N92-21520

ADVANCED RAMJET CONCEPTS PROGRAM

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Uniquely advantageous features, on both the performance and weight sides of the ledger, can be achieved through synergistic design integration of airbreathing and rocket technologies in the development of advanced orbital space transport propulsion systems of the combined cycle type. In the context of well understood advanced airbreathing and liquid rocket propulsion principles and practices, this precept of synergism is advanced mainly through six rather specific examples. These range from the detailed component level to the overall vehicle system level as follows:

- Utilizing jet compression, as a specific air-augmented rocket mode approach
- Achieving a high area-ratio rocket nozzle through innovative use of air-handling ducting
- Ameliorating gas-generator cycle rocket system deficiencies while meeting, (1) ejector mode afterburner and, (2) rocket-mode internal aerodynamic nozzle operating needs
- Using the in-duct special rocket thrust chamber assembly as the principal scramjet fuel injection station
- Using the unstowed, covered fan as a duct closure for effecting high area-ratio rocket-mode operation
- Creating a unique airbreathing rocket system via the onboard, cryogenic hydrogen-induced air liquefaction process.

ILLUSTRATIVE CASES-IN-POINT: AIRBREATHING/ROCKET SYNERGISM

JET COMPRESSION

Jet compressors resemble conventional ejectors as used in industrial applications of yesterday (steam-locomotive smokestacks), and today (steam-cycle electric powerplant vacuum condensers). A propulsion-oriented application familiar to rocket test personnel are steam ejector systems used to initially "pull down" and/or actively exhaust a rocket altitude-simulation facility (e.g., as used in RL-10 engine testing).

In the combined cycle engine context, jet compressors are made up of a supersonic primary flow unit installed in a duct with an inlet providing the secondary flow to be compressed (air in this case). The downstream portion of the duct is divided into a mixer (usually of constant area), followed by a diffuser having diverging geometry for diffusing the mixed high-subsonic flow. Compression is achieved in the mixed stream by virtue of the high "driving" enthalpy of the primary flowstream, in this case a rocket.

Such jet compressors are characterized as "effective," if not necessarily "efficient" compressors, having characteristic advantages and disadvantages. They are lightweight, rugged, and highly tolerant of flow-distortion profiles, on the one hand. On the other, in contrast to conventional turbomachines, they have relatively high propellant consumption rates, and require considerable mixing duct lengths, hardware which usually must be actively cooled.

AIRBREATHING-MODE DUCTING USED AS HIGH AREA-RATIO ROCKET NOZZLE

The obvious technical approach is to utilize part of the air-handling duct and the airbreathing mode(s) combustor/nozzle assemblies to this end. Although this approach remains yet undemonstrated in "the problem has been solved" sense, there is analytical and even experimental evidence that this is, indeed, a feasible design approach. This evidence will be summarized below.

Specifically, the objective is to so configure and operate the engine (in rocket mode) as to provide an "aerodynamic or virtual" nozzle extension for the physical rocket unit with its low expansion-ratio nozzle. This aerodynamic extension would "control" the underexpanded supersonic exhaust plume such that it would smoothly attach to the engine's specially configured airhandling duct now serving as a physical nozzle extension. The objective is to minimize shock losses and otherwise non-optimum intermediate exhaust expansion processes.

Once attached, the divergent final section of the duct would continue the nozzle expansion process to very high area ratios of, say, several hundreds-to-one. The following flow exiting from the engine duct, or nozzle, further expansion of the rocket exhaust would take place on the vehicle aft-body. Nozzle aerodynamics-wise, the mechanization of this latter approach is seen to be a fortunate carry-over from the supersonic combustion ramjet mode (where aft-body expansion is a virtual necessity), assuming that the scramjet mode is to be used.

ADVANTAGEOUS DISPOSITION OF ROCKET SUBSYSTEM TURBOPUMP GAS GENERATOR EXHAUST

CONVENTIONAL LIMITATIONS OF GAS-GENERATOR TYPE ROCKET ENGINES

Historically, hydrogen/oxygen rocket engines have utilized turbopump propellant feed delivery systems. This achieves high combustion pressures leading to advantageous high area-ratio nozzle operation, without the structural weight penalties which might accompany pressure-fed systems, which are inherently difficult to execute with liquid hydrogen fuel.

Various turbopump drive approaches have been selected in hydrogen-oxygen rocket engines developed to date. For U.S. engines developed so far, the following turbopump drive approaches have been used:

RL-10	Expander Cycle
J-2	Gas Generator Cycle
SSME	Staged Combustion (Topping) Cycle

Looking ahead, it would seem to be the case that the staged combustion system will continue to be favored for large rocket engines, e.g., possible successors to the SSME (which pioneered this turbopump drive cycle). For smaller engines, such as those which might be applied to orbital transfer vehicle (OTV) systems, the expander cycle appears to be favored. Why not the gas generator cycle for these applications? The answer is, its lower special impulse performance, as nest discusses.

Although the gas generator cycle has a number of technical advantages (e.g., low pump-out pressures, achievement of high engine thrust/weight ratios, reduced interfacing difficulties), it has one salient and intrinsic disadvantage: a low-pressure fuel-rich turbopump turbine exhaust, which is not very effective in producing additive thrust. This leads to an overall engine specific impulse deficiency in comparison with the competing turbopump-drive approaches.

GAS-GENERATOR CYCLE IN THE COMBINED CYCLE ENGINE

In addition to its advantages in the conventional rocket engine context, which should carry over into a combined cycle system, the gas generator cycle may actually be strongly preferred in selecting the design of the engine's rocket subsystem. The main reason lies in unique uses of the turbopump-drive fuel-rich exhaust flow. A secondary reason for this preference lies in what is probably a poor physical design integration prospect for both the staged-combustion, and expander cycle alternative.

ROCKET MODE

Taking the second-name mode first, it has been previously treated as the probable need for a finite secondary flow (Rocketdyne's "basebleed") to maximize "aerobell" nozzle performance. The turbopump exhaust of the gas generator cycle seems to be a natural source of the base-bleed flow. In fact, the secondary flow might well have to be otherwise created in the case of the staged combustion cycle, for instance.Whether the gas generator exhaust flow "naturally matches up" with the special aerobell configuration needs in its quantity available, or in its flow properties, has yet to be examined. perhaps some system optimization effects will be devised to steer the turbopump component design in one direction, or another. But the point remains that there is what appears to be a good "fit" for the gas generator cycle (and not the other alternatives) in mechanizing the combined cycle rocket mode. After all, this is the implied expectation supported by the effort reported by the Rocketdyne researchers.

EJECTOR MODE

Proceeding now to the first of the two rocket-operating modes, the ejector mode, some discussion of the thermodynamic process nature of this operation is called for. This has to do with the specific type of air-augmented rocket to be selected, as covered in the earlier discussion of the jet compressor process.

Alternative concepts proposed for accomplishing air augmentation of rockets, presumably to raise specific impulse and/or thrust levels, are several. They range from affixing a simple, lightweight fixed-geometry duct or shroud around an otherwise conventional rocket engine, to more complex, but are judged to be more workable systems. Unfortunately, space here does not permit even a summary review of the possibilities. Alternatively, let us go to one leading-candidate type system as evidenced in previous assessments the afterburning cycle air augmented rocket.

DIFFUSION AND AFTERBURNING CYCLE AIR AUGMENTED ROCKET

The stipulation of a stoichiometric rocket (departure from the usually fuel-rich setting) now joins the earlier specification of an unconventional "distributed" high shear-area configuration, as a design precept, or hardware determinant. This is, of course predicated on the selection of the DAB cycle.

Now the afterburning aspect of this cycle implies making fuel available in the afterburner combustor, downstream of the ejector's mixer and diffuser. Typical engine designs provide for conventional afterburner (or ramburner) fuel supply and injection means. Here is how the gas generator cycle again appears to fit in well. Recall its performance detriment, as a rocket, stemmed from its characteristic fuel-rich (to control turbine temperatures) low pressure (turbine-drive enthalpy extraction) exhaust.

This gas can now be very usefully combusted in the (relatively) low-pressure afterburner. Calculations check that, even with this hot gaseous fuel supply, substantial amounts of additional fuel--hydrogen in this illustration--are required to burn stoichiometrically with air, for full-engine-power conditions, as needed for high-thrust acceleration propulsion operation. Hence, the gas generator cycle's turbine exhaust "detriment" is largely removed and, as we have seen, the gas generator cycle becomes uniquely and naturally the system of choice in the type of combine cycle engine under discussion.

IN-DUCT SPECIAL CONFIGURATION ROCKET THRUST CHAMBER ASSEMBLY FOR SCRAMJET FUEL INJECTION

A specialized "rocket engine subsystem" was designed by Rocketdyne for the 1966-1967 "Composite Engine Study". The special combined cycle engine for which this subsystem was designed was a circular cross-section version of what is referred to as the ScramLACE engine. It turns out that the final version of this engine had a rectangular cross-section duct configuration to more optimally match up with the Lockheed-designed first-stage vehicle of the study. The 2-ring configuration was, in the final version of this engine accordingly replaced with eleven vertically-mounted "linear" thrust chambers of equivalent propellant flow-rate and thrust. These hydrogen-fueled units used liquid air (LAIR) as oxidizer.

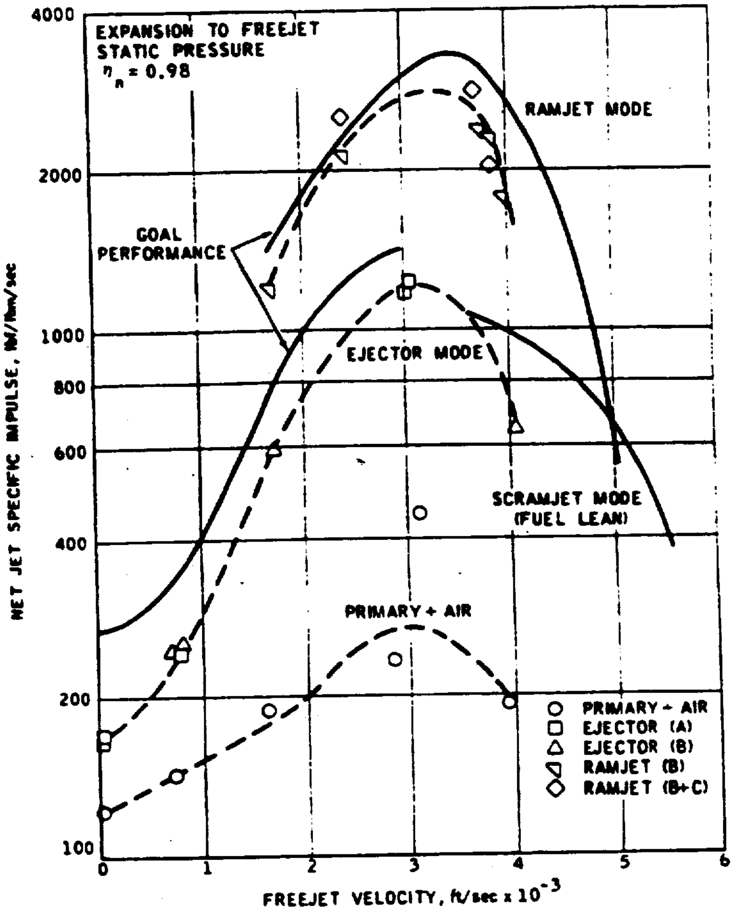
TRUE AIRBREATHING ROCKET SYSTEMS VIA AIR LIQUEFACTION

Onboard cryogenic hydrogen-induced air liquefaction was fairly heavily dealt with in the original aerospaceplane R&D of the 1960s in both analysis and design experimental set-ups. Some of the design consequences of this refrigerative capacity limitation are gone into below. As the available airbreathing achieves testify, a number of air liquefaction cycles were vigorously explored in the early 1960s, ranging from "BasicLACE through SuperLACE" to "ACES (Air Collection and Enrichment System)."

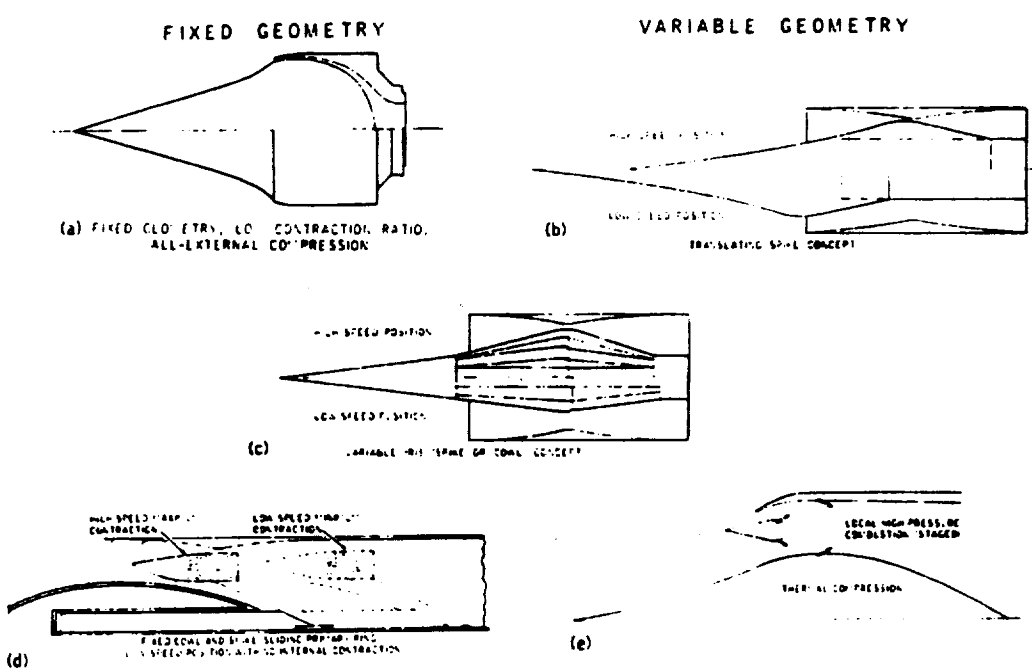
These earlier developments were substantially more extensive than today's propulsion engineers seem to be aware, not just in terms of dollars and manhours expended, but in depth and sophistication of the design analyses conducted, and the experiments which were run. All in all, there was considerable success in the early developmental experiments conducted.

By the late 1960s, when there was a cessation of further research and development work in air liquefaction systems and subsystems, a substantial level of technology had been documented. Particularly important, some of the salient design and operating challenges became fairly well-known, and design solutions accordingly brought forward.

One of the salient objectives of liquefying air in such systems was to provide a means for operating very lightweight, compact engines, which could gain the "airbreathing advantage" while maintaining rocket-like qualities of low weight and compactness. Such was achievable in principle and technically illustrated in small-scale test rigs encompassing: (1) a liquid air pump of the rocket engine type, and (2) a high-pressure rocket-type thrust chamber. These equipment items were much more compact and less massive than the corresponding conventional ambient-temperature air compressor and its drive turbine and, lower combustion pressure turbojet-type combustor or afterburner assemblies.

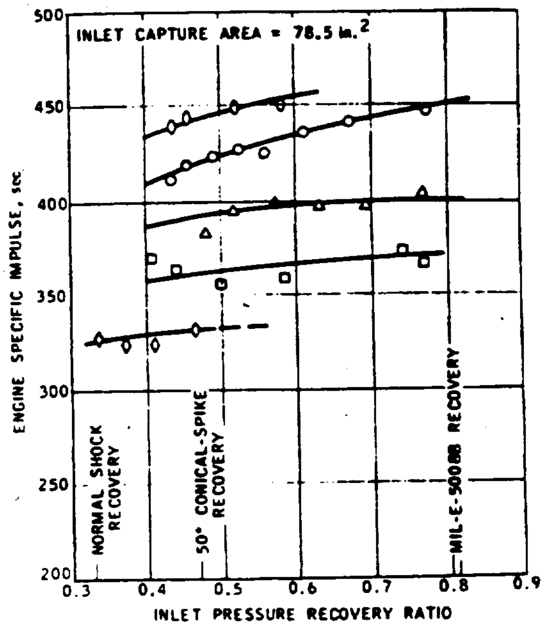


EJECTOR SCRAMJET TEST ENGINE PERFORMANCE



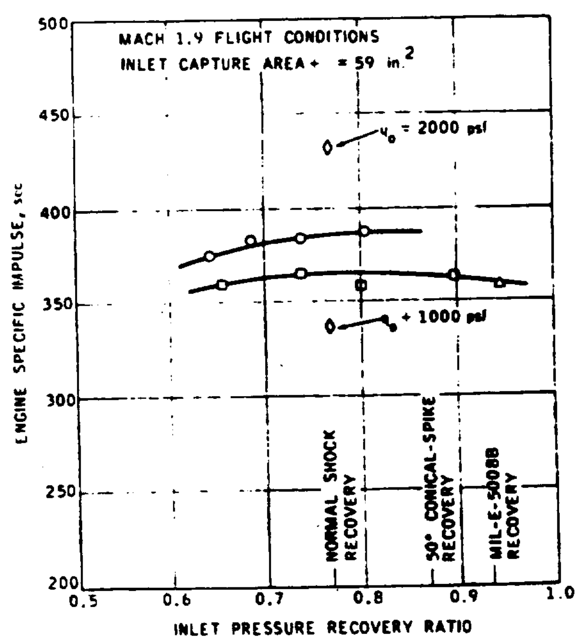
SPECTRUM OF POTENTIAL EJECTOR SCRAMJET INLET CONCEPTS

SYM.	RUN	SEC/PRI. FLOW RATIO	PRI. PRESS.	PRI. EQUIV. RATIO	A/B EQVUI. RATIO	SIM. ALTITUDE	DYNAMIC PRESS.
◇	37	3.24	4.94	0.97	0.84	45,200	1920
○	25	3.12	4.96	0.96	0.85	46,500	1800
△	37	2.70	4.95	0.96	0.83	50,000	1525
□	37	2.35	4.95	0.95	0.85	52,300	1360
◇	22	1.70	4.95	0.95	0.82	59,000	990

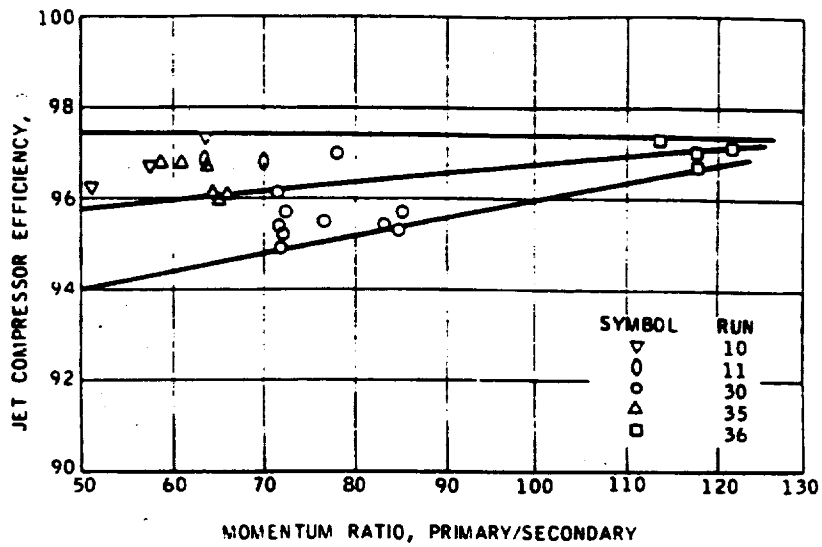


MA191-B1 EJECTOR RAMJET ALTITUDE PERFORMANCE
MACH 3.0 EJECTOR MODE

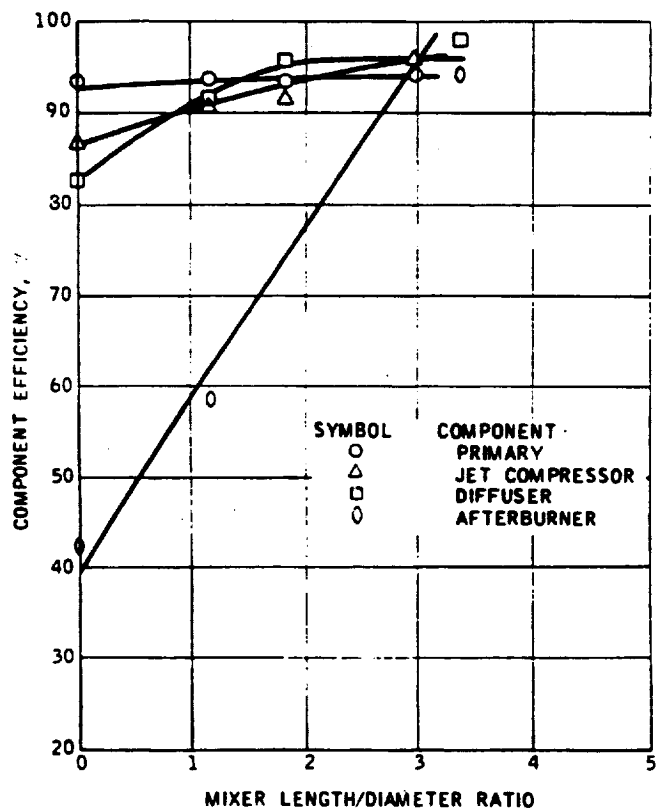
SYM.	RUN	SEC/PRI. FLOW RATIO	PRI. PRESS.	PRI. EQUIV. RATIO	A/B EQUIV. RATIO	SIM. ALTITUDE	DYNAMIC PRESS.
△	37	1.82	496	0.94	1.00	42,000	900
□	28	2.08	504	0.94	0.87	38,500	1060
○	28	2.65	501	0.93	0.87	33,000	1370
◇	PREDICTED						



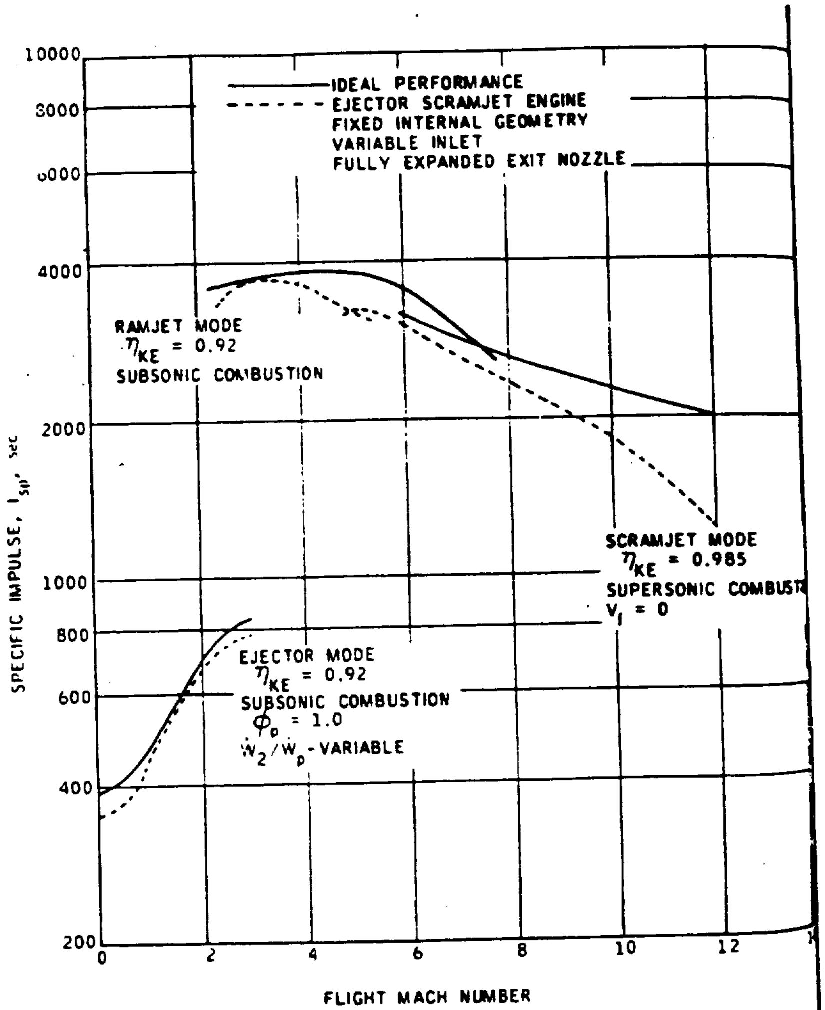
MA191-B1 EJECTOR RAMJET ENGINE PERFORMANCE
EJECTOR RAMJET OPERATING MODE
ALTITUDE OPERATION



MA191-B1 EJECTOR RAMJET ENGINE PERFORMANCE
JET COMPRESSOR PERFORMANCE VERSUS MOMENTUM RATIO - PRIMARY/SECONDARY



MA191-B1 EJECTOR RAMJET ENGINE PERFORMANCE
COMPONENT EFFICIENCY VERSUS MIXER LENGTH/DIAMETER RATIO



COMPARISON OF EJECTOR SCRAMJET CYCLE PERFORMANCE CHARACTERISTICS