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An Integration of the Turbojet and Single-Throat Ramjet

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ABSTRACT

A turbine-engine-based hybrid propulsion system is described. Turbojet engines are integrated with a single-throat ramjet so as to minimize variable geometry and eliminate redundant propulsion components. The result is a simple, lightweight system that is operable from takeoff to high Mach numbers.

Non-afterburning turbojets are mounted within the ramjet duct. They exhaust through a converging-diverging (C-D) nozzle into a common ramjet burner section. At low speed the ejector effect of the C-D nozzle aerodynamically isolates the relatively high pressure turbojet exhaust stream from the ramjet duct. As the Mach number increases, and the turbojet pressure ratio diminishes, the system is biased naturally toward ramjet operation. The common ramjet burner is fueled with hydrogen and thermally choked, thus avoiding the weight and complexity of a variable geometry, split-flow exhaust system. The mixed-compression supersonic inlet and subsonic diffuser are also common to both the turbojet and ramjet cycles. As the compressor face total temperature limit is approached, a two-position flap within the inlet is actuated, which closes off the turbojet inlet and provides increased internal contraction for ramjet operation. Similar actuation of the turbojet C-D nozzle flap completes the enclosure of the turbojet.

Performance of the hybrid system is compared herein to that of the discrete turbojet and ramjet engines from takeoff to Mach 6. The specific impulse of the hybrid system falls below that of the non-integrated turbojet and ramjet because of ejector and Rayleigh losses. Unlike the discrete turbojet or ramjet however, the hybrid system produces thrust over the entire Mach number range.

An alternate mode of operation for takeoff and low speed is also described. In this mode the C-D nozzle flap is deflected to a third position, which closes off the ramjet duct and eliminates the ejector total pressure loss.

NOMENCLATURE

A	cross-sectional area or flow area
C_{fg}	nozzle gross thrust coefficient; ratio of axial gross thrust to ideal thrust
F	axial force
H	altitude
I_{sp}	specific impulse based on net axial force; ratio of net force to total fuel flow, lbf-sec/lbm
M	Mach number
$\dot{m}$	airflow
NPR	nozzle pressure ratio; ratio of nozzle total pressure to free-stream static pressure, P_{t8}/P_0
P	pressure
q	free-stream dynamic pressure (lbf/ft^2)
γ	ratio of specific heats
ϕ	fuel-air equivalence ratio; fuel-air ratio normalized by stoichiometric fuel-air ratio

SUBSCRIPTS

0, ..., 8	station numbers in flow (0 = free-stream condition; see fig. 4)
e	exit
ej	ejector mode
g	gross thrust
i	inlet projected
max	maximum
n	net

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opt	optimum
ram	ramjet
sep	flow separation
t	total or stagnation condition
turbo	turbojet

<u>SUPERSCRIPT</u>

*	sonic condition

INTRODUCTION

The supersonic-combustion ramjet or "scramjet" cycle provides efficient propulsion at cruise speeds in excess of about Mach 6. Scramjet-powered vehicles must be accelerated by some other means, however, to at least ~Mach 3, where sufficient ram compression exists for operation of the scramjet flow path in a subsonic-combustion "single-throat" ramjet mode[1]. In this mode, thermal choking in the initial portion of the scramjet nozzle is used in lieu of a conventional throat to "back-pressure" and control the airflow through the engine. As the Mach number increases, transition to supersonic combustion mode is complete when all of the fuel can be injected near the minimum area without causing a thermal choke.

The turbojet engine is a reliable and efficient means to provide acceleration to Mach 3. Turbojets also afford low-speed cruise capability for ferry or abort scenarios and can operate with a variety of fuels, including those available in the present airport infrastructure. However, since the turbojet is a weight and volume liability at Mach numbers above which it can operate, a high degree of integration between the turbojet and ramjet propulsion systems is desirable. This paper describes the integration of a turbojet and a single-throat ramjet. The resulting turbine-based combined-cycle (TBCC) propulsion system is intended to operate from takeoff to high Mach numbers by using the single-throat ramjet flow path in both subsonic-combustion (thermally choked) and supersonic-combustion (scramjet) modes.

Figure 1 shows the proposed arrangement of the turbojet and single-throat ramjet. A row of turbojet engines is installed in a recessed bay above the ramjet throat region. As shown in figure 2, the high-pressure turbojet exhaust stream is isolated aerodynamically from the ramjet stream by ejector action. This enables the use of a common, fixed-geometry hydrogen afterburner and nozzle for the turbojet and ramjet streams. A common inlet and subsonic diffuser, in which a two-position panel would shield the turbojet at high Mach number and provide increased internal contraction appropriate for the ramjet cycle, is also proposed. The ejector allows a minimal amount of air through the ramjet throat passage at low speed. The ejector combines this ram air with the oxygen-deficient turbojet exhaust stream to form a mixture that is burned stoichiometrically with hydrogen to produce a thermal choke in the afterburner region. The area ratio at which the thermal choke occurs is assumed to be controllable and determines the operating pressure ratio, or back-pressure, of the TBCC system. As flight speed and ram compression increase, turbojet airflow demands diminish, and the ejector provides for a gradual, aerodynamic conversion to the ramjet cycle. As the free-stream total temperature approaches the turbojet inlet temperature limit, the inlet and ejector flaps close off the turbojet bay. The thermally choked afterburner continues to operate until about Mach 6, where the scramjet cycle is viable. The hydrogen spraybars are then either retracted or sacrificed.

CONCEPTUAL DESIGN AND ANALYSIS METHODS

To maximize range, the design of an integrated propulsion system is biased toward high efficiency at the cruise Mach number. Therefore, the profile of a generic scramjet flow path was used as a reference into which the TBCC system was integrated. The overall dimensions and the design features of the vehicle and propulsion system are shown in figure 3. The 10 to 1 contraction ratio and forebody ramp geometry is appropriate for about Mach 8. Beyond Mach 8, additional contraction may be necessary. The scramjet combustor cross-sectional area is 1.33 that of the throat. (This constant-area combustor was postulated merely to simplify the ejector analysis and is not a constraint of the TBCC concept.) A detailed design and analysis of the scramjet flow path is beyond the scope of this study.

The primary nozzle inflow angle of 20^{o} was chosen to minimize the length of the ejector region while avoiding the momentum loss associated with too great an injection angle. The primary nozzle throat area, which is fixed, was based on the turbojet airflow requirements. The primary nozzle area ratio of 1.15 (Mach 1.46) was determined as follows. Maximum turbojet engine size was desired because of takeoff and transonic thrust requirements. The maximum allowable engine size (to avoid reverse flow in the ejector) increases slightly with the primary nozzle area ratio. Too large an area ratio, however, causes flow separation and also lengthens the ejector region. At the Mach-3 flight condition, the 1.15 area ratio is the maximum advisable since the primary nozzle exit static pressure is 0.35 that of the surrounding flow. For these primary nozzle characteristics, the compressor face area is 2.26 times the ramjet throat area.

<u>ANALYSIS METHODS</u>

A rudimentary cycle analysis was performed to determine the feasibility of the TBCC integration. The analysis was limited to subsonic-combustion modes from sea-level static conditions to about Mach 6, where, presumably, the vehicle further accelerates and then cruises on the scramjet cycle. The cycle is analyzed from the free stream through the stations in the flow path depicted in figure 4. Calorically perfect air is assumed throughout, and fuel mass is neglected.

The stagnation pressure at stations 2 and 3 (also referred to as inlet recovery) is calculated by considering the inviscid shock losses from the 7^{o} and 13^{o} forebody ramps and including an additional 5-percent total pressure loss. In ejector mode, the inlet is assumed to be unstarted. The Mach number at station 2 is determined by the turbojet airflow demand, and at station 3 by the ejector analysis. In ramjet mode, the inlet is assumed to be started and the recovery to be dependent on the back-pressure and terminal shock location imposed by the thermal throat.

The equations of reference 2 were used to determine a typical turbojet total pressure ratio, total temperature ratio, and fuel-air ratio variation with Mach number and altitude. The burner exit area, primary nozzle throat area, and component efficiencies were picked to match the performance of a currently available low-bypass-ratio turbofan engine at sea-level static conditions. It was assumed that the turbojet always operated at a constant turbine inlet temperature of 3120 ^{o}R and would survive to Mach 3.0. The fuel-air equivalence ratio that this requires is a fairly constant 0.28 over the Mach range. A fuel heating value of 18 900 Btu/lbm was assumed.

The ejector region was modeled by using the control volume depicted in figure 4 and following the constant-area analysis of reference 3, which was modified to account for nonzero inflow angles. It was assumed that the turbojet and ramjet flows are completely mixed and uniform at station 6. This assumption was made to facilitate the analysis; it may not be a necessary condition for successful TBCC system operation. The ejector is used mainly to isolate the two flows, not necessarily to supercharge the ramjet flow. Inputs to the analysis include the primary nozzle Mach number and flow angle, the area ratio between stations 4 and 6 (also referred to as the scramjet combustor divergence), and the total pressure and temperature ratios between stations 4 and 5 (which are the turbojet engine total pressure and temperature ratios). The conditions at the ejector "critical point" are calculated first. At the critical point, the ramjet airflow is maximum for the given geometry and turbojet pressure and temperature ratios. Depending on the input values, the ramjet airflow is limited by sonic conditions at station 4, or station 6, or an intermediate choked aerodynamic area. The static pressure at station 6, or back-pressure, consistent with the maximum ramjet flow is then determined. This represents the highest back-pressure that the ejector process can support before a reduction in airflow at station 4 occurs. The critical point back-pressure is then compared to an input back-pressure from the thermally choked ram burner calculation. If the input back-pressure is higher than the critical back-pressure (a condition referred to as "subcritical" operation), the airflow at station 4 must be reduced, and an iteration including the inlet, ejector, and ram burner analysis must be invoked. If the input back-pressure is less than the critical value, the ejector would be operating in a "supercritical" state; that is, the airflow at station 4 remains at the critical value, but the Mach number at station 6 increases with a corresponding reduction in total pressure. In general, the specific gross thrust of the TBCC system in ejector mode reaches a maximum when the ejector is back-pressured to a zero ramjet flow condition, since the turbojet stream experiences a greater pressure ratio than does the ramjet stream. However, the specific net thrust and specific impulse (I_{sp}) of the TBCC system are always at a maximum at the ejector critical point—when the product of back-pressure and airflow is at a maximum and inlet spillage drag is at a minimum.

Given the total temperature at station 6 and the ratio of ramjet-to-turbojet airflow, we can evaluate the hydrogen-fueled ram burner. The airflow ratio is required because the turbojet flow is oxygen deficient; thus, the oxygen mole fraction of the mixture depends on the relative amounts of ramjet and turbojet airflow. The stagnation temperature rise is calculated on the basis of hydrogen-air equilibrium at 1 atm for a given initial stagnation temperature, equivalence ratio, and 95-percent combustion efficiency. An effective maximum equivalence ratio (see eq. (1)) was used to account for the oxygen deficit in the turbojet stream:

$$\phi_{\mathrm{ram,\,max}} = 1 - \frac{\phi_{\mathrm{turbo}}}{1 + \dfrac{\dot{m}_{\mathrm{ram}}}{\dot{m}_{\mathrm{turbo}}}} \tag{1}$$

The hydrogen-fueled ramjet burner is always operated at this maximum equivalence ratio; thus, the oxygen in the mixture of turbojet and ramjet streams is completely consumed. Once the stagnation temperature ratio is known, equations for Rayleigh flow are used to determine the Mach number at station 7 and the stagnation pressure loss, or Rayleigh loss, due to the heat addition. In the Rayleigh flow analysis, the heat addition takes place at constant area, or for the present case, in an infinitely thin region of the diverging channel. The thermal throat location is assumed to be controllable over the flight Mach number range. It is placed at an area ratio that supports the ejector critical point Mach number.

The normalized gross thrust is a function of the nozzle area ratio (which varies with thermal choke location), the nozzle pressure ratio (NPR), the exit pressure ratio (P_e/P_{t8}, based on isentropic expansion to the nozzle exit area), and the ratio of specific heats (taken to be 1.4 throughout the cycle analysis):

$$\frac{F_g}{P_{t8}A^*} = \gamma\sqrt{\frac{2}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}\sqrt{1-\left(\frac{P_e}{P_{t8}}\right)^{\frac{\gamma+1}{\gamma}}} + \frac{A_e}{A^*}\left(\frac{P_e}{P_{t8}} - \frac{1}{\mathrm{NPR}}\right) \tag{2}$$

This relationship[4] accounts only for overexpansion losses. Losses due to divergence, friction, and non-equilibrium are neglected, the latter two usually being of little consequence at flight Mach numbers below 6. For the present configuration, the entire aft end of the vehicle is used as the expansion surface of a single expansion ramp nozzle (SERN). The large area ratio provided by this arrangement results in a highly overexpanded condition over much of the ramjet Mach number range. In practice, three-dimensional flow effects and flow separation result in a thrust level higher than that calculated by the one-dimensional analysis[5]. To account for this in an approximate fashion, it was assumed that the exhaust stream separates at a pressure ratio given by the following:

$$\frac{P_{\mathrm{sep}}}{P_0} = 0.3 + \frac{0.2}{\left(M_0+1\right)} \tag{3}$$

If this pressure ratio is greater than that calculated for an isentropic expansion to the actual nozzle exit area, then it is taken to be the exit-to-ambient pressure ratio. The area ratio corresponding to an isentropic expansion to the prevailing exit pressure is then used to determine the gross thrust. The gross thrust is ratioed to the maximum gross thrust (obtained by setting $P_e = P_0$ in eq. (2)) to form the nozzle gross thrust coefficient C_{fg}, which is reported in a subsequent section. This method gives C_{fg} as a function of nozzle area ratio, nozzle pressure ratio, and free-stream Mach number.

Only axial forces were considered in this study. The axial net thrust of the propulsion system is taken to be the axial gross thrust minus the ram and spillage drags. The ram drag is calculated simply as the momentum of the free-stream-captured streamtube. The spillage, or additive drag, is the change in axial momentum from the free stream to the inlet cowl lip; it is charged to the propulsion system since it depends on engine airflow conditions. The spillage drag is computed by integrating the static pressure along the capture streamline from the free stream to the cowl lip. For supersonic portions of the flow, the capture streamline is a straight line inclined parallel to the compression ramps, so the static pressure can be found from the oblique shock relations. For subsonic portions of the flow, some additional modeling is required. Subsonic flow exists downstream of the normal shock when the inlet operates in external compression mode and exists along the entire inlet when the free-stream Mach number is subsonic. In this simulation, static pressure is assumed to be a constant equal to the average of the free-stream and cowl-lip pressures. The capture streamline is assumed to be a straight line from the free stream to the cowl lip. In reality, the static pressure varies from the free stream to the cowl lip along a curved capture streamline. A correction factor equal to 0.5 is applied to the integrated subsonic spillage drag to account for pressure variation, streamline curvature, and lip suction effects which occur at subsonic conditions. The value of the correction factor is based on experimental correlations from reference 6.

Finally, the I_{sp} of the TBCC system is calculated as the ratio of the net axial thrust to the sum of hydrogen and hydrocarbon fuel flows.

SEA-LEVEL STATIC CYCLE ANALYSIS EXAMPLE

The table in figure 4 shows cycle analysis results at sea-level static conditions with the ejector operating at its critical point. The increase in total pressure and temperature across the turbojet is reflected in the values from stations 2 to 5. Even with no ramjet flow, the ejector process yields a total pressure loss between stations 5 and 6 as the Mach-1.46 primary nozzle flow decelerates to Mach 0.605. The total temperature ratio and the Rayleigh loss across the ramjet burner are evident from station 7 to station 8, where sonic conditions exist. The thermal throat at station 8 was placed at an area ratio (A_8/A_6) that allows subsonic diffusion from the ejector exit (station 6) critical point Mach number of 0.605 to the Mach-0.346 condition required for a Rayleigh flow process with stoichiometric hydrogen combustion. Note that had the station 6 Mach number been less than 0.346, a thermal choke would not have been possible at the ejector critical state, even with the heat addition occurring at station 6. For this situation the ejector would operate supercritically.

FLIGHT TRAJECTORY

The flight trajectory used is depicted in figure 5. It was assumed that the vehicle begins to climb at 200 kn. To maximize I_{sp} and minimize sonic boom during initial acceleration, the highest possible altitude is desirable. A constant total temperature trajectory is therefore used to maintain the sea-level static turbojet temperature and pressure ratios, and thus, the ejector critical state of zero ramjet flow. This trajectory represents the highest altitude possible for a given Mach number if ejector back flow is to be avoided. A level flight acceleration at 19 500 ft from Mach 0.878 to Mach 1.2 joins the constant total temperature trajectory to a 1 000 lbf/ft^2 free-stream dynamic pressure trajectory that was arbitrarily chosen. No attempt was made to optimize the trajectory for this feasibility study.

BASELINE PERFORMANCE RESULTS

Results for the baseline configuration over the flight trajectory are presented in this section. These results will be compared to those for an alternate configuration that eliminates the ejector loss at the expense of additional variable geometry. Inlet recovery and total mass capture ratio (A_0/A_i) are shown in figure 6. The inlet starts at Mach 3 as the inlet and ejector flaps close off the turbojet duct, and the system becomes a single-throat ramjet. In ejector mode the mass capture ratio is at the maximum possible because the ejector is operating at its critical point. Beyond Mach 3, the mass capture is determined by the forebody and inlet geometry since the inlet is started. A standard inlet recovery schedule (Mil-E-5007D) is shown for comparison.

Turbine engine characteristics are plotted in figure 7. Free-stream total temperature is the only variable affecting these parameters, since the engine is always operated at maximum dry power. The pressure and temperature ratios are roughly constant from takeoff to Mach 0.88 because of the constant free-stream total temperature trajectory. The reductions from takeoff to Mach 0.4 are caused by the slight increase in total temperature during the ground roll. At Mach 3 the engine pressure ratio is unity, at which point there is no thermodynamic advantage to operating the turbojet. The equivalence ratio decreases only slightly as free-stream total temperature increases and the amount of fuel required to heat the compressor discharge air to the turbine inlet temperature of 3120 ^{o}R diminishes.

Performance of the SERN is depicted in figure 8. As explained in the section on analysis methods, C_{fg} is calculated as a function of the NPR, the free-stream Mach number, and the overall area ratio, which varies with thermal throat location. The combination of high area ratio and low NPR causes the relatively poor performance from takeoff to Mach 3.

Ejector characteristics at the critical point are plotted versus free-stream Mach number in figure 9. The ratio of ramjet to turbojet airflow remains near zero until Mach 0.88 because of the constant total temperature trajectory. This ratio increases to about 1.3 at the Mach-3 point of transition to single-throat ramjet. The total flow through the TBCC system is not reduced during the mode change and in fact increases slightly (see fig. 6) because of the assumption that the inlet starts at Mach 3. The ratio of the total pressure at the ejector exit (station 6) to that of the turbojet exhaust (station 5) is about 0.70 between takeoff and Mach 0.88. Since no ramjet flow is admitted during this period, this represents a significant loss in performance. By Mach 3, the station 6 total pressure is slightly greater than that of the turbojet. The Mach number at station 6 increases from 0.6 at takeoff to ~1 at Mach 3. With the present geometry and operating conditions, the ejector always operates in a regime where the ramjet, or secondary, flow is limited by aerodynamic choking within the ejector control volume.

The specific net thrust appears in figure 10 as the region bounded by the gross thrust and the sum of the ram and spillage drags. Its minimum is at Mach 1.3, where the spillage drag peaks. There is only a slight step change in net thrust at Mach 3, where the inlet and nozzle flaps are closed, the inlet starts, and the system becomes an all-hydrogen, subsonic-combustion single-throat ramjet. The smooth transition is a consequence of the increasing proportion of ramjet airflow admitted as the turbojet airflow demand and pressure ratio diminish. For the present geometry and inlet modeling, Mach 5.71 is the upper limit, beyond which the subsonic, stoichiometric combustion of hydrogen at the minimum area (the scramjet combustor area) is no longer sufficient to cause a thermal choke. Transition to scramjet mode would therefore be required at or before this point. In practice, this transition could be effected at a lower Mach number, if a net benefit to the vehicle system were to result, by taking into account structural and thermal loads as well as net thrust.

The specific gross thrust of just the turbojet with hydrogen afterburning (uninstalled) is also plotted in figure 10 for comparison to the TBCC system. The difference between these gross thrusts is attributable to the SERN losses (the turbojet gross thrust is based on a C_{fg} of unity), the Rayleigh loss associated with the thermal choke, and the total pressure loss caused by the ejector action. These factors are losses only in the sense that they reduce the specific net thrust of the TBCC system from that of an isolated turbojet. They represent the “cost of integration” that allows operation to high Mach number at reduced mechanical complexity and weight, over a system with discrete turbojet and ramjet engines. Of the three loss mechanisms, the dominant one is the ejector total pressure loss, which can be eliminated with an additional ejector flap movement, as described in the next section.

ALTERNATE TAKEOFF AND LOW– SPEED CONFIGURATION

The ejector total pressure loss can be eliminated by mechanically isolating the turbojet stream from the ramjet duct. Since the ejector flap is already mechanized to close off the turbojet bay, it can conveniently provide this isolation (see fig. 11). In this position, the primary nozzle is disabled and the turbojet back-pressure and airflow are controlled by the thermal throat. Therefore, in this mode, the thermal throat location is set such that it provides the same airflow as the fixed throat of the primary nozzle in ejector mode. As will be shown in a subsequent section, the thermal throat must be located at an area very near the ejector exit (or scramjet combustor) area. This is a consequence of the choice of turbojet characteristics and the scramjet geometry. If the ejector exit area were larger, or if less oxygen were available in the turbojet stream, a thermal throat alone would not provide the required back-pressure.

Since the baseline design and trajectory admitted no ramjet airflow until about Mach 0.9, this alternate mode of operation has no aerodynamic disadvantages at low speed. As seen in figure 12, the gross thrust is now nearer that of the uninstalled turbojet, being separated only by the SERN and Rayleigh losses. A comparison to figure 10 shows that at takeoff the ejector loss is roughly equal to the SERN and Rayleigh losses combined. The net specific thrust in this mode is as much as 20 percent greater than in the ejector mode at low speed (see fig. 13), but degrades quickly at supersonic flight conditions as the inlet spillage drag rises. At Mach 1.1, the reduction in spillage drag afforded by the ramjet airflow compensates for the ejector loss, and a switch to the ejector mode is indicated.

In figure 14 the I_{sp} is plotted to Mach 3 for both the ejector and takeoff modes, and for the ramjet mode from Mach 3 to 6. The precipitous decline seen in figure 13 for net thrust benefit beyond Mach 0.8 in takeoff mode is not as apparent in terms of the I_{sp}, since the hydrogen fuel flow increases in proportion to the additional ramjet airflow.

The I_{sp} of the uninstalled TBCC propulsion system is compared to those of its component parts in figure 15. The uninstalled I_{sp} is the gross thrust minus the ram drag, divided by the sum of the hydrocarbon and hydrogen fuel flows. For an equitable comparison, spillage drag and nozzle losses are not included, and a standard inlet recovery schedule is used for all systems. The TBCC system in ejector mode falls below that of the hydrogen-afterburning turbojet at takeoff because of the ejector and Rayleigh losses. By Mach 2.1, however, the influence of the ramjet cycle is apparent as the I_{sp} of the TBCC in ejector mode climbs above that of the discrete turbojet. At Mach 3, the turbojet provides no additional pressure ratio to the system; it only adds heat by combustion of the hydrocarbon fuel. The step change in uninstalled I_{sp} is due to the switch to all-hydrogen fuel in ramjet mode. In takeoff mode, the TBCC system is a thermally choked, hydrogen-afterburning turbojet, so the I_{sp} deficit is only that due to the Rayleigh losses. Similarly, in ramjet mode the difference between the TBCC system and the dual-throat, or mechanically choked, ramjet is the Rayleigh loss. The Rayleigh loss accounts for approximately a 3-percent reduction in I_{sp} across the Mach range. The ejector total pressure loss results in a 15-percent I_{sp} deficit at takeoff, and strictly, is only a loss from takeoff to Mach 0.88 (the range where no ramjet air is admitted). The premise of this paper is that the benefits to a vehicle using a simple, fixed-geometry integrated propulsion system will outweigh the I_{sp} deficits shown here.

EFFECT OF THERMAL THROAT LOCATION ON PERFORMANCE

To this point, it has been assumed that the TBCC system operates with the thermal throat positioned such that the optimum back-pressure is maintained over the Mach number range. In ejector mode this corresponds to the critical point; in ramjet mode, to an operating point just prior to inlet unstart; and in takeoff mode, to a condition that mimics the fixed primary nozzle throat. In practice, the required level of control may be difficult to accomplish without providing a number of spraybars or somehow modulating the rate of mixing of the hydrogen fuel. It is of interest, therefore, to examine the sensitivity of net thrust to excursions from the optimum thermal throat area ratio. Also of interest is the feasibility of operating the system with the thermal throat fixed at one location, a situation that may approximate a single, fixed hydrogen spraybar array.

The optimum position of the thermal throat is shown in figure 16 for the three operating modes discussed thus far. The ejector mode requires that the thermal throat move aft from an area ratio of about 1.5 at takeoff to 2.1 at Mach 3 to keep the ejector at the critical point. If the thermal throat is moved too far aft, the total pressure loss due to supercritical ejector operation reduces the NPR to below 1.89, and the minimum NPR limit is reached. If the thermal throat is moved forward of the optimum location to a smaller area, the ejector operates subcritically and reverse flow can result. The ejector backflow limit shown is the thermal choke area at zero ramjet flow. From takeoff to Mach 0.88, it is generally coincident with the critical point. In takeoff mode, the thermal throat remains at an area ratio of about 1.04, moving aft slightly from Mach 2 to 3. In ramjet mode, the thermal throat is positioned slightly aft of the critical inlet limit to provide some stability margin. It must move forward from an area ratio of 2.0 at Mach 3 to 1.0 at Mach 5.71, at which point it reaches the thermal choke limit, beyond which hydrogen combustion can no longer sustain the thermal throat.

A two-position thermal throat area ratio schedule is also depicted in figure 16. In this scenario, the TBCC system operates in takeoff mode to Mach 1.1, in ejector mode from Mach 1.1 to 3, and in ramjet mode above Mach 3. A fixed area ratio of 1.8 was chosen for the ejector and ramjet modes (after considering the transonic net thrust minima), to be as close as possible to the optimum value at Mach 1.3 without violating the critical inlet limit at Mach 3. Note that the 1.8 area ratio would penetrate the minimum NPR region at takeoff if the ejector mode were used. The area ratio from takeoff to Mach 1.1 is 1.04, as required by the turbojet.

The sensitivity of net thrust to thermal throat location is shown in figure 17. In ejector mode, both supercritical (thermal throat aft of critical), and subcritical (thermal throat forward of critical) operation is considered from Mach 0.88 to Mach 3. From takeoff to Mach 0.88, only an aft excursion of the thermal throat (to a larger area ratio) is relevant, since forward movement would result in ejector backflow. The sensitivity of net thrust to forward (subcritical) thermal throat excursions is more than twice the value for aft (supercritical) excursions. This is due to the reduction in airflow and increase in spillage drag in the subcritical state. In the supercritical state, a reduction in NPR is the only detriment to net thrust. In ramjet mode, only aft excursions are considered since only a very limited forward excursion is possible before inlet unstart. Sensitivity to aft excursions is less than percent of net thrust per 1 percent of thermal throat area increase in ramjet mode.

In the two-position scheme, with the thermal throat fixed at an area ratio of 1.8, the ejector operates supercritically from Mach 1.1 to 1.8, and subcritically from Mach 1.8 to 3. Figure 18 shows the percent difference in net thrust between the two-position and optimum thermal throat schedules plotted versus flight Mach number. A 25-percent net thrust loss is apparent at Mach 3, and a 30-percent loss occurs near the maximum ramjet Mach number because the 1.8-area-ratio thermal throat is 80 percent larger than the optimum value of 1.0. These levels of thrust deficit indicate that scheduling and control of the thermal throat location is an important aspect of the TBCC system development.

SUMMARY AND CONCLUSIONS

This paper describes a turbine-based combined-cycle (TBCC) propulsion system concept in which a turbojet and single-throat ramjet were integrated to provide thrust over a wide range of Mach numbers. Ejector action was used to isolate and modulate the turbojet and ramjet streams. This approach minimizes system weight and mechanical complexity by allowing the use of a common afterburner and nozzle, and provides a gradual, aerodynamic transition between turbojet and ramjet cycles.

The results of a rudimentary cycle analysis indicate that the integration of a present-day turbojet with a generic single-throat ramjet is feasible. The scramjet combustor area is sufficient to accommodate the ejector process that combines the turbojet and ramjet flows. A reduction in turbojet airflow demand with increasing flight Mach number results in complete transition to ramjet mode at Mach 3 with little step change in net thrust. Stoichiometric combustion of hydrogen with the remaining oxygen in the combined turbojet and ramjet streams is sufficient to effect a thermal throat over the Mach number range considered.

Two loss mechanisms are inherent to the TBCC system. The ejector total pressure loss results in 15-percent less specific impulse at takeoff in comparison to an ideal turbojet. The total pressure, or Rayleigh, loss associated with the thermal throat causes a 3-percent specific impulse deficit across the entire Mach range considered. The ejector loss can be eliminated at takeoff and low-speed conditions by moving the ejector flap to mechanically isolate the turbojet stream. However, inlet spillage drag considerations dictate that the system be switched to ejector mode by about Mach 1.1.

The net specific thrust of the TBCC system is maximized when the ejector operates at its critical point. This requires a prescribed variation in the thermal throat location with flight Mach number. Operation at a fixed thermal throat area is possible, but net thrust is significantly reduced. Scheduling and control of the thermal throat location thus emerges as an important aspect in the further development of this system.

The scramjet combustor length may be too short for complete mixing of the turbojet and ramjet flows. Complete mixing of these flows may not be critical, however, since the ejector process is employed more to isolate and modulate the two streams than to supercharge the ramjet flow. A bifurcated stream will, however, complicate the hydrogen-fueling process. A numerical and experimental investigation into this issue is recommended.

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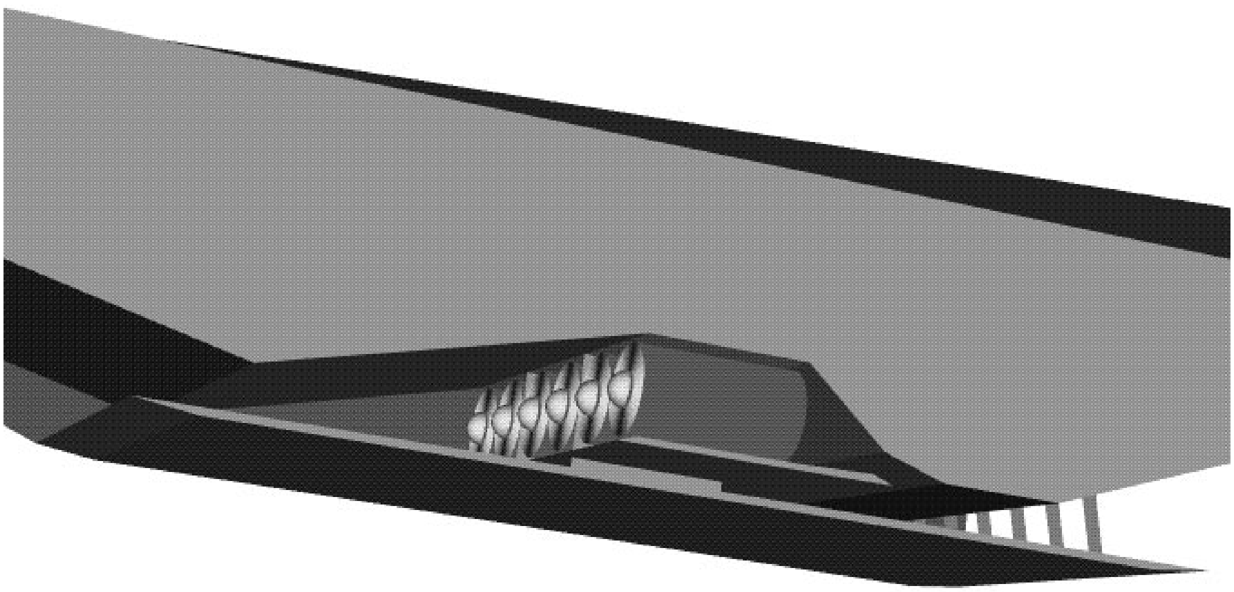


Figure 1.—Turbine-based combined-cycle (TBCC) propulsion system (sidewall removed).

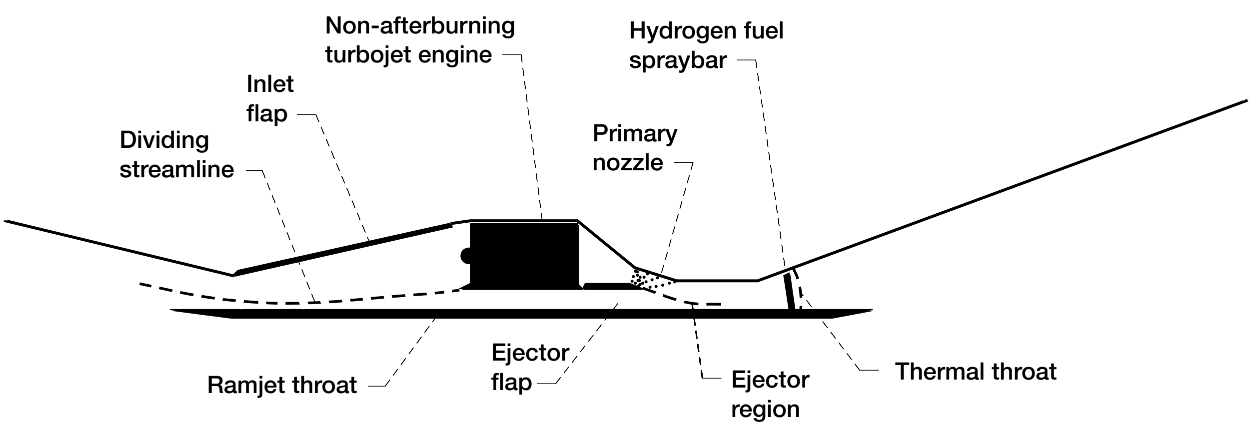


Figure 2.—Features of TBCC propulsion system.

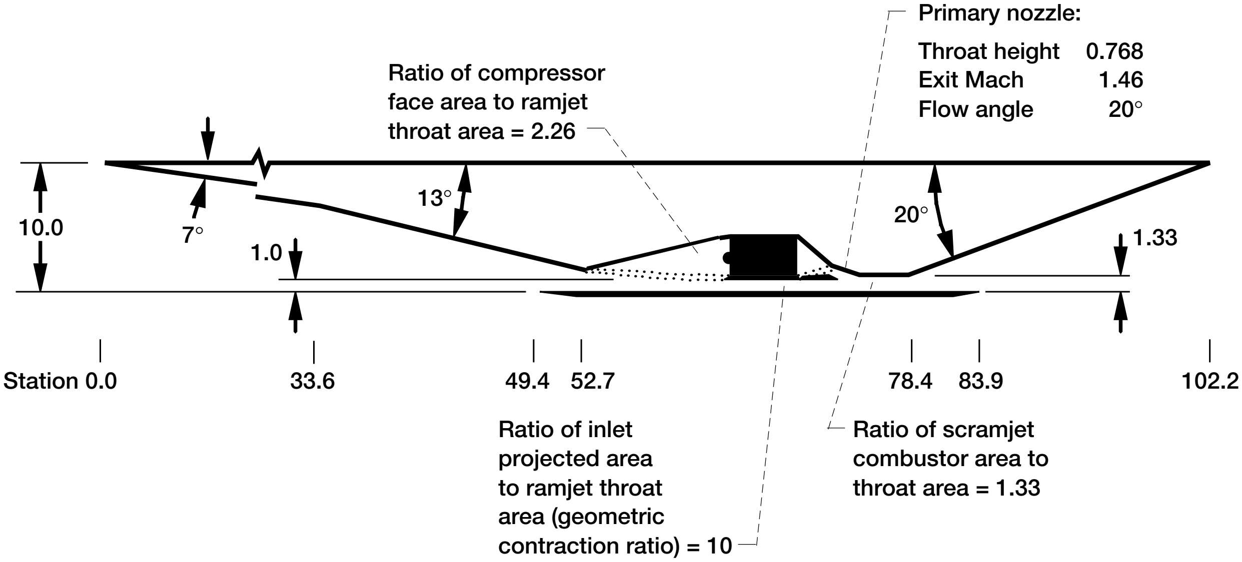
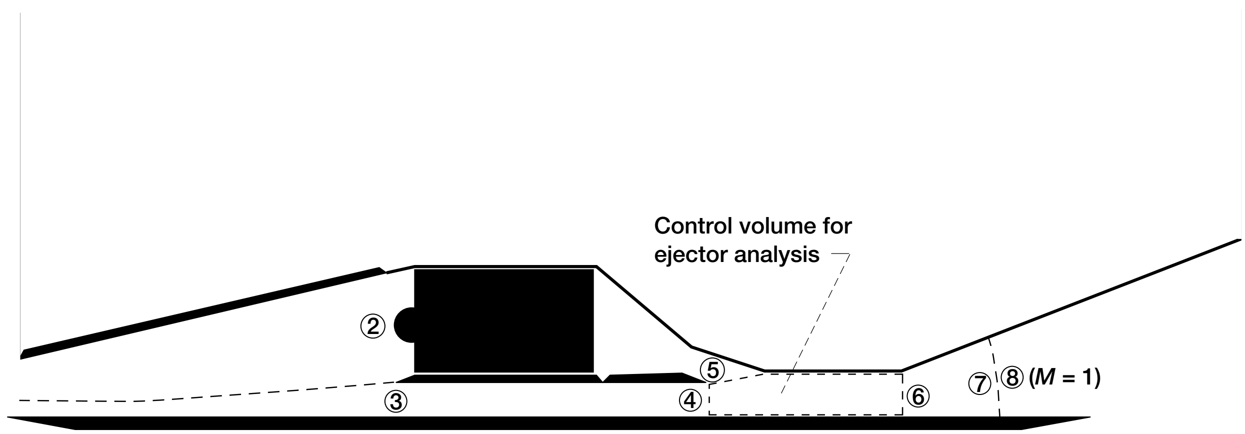


Figure 3.—Baseline configuration.



Flow properties

Station:	2	3, 4	5	6	7	8
P_t/P_{t0}:	0.95	0.95	3.63	2.47	2.47	2.09
T_t/T_{t0}:	1.00	1.00	3.53	3.53	3.53	8.19
Mach:	.450	.00	1.46	.605	.346	1.00

Figure 4.—Sea-level static cycle analysis example.

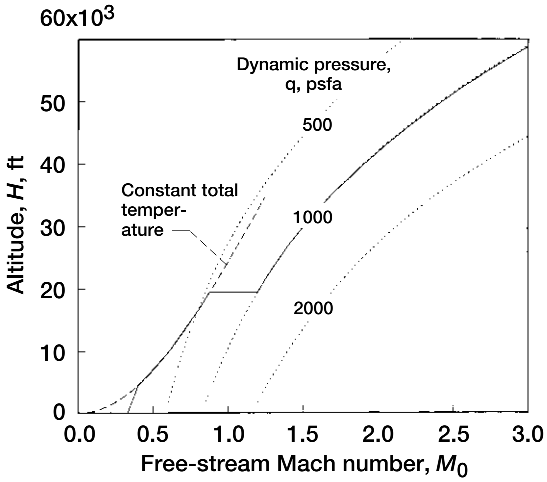


Figure 5.—Flight trajectory.

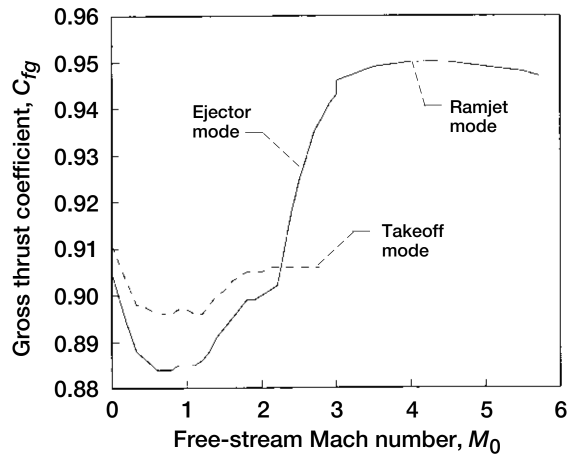


Figure 8.—Nozzle thrust coefficient.

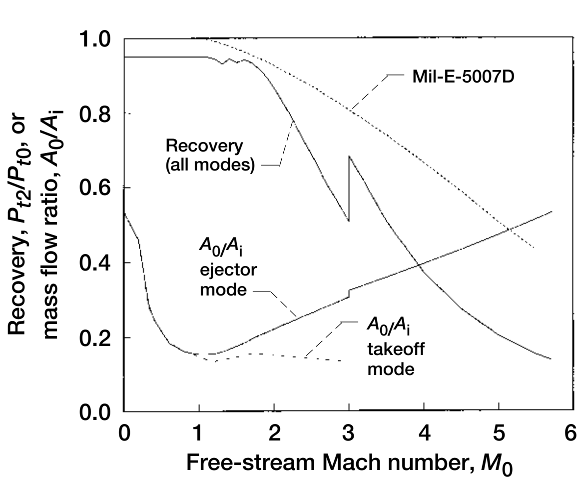


Figure 6.—Inlet performance characteristics.

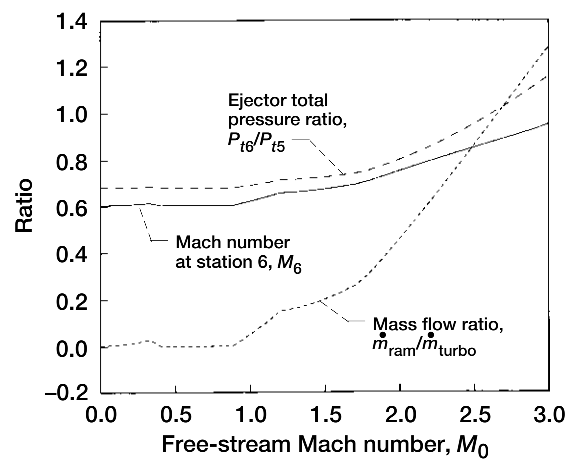


Figure 9.—Ejector characteristics (locus of critical points).

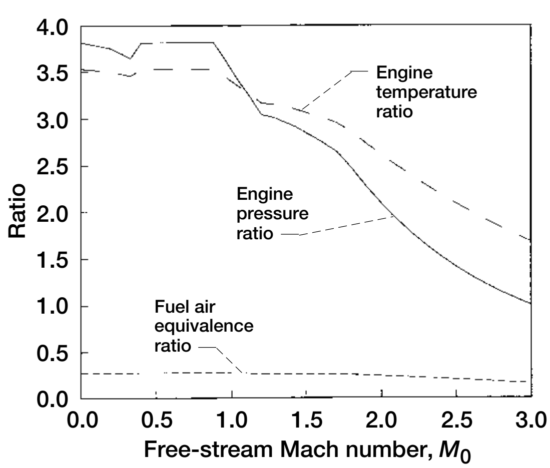


Figure 7.—Turbojet engine characteristics.

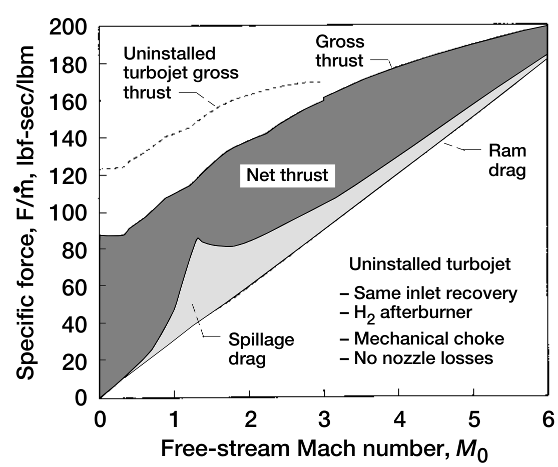


Figure 10.—Specific forces for baseline configuration.

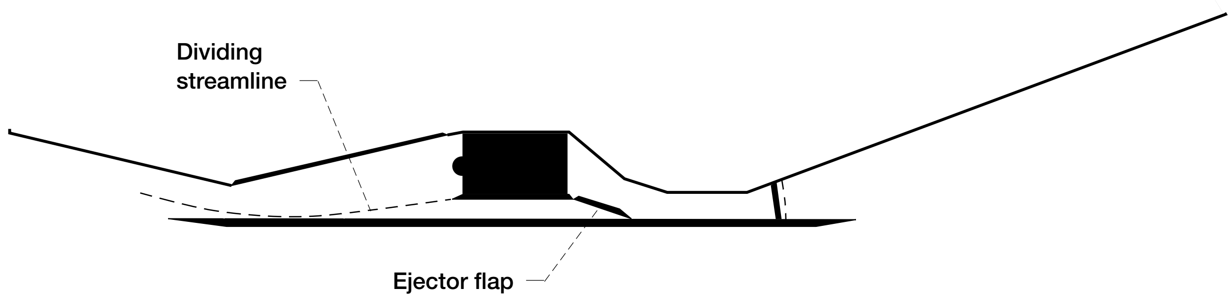


Figure 11.—Ejector flap in alternate position for takeoff and low-speed operation.

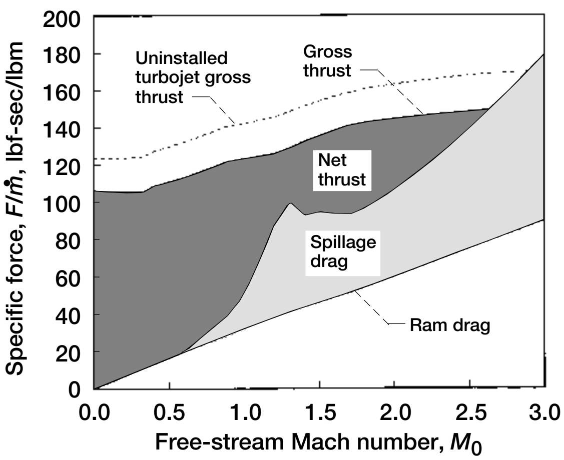


Figure 12.—Specific forces for alternate takeoff and low-speed configuration.

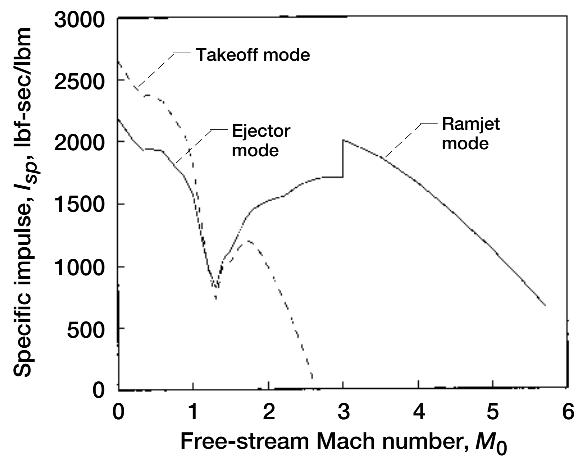


Figure 14.—TBCC system specific impulse.

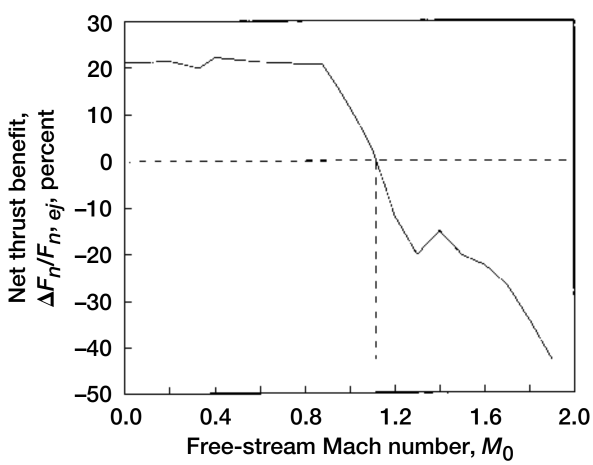


Figure 13.—Net thrust benefit for takeoff mode.

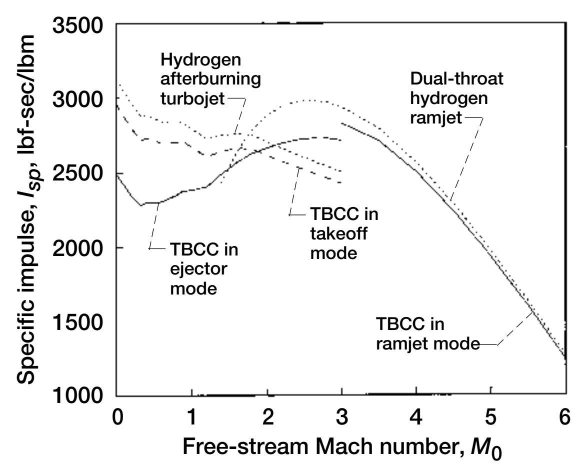


Figure 15.—Uninstalled specific impulse compared to ideal cycles (MIL-E-5007D inlet pressure recovery assumed for all).

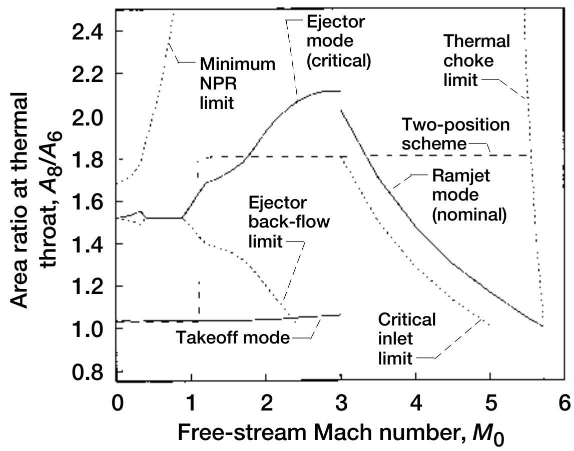


Figure 16.—Area ratios at thermal throat.

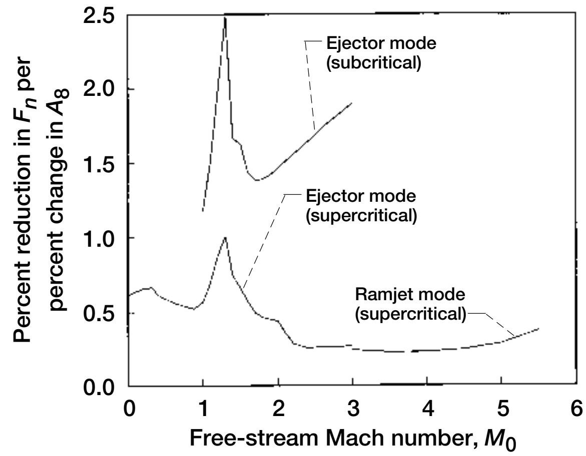


Figure 17.—Sensitivity of net thrust to thermal throat location.

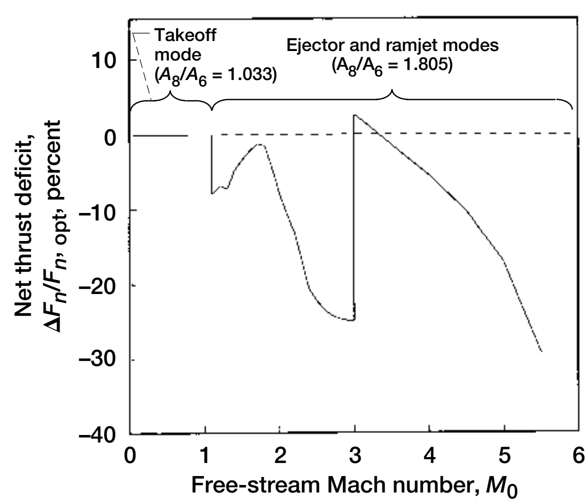


Figure 18.—Net thrust deficit for two-position thermal throat.

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13. ABSTRACT *(Maximum 200 words)*

A turbine-engine-based hybrid propulsion system is described. Turbojet engines are integrated with a single-throat ramjet so as to minimize variable geometry and eliminate redundant propulsion components. The result is a simple, lightweight system that is operable from takeoff to high Mach numbers. Non-afterburning turbojets are mounted within the ramjet duct. They exhaust through a converging-diverging (C-D) nozzle into a common ramjet burner section. At low speed the ejector effect of the C-D nozzle aerodynamically isolates the relatively high pressure turbojet exhaust stream from the ramjet duct. As the Mach number increases, and the turbojet pressure ratio diminishes, the system is biased naturally toward ramjet operation. The common ramjet burner is fueled with hydrogen and thermally choked, thus avoiding the weight and complexity of a variable geometry, split-flow exhaust system. The mixed-compression supersonic inlet and subsonic diffuser are also common to both the turbojet and ramjet cycles. As the compressor face total temperature limit is approached, a two-position flap within the inlet is actuated, which closes off the turbojet inlet and provides increased internal contraction for ramjet operation. Similar actuation of the turbojet C-D nozzle flap completes the enclosure of the turbojet. Performance of the hybrid system is compared herein to that of the discrete turbojet and ramjet engines from takeoff to Mach 6. The specific impulse of the hybrid system falls below that of the non-integrated turbojet and ramjet because of ejector and Rayleigh losses. Unlike the discrete turbojet or ramjet however, the hybrid system produces thrust over the entire Mach number range. An alternate mode of operation for takeoff and low speed is also described. In this mode the C-D nozzle flap is deflected to a third position, which closes off the ramjet duct and eliminates the ejector total pressure loss.

14. SUBJECT TERMS

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