Design and Experimental Study of an Over-Under TBCC Exhaust System

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Turbine-based combined-cycle (TBCC) propulsion systems have been a topic of research as a means for more efficient flight at supersonic and hypersonic speeds. The present study focuses on the fundamental physics of the complex flow in the TBCC exhaust system during the transition mode as the turbine exhaust is shut off and the ramjet exhaust is increased. A TBCC exhaust system was designed using methods of characteristics (MOC) and subjected to experimental and computational study. The main objectives of the study were: (1) to identify the interactions between the two exhaust jet streams during the transition mode phase and their effects on the whole flow-field structure; (2) to determine and verify the aerodynamic performance of the over-under TBCC exhaust nozzle; and (3) to validate the simulation ability of the computational fluid dynamics (CFD) software according to the experimental conditions. Static pressure taps and Schlieren apparatus were employed to obtain the wall pressure distributions and flow-field structures. Steady-state tests were performed with the ramjet nozzle cowl at six different positions at which the turbine flow path were half closed and fully opened, respectively. Methods of CFD were used to simulate the exhaust flow and they complemented the experimental study by providing greater insight into the details of the flow field and a means of verifying the experimental results. Results indicated that the flow structure was complicated because the two exhaust jet streams interacted with each other during the exhaust system mode transition. The exhaust system thrust coefficient varied from 0.9288 to 0.9657 during the process. The CFD simulation results agree well with the experimental data, which demonstrated that the CFD methods were effective in evaluating the aerodynamic performance of the TBCC exhaust system during the mode transition. [DOI: 10.1115/1.4025314]

1 Introduction

Compared to the rocket, the air breathing propulsion system does not need to carry its oxidizer on board so that it can provide a significant specific impulse advantage over the entire range of the vehicle's Mach number. Thus, air breathing propulsion systems are attractive for hypersonic flight vehicles for applications such as high-speed, long-distance strike or access to orbit. However, no single air breathing engine can provide consistent performance across as wide a range of Mach numbers as that of a rocket. Rather, air breathing propulsion systems require a combination of propulsion systems to provide this capability [1,2].

A turbine-based combined-cycle (TBCC) propulsion system employs a gas turbine engine to accelerate the vehicle to supersonic speed at which a ramjet or a scramjet can then operate to provide propulsion. The TBCC concept can be dated back to the 1950s when a turbo ramjet (combining a turbojet with a ramjet engine) was used to power the French-built Griffon II airplane that reached a speed of Mach 2.1. It demonstrated the feasibility of TBCC engines [3]. From 1960 to 1990, only some research on the combined cycle was conducted. The most successful TBCC example during this period was the Lockheed's SR-71 program, which ran from the early 1960s to 1989. The SR-71, powered by two J58 "bleed bypass" engines, was able to take off and fly up to a top speed of Mach 3+ [4]. After the 1990s, the combined-cycle engines were examined again to meet the demand for cheaper and safer launch vehicles. Consequently, many aspects of the TBCC engines were researched. Reference [3] indicated that for operating over the entire speed range, the TBCC flow path must be optimized, and variable geometry inlets and nozzles are required.

Further, weight and volume selections were major considerations in the design process. Reference [5] described the design and optimization of the integration of propulsion and airframe; both were addressed from aeropropulsive and mechanical perspectives. Reference [6] provided a thrust-matching analysis for a smooth transition between the two engines. Using a method to achieve valid propulsion/airframe integration at the initial design stage can establish the desired thrust matching at transition speed. In Ref. [7], one of the current TBCC design concepts was proposed: The turbine flow path was located parallel to and above the high-speed ramjet/scramjet flow path and it was shut off after the ramjet takeover. The main components of this "over-under" TBCC engine are shown in Fig. 1; a diverter door will rotate to close the low-speed flow path as the high-speed engine takes over. This low-speed mode transition is very critical since unstart of the low-speed inlet and the high-speed inlet might occur [8]. The opening and closing of the nozzle doors of the separate flow paths as well as the interaction of the two exhaust streams are the operational challenges that are inherent in this type of TBCC configuration.

For this over-under TBCC concept, the exhaust nozzle/aft body configuration was one of the most important components, which could influence considerably the stability of the vehicle during mode transition as well as the force vector and the moment balance of the total aircraft over the entire flight regime. The most important issue that existed in the designing procedure was that the nozzle system of the TBCC engine had to be operated from the low-speed region up to the required maximum flight Mach number. Therefore, the nozzle pressure ratio (NPR, i.e., the ratio of internal total pressure at the nozzle entrance to the static ambient pressure) would range from 2 to 300, which would inevitably require the variation of the area ratio of the nozzle throat and the nozzle exit. Thermal loads and dynamic loads also increased, especially in the ramjet mode, which resulted in the need of new cooling schemes and innovative structural design approaches [9].

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Fig. 1 Over-under TBCC concept (Ref. [7])

For the aerodynamic performance, using the computational method and inviscid relationships, Ref [10] described the flow field of the turbojet and ramjet interaction at the end of a common two-dimensional cowl. Reference [11] investigated the ejector process within a turbine-based combined-cycle propulsion and the impact of incomplete mixing on the overall propulsion system performance. In Ref. [12], methods of computational fluid dynamics (CFD) were used for both inviscid and viscous analysis of the effect of geometric variation on transonic performance of a dual-throat single expansion ramp nozzle (SERN). Results indicated that low nozzle performance and a highly three-dimensional nozzle flow field at transonic conditions would inevitably occur. To improve nozzle performance and reduce off-axis thrust at off-design flight conditions, fluidic injection might be an effective method. References [13,14] indicated that better performance was achieved by separating the overexpanded nozzle exhaust flow from the SERN ramp with oblique shocks generated through fluidic injection from the trailing edge of the SERN cowl and the ramp of the SERN. A similar method also had been used to improve the SERN's off-design performance in the Sanger Space Transportation System [15,16], the numerical and test results indicated that the gross thrust vector can be adjusted to the requirements of the inlet forces and external aerodynamics and high thrust efficiency in flight direction can be achieved with the injection of secondary air into the nozzle flow [17]. Reference [18] focused on an over-under installation concept and discussed various propulsion/airframe design, integration, performance, and operability considerations. The hydraulic and kinematic system models were developed by Gamble [19] to actuate a variable surface for propulsion control, and an over-under nozzle model was established and verified. However, to the knowledge of the authors, most research about the TBCC focused on the inlet and integration of the propulsion system and very little information, particularly experimental data, was available in the literature that was related to the TBCC exhaust system design and its detailed flow-field structure and aerodynamic performance during the mode transition phase.

In this paper, a TBCC exhaust system model was designed based on the method of characteristics (MOC) [20]. This paper reports on experimental and computational studies that examined the fundamental physics of the complex flow in the exhaust system during mode transition at Mach numbers of 2.0 and 2.5. Static pressure taps and Schlieren apparatus were employed to obtain the wall pressure distributions and flow-field structures. Steady-state tests were performed with the ramjet nozzle cowl at six different positions, at which the turbine flow path varied from half closed to fully opened, respectively. To provide a further understanding of the flow field during the TBCC exhaust-system mode transition, methods of CFD were adopted to simulate the flow field within the turbine flow path and ramjet flow path under the same test conditions. Comparison between the experimental and

simulation results was conducted to validate the simulation ability of the CFD software.

2 Nozzle Description

In Table 1, the conditions for the TBCC exhaust system design point are given. Based on the total inflow parameters of the ramjet nozzle, the ramp and flap contour at the design point of Mach 4 (cruise Mach number, $H = 28$ km) in ramjet mode was obtained by the MOC [21], as shown in Fig. 2(b). On the basis of this design point, the nozzle had an overall area ratio of 3.76. The isentropic ramp and flap contour would ensure the optimum thrust performance for the exhaust system at the cruise Mach number. Then, an appropriate position for the turbojet nozzle located on the ramjet expansion ramp contour was determined according to both the design point condition of the turbojet nozzle and the flow-field structure in the ramjet flow path before the mode transition. The exhaust system thrust performance and reliability at the turbojet nozzle design point of Mach 2.0 would be deeply influenced by the location of point P , as listed in Table 2. The length of splitter varied from $0.337L$ to $0.687L$, and an optimum axial thrust coefficient of the exhaust system was produced when the length of splitter was $0.587L$, then the location of point P would be determined. The surface S3 that served as the cowl contour of turbojet nozzle was shaped based on the turbojet nozzle design point listed in Table 1, while the surface S2 served as a portion of the ramjet expansion ramp-contour and it was designed using MOC at a design point Mach number of 4. Table 3 shows the typical geometrical parameters of the TBCC exhaust system. The ramjet will be started at approximately Mach 2.0. When the turbojet shuts down at Mach 2.5, the splitter cowl will be deflected upward to the turbojet nozzle flow path and turbojet nozzle exit doors will seal shut.

3 Experimental and Simulation Methods

3.1 Test Facility and Model. The nozzle model was tested at the Internal Flow Research Center at the Nanjing University of Aeronautics and Astronautics (NUAA). This facility is a cold blowdown wind tunnel. Compressed cold air of ambient temperature was supplied from a high pressure vessel, then it was expanded and speeded up through the ramjet nozzle model. Also,

Table 1 The design point conditions of the TBCC exhaust system

	$P_t/(kPa)$	$T_t/(K)$	NPR
Ramjet nozzle	150.6	1600	94
Turbojet nozzle	120.4	910	10.5

Fig. 2 Sketch of the TBCC exhaust system

Table 2 Axial thrust coefficients for different point P

L_p	$0.337L$	$0.462L$	$0.587L$	$0.687L$
Thrust coefficients	0.928	0.931	0.943	0.927

Table 3 Typical geometrical parameters of the TBCC exhaust system model

Parameters	$H_t/(mm)$	$H_e/(mm)$	$H_{t1}/(mm)$
	8.5	32	4.5

the compressed cold air was supplied to the turbojet nozzle plenum by a pipe and finally both jets were exhausted into a $1.8\text{ m} \times 1.7\text{ m} \times 1.7\text{ m}$ cubic vacuum chamber. The pressure ratios of both jets could be varied independently. A photograph of the test facility is shown in Fig. 3. The facility is typically operated at stagnation pressures ranging from 0.4 to 0.6 MPa, corresponding to the back pressure ranging from 0.02 to 0.05 MPa. Quartz windows are installed in the opposite walls of the test section to provide visual access for Schlieren and other nonintrusive techniques. A total-pressure rake consisting of five probes was located upstream of the ramjet nozzle to record the inflow total pressure of ramjet nozzle, and the inflow total pressure of turbojet nozzle was calculated by averaging two static pressure values that were measured by pressure taps located on each sidewall of the turbojet nozzle plenum. Other data measured included the static pressures along the surfaces of the turbine engine nozzle and ramjet engine nozzle on the symmetry plane of the flow path. All the static pressures were acquired by the electronically scanned pressure transducer system (ESP) of PSI Inc., with a margin of error of 0.05% of the full scale measurement. The tests were conducted at a stagnation temperature of approximately 300 K and the ambient temperature is nominally at 288 K.

Figure 4 is a cut-away image of the rectangular cross-sectional experimental model of a TBCC exhaust system that consists of a flange, circular-to-rectangular transition, ramjet flow path, cowl of ramjet nozzle, high-pressure plenum, turbine engine nozzle flow-path, expansion ramp, splitter cowl, and sidewall. The model had a rotating splitter cowl that could rotate to change the throat area of the turbine nozzle. And the cowl of the ramjet engine nozzle

Fig. 3 Blowdown wind tunnel facility

Fig. 4 Cut-away perspective view of the over-under TBCC exhaust nozzle model

could rotate to control the ramjet throat area during the mode transition. Both the turbojet and ramjet nozzles incorporated a variable geometry splitter cowl that could be manually regulated during the experiment. A total of 24 static pressure taps were set on the surfaces of the expansion ramp of the turbojet engine, on the bottom of the splitter cowl and the cowl of the ramjet nozzle.

Figure 5 shows the test points of the TBCC exhaust system during the mode transition, from which it was shown that the mode transition started from Mach 2 and finished at about Mach 2.5. The test conditions for all test points are listed in Table 4.

3.2 Computational Fluid Dynamics (CFD) Methods. To achieve a greater understanding of the exhaust system flow field during the mode transition process, FLUENT software was used to simulate the flow field. The effectiveness of this solver had been verified for the calculations of the hypersonic nozzle flow [22]. The Reynolds-averaged Navier–Stokes (RANS) equations were discretized upon a structured, multiblock, three-dimensional grid. In solving the flow equations, a point Gauss–Seidel scheme was used to perform time marching, and the Roe’s scheme in combination with the monotone upstream-centered schemes for conservation laws (MUSCL) interpolation method [23] was used to obtain the inviscid fluxes at the control surfaces. To alleviate the numerical stiff problem that would be possibly encountered in the calculations of the chamber flow, a time-derivative preconditioning technique was also adopted. The turbulent flow was modeled by the nonlinear renormalization group (RNG) $k-\epsilon$ model, of which

Fig. 5 Test points during TBCC exhaust-system mode transition

Table 4 Typical flow properties of the test points

Test points	NPR _{ramjet}	NPR _{turbojet}	He2/Ht2	He1/Ht1
(a) The splitter cowl in the half-closed position				
Transition point I (lower)	6.63	10.57	2.59	3.2
Transition point I	6.68	10.49	2.64	3.2
Transition point I (upper)	6.64	10.48	2.6	3.2
Transition point II (lower)	14.00	15.39	3.1	3.2
Transition point II	13.86	15.37	2.93	3.2
Transition point II (upper)	13.36	15.38	2.75	3.2
(b) The splitter cowl in the fully opened position				
Transition point I (lower)	6.64	10.47	2.34	2.44
Transition point I	6.67	10.50	2.38	2.44
Transition point I (upper)	6.64	10.51	2.35	2.44
Transition point II (lower)	13.84	15.20	2.61	2.44
Transition point II	13.61	15.88	2.54	2.44
Transition point II (upper)	13.78	14.26	2.43	2.44

the governing equations were discretized by a second-order upwind scheme. In addition, standard wall functions were introduced to model the near-wall region flow. The computational domain included the flow fields surrounding the model. The y^+ values of the first cell near to the wall were found to be about 30. For more details about the software, please refer to the user's guide [24].

Figure 6(a) shows a typical three-dimensional computational grid generated by a commercial mesh generation tool, ANALYSIS ICEM-CFD, and used for this study, and Fig. 6(b) shows a local three-dimensional grid. Only half the nozzle was modeled because the nozzle was symmetric along the expansion ramp. The three-dimensional grid had about 540,000 grid points.

Grid sensitivity analysis was performed with three grids of increasing resolution. For each grid size, the grid topology was the same. The obtained results are shown in Fig. 7(a) and listed in Table 5. It could be seen that the difference between the fine grid result and the coarse grid result was quite small (the maximum was about 0.015). Especially for the medium grid and the fine grid, the maximum was only about 0.007. Thus, a grid of 538,898 cells was adopted in the subsequent numerical analysis. The sensitivity to a different turbulence model was also addressed for this simulation. One and two equation turbulence models (Spalart–Allmaras, $k-\epsilon$ and $k-\omega$ shear stress transport (SST) models) were used in this study and the results are shown in Fig. 7(b), which indicated that the effect of turbulence model is quite limited and could be neglect in this simulation.

The statistics of the error of the residuals are shown in Table 6, from which one could see that the maximum relative error of the residuals had even dropped below 0.001 and when the residuals approached stability, the iteration could be considered as converged.

Fig. 6 The typical computation grid of the over-under TBCC exhaust system

3.3 Thrust Definition of the TBCC Exhaust System. A postprocessing program was used to calculate the parameter that was used to quantify the nozzle performance—the axial thrust coefficient. The thrust of the nozzle was defined as the sum of the axial component of flow momentum and the gauge pressure force at nozzle exit plane, namely the relative impulse at the nozzle exit plane. Therefore, the whole thrust of the over-under TBCC exhaust system was the sum of the turbojet and ramjet engine thrust. As shown in Fig. 8, the exit plane of the turbojet nozzle was plane AB, and the thrust could be decomposed into the axial force and the normal force. The axial thrust vector was the x (longitude) direction component, and could be calculated as

$$F_{xt} = \int_{AB} \rho V_x (\vec{V} \cdot \vec{n}) dA + \int_{AB} (p_{AB} - p_{amb}) dA \sin \theta \quad (1)$$

where θ was the slope angle of the AB section, p_{AB} was the absolute static pressure on it, and p_{amb} was the ambient pressure. The BC plane was defined as the exit plane of the ramjet nozzle. The axial thrust defined as F_{xr} on it could be calculated in the same way as described above.

The ideal thrust F_S is used to calculate the thrust coefficient and to estimate the performance of the nozzle, which can be expressed as the following:

$$F_S = \dot{m} \sqrt{\frac{2\gamma}{\gamma-1} RT_t \left[1 - \left(\frac{p_{amb}}{P_t} \right)^{\frac{\gamma-1}{\gamma}} \right]} \\ = \dot{m} \sqrt{\frac{2\gamma}{\gamma-1} RT_t \left[1 - \left(\frac{1}{NPR} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (2)$$

where, $\dot{m}$, T_t , and NPR are the mass flow rate, the total temperature, and the nozzle pressure ratio, respectively. Then the axial

Fig. 7 Pressure distributions along ramp relative to different grids and turbulence models

Table 5 Grid independence analysis

Run	No. of cells	Grid density factor (r)	Maximum relative error
Coarse	244,893	0.45	0.015
Medium	538,898	1	0
Fine	923,502	1.7	0.007

thrust coefficient can be introduced as Eq. (3) to measure the deviation of the real axial thrust from the ideal one and to quantify the performance of an exhaust system (F_{xt} , F_{xr} , F_{st} , and F_{sr} are the axial thrust and ideal thrust of the turbine nozzle and ramjet nozzle, respectively)

$$C_{fx} = (F_{xt} + F_{xr}) / (F_{st} + F_{sr}) \quad (3)$$

4 Analysis of Results

4.1 Comparison Between Experimental and Simulation Results. The static pressures at the turbojet nozzle upper wall and ramjet nozzle lower wall obtained by experiments and numerical simulations are compared in Figs. 9, 10, and 11, where Figs. 9 and 10 correspond to the transition point I and transition point II,

Fig. 8 Control plane used for nozzle thrust calculations

Fig. 9 Comparison of experimental static pressure data with CFD prediction at transition point I corresponding to the splitter cowl in the half-closed position

respectively, with the splitter cowl in the half-closed position. Figure 11 corresponds to the transition point II with the splitter cowl in the fully opened position. The comparisons of the pressure distribution measurements and simulation results are given in the form of the nondimensional pressure. As presented in Fig. 9(a), in the subsonic convergent part of the turbojet nozzle duct, the static pressure decreases and reaches a minimum in the throat area. The static pressure increases again downstream of the throat is due to

Table 6 Residuals after the iteration converged

Continuity	x-velocity	y-velocity	z-velocity	Energy	k	Epsilon
1.2416×10^4	2.4552×10^4	3.1586×10^5	6.3754×10^4	4.5166×10^5	7.2783×10^5	8.7882×10^5

Fig. 10 Comparison of experimental static pressure data with CFD prediction at transition point II corresponding to the splitter cowl in the half-closed position

the large geometrical curvature of upper wall of the turbojet nozzle. Recompression effects occurred in this region and a weak shock in this region results in higher pressure. These phenomena occurring in the measured data are excellently reproduced by the numerical simulation. The numerical static pressure distributions along the walls of the nozzle model agreed well with the experimental data in both the values and the trends.

4.2 Flow Features. The shock waves, expansion waves, shock–shock interaction, shock–expansion wave interaction, viscous shear layers, and other flow features were all clearly demonstrated in the Mach number contours and static pressure contours by CFD simulations. This provided a thorough understanding of the aerodynamic phenomena involved in the nozzle mode transition phase. Though the flow field of the exhaust system of the over–under TBCC engine was very complex, the numerical code could provide a satisfactory result. Furthermore, as shown in Fig. 12, a good agreement between the measured and the calculated nozzle flow-field structures had been observed. Flow features and shock positions matched with each other quite well, which ensured that the CFD methods could be applied to simulate the exhaust system flow field during the mode transition phase satisfactorily. Further information (the structure of the flow field and the performance of the exhaust system) about the flow could be obtained with the CFD methods.

In Fig. 13, the Schlieren picture of the TBCC exhaust system at the mode transition point II with the splitter cowl fully opened is presented, which imparted an impression of afterbody/nozzle flow field. The wind tunnel's Schlieren window only allowed the observation of a part of the afterbody region. Basic flow features

Fig. 11 Comparison of experimental static pressure data with CFD prediction at transition point II corresponding to the splitter cowl in the fully opened position

are explained with the aid of Fig. 14. For the direction and the static pressure mismatch between the exit planes of the turbojet nozzle and the ramjet nozzle, the supersonic exhaust jets from the turbojet engine and ramjet engine interacted with each other, and therefore, several shocks were evident in this image. The interpretation and description of these features were shown in the profile illustration of Fig. 14.

Shear layer 1 separated the turbojet and ramjet nozzle flow-paths. Actually, the two supersonic flows encountered at point B and interacted with each other. The static pressure of the turbojet nozzle at point B was much higher than that of the ramjet nozzle at point C, hence the streamlines turned towards the ramjet flow path. Thus, a strong oblique shock was generated at the trailing edge of the ramjet expansion ramp. At the cowl trailing edge of the ramjet nozzle, the overexpanded flow resulted in a shear layer that turned towards the ramjet internal flow field and generated a weak plume shock (DN) at the point D, which intersected with the strong shock CN and then impinged on the shear layer 1. The strong shock NH reflected at the shear layer 2 and formed an expansion wave.

Because of the strong curvature of upper wall downstream of the turbojet nozzle throat, an internal shock (AM) formed at the point A. The shock BM generated at point B impinged on the expansion ramp surface and then created a reflected shock from the afterbody. In some cases, the impinging shock was strong enough to create a separation region on the expansion ramp surface, which would result in losses in performance of the exhaust system. Therefore, the location and contour of the splitter cowl should be carefully designed and optimized in order to alleviate the interaction between the turbojet and ramjet nozzle flow-paths.

Fig. 12 Comparison of Schlieren image with computational Mach contours at transition point II

Fig. 13 Experimental Schlieren image at transition point II

Fig. 14 Computational flow field structure of the TBCC exhaust system at transition point II

4.3 Thrust Coefficient. The exhaust axial thrust coefficients calculated using Eq. (3) are listed in Table 7. In most of the cases, the exhaust thrust coefficient corresponding to the splitter cowl in the fully opened position was better than that corresponding to the splitter cowl in the half-closed position. With the splitter cowl in the half-closed position, the exhaust thrust coefficient had a minimum value of 0.9288 at transition point I (upper). For this test point, the turbojet flow and the ramjet flow are both overexpanded, and oblique shock appears at the trailing edge of the turbojet nozzle ramp and ramjet nozzle ramp, respectively. The phenomena described here could also be seen in the Fig. 9 as the pressure rising at the trailing edge of the ramp. For the transition point II (upper), the ramjet is operating at an NPR of around 13.

Table 7 Thrust coefficient of the TBCC exhaust system corresponding to the splitter cowl in the half-closed and fully opened positions

	$C_{f,half}$	$C_{f,full}$
Transition point I (lower)	0.9371	0.9413
Transition point I	0.9298	0.9406
Transition point I (upper)	0.9288	0.9362
Transition point II (lower)	0.9591	0.9553
Transition point II	0.9649	0.9657
Transition point II (upper)	0.9651	0.9600

Compared with an NPR of 6.6 at transition point I (upper), the ramjet flow is no longer overexpanded and oblique shock at the end of the ramjet nozzle ramp was eliminated. Also, the NPR of turbojet nozzle increased from 10.5 to 15.3; the turbojet nozzle performance at this test point would be better than that at transition point I (upper). Consequently, the axial thrust coefficient increased from 0.9288 at transition point I (upper) to 0.9651 at transition point II (upper). Similarly, the minimum and maximum values with the splitter cowl in the fully opened position were 0.9362 (transition point I) and 0.9657 (transition point II), respectively. For the splitter cowl in the fully opened position, as the splitter cowl was deflected downwards to the ramjet flow path, both the area ratios of ramjet nozzle and turbojet nozzle had decreased compared with that of the splitter cowl in the half-closed position. The turbojet nozzle flow and ramjet nozzle flow were no longer overexpanded, and the oblique shock at the end of the ramp of each nozzle flow was eliminated (as shown in Fig. 11), thus the exhaust thrust performance corresponding to the splitter cowl in the fully opened position would be better than that corresponding to the splitter cowl in the half-closed position.

5 Conclusions

To provide insight into the fundamental physics of the complex flow in a TBCC exhaust system during the critical transition mode regime, an experiment was conducted to evaluate the performance of the TBCC exhaust nozzle. Static pressure taps and Schlieren apparatus were employed to obtain the wall pressure distributions and flow-field structures. Steady flow fields with the ramjet nozzle cowl at six different positions were measured, with the turbojet flow path in half-closed and fully opened positions, respectively. To provide a thorough understanding of the flow during the TBCC exhaust system mode transition, methods of CFD were also adopted to simulate the aerodynamics within the turbine and ramjet flow paths under the same test conditions. Results indicated that the flow structure is complicated and the two exhaust jet streams interacted with each other during the exhaust system mode transition. The exhaust system thrust coefficient varied from 0.9288 to 0.9657 during the mode-transition process, which seemed to be a good performance. The CFD simulation results agreed qualitatively with the experimental data, which demonstrated that the CFD methods were effective in evaluating the aerodynamic performance of the TBCC exhaust system during the mode transition.

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Nomenclature

A = area
 $C_{f, full}$ = axial thrust coefficient of the exhaust system corresponding to the splitter cowl in the fully-opened position

$C_{f, \text{half}}$ = axial thrust coefficient of the exhaust system corresponding to the splitter cowl in the half-closed position
 F_{sr} = ideal thrust of ramjet nozzle
 F_{st} = ideal thrust of turbine nozzle
 F_{xr} = axial thrust of ramjet nozzle
 F_{xt} = axial thrust of turbine nozzle
 $\dot{m}$ = mass flow rate
 MOC = method of characteristics
 NPR = nozzle pressure ratio
 p_{amb} = ambient pressure
 P_t = total pressure
 SERN = single expansion ramp nozzle
 T_t = total temperature
 TBCC = turbine based combined cycle
 V_x = axial velocity
 x = coordinates
 θ = angle between the velocity and the direction at nozzle exit
 ρ = density

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