A Tutorial
for the

Rocket Based Combined Cycle Propulsion Technology Workshop

March 23-27, 1992

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Acknowledgements

We would like to thank Dick Foster, Bill Escher, and Eric Hyde for reviewing early drafts of this tutorial. A special thanks goes to John Leingang for his contribution.

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About the Tutorial

The tutorial presented is based on a thorough review of the literature. It is designed to address the key technological issues involved in Rocket-Based-Combined-Cycle propulsion systems and Earth-to-Orbit vehicles that utilize these engines. Each section is designed to be a "stand alone" assessment of a specific critical technology or issue. The tutorial as a whole is intended to give the reader an idea of where the technology stands and where the technology base must be increased.

The tutorial is envisioned to give a starting point for discussions at the RBCC Workshop. It is also proposed as a basis for the workshop final report incorporating inputs from the workshop participants.

The tutorial is organized into the following sections:

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INTRODUCTION

Objective

Current studies are assessing options for the low cost, routine manned/cargo access to Low Earth Orbit (LEO). Parallel related technology development programs are in place. However, one class of vehicles for which comparable concept studies and technology development have not been accomplished is that employing the Rocket-Based-Combined-Cycle (RBCC) propulsion system alternative. For the proper decision to be eventually made regarding development of an advanced propulsion system for Earth-to-orbit missions, it may be desirable to bring the technical base for the RBCC approach to a higher level of maturity.

The objective of the Rocket-Based-Combined-Cycle Propulsion Systems Workshop is to assess the desirability of developing a higher level of maturity in the technology base for the Rocket-Based-Combined-Cycle propulsion system approach through an examination of the feasibility, practicality, and potential benefits of this concept for Earth-to-orbit transportation.

Air-Breathing Terminology

The issue of air-breathing terminology versus rocket terminology has been a sticking point in previous discussions about RBCC systems. To eliminate any confusion between the terminologies, we have included a section to recommend a standard on how rocket and air breathing system performance should be calculated for unbiased assessment.

Additionally, the following definitions will be used throughout the tutorial:

Isp: Propulsive specific impulse defined as Thrust/(Weight Flow of on board propellant).

Net-Jet Isp: (Thrust - Ram Drag)/ (Weight Flow of on board propellant). The ram drag is the drag associated with bringing the air on board. For all-rocket systems, Isp is the same as Net-Jet Isp.

Isp_{eff}: Net accelerative force/(Weight Flow of on board propellant). This instantaneous value accounts for the total vehicle drag and the effects of gravity.

I^*: Total mission effective specific impulse. It is the Isp_{eff} integrated over the mission profile. This value can be substituted directly into the ideal rocket equation to relate the change in velocity to the vehicle weight ratio.

Specific Impulse and Thrust-Drag Relations for Airbreathing Propelled Vehicles

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The following discussion provides some insight to specific impulse, effective specific impulse, and thrust-drag relations in airbreathing accelerative vehicles. Comparison with rocket-propelled vehicles is given. Definitions of symbols used in the discussion are given in Table 1.

The concept of an effective specific impulse is applied to an accelerating vehicle for the purpose of assessing losses to the propulsion system due to aerodynamic drag and flight path ascent. For horizontal flight and no air resistance the effective specific impulse would be identical to the propulsion specific impulse. For a vehicle accelerating essentially horizontally in the atmosphere, Newton's law of motion gives the net acceleration force as:

$$\sum F = T - D = \frac{W}{g_c}\frac{dV}{dt} \tag{1}$$

The effective specific impulse is the net accelerative force divided by the on-board propellant flow rate:

$$I_{sp_{eff}} = \frac{T - D}{dW/dt} \tag{2}$$

Combining (1) and (2) gives:

$$I_{sp_{eff}} = \frac{W}{g_c}\frac{dV/dt}{dW/dt} \tag{3}$$

Equation (3) states that at any instant of a vehicle's acceleration the effective specific impulse is proportional to the velocity gain divided by the propellant expenditure rate. It is important to confine the propellant expenditure to the propellants that are carried by the vehicle. For the case of a rocket, this is straightforward. Both the fuel and the oxidizer are carried by the vehicle. For an airbreathing vehicle, the oxidizer is obtained from the atmosphere, and the drag associated with bringing it into the propulsion system must be accounted for. Only the on-board fuel is treated as propellant expenditure.

The most straightforward way to compare rocket and airbreathing propulsion systems is to express the performance of each by the familiar C_F method. Thus a rocket will have a gross thrust given by:

$$F_G = P_c A_T C_F = \frac{\dot{m}_p c^* C_F}{g_c}$$

For an airbreather, the gross thrust must be reduced by the ram drag (momentum change) required to decelerate the air from free stream velocity, Vo, to the thrust chamber conditions. Thus, for an airbreather, the net thrust is given by:

$$Tnet = F_G - \frac{\dot{m}o\ Vo}{g_c}$$

Two examples are given to show the effect of drag on specific impulse during acceleration and the difference in force accounting between the rocket and the airbreather. Consider the two vehicles in figure 1. They are accelerating essentially horizontally at a velocity of 3000 ft/sec. Each weighs 300,000 lbs. Each has a drag of 100,000 lbs. Each has a net accelerative force of 50,000 lbs. From Newton's law the acceleration of each would be:

$$a = \frac{T - D}{W/g_c} = \frac{50,000}{300,000/32.17} = 5.36 ft/\text{sec}^2$$

Vehicle A, the rocket, can develop this acceleration if a gross thrust of 150,000 lbf is used. Assuming a H_2/O_2 rocket with a propulsion specific impulse of 450 secs, this requires a propellant flow rate of 333 lbm/sec. The effective specific impulse is 150 secs at this acceleration.

Vehicle B, the airbreather, has a propulsion specific impulse, based on the fuel flow, of 3000 secs. Assuming hydrogen fuel and stoichiometric operation; 34.2 air-to-fuel ratio, the net thrust requirement of 150,000 lbf can be developed at a fuel flow of 50 lbf/sec. The air flow required is 1712 lbm/sec and the ram drag for capturing this air is 159,652 lbf. The nozzle gross thrust is 309,652 lbf. The total of fuel and air flow is 1752 lbm/sec. The effective specific impulse is 1000 secs at this acceleration. The net thrust is given by:

$$Tnet = F_G - \frac{mo\ Vo}{g_c}$$

$$= 309,652 - \frac{1712 \ x \ 3000}{32.17} = 150,000 lbf$$

In each case, the effective specific is reduced to 1/3 of its propulsion value. This is because the net accelerative force is reduced to 1/3 of the propulsive thrust.

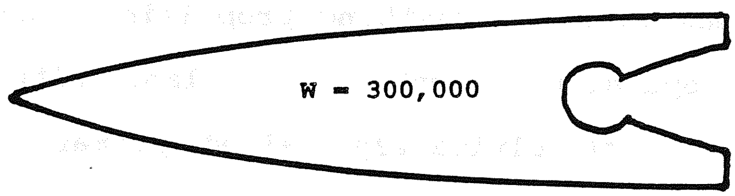
Table 1 - List of Symbols

a	Vehicle acceleration	ft/sec^2
A_T	Engine throat area	ft^2
c°	Propulsion characteristic velocity	ft/sec
C_F	Exhaust nozzle thrust coefficient	
D	Vehicle drag	lbf
F	Net accelerative force on vehicle	lbf
F_G	Gross nozzle thrust	lbf
g_c	Gravitation constant	$\frac{lbm\ ft}{lbf\ sec^2}$
Isp	Propulsive specific impulse	$\frac{lbf}{lbm/sec}$
Isp_{eff}	Effective specific impulse	$\frac{lbf}{lbm/sec}$
$\dot{m}o$	Airbreather inlet air flow	lbm/sec
$\dot{m}p$	Propellant flow rate	lbm/sec
P_c	Chamber pressure of engine	lbf/ft^2
t	Time	sec
T	Propulsive thrust	lbf
Tnet	Airbeather net thrust	ft/sec
Vo	Vehicle velocity	ft/sec
V	Velocity	ft/sec
W	Vehicle weight	lbm

Table - 1 Thrust and Drag Forces on Accelerating Vehicles

A - Rocket Vehicle

Flight Velocity = 3000 ft/sec



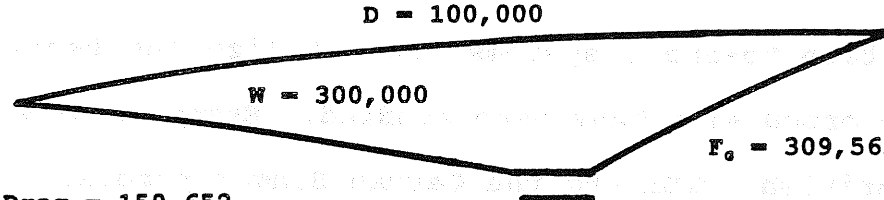
$$I_{sp_{eff}} = \frac{150{,}000-100{,}000}{333}$$

$$= 150 \text{ secs}$$

I_{sp} = 450
$\dot{m}_p$ = 333

B - Airbreathing Vehicle

Flight Velocity = 3000 ft/sec



$$I_{sp_{eff}} = \frac{309{,}562 - 159{,}652 - 100{,}000}{50}$$

$$= 1000 \text{ secs}$$

I_{sp} = 3000
$\dot{m}_p$ = 50

Background

The attraction of single-stage, fully reusable vehicles for access to space is evident. In fact, single-stage-to-orbit (SSTO) concepts date back to 1960 when the original Aerospaceplane was proposed. The Aerospaceplane was "envisioned to take off horizontally from conventional runways, climb and accelerate to hypersonic speeds with airbreathing propulsion, boost into orbit with rocket propulsion, reenter, and land with airbreathing power."[1] This project ultimately failed primarily due to the technical immaturity of hypersonic propulsion and the lack of low weight/high strength materials.[2] Advances in materials technology (carbon and metal matrix composites, for example), improvements in hypersonic propulsion technologies, and the advent of computational fluid dynamics (CFD) have again raised the possibility of SSTO missions. Currently, single-stage-to-orbit studies are underway in France[3], Japan[4], Russia[5], and Germany[6]. The U.S. effort is focused on the well-known National Aerospace Plane (NASP) program.[7,8,9] (The references given for these programs are only examples of the countless number of papers available on the subject.) Even some two-stage-to-orbit systems that utilize the benefits of combined cycle propulsion have been studied. Examples of such programs are the British HOTOL and the German Sanger programs.

Not all SSTO concepts are aerospace planes. The RBCC Workshop will focus on extensively axisymmetric (slender-cone configured), minimal lifting surface SSTO vehicles. In his presentation to the National Research Council Panel on Strategic Assessment of Earth-to-Orbit Propulsion Options[10], Escher lists the advantages of such a system as: lightest weight pressure vessel design; simplest to fabricate (circular cross section); multi-use parts, not special one-use-only-pieces; reduced tooling requirement; maximum feasible airflow capture area (for scramjet); maximum feasible exhaust-nozzle expansion; intrinsically balanced thrust vector location; power-on attitude control via differential throttling; efficient lightweight load-transfer means; minimum drag configuration at high-speed conditions; and minimal exposed area requiring thermal protection.

The vehicles will be propelled by rocket-based combined cycle engines. Combined cycle engines integrate airbreathing propulsion and rocket propulsion into a single engine assembly. Therefore, there is a single engine assembly operating in various engine modes (cycles). The RBCC engine system that is the focus of the workshop transitions from initial air-augmented rocket mode takeoff and initial acceleration to ramjet mode to scramjet mode and finally to all-rocket mode for orbital insertion.

Many of the concepts relevant to RBCC engines were first studied in the mid-1960's by a team led by the Marquardt Corporation. Under NASA contract NAS7-377[11], they began research on composite (combined-cycle) propulsion systems. The study objectives were: 1) To systematically appraise the significance of Synerjet (combined cycle) engines to potential advanced launch vehicle missions in the period, post 1975; 2) To determine the technology ramifications of Synerjet engines with particular emphasis on delineating critical or pacing technology requirements; 3) To systematically and comprehensively document technical data which would be useful for further studies involving Synerjet engines, with emphasis on vehicle/mission applications. During the study, Marquardt examined thirty-six candidate engine concepts. After initial screening, twelve were chosen for further analysis. From these twelve, the two most promising were selected: the Supercharged Ejector Ramjet, and the ScramLACE. Details of these systems will be given later.

This study, however, dealt almost exclusively with two-stage systems in which the first stage utilized these engines and the second stage was all-rocket.[12] The study showed increased performance (payload/TOGW) over all-rocket-powered vehicles for fully recoverable, orbital launch systems (about 4 times the ratio of payload/TOGW for the Supercharged Ejector Ramjet, and about 8 times for the ScramLACE)[13]. They determined that the more attractive combined cycle propulsion systems are characterized as ejector or advanced air-augmented rocket systems which are capable of ramjet operation following the initial acceleration phase.[14]

These air-augmented concepts were revisited by the Astronautics Corporation of America (ACA) under Air Force contract in 1986 and the results reported in AFAL-TR-88-004.[15] This study "focused on the analysis of past work in the field of rocket-based combined cycle engine systems, the selection of five RBCC engines for further evaluation and investigation of design approach alternatives which integrate these concepts into a vehicle design."[16] The five RBCC engines selected were: the Ejector Scramjet; the Supercharged Ejector Scramjet; the ScramLACE; the Supercharged ScramLACE; and the Recycled Supercharged ScramLACE. A brief description of each follows.

Ejector Scramjet

The Ejector Scramjet (ESJ) (Figure 1) is the simplest design of the five. The other four are variations of this design as will be discussed later. The ESJ is designed with variable inlet, fixed combustion geometry and fixed exit geometry. The use of thermal choke eliminates the need for physically variable exit geometry which is required in the Ramjet mode to control the inlet shock position.

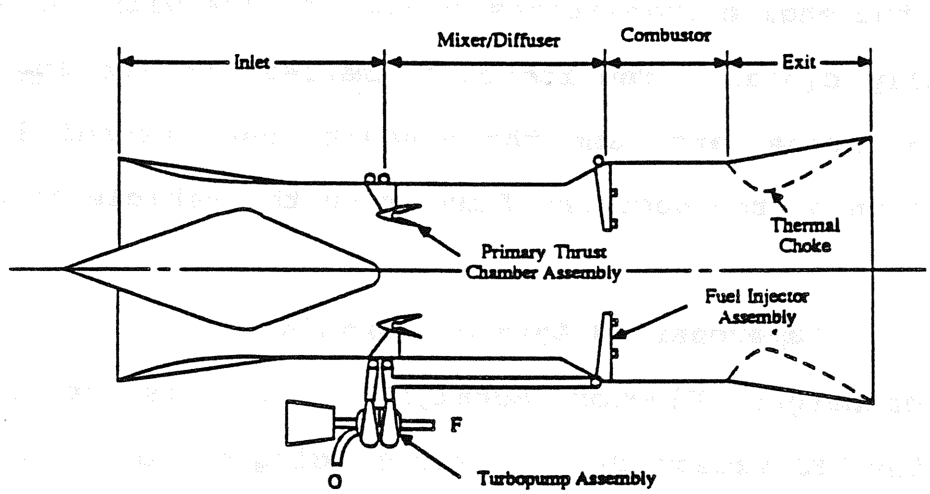


Figure 1: Ejector Scramjet

In Ejector mode, the engine operates at high thrust for liftoff and acceleration to the Mach 2 or 3 range. The rocket primaries are at full thrust using hydrogen/oxygen propellants, and the afterburner is operating at local stoichiometric conditions at full flow.

The engine transitions to Ramjet mode at about Mach 3 at an intermediate thrust level for supersonic to hypersonic acceleration continuing with the thermal choke expansion mechanism. The rocket primaries are off, while the ramjet combustor operates at near stoichiometric conditions.

In the range of Mach 6-8, the engine transitions to the Scramjet mode. Hydrogen fuel is injected in the forward part of the duct, now flowing all-supersonically. Combustion takes place in the constant area and diverging duct section. Exhaust gas expansion initiates in the nozzle and completes on the aft-body of the vehicle.

At some flight Mach number, depending upon vehicle and mission requirements, the engine transitions to rocket mode with the inlet being physically closed. The rocket primaries operate again on hydrogen/oxygen propellant, and the exhaust gases expand in the divergent portion of the duct and finally on the vehicle aftbody.

Supercharged Ejector Scramjet

The Supercharged Ejector Scramjet (SESJ) is configured similarly to the ESJ except that a supercharging turbofan is added between the subsonic diffuser section of the inlet system before the rocket ejector station (Figure 2). The advantages of this design include an increased Ejector mode effective specific impulse, and the availability of a Fan-Ramjet mode which means sharply decreased fuel consumption in flyback, landing, go-around and self-ferry modes when the fan is operated alone as a high bypass ratio turbofan.

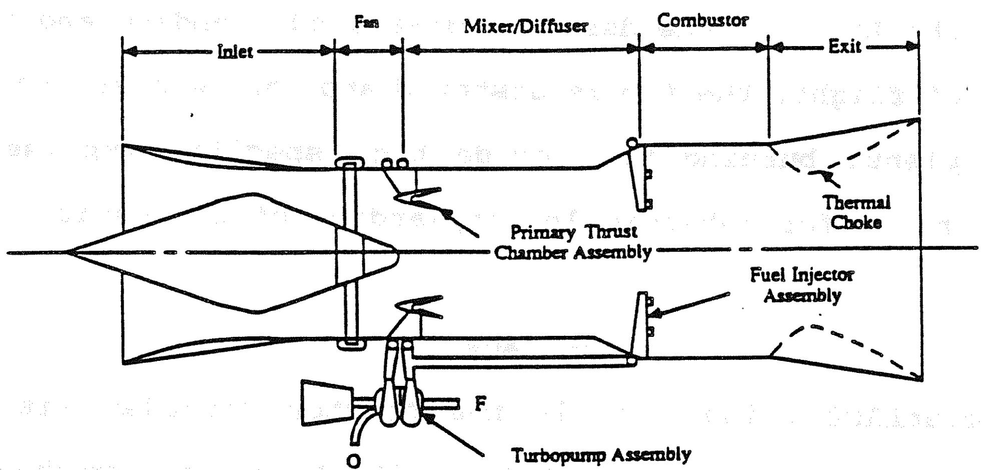


Figure 2:Supercharged Ejector Scramjet

Obviously, these advantages are gained at the expense of additional weight and complexity. These complications include the fan itself, the turbomachinery required to drive the fan system, and the machinery required to remove the fan from the flowpath during scramjet operation. These factors increase vehicle complexity and increase vehicle inert weight.

Initially, the system operates similarly to the Ejector Scramjet with the exception that the fan is operating at full power. In the Fan-Ramjet/Ramjet mode, the engine transitions from Supercharged/Ejector mode to Fan-Ramjet mode in which only the fan and the ramjet systems are operating. As engine inlet recovery temperature rises and the pressure contribution of the fan drops as flight velocity further increases, the fan system is shut down and stowed out of the flowpath. The engine then transitions to Scramjet mode, followed by Rocket mode for orbital insertion as in the case of the ESJ. In the descent, go-around, landing and self-ferry modes of flight, the fan is unstowed and can be operated with or without plenum burning to provide high specific impulse and sufficient thrust for subsonic loiter/landing of the vehicle.

ScramLACE

The ScramLACE (Figure 3) is the Ejector Scramjet with the tanked liquid oxygen replaced in flight with liquid air produced by an on-board air liquification system. The LACE (Liquid Air Cycle Engine) subsytem only operates during the ejector portion of the flight, and then the engine operates identically to the ESJ. Operation of the air liquification system is initiated and liquid air is supplied to the rocket primaries which operate on hydrogen/liquid air throughout the Ejector mode. The system operates extremely fuel-rich because the hydrogen required to cool and liquify the air exceeds that needed for combustion.

The primary advantage of the LACE subsystem is that it reduces the amount of liquid oxygen that the vehicle must carry and thus increases the effective specific impulse of the engine. The disadvantages of the LACE system are the additional weight and complexity of the air liquification subsystem, and the reduced performance of the engine due to operation at an extremely fuel-rich level instead of at near stoichiometric conditions.

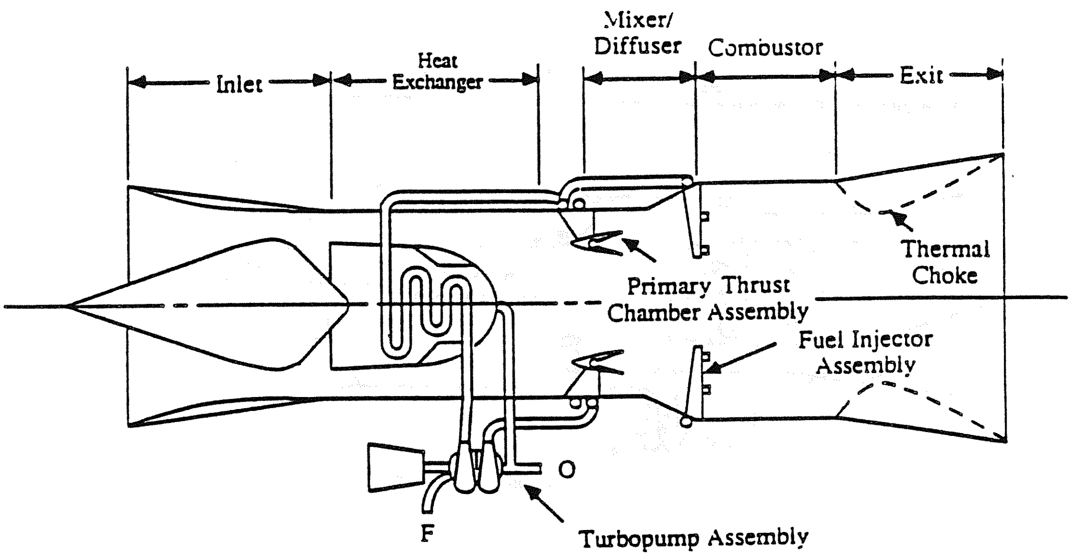


Figure 3: ScramLACE

Supercharged ScramLACE

The Supercharged ScramLACE (Figure 4) is the ScramLACE engine described above with the addition of a supercharging fan. At liftoff, both the fan and air liquification system are fully operating. The air liquification system shuts down as the engine transistions to Ramjet mode. The fan continues to operate at full power in the Ramjet mode until its pressure contribution drops. The fan is then stowed out of the way of the flow. The engine continues on in Scramjet mode and then Rocket mode for orbital insertion. The Supercharged ScramLACE has the advantages and disadvantages of both the fan subsystem and the air liquification subsystem previously discussed.

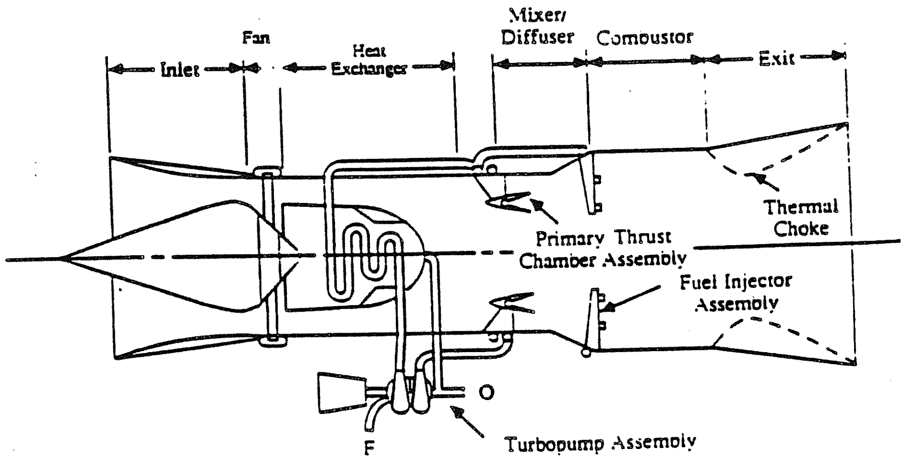


Figure 4: Supercharged ScramLACE

Recycled Supercharged ScramLACE

A disadvantage of the ScramLACE engine is that it operates at a non-optimum fuel-rich level. The Recycled Supercharged ScramLACE returns the excess hydrogen to the hydrogen tank where "slush" hydrogen (a 50/50 mixture of liquid and solid hydrogen) resides. This process results in the recycling of hydrogen at approxiamtely 120 R to the main hydrogen tank where the greater thermal sink of the slush hydrogen is required to recool the hydrogen. The engine is then able to operate at near optimum mixture ratios Ejector mode.

The Recycled Supercharged ScramLACE provides the highest specific impulse of the five choices, but it is attained at the price of added complexity. Slush hydrogen is quickly becoming a developed technology due to the work being done in support of the NASP Technology Maturation Program. Dr. Robert Barthelemy, director of the NASP program, reports that slush hydrogen is now considered a mature technology.[17]

Program Conclusions[18]

1) The study findings indicate that extensively axisymmetric, single stage to orbit vehicles powered by RBCC propulsion systems exceed most and are comparable to the best (Saturn V) multi-stage all-rocket powered vehicles measured in terms of payload to takeoff gross weight (TOGW).

2) While all five engines can provide for powered descent, the inclusion of a fan subsystem allows for a go-around and self-ferry capability. The addition of the fan subsystem also provides for a "supercharger" in the Ejector mode and can act during early Ramjet mode to enhance performance. The challenge of fan stowage for Scramjet mode remains to be addressed.

3) Airbreathing thrust augmentation in the Ejector mode makes a major contribution to final payload capability compared to non-air-augmented rocket engine performance.

4) Air liquification is not required to achieve the target mission case, and its use does not appear to be particularly advantageous

in terms of payload capability for a SSTO vehicle.

5) Air liquification systems required in the above engine designs are not presently available. The problems in air liquification system design include safe, reliable partitioning of the liquid hydrogen coolant from the air flow, and more significantly, prevention of fouling of the heat exchanger by ice formation from the inducted air. Compact, lightweight, reliable heat exchanger subsystems will be needed for a liquid air system to outperform LOX based engines.

6) A substantial portion of the payload performance enhancement achieved through the use of slush hydrogen was due to the 15% higher density of the slush hydrogen. The higher density of the slush hydrogen results in reduced wetted tank area and thus reduced hydrogen tank assembly (structural, tank, and insulation) weight.

7) The reduced vehicle weight achieved through the incorporation of manufacturing materials and processes that are assumed to become available for development applications in the 1995 time frame is important to achieving the high payload fractions projected in this study.

CRITICAL TECHNOLOGY ASSESSMENT

INLET

Inlet design is a matter of aerodynamics and, at high temperatures, real gas kinetics. The challenge of inlet design is that the inlet serves many purposes over the flight regime of the engine. At takeoff, the inlet serves to ingest air for the air-augmented rocket engine. As Mach number increases, and the propulsion system switches to ramjet operation, the inlet compresses the airflow for the combustor section. Finally, for the all-rocket mode orbital insertion, the inlet probably will need to be closed to prevent recirculating flow from developing as a result of engine operation in the main engine duct, and to prevent the need for active cooling of the engine system during reentry[19].

During ramjet and scramjet operation, the primary purpose of the inlet is to capture and compress air for the combustor section of the engine. At high speeds, the engine is extremely sensitive to losses within the inlet. Describing the NASP inlet, Van Wie states "the design of an inlet for NASP involves defining an inlet shape that operates efficiently at high speed but also provides adequate performance at lower speeds."[20] This seems to suggest that a minimum performance standard be set for air-augmented rocket mode inlet operation, but inlet design should concentrate on efficient performance at high speed. The performance of scramjet inlets can be discussed in terms of the amount of air captured by the inlet and the efficiency of the process used to compress the flow.[21] The parameters most often used to describe the inlet performance are the air capture ratio, kinetic energy efficiency, and enthalpy ratio.[22] Together with a geometric description of the inlet, these parameters can be combined to calculate the mass, momentum, and energy of the flow entering the combustor.

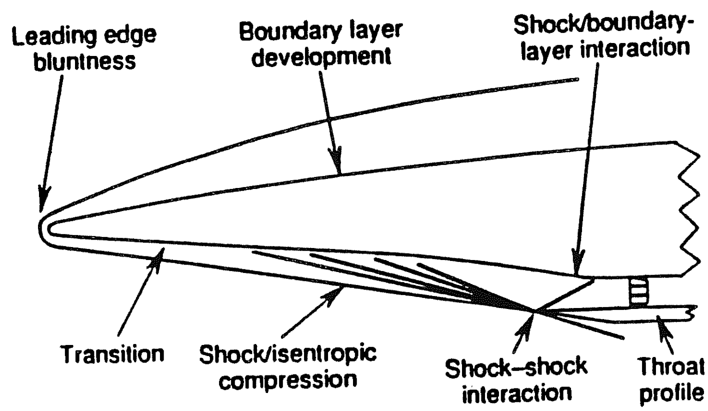


Figure 5: Inlet Schematic[23]

For a slender, axisymmetric vehicle, compression is a two step process (Figure 5). The first step takes place through a bow shock from the forebody of the vehicle. The second step is a result of the compression waves at the inlet entrance. It is preferable to capture all of the air that the forebody and inlet have compressed. The air capture ratio is the measure of this efficiency. Physically, it is the ratio of the amount of air that the engine captures to the total amount of air compressed by the bow shock.

Full capture of the inlet (an air capture ratio of one) corresponds to the inlet capturing all of the compressed air, which means the forebody compression waves fall inside the cowl lip.[24] Kors suggests, "To obtain the required thrust at the higher Mach numbers, the inlet must capture nearly all of the air processed by the vehicle bow shock."[25] Compressed air that is not captured by the inlet shows itself as a drag force called additive (or spillage) drag.

Friction, shocks, and heat loss are all losses associated with the inlet. Kinetic energy efficiency and enthalpy ratio are measures of these losses. The prediction and measurement of these losses represent the major challenges in predicting inlet performance.[26] The kinetic energy efficiency is the ratio of useful energy remaining in the stream to the freestream kinetic energy. The enthalpy ratio is related to the amount of energy lost from the captured streamtube as a result of heat losses. Heat transfer and friction losses within an inlet flow field are generally restricted to the boundary layers. These boundary layers are strongly influenced by the variable entropy layer produced by the curved forebody bow shock, by the transition from laminar flow to turbulent flow, by the effects of adverse pressure gradients, and by interactions with the shock waves.[27]

The prediction of the transition point from laminar to turbulent flow is important because of the rapid increase in local heating rates near the point of transition. According to Van Wie, "Currently, a large degree of uncertainty exists in the prediction of transition from laminar flow to turbulent flow in hypersonic boundary layers. The uncertainty exists both in the prediction of the onset of transition and the prediction of the length of the transition zone."[28]

Another important consideration in the design of a hypersonic inlet is the interaction of the shock waves and boundary layers. These interactions can result from compression corners, reflected shocks, shock cancellations or sidewall compression corners. In these cases the boundary layer cannot negotiate the associated pressure rise and a separated boundary layer results. Separated boundary layers cause large local heating rates and unsteady flow effects which are very undesirable. Van Wie states, "The goal of most inlet designs is to produce an inlet that does not result in any separated boundary layers. The designer must understand the magnitude of the pressure rise that will separate a boundary layer for various types of interactions, and then design the inlet so that this pressure ratio is never exceeded."[29]

In addition to the boundary layer losses described above, at high speeds the interaction of the viscous and inviscid segments of the flow become important. At lower speeds , the two areas can be calculated separately and the boundary layer solutions superimposed on the inviscid flow field. But as the speed increases they must be calculated together, which requires an iterative design procedure.[30]

In addition to being designed for highly efficient operation over the entire Mach number range, the inlet must be designed to withstand perturbations in the flow. These perturbations can be a result of atmospheric anamolies, or can be the result of vehicle operation. In the RBCC vehicle under consideration, there are engines around the entire circumference of the vehicle. The flow seen by the engines on the top surface and the bottom surface will be different because of variations in the angle of attack during vehicle ascent. It is essential that inlet operation remain stable for a wide range of flow conditions. Stable (or started) flow is defined as supersonic flow at the inlet throat location. A sudden flow disturbence can occur downstream of a started inlet due to a transient increase in heat addition or decrease in flow area. This can result in an unstart condition with a normal shock and flow spillage upstream of the cowl.[31]

Test of a full scale inlet model will not be possible because of the large size of the craft under consideration and the enormous power levels that would be required to operate a facility.[32] Limited capability exists for simulating the high temperature aspects of the hypersonic inlet flow fields in ground based facilities because of either their excessive energy requirements, temperature limits of materials, or pressure limitations.[33] Because of the limitations, ground based tests are used to investigate particular features of the flow, such as boundary layer/shock interactions, rather than provide full simulation. Van Wie reports that "existing facilities can match free stream Mach Number, specific heat ratio, Reynolds number, and the ratio of wall to free stream temperature for speeds up to about Mach 15."[34]

This contradicts the National Research Council Committee on Hypersonic Technology for Military Application which says, "capabilities for aerodynamic and propulsion testing in the hypersonic regime are extremely limited below Mach number 10 and virtually non-existant above Mach number 10."[35] This is a contradiction that must be resolved at the workshop.

Where testing is either expensive or diffficult, CFD codes have been providing useful data. White reports, "...the maturing of capabilities in CFD have reduced the dependence on testing by nearly eliminating the need for parametric experimental investigations in the preliminary design phase. CFD techniques have also proven to be a valuable diagnostic tool with which to compliment and interpret experimental data."[36]

NOZZLE

The nozzle is the thrust producer of the engine. Without an efficient expansion of the gas, the inlet drag will be greater than the thrust produced by the nozzle and acceleration will not occur. Nozzle design is especially difficult because the same nozzle must serve each of the engine modes over the entire flight regime of the vehicle. Initially, the nozzle serves the air-augmented rocket mode. One option for air-augmented operation is the diffusion and afterburning (DAB) approach. The idea of DAB is to mix a non-fuel rich supersonic rocket ejector drive jet with a subsonic air stream (ingested by the engine inlet) and to further expand the combined flow stream in such a manner as to increase static pressures and, at this point, to introduce additional fuel into that combined flow stream and then to expand the total flow through a convergent/ divergent nozzle. While this is the most efficient operation for air-augmented rocket mode, a convergent nozzle section is undesirable during scramjet mode, where the entire flow stream is always at a supersonic velocity.[37]

One option to resolve the difference is to use a variable geometry nozzle assembly. However, the variation in the nozzle assembly would not just be one of changing exhaust area ratio. It would involve changes in which the convergent section of the nozzle is removed completely for scramjet mode. An alternative is the use of a thermal choke (Figure 6). A thermal choke is created by scheduling the heat release rate in the subsonic, constant pressure mixed flow combustion passage in such a fashion as to establish local sonic conditions. Supersonic expansion occurs on the remaining portion of the divergent nozzle. The advantage of the thermal choke is that it establishes supersonic flow in the air-augmented rocket mode without the use of a convergent section that would impede scramjet mode operation. While the total inert weight per unit pound of payload is slightly higher for the case of the thermal choke (22/1 for thermal choke, 21/1 for variable exit geometry)[38], the use of variable exit geometry introduces increased system weight and complexity. The principal investigators of AFAL-TR-88-004 concluded that "the use of a variable exit geometry is not justified in terms of increased system weight and complexity."[39]

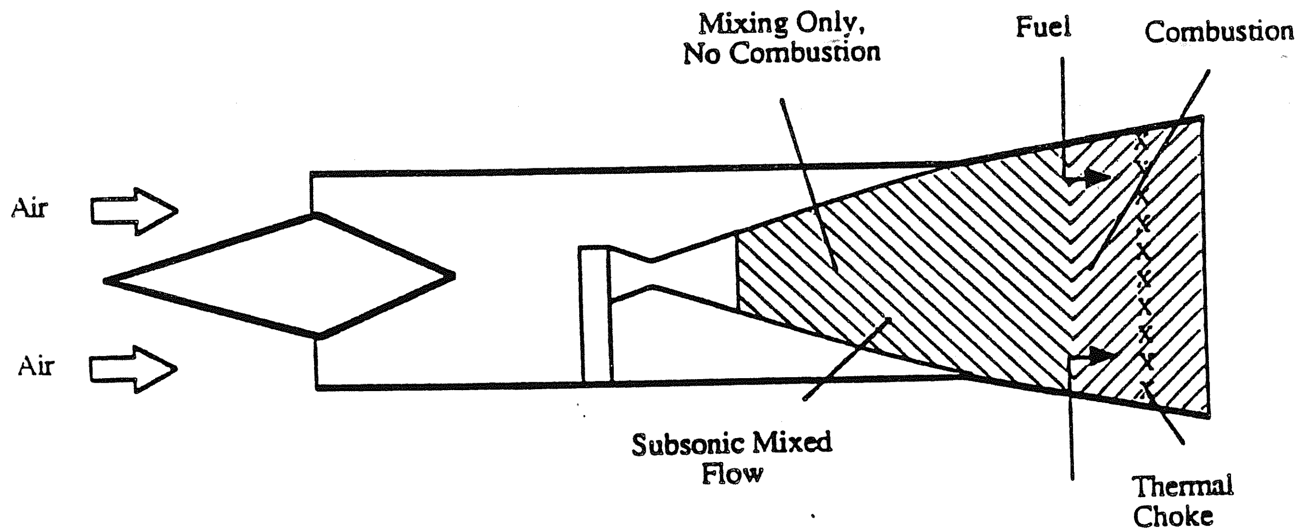


Figure 6: Thermal Choke

The nozzle receives hot gas from the combustor and acts as the expansion surface for that gas. The expansion process essentially changes the pressure and heat energy in the gas to kinetic energy. Fluid dynamic losses in this process include skin friction, heat transfer losses and non-isentropic expansion of the gas. As the gas expands in the nozzle, boundary layers are formed along the nozzle walls as a result of the friction between the walls and the gas. The loss of momentum in the boundary layers reduces the overall thrust available. Heat transfer losses are generally confined to the boundary layers as well. Heat transfer losses reduce the total amount of heat energy in the flow and thus the efficiency of the expansion. If the expansion surface contour is poorly designed, internal compression waves can form in the nozzle. As the flow crosses each of the waves, the flow experiences a loss in total pressure. The presence of these compression waves results in a non-isentropic expansion of the gas, and thus reduced performance.

A major thrust loss mechanism in supersonic nozzles at high Mach numbers is the thermochemical energy retained by dissociated species when subjected to rapid expansion.[40] Energy retained by the dissociated species reduces the total amount of energy available as kinetic energy. This obviously reduces engine performance. An added complication during high speed operation is the presence of a reacting gas in the nozzle. This is as a result of the relatively high temperatures and short residence time in the combustion chamber. The NRC report on Hypersonic Technology for Military Applications states, "It is important that these reactions be completed to the maximum degree during expansion in the nozzle. Because it is most unlikely that crucial experiments can be done, a high degree of reliance must be placed upon computation. These must, at the least, be for two-dimensional reactive flow and preferably three-dimensional."[41]

The NRC reports, "Even approximate calculation of the nozzle expansion process makes it clear that the pressure distribution over the nozzle surface and, consequently, the calculated performance of the engine, is very sensitive to the details of the gas state distribution at the start of expansion."[42] Nonuniform entrance conditions can generate internal compression waves resulting in reduced nozzle performance.[43] The main objective in developing the CFD capability for nozzle analysis is to evaluate the effect of nozzle inflow nonuniformities and the turbulent mixing of those non-uniformities on nozzle performance. CFD analysis has shown that non-uniform nozzle inflow actually produces a more uniform Mach number profile at the exit. However, the exit mass flow distribution for the non-uniform inflow is more non-uniform than for the uniform inflow case.[44]

Limited research has been done on hypersonic nozzles due to limited test facility capabilities. Experimental research must be augmented by analytical investigation to demonstrate nozzle flow behavior. The goals of nozzle development should be a better understanding of the expansion process, development of a data base for nozzle performance over a wide range of Mach numbers, and eventually design of an "optimum" nozzle contour for operation over the entire flight regime.

The large nozzle expansion ratio required for efficient hypersonic operation leads to serious base drag problems at transonic speeds, where the nozzle pressure ratio is far too low to fill the entire base area. Full annular flow of engine exhaust products of combustion on the aftbody is expected to significantly reduce the base drag.[45] However, base burning is a possible alternative, but one whose heating and fuel consumption implications have not been fully explored.[46]

A combination of factors makes the task of nozzle design a difficult one. Nonuniform inlet conditions, the expansion of a reacting flow, and the need to operate efficiently over the entire Mach number regime are all considerations that must be accounted for in nozzle design. The problem becomes greater when one considers test facility limitations and the shortcomings in present CFD technology. The NRC concludes, "At present this magnitude of numerical calculation is impossible and the situation is unlikely to change very soon. The nozzle may be one aspect of NASP that undergoes significant development during flight test."[47]

COMBUSTOR

The objective in combustor design is to maximize the amount of heat release with a minimum of total pressure and thrust losses. In the RBCC engine under consideration, the combustor is the section of the engine where the rocket ejector resides, the fuel and ingested air are mixed, and combustion occurs. The combustor must be designed for high combustion efficiency and maximum possible heat release.

Initially, the combustor acts as the chamber where the ingested subsonic air and the supersonic ejector jet are mixed. In the preferred diffusion and afterburning (DAB) approach, a non-fuel-rich supersonic ejector jet and the subsonic airstream are mixed and expanded in such a manner as to increase static pressures. At this point additional fuel is introduced and then the flow stream is expanded through a convergent/divergent nozzle (or through a thermal choke nozzle as explained in the nozzle section). Extensive work on this subject was carried out under

NAS7-377[48] in the mid-sixties, but not much work has been done since. The issue of air-augmented rocket combustion will have to be re-examined for RBCC applications.

During ramjet mode, the rocket primaries are not operating. Fuel is supplied from injection assemblies in the aft section of the combustor. Ramjet technology is supported by a very extensive database, covering both analytical methods and actual engine development, manufacturing and flight use.[49] Succesful ramjet combustion has been demonstrated in very short combustion lengths on the order of less than one foot. Experimental work has shown high combustion efficiencies approaching 98%.[50] The only adaptation of current ramjet technology needed for the RBCC engine concept is that of the flameholder/fuel injector. These must be retractable so that they will not be in the flowpath during scramjet operation.

Due to the relative maturity of rocket and ramjet combustor technology, this tutorial will focus primarily on the scramjet mode combustor operation. The combustion efficiency for scramjet operation is based on several factors including: the shock-on-lip-Mach number which establishes the shock structure in the combustor; the thermal management of the engine (and vehicle) which determines how much fuel must be circulated to cool the engine; the efficiency of the air and fuel mixing; and the chemical kinetics of the fuel-air reactions.[51] These processes are dependent on the flow profile and properties provided by the inlet.

Associated with a scramjet combustor is an isolator. An isolator is section of constant area duct between the inlet and the combustor which confines the shock train to prevent undesirable combustor-inlet interactions. Billig reports that "failure to account for the presence of strong shock trains in analysis of engine test data leads to spurious results and erroneous solutions. In particular, unrealistically fast mixing and heat release rates are generally deduced."[52]

At high flight Mach numbers, protection of the combustor wall is of paramount importance due to the high enthalpies of the incoming flow.[53] Film cooling offers a solution to the problem of wall heating while offering the potential of reducing wall shear. In film cooling, a portion of the air ingested in the inlet does not participate in the combustion process and therby provides a film barrier to mitigate the extremely high heat transfer rates in the combustor (and the nozzle).[54]

Flame stabilization and combustion is a major process of concern. It is tied directly to the processes of fuel injection, vaporization, mixing and ignition processes as well as combustor geometry. This is significant because nearly all of the experimental combustion data is configuration dependent.[55] Billig suggests that "to assure engine relight in the event of flameout or engine unstart, an ancillary ignition source will probably have to be provided up to (free stream) Mach number=9."[56]

The NRC reports, "A serious concern among workers experienced in the field is the stability of the hypersonic flow in the combustion chamber during combustion. Would a small disturbance imposed on a flow field decay, diverge, or lead to a pulsating combustion process? A central obstacle to understanding the result of such a time-dependent disturbance to the combustion chamber flow is our current incapability (sic) to either experimentally measure or to compute this chemically reacting flow."[57]

The process of injecting the hydrogen fuel and mixing it with the air in the scramjet appears to be the most difficult obstacle to the realization of a successful engine.[58] In the RBCC engine, one of the options for fuel injection is utilizing the rocket ejector (operating extremely fuel rich) for co-axial fuel injection. Co-axial injection has certain advantages. Billig states that fuel injection "must approach coaxial at high (free stream) Mach numbers. At (free stream) Mach numbers > 10 the momentum of the fuel becomes an increasingly important element in the thrust potential of the engine."[59]

Certain physical properties of the combustion process need to be studied. These include a knowledge of the mixing mechanisms in the free stream and along the wall, a knowledge of compressible fluid dynamics in the combustor region, and for axial fuel injection the free shear layer characteristics need to be understood. This work should be carried out analytically and experimentally and a data base of the results as a function of Mach number and other inlet parameters must be gathered. These results are necessary to acurately assess combustor performance and to predict the conditions at the nozzle inlet to aid in nozzle design and development.

TEST OPERATIONS

Equal to the challenge of designing and building a RBCC vehicle is testing it. The National Research Council's Report on Hypersonic Technology for Military Applications reports, "... it should be noted that many of the (test) facilities are 20 to 40 years old and do not have flow conditions of sufficiently good quality for aerodynamic and chemical kinetic testing and code validation."[60] Associated with any hypersonic propulsion initiative will have to be a "substantial and continuing development of new facilities and instrumentation."[61]

It is unlikely that all system components will be able to be tested on the ground. The facilities for hypersonic testing are inadequate to meet the needs of such systems. The NRC reports, "For stream Mach numbers above about 10, test requirements become even more severe for slender vehicles because Mach number, free flight Reynolds number, and free enthalpy must be achieved in a single test facility as real gas effects become important."[62] An added difficulty in testing is that testing facilities using air as the working fluid are required since "other gas media (freon, helium, pure nitrogen) do not provide the right quantitative simulation."[63] As a result of these difficulties much of the "ground testing" will need to be done on a computer using CFD codes. Some testing will need to be done, however, for code validation.

The above testing requirements are only for the aerodynamics and propulsion system. The vehicle's materials and structures also need testing. A SSTO vehicle requires ultra-lightweight materials that are capable of withstanding the hypersonic environment. The Air Force conducted a study on the high temperature materials test technology needed for hypersonic vehicle applications. The NRC reports, "They found heating capability above 1400 C will be difficult to achieve and that instrumentation is not availible for use above 800 C. Also high-temperature strain gauges are not availible for temperatures above 800 C. When testing must combine flowing air and thermal cycling to obtain the necessary data for structural design, one must conclude that additional testing capabilities are needed to get the data in an expeditious manner."[64] A full scale test facility to accomplish the above mission was estimated to cost $462 million. In the National Space Transportation and Support Study prepared by the Joint DOD/NASA Transportation Technology Team the total structures/materials facilities were estimated to cost $554 million.[65]

Overall , the facilities for testing a hypersonic propulsion vehicle do not exist and an effort is needed to design and build (or at least refurbish) facilities capable of accomplishing this mission. Currently, this effort is underway for the NASP program. Even with this effort, much of the data will need to come from flight testing. The National Aerospace Plane is described in the literature as being "a flying research facility, proving propulsive concepts at speeds up to Mach 25 that cannot be duplicated in existing ground faciilities."[66] Therefore, the NASP test vehicles can provide a major part of the technology research base for high speed operation of RBCC/SSTO vehicle systems.

Controls

The control system for the RBCC vehicle will play a key role in its success. The system must control almost every crucial element in the vehicle, such as the inlets, the thermal protection system, the vehicle trajectory, both the vehicle structural and slosh modes, the nozzle thermal choke, as well as the constant monitoring of the engine performance.[67]

As mentioned in the section on inlets, inlet unstart is a condition where a strong normal shock is moved in front of the inlet throat resulting in flow spilling out of the cowl. The two principal factors affecting unstart are inlet disturbances (atmospheric anamolies) and back pressure from the combustor or engine geometry.[68] Control of the disturbance effects and combustor back pressure is important to mission success. This can be done by having highly responsive control systems that prevent inlet unstart by rapidly decreasing fuel flow and/or increasing downstream flow areas.[69] The control system must also control the variations in inlet geometry as a function of flight conditions (altitude, speed, etc.)

High speed ramjet and scramjet operation produce high total temperature air that, even in the inlet before combustion, may greatly exceed the safe operating limits of the flow path materials. Standard design practice requires that each coolant circuit be designed to provide constant flow to accomodate the peak heat flux at that location.[70] During periods of high heat flux, the hydrogen rate required for cooling greatly exceeds that needed for combustion. Excess hydrogen results in non-optimum performance of the scramjet. Therefore, the amount of excess hydrogen used for cooling must be kept to a minimum. This suggests the use of an actively controlled thermal protection system to balance the flow of coolant to various parts of the vehicle during flight in order to maintain thermal margins and minimize the use of excess hydrogen for cooling.[71]

Throughout the flight the engine will be subject to perturbations from atmospheric anamolies and as a result of the trajectory (angle of attack, and sideslip of flow from one engine to another). These factors effect the total amount of air captured by the inlets, and therefore engine performance. Engine controls will include the valves and effectors manipulating the flows and geometry of the engine, the controlling logic embodied in the real time software, the implementing controllers/computers and the suite of instruments feeding back control parameters from the engine.[72] These controls must be compatible with and interact with the vehicle management system to configure the engines for delivery of the commanded thrust while imparting the desired lifts and moments to the vehicle -including correction of imbalances for each engine. The controls must also maximize engine performance by controlling and maintaining proper engine fuel-air-equivalence ratios. And finally, they must ensure engine and vehicle safety by controlling coolant flows, providing smooth mode transitions, minimizing unstart/ restart transients, effectively monitoring engine condition and stating parameters for signs of degradation and decoupling/desensitizing outlet flows to inlet conditions.[73]

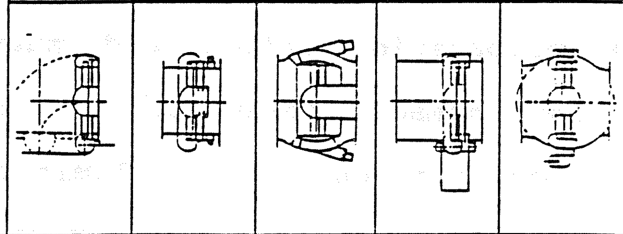
Engine Integration

In AFAL-TR-88-004, the principal investigators concluded that "the technology status of RBCC engine systems is nearly adequate at the component/subsystem level." "This knowledge of component/ subsystem technology is adequate to support an advanced development program with the objectives of investigating the technology of INTEGRATION of these components and subsytems into a single RBCC candidate engine configuration."[74] Integration of this kind has a limited history. One example was Marquardt's early 1960's test of an ejector ramjet system. Engine tests were successfully conducted at sea level/static conditions and direct-connect simulated flight conditions up to Mach 2.2. Predicted performance was demonstrated in ejector mode through transition to ramjet and in ramjet mode, and the capability for a controllable mode transition from ejector to ramjet mode were experimentally demonstrated.[75]

The essence of combined cycle propulsion is a single engine assembly operating in various modes. This implies that at least some of the components are used for more than one of the cycles. Examples of this "multimode implementation" are the ramburner that doubles as the air-augmented rocket mode afterburner, the rocket ejector that injects fuel during scramjet operation, and if a LACE system is used, the injector that performs as a dual-oxidizer (LOX and LAIR) injector. The degree of multi-use hardware that can be used is still in question. However, to arrive at the highest performance, lightest weight design, multiple use of the hardware is mandatory.[76]

Engine integration must consider the selection of an ejector system. In the Introduction, the five candidate RBCC engine designs were described. Each system is capable of performing the required mission (10klbm to 100nmi orbit).[77] However, mission flexibility (the inclusion of an efficient self-ferry, go-around and loiter capability) suggests that the inclusion of a supercharging fan is necessary. It was also the conclusion of AFAL-TR-88-004 that an air liquification subsystem is not particularly advantageous for a single stage mission.[78] These factors suggest that the Supercharged Ejector Scramjet may be the most attractive ejector design.[79]

If a supercharged version is chosen, the choice of a fan stowage system becomes important. The Marquardt Corporation studied five different stowage options.[80] Five different criteria were used to rank the options: weight of the assembly, reliability, maintainability, recurring cost, and adaptability to either fan drive tip turbine or shaft. The two designs with the best overall scores (the windmilling and bypass options) were eliminated due to the requirement that the fan and its assembly be completely out of the way of the flow path during scramjet operation. The results are summarized in Figure 7.



NOTE: THE LOWER THE POINTS ASSIGNED THE HIGHER THE PREFERENCE INDICATED	OFF-AXIS SWINGING	WINDMILLING	BYPASS	IN PLANE ROTATION	IN PLACE ROTATION
WEIGHT ASSESSMENT	5	1	2	3	4
RELIABILITY	3	1	2	3	3
MAINTAINABILITY	3	1	2	3	3
RECURRING COST	4	1	2	3	3
ADAPTABILITY TO EITHER FAN DRIVE TIP-TURBINE OR SHAFT	3	1	2	3	3
EFFECT OF INSTALLED PERFORMANCE	5	1	2	2	3

SOURCE. THE MARQUARDT CORPORATION

Figure 7: Fan Stowage Options

Since studies thus far (primarily AFAL-TR-88-004) have investigated only the five engine variants mentioned in this tutorial, the initial mode of the RBCC engine cycle must be critically compared with other air-augmented rocket concepts as well as non-air augmented ideas within a common set of ground rules.[81] This comparison would result in a "best" option for the initial takeoff mode for a given set of mission requirements. For such a comparison to take place, however, a greatly expanded, updated database for air-augmented engine performance and sizing is needed.[82]

Because the vehicle in question is single-stage, the weight of the engine is a significant driver in engine design. To reduce the weight of the engine, advanced materials will have to be incorporated in the structure design. Materials must be resistant to chemical attack, oxidation and corrosion resistant, have interphase stability at high temperatures between fibers and matrix (in the case of composite materials), have high fatigue resistance, have low crack propagation rates, and must be inspectable, refurbishable, and damage tolerant.[83] Currently, material development is focusing on metal aluminides, metal matrix composites, and carbon reinforced composites (to include carbon/carbon). A related concern is that of joining techniques in the engine assembly. Escher lists several alternatives[84]: conventional welding and brazing; diffusion bonding; electron beam welding; laser welding; and advanced adhesives. Research and development on the materials and the joining techniques is currently underway.

A source of concern in engine assembly is the degree of geometric variations that must occur during the flight. AFAL-TR-88-004 called for a variable geometry inlet, fixed combustor, and fixed nozzle. What machinery will be needed to implement the geometric changes in the inlet? The machinery must be able to operate in the hypersonic environment, and must not interfere with the thermal control system required by the inlet section. The National Research Council Committee on Hypersonic Technology for Military Applications concludes, "The design of cooling passages in the engine and assurance of their effectiveness are made much more difficult, and may perhaps be compromised by the extensive geometric changes required of the engine over its Mach number range."[85]

In the scramjet mode particularly, the flow profiles (including the nature of the boundary layer) coming from one component will effect the performance of subsequent components. Therefore, component integration addresses the key issues of forebody effects on performance, combustor/inlet interaction, and combustor flow profile effects on nozzle expansion.[86] For airframe integrated (annular) engine design, investigation is needed on the effects of a simulated forebody flow on the performance of the engine module.[87] An isolator (constant area duct) between the inlet and the combustor is needed to control a combustor-induced shock train upstream of the combustor. Without the isolator, the shock train system could disrupt inlet operation and cause possible engine unstart (the movement of a normal shock to the cowl entrance). Nozzle performance must be measured for a wide range of combustor flow profiles.

The RBCC engine must incorporate means for mode transition. For example, from air-augmented rocket mode to ramjet mode, the rocket ejector must be shut down, and injector/flameholders for ramjet operation must be moved into the flowstream. Moving from ramjet operation to scramjet mode, the ejector must be restarted, and the injector/flameholders retracted. Finally, for scramjet to all-rocket mode, the rocket primaries must resume full operation and the inlet closed. The transition management of the engine and the vehicle must be analytically and experimentally investigated, and the results used to ensure "smooth" transition from mode to mode.

Vehicle Integration

Maximum vehicle performance requires operating at the highest possible effective specific impulse through the entire flight regime of the vehicle. Figure 8 shows a typical performance profile of a Supercharged Ejector Scramjet. The solid lines represent individual mode net-jet specific impulses (specific impulse accounting for ram-drag in the case of the airbreathing modes) as a function of flight velocity, and the dashed line represent the chosen multi-mode operation of the RBCC engine to maximize performance.

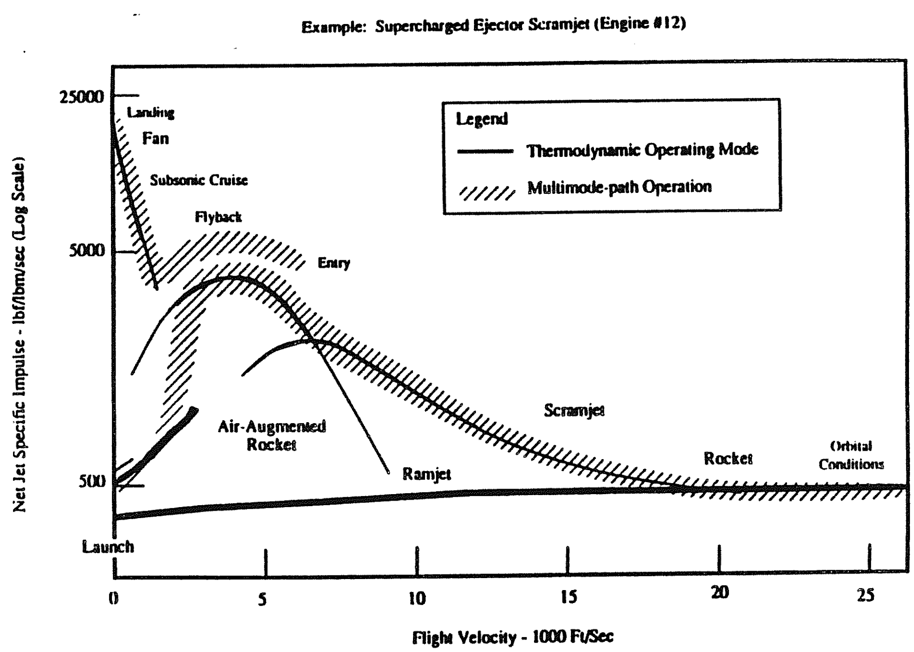


Figure 8:Multi-Mode Flight Performance[88]

A measure of vehicle performance over the entire flight is the total mission effective specific impulse (I^*). This term can be calculated by integrating the instantaneous effective specific impulse over the mission profile over the mission profile (see the discussion of Specific Impulse and Thrust-Drag Relations for Airbreathing Propelled Vehicles by John Leingang in this tutorial for a discussion of effective specific impulse). The value I^* can be put directly into the ideal rocket equation to calculate final velocity while accounting for both gravity and drag losses:

$$V_2-V_1=g_c*I^*\ln(W_1/W_2) \qquad (1)$$

This defines I^*, but a numerical integration of the coupled equations of motion are needed to relate the change in velocity to a value of W_1/W_2.[89]

The two extreme alternatives of vehicle/engine integration are fully annular engines and engine pods. Fully annular engines are the highest degree of integration possible. In the annular approach, the engine captures the largest geometrically possible amount of the air processed by the forebody bow shock. In the pod, or discrete approach, engine modules are positioned away from the body of the vehicle. Figure 9 illustrates the range of integration options.

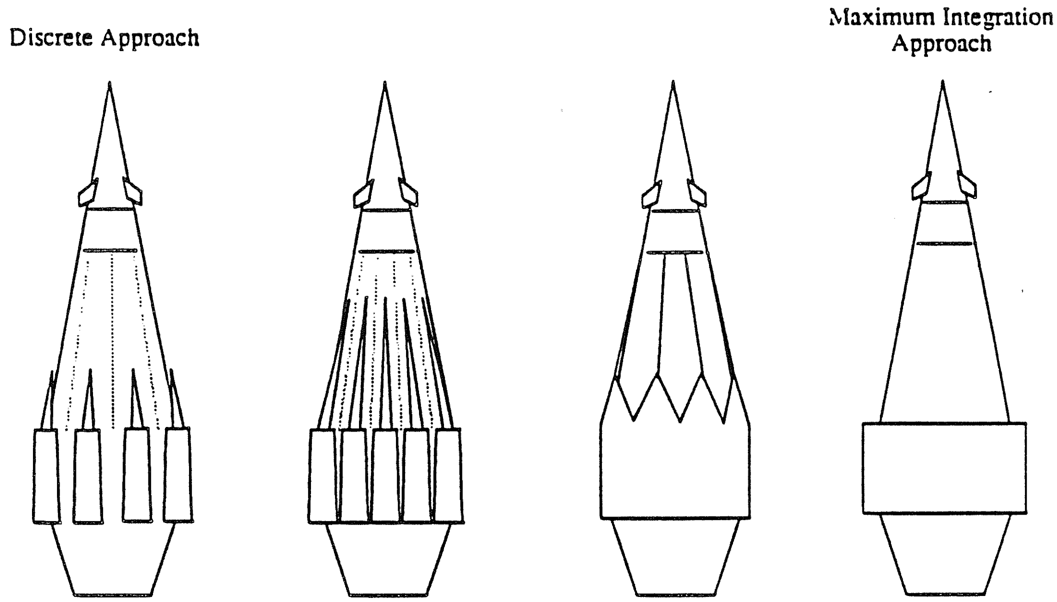


Figure 9: Vehicle/Engine Integration

As described in previous sections, the hypersonic acceleration phase of vehicle flight requires a highly integrated forebody, engine, and aft body to achieve the performance goals of a SSTO vehicle. Therefore, in any vehicle integration, experimentation and CFD investigation is required to predict the degree of forebody compression and aft body expansion that can be expected. Full annular flow of engine exhaust products of combustion on the aftbody is expected to reduce base drag, but this must be examined. A projected problem that warrants investigation is reduced forebody compression on the top side of the vehicle as angle of attack increases (Figure 10). This is accompanied by a thickened boundary layer which may cause inlet flow distortions and unstart conditions in the engine itself. Additionally, there is a problem of conical crossflow around the forebody of the vehicle at sustained angles-of-attack.[90]

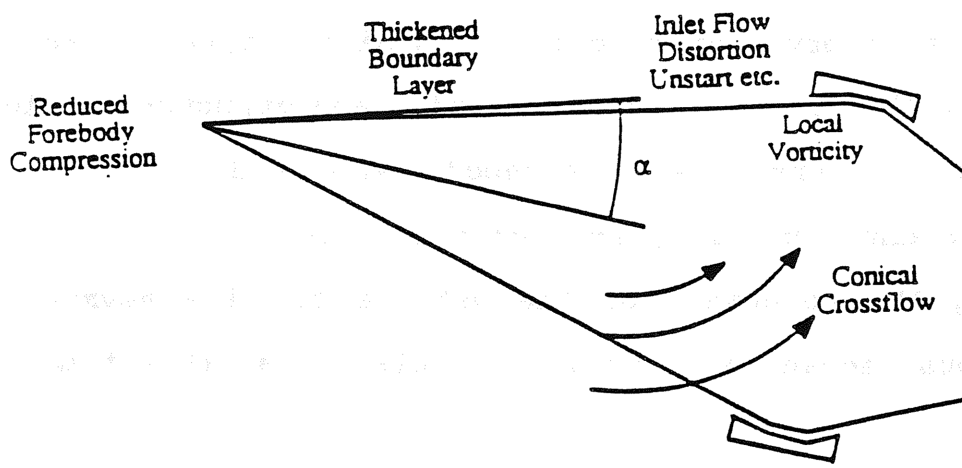


Figure 10: Effects of Angle of Attack on Vehicle Performance

In AFAL-TR-88-004, the vehicle systems studied centered on propulsion systems with thrust vector co-linear with the chord lines of the lifting surfaces with both the thrust line and chord line parallel to the longitudinal axis of the vehicle.[91] This assumption resulted in significant angles-of-attack being required over the full ascent profile. The principal investigators of AFAL-TR-88-004, suggested that additional study should be given to vehicle systems where the thrust vector and lifting surface chord lines could be varied from the longitudinal axis of the vehicle.[92] Other solutions to the problems of thrust and lift scheduling suggested in the report were[93]:

- The use of additional lifting surface area with the drawbacks of weight and drag.

- Fixed positive angle of incidence or variable angle of incidence with the drawback of additional weight and complexity.

- Flying a trajectory characterized with higher dynamic pressure which has significant impact on aeroheating and structural weight.

- The use of an articulated forebody with additional system complexity, weight penalties, and unknown effectiveness.

- Increasing the tolerance of the vehicle to flow asymmetries through various mechanisms that are speculative at this time.

The mode of take-off and landing will significantly effect the design of the vehicle. Escher lists the advantages of vertical take-off as[94]: no runway required for launch; no undercarriage/ runway-constrained TOGW upper limit; readily accessible payload loading/unloading; lightweight takeoff gear, good load distribution; and no need for large takeoff-sized wings. Additional advantages are gained if vertical landing is also employed[95]: no runway requirement at all; emergency landing capability at any level, firm site; self-ferry capability intrinsic; utilizes takeoff gear for landing; and ground taxi requirements are eliminated. Escher lists three disadvantages[96]: often-perceived early-abort unavailability; "aircraft imagery" problems; and vertical landing requires very high Isp landing propulsion mode.

Beyond speculative arguments, each of the takeoff and landing modes (horizontal and vertical) need to be compared in depth. It is recommended that this work be done with respect to[97]: stability and control requirements for each mode; the flight dynamic characteristics of the transition maneuvers and the requirements of optimum transition maneuvers; the fuel weights required for each takeoff and landing mode; vehicle behavior in high wind conditions that could effect takeoff and landing maneuvers; the structural implications of each mode, and any practical combination of takeoff and landing modes particularly regarding landing gear requirements; and the safety of each mode of takeoff and landing and the means for achieving an acceptable level of system safety.

Finally, revised weight estimates must be done for all candidate vehicle designs. Vince Weldon suggests that weight increases during the design phase can often be considered a technological "show stopper"[98]. Weight reduction plays a vital role in the success of a SSTO mission, and thus any design choices for the vehicle must make weight a prime consideration.

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for the

Rocket Based Combined Cycle Propulsion Technology Workshop

March 23-27, 1992

Kent Chojnacki & Clark Hawk

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