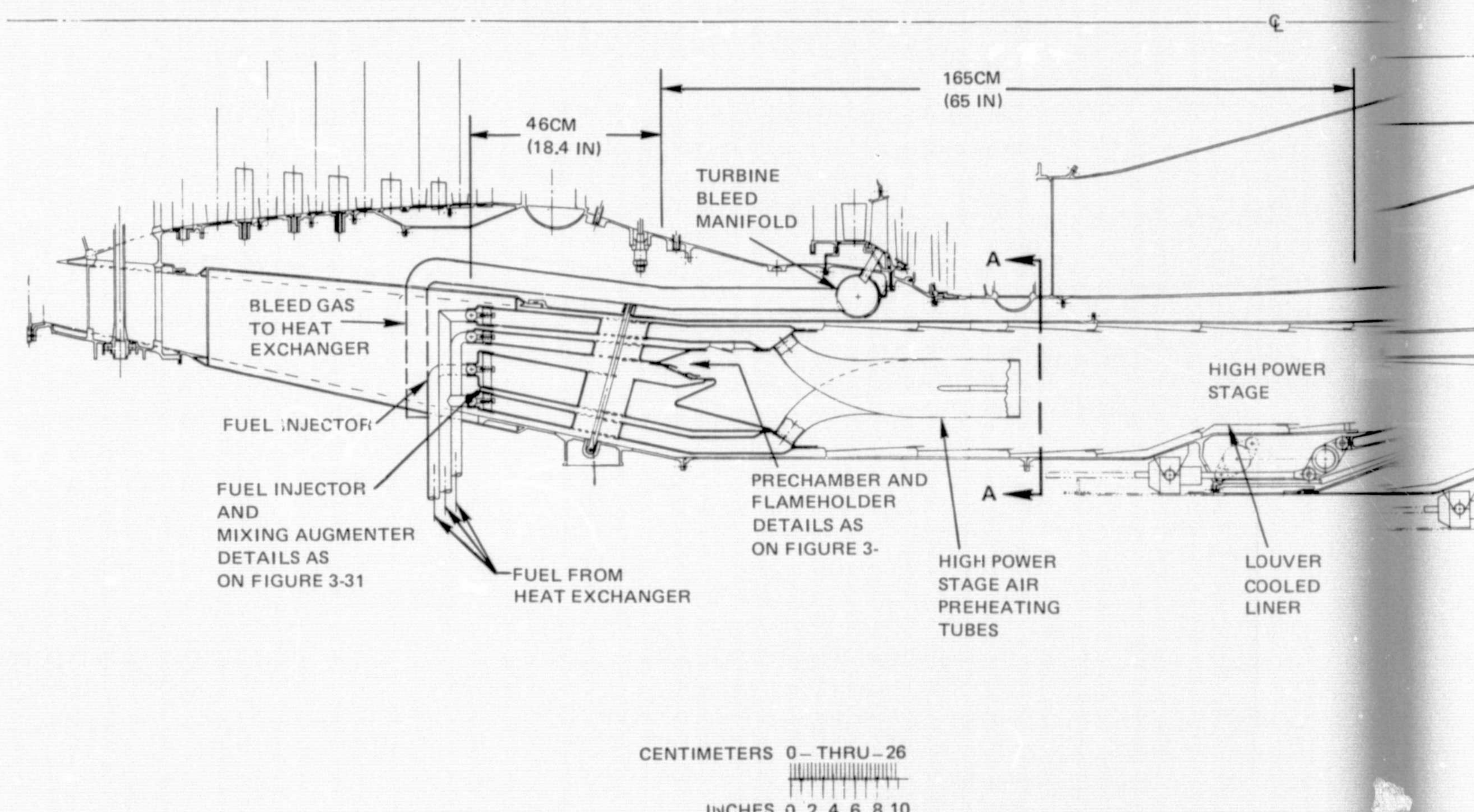
TABLE 3-X

MECHANICAL DESIGN PROBLEMS WITH PREMIXED-PREVAPORIZED CONFIGURATIONS

Problem	Operational Concern	Comments
Fuel nozzles integral with premix passages	• Maintenance	• Assembly problem
Multiple annular premixing passages	• Initial cost	• Assembly problem • Requires close tolerances
Isolation of premixing passages		• Difficult to implement serious design problem - potential safety hazard
• Sealed areas at interface of hot flameholder and cold premixed passages • Numerous mechanical joints	• Burner life • Initial cost	• Severe thermal gradients between hot flameholders and adjacent structure could cause cyclic fatigue problems.
Ignitor location	• Maintenance	• Limited accessibility • Requires penetration of premixed passages
Mechanical support of core engine through premixed passages	• Maintenance • Initial cost	• Assembly problem



SECTION A-A



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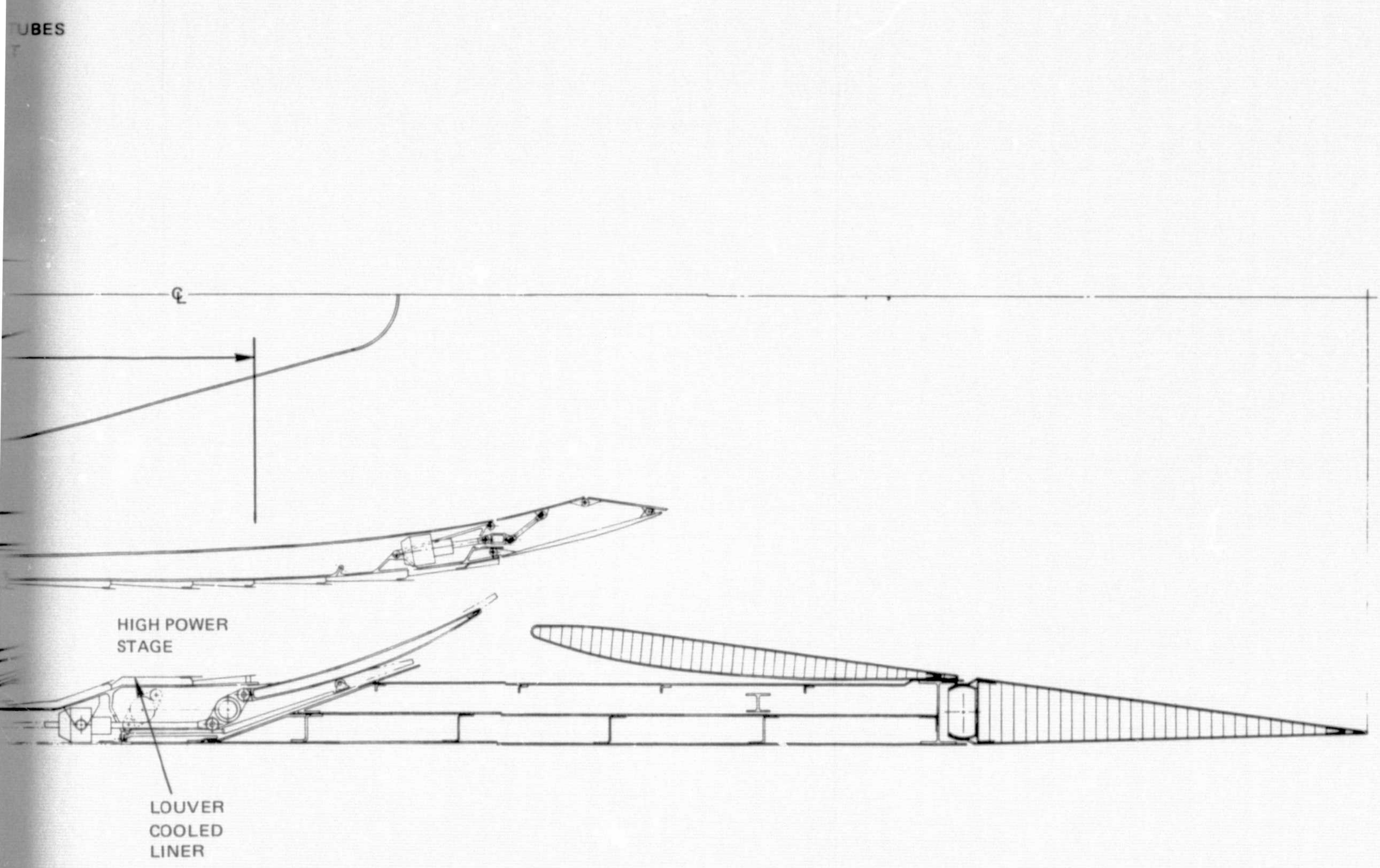

Figure 3-32 Mechanical Configuration of the Premixed-Prevaporized Duct Burner with Turbine Bleed Fuel Heating and High Power Stage Air Heating in the VSCE-502B Engine

FOLDOUT FRAME 2

The regenerative fuel heating system employed on the configuration of Figure 3-31 was sized to provide adequate heat transfer capacity to preheat the sea level takeoff fuel flow of the duct burner to 560K (550°F). As described in Section 3.3.6, this system consists of seventy-five tubes, each of which is wrapped around the circumference of the burner four times before discharging to a collecting manifold. This leads to the need for numerous connections and complex manifolding of the high pressure fuel lines. Potential interference exists between the fuel manifold on the outer liner and the exhaust nozzle actuator mechanism; fuel lines to the walls of the pilot stage crossing the premixing passages for the high power stage; and the need for fuel lines to the inner wall of the burner to span the entire radial height of the fan duct lead to increasing complexity of the system and make maintainability and reliability very questionable. Differential thermal growth of the regenerator tubes could also lead to cyclic fatigue problems particularly at the ends where the tubes join the manifolds. Finally, the potential problem of fuel deposition on the heat transfer surfaces of the regenerator tubes, with the attendant reduction in heat transfer capability and burner life or possible clogging must also be addressed. In view of these problems, the regenerative fuel heating concept appears to be an extremely complex and high risk approach to fuel preheating.

The alternative configuration of Figure 3-32, employs turbine bleed gas to preheat the fuel in an external heat exchanger and a conventional louver cooled liner. The general arrangement of the premixing passages and the stage combustion zones and the problems associated with these features are identical to those of Figure 3-31. The bleed gases used to heat the fuel are collected in a manifold around the turbine case and are brought out through pipes that are routed through the service struts to the heat exchanger modules that are assumed to be located on the outside of the fan duct diffuser case. The material used for the construction of these bleed pipes must be considered a problem area because the temperature of the bleed gases are as high as 1375K (2019°F) at supersonic cruise. In addition, these pipes must pass through regions occupied by the main burner fuel manifolds and the high compressor vane actuating mechanism. Once the bleed gases are delivered to the heat exchangers, the problems associated with fuel preheating are not as complex as those encountered with the regenerative heating concept. The problem of fuel deposition in the heat exchanger is also of concern, and could have substantial impact on the reliability and maintainability of this component. A more elaborate control system would also be required to maintain the bleed flow at levels consistent with the fuel heating requirements over the entire flight envelope.

This configuration also employs immersed premixing tubes in the high power stage to provide preheating of the combustion air entering this stage. Rather than the cylindrical tubes evaluated in Section 3.3.6, the tubes are shown to transit to an elliptical cross section at the downstream end to enhance the dispersion of the tube flow over the cross section of the burner. The tubes are interconnected at the downstream end to relieve some of the load at the cantilevered support and provide rigidity against vibratory loads. Despite these design features, the long term durability of the immersed premixing tubes remains an area of concern.

3.4.5 Overall Evaluation

A final evaluation of the four duct burner concepts was made following the completion of the mechanical design studies. In the moderate development risk category, the design evaluation revealed common characteristics in both the prechamber vorbix and single stage pilot vorbix concepts. The single stage pilot vorbix configuration is projected as having a minimum ignition fuel-air ratio of 0.0028 as opposed to a goal of 0.002. While this deficiency might be responsive to development, it is also possible that more elaborate approaches, such as variable geometry air swirlers or duplex pilot fuel systems, might be necessary to circumvent this problem. In this instance, the single stage pilot loses its principle advantage – a simpler fuel and control system – over the prechamber pilot concept. The problem of inadequate ignition capability should not be minimized because ignition at higher fuel air ratios can cause pressure pulses that could stall the fan and initiate duct flow instabilities. If the duct burner were deficient in another performance goal, the testbed program could still be conducted with meaningful results, but inadequate ignition capability could compromise the entire program.

While the single stage pilot vorbix duct burner is a less complex configuration, the prechamber vorbix concept offers superior ignition characteristics, lower NO_x emissions and better integrating with the fan duct envelope of the VSCE-502B engine. Consequently, it is recommended that the prechamber vorbix duct burner concept be selected for incorporation into the VCE Critical Technology Testbed Engine program.

In the high risk category the two advanced premixed-prevaporized concepts evaluated in the mechanical design studies differed primarily in the means of fuel preheating and the use of air preheating in the high power stage, as opposed to any basic difference in the combustor definition. With the uncertainties present in regard to the extent of fuel and air preheating necessary, and the ability to reproduce the inlet conditions of the idealized research burners upon which the performance is projected, establishing a preference for a particular configuration would be premature.

In general, the advanced premixed-prevaporized combustors are projected to offer potentially lower emissions than the more moderate risk configurations, producing NOx emissions levels about 50 percent below the program goal at supersonic cruise. The benefits associated with the premixed-prevaporized concepts are substantially less at higher augmentation levels. The sea level takeoff NOx emissions of these advanced concepts are about 20 percent above the program goal whereas the more moderate development risk concepts are about 40 percent above the goal. The introduction of premixed-prevaporized duct burners increases aircraft takeoff gross weight 2 to 3 percent,over that with less advanced duct burner concepts, with the increases being due to the weight of fuel preheaters and the premixing components. Since most of the weight and complexity of the premixed-prevaporized duct burner results from the requirement to operate in this combustion mode at high augmentation level, where the emissions advantage of this concept is small, a hybrid duct burner system appears to be a potential alternate concept. This system would employ a premixed-prevaporized pilot stage in combination with a more conventional high power stage such as the vorbix configuration. This combination would offer the capability of achieving the low supersonic cruise NOx emissions of the premixed-prevaporized concepts with smaller takeoff gross weight penalties and only slight increases in NOx emissions at sea level takeoff.

4.0 CONCLUDING REMARKS

Based on the results of this study it is concluded that many of the concepts that may be applicable to duct burners for advanced supersonic engines to maintain low emissions levels are also essential to meeting the performance and operational requirements for this component. The combined requirements of ignition at low duct fuel-air ratio to prevent pressure perturbations that might cause fan instability, high thrust efficiency at low to moderate fuel-air ratios to minimize fuel consumption at supersonic cruise, and the need for higher augmentation levels during take-off and transonic acceleration dictate the need for a duct burner having at least three stages or incorporating variable geometry.

Evaluation of a piloted radial V-gutter concept, based on technology established in previous augmentor development programs, indicates that while this configuration offers the potential for relatively low NOx emissions, it is severely limited in combustion and thrust efficiency at the critical supersonic cruise condition. The aircraft takeoff gross weight penalty for this deficiency is more than six percent over a configuration which meets the performance goals.

Axially staged burners, exploiting technology recently evolved in the definition of low emissions main burners, have the potential of meeting all of the performance goals and requirements, and are geometrically compatible with the prescribed fan duct envelope of the VSCE. While these concepts are projected to provide combustion efficiency levels that meet the program emissions goal and are consistent with the high thrust efficiency requirements, the analysis indicates they will not satisfy the study goal set by NASA for NOx emissions. Of the various configurations evaluated in this study, a three stage duct burner, employing the Vorbix combustor concept in the latter two stages appears to offer the greatest potential for relatively near term application with moderate development risk. It is recommended that this concept be incorporated into the VCE Critical Technology Testbed Engine program.

Advanced concepts, employing premixed-prevaporized combustion technology, may offer potentially lower emissions while meeting all performance and size goals, but would require further substantiation of the projections of this study and extensive research and development to produce a practical duct burner. Because of these risks, these advanced concepts are not considered candidates for the testbed program.

These studies also indicated that the use of variable geometry to enhance the performance, reduce emissions or to simplify the configuration of duct burners was of little value. Evaluation of the potential use of variable geometry in both the moderate development risk staged Vorbix concept and an advanced premixed-prevaporized burner indicated that the strong interaction between pressure loss and frontal area constraints unique to duct burners produced excessive penalties when these features were introduced.

Most of the prior development effort on augmentors has been concentrated on military applications, whereas the duct burners under evaluation in this study have ultimate application in commercial supersonic transport aircraft. In this context, problems associated with durability and maintainability of the components become increasingly more significant because of the requirement for long life. Concerns over the long term structural integrity of large diameter sheet metal liners and cases subjected to potentially high vibratory stresses must be addressed in future design studies.

It should be noted that, as in any study of an analytical nature, uncertainties enter this evaluation because of inadequate data bases and assumptions regarding the extrapolation of data and experience to new and unique conditions. While there is uncertainty in some of the projections made in this study, efforts have been made to maintain the analysis internally consistent. For this reason, the relative magnitudes of the computed parameters used in the evaluation, and consequently the conclusions reached regarding the identification of the most promising concepts, are considered to be valid.

NOMENCLATURE

A	Flow Area ~ cm^2 (in^2)
C_D	Discharge Coefficient
CO	Carbon Monoxide
D	Engine Diameter ~ cm (in)
F/A	Fuel to Air Ratio
Mref	Mach Number at Maximum Cross-Sectional Area of Fan Duct Neglecting Blockage by the Duct Burner
NO_X	Oxides of Nitrogen
P_T	Total Pressure ~ N/m^2 (psia, atmospheres)
THC	Total Unburned Hydrocarbons
TOGW	Aircraft Takeoff Gross Weight ~ kg (lb)
TSFC	Thrust Specific Fuel Consumption ~ kg/Hr/N (lb/hr/lb)
T_T	Total Temperature ~ K (°F)
V	Volume ~ m^3 (ft^3)
Vref	Velocity at a Cross-Section of the Duct Burner in the Absence of Combustion ~ m/sec (ft/sec)
W	Engine Weight ~ kg (lb)
WA	Fan Duct Air Flow ~ kg/sec (lb/sec)
W_{AE}	Core Engine Air Flow ~ kg/sec (lb/sec)
W_{FUEL}	Duct Burner Fuel Flow ~ kg/sec (lb/sec)
X	Axial Distance from Fan Duct Diffuser Exit ~ cm (inches)
η_C	Chemical Combustion Efficiency
η_T	Thrust Efficiency
ϕ	Equivalence Ratio

Duct Burner Component Flows in Percent of Fan Duct Flow:

W_C	Combustion or Dilution Air
W_F	Exhaust Nozzle Flap Cooling Air
W_N	Air Flow Through or Around Fuel Nozzles
W_L	Liner Cooling Air

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