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Effect of Non-uniform Flow on the Ignition and Combustion in Scramjet Combustor

(Recommended by Prof. Sree Hari Rao V.)

The uniform and nonuniform flow conditions are created at the entry of the combustor by incorporating a Circle to Rectangle Transition duct (CR-Transition duct, for short) in the subsonic and supersonic flows ahead of the combustor. Computational and experimental studies have been carried out on a ramp-cavity based supersonic combustor with supersonic and subsonic transitions. It is observed that the non-uniform three-dimensional flow caused by the presence of the CR-Transition duct, help in ignition of kerosene, while the same could not be achieved with uniform flow at the combustor entry. The experimental and computational investigations are discussed in this paper.

Условия однородного и неоднородного потоков создаются введением круглого и прямоугольного соединительных воздухопроводов в дозвуковой и сверхзвуковой потоки перед камерой сгорания. Численные и экспериментальные исследования проводились на основе ramp-cavity со сверхзвуковыми и дозвуковыми соединениями. Установлено, что неоднородный трехмерный поток, вызываемый наличием соединительного воздухопровода, способствует зажиганию керосина, что не достигается при однородном потоке на входе камеры сгорания.

Key words: kerosene fueled scramjet combustor, supersonic combustion ramjet combustor, mixing, combustion efficiency, pilot injection, barbotage, injection. или supersonic combustion ramjet combustor, injection schemes, non-uniform flow.

1. Introduction. Defence Research and Development Laboratory (DRDL for shot) is currently developing flight focused SCRAMJET engine. To pursue basic studies on supersonic combustion, SCRAMJET combustor facility has been established to conduct experiments on different types of combustor configurations. It is very essential to exactly simulate the combustor entry conditions in the ground testing. Also the entry flow characteristics greatly influence the combustor performance and validity of the ground testing for the realization of flight hardware. Towards this, two areas of studies are being focussed. The first area of studies pertains to the kerosene ignition characterisation using two different injection schemes and the second area of studies pertains to kerosene ignition characterisation under two different entry flow conditions. Important results of the studies on the injection schemes have been presented in [1]. In the present study, two-dimensional ramp and cavity based combustor testing has been carried out on two different facility configurations (supersonic transition and subsonic transition). The effect of inlet flow conditions on the kerosene ignition and sustained combustion has been investigated.

2. Experimental setup. The experiments were conducted in two different facility configurations. In the first configuration, CR transition duct placed in the supersonic flow as shown in Fig. 1, *a*. In the second case the CR-transition duct is placed in the subsonic flow as shown in Fig 1,*b*. The test facility consists of a hydrogen burner, to simulate the high enthalpy flow conditions, by burning hydrogen in the air, with oxygen replenishment. The composition of the vitiated air is controlled to achieve a 21 % oxygen mole fraction, with regulated gas flows, by means of operating pressure and injection orifices. The hydrogen burner pressure and temperature are monitored for the calculation of total pressure and total temperature. The high enthalpy gas is then accelerated to reach the required Mach number and flow conditions. The burner is axis-symmetric in construction, for ease of fabrication and to avoid structural problems associated with rectangular constructions. Rectangular flow field has been established by suitable arrangement at the entry to the SCRAMJET combustor.

Circle to rectangle transition duct. The combustors focused towards the hypersonic flight operations are rectangular in cross section. The flow field will be highly depending on the combustor flow path, thus it is essential to carry out ground testing on two-dimensional combustor. As mentioned earlier, conventional vitiated air heaters are axis-symmetric in nature. Thus it is required to convert the flow from axis-symmetric to rectangular flow, which requires a CR-transition duct to be positioned in the facility flow path. The CR-transition duct has been designed from converting the circle to ellipse and then to a rectangle [2]. The major and minor axis distance is equal near the entry where the cross section is circular, and they are varied slowly to the rectangular cross section at the entry of the combustor. The internal flow path has been developed from two-dimensional contours in two normal planes intersecting at the centre of each of the cross sectional planes [2, 3]. Thus the transition duct has two planes of symmetry. The two dimensional contours have been designed considering the supersonic and subsonic flow properties. For the case of supersonic transition duct, contour design has been made for minimum losses [2, 4]. Here an attempt has been made to position the CR-transition duct both in the supersonic flow and

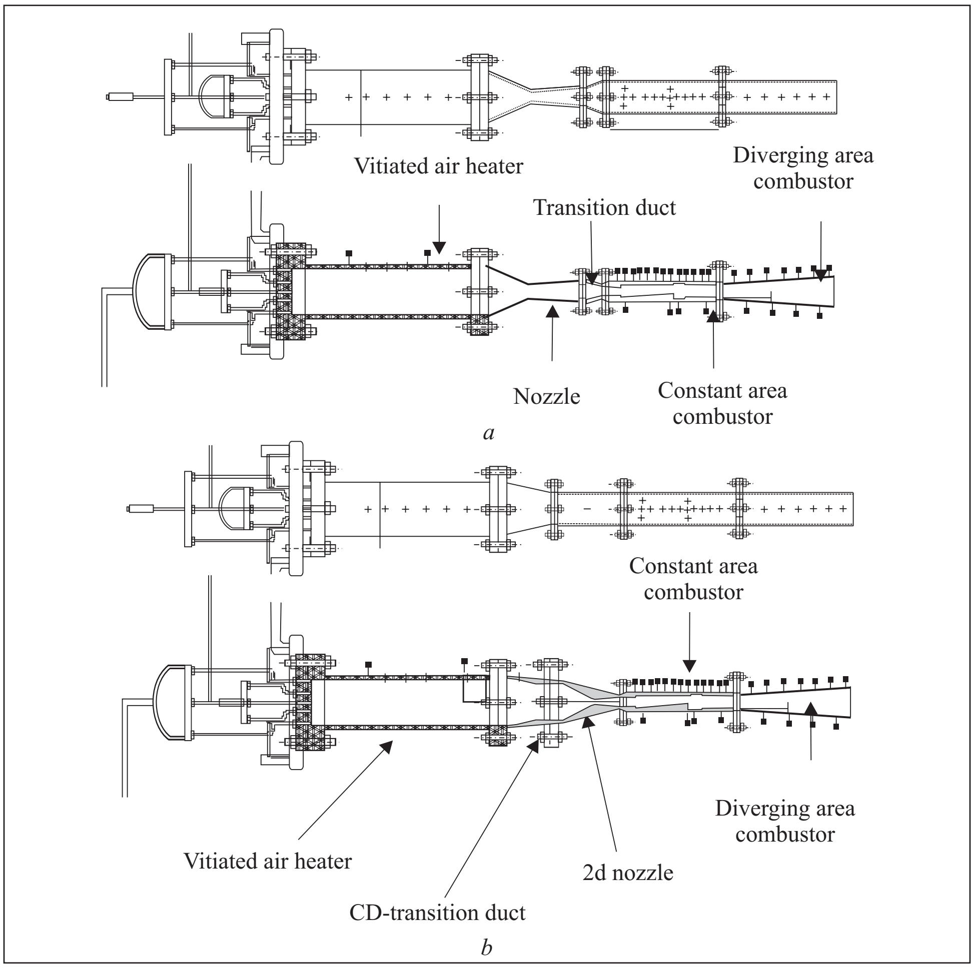

Fig. 1. SCRAMJET combustor test facility (schematic)

in the subsonic flow and to investigate the effect of the change of the flow field at the entry of the combustor.

CR-transition duct in the supersonic flow. The solid model of the supersonic transition duct and the schematic of the flow path is shown in Fig. 2. The vitiated air from the hydrogen burner is expanded to supersonic flow by an axis-symmetric nozzle. The nozzle is designed to operate at a Mach number of 2.6. The axis-symmetric flow is converted in to a rectangular flow using a CR-transition duct at the exit of the nozzle. The two dimensional combustor follows the CR-transition duct.

CR-transition duct in the subsonic flow. The schematic of the flow path is shown in Fig 3 (see color inset). This configuration is shown in Fig. 1, *b*.The viti-

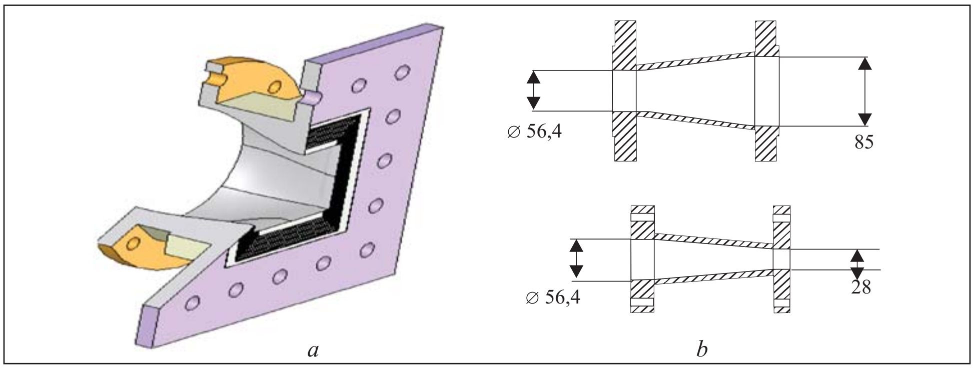

Fig. 2. Transition duct: solid model (*a*) and schematic drawing (*b*)

ated air from the hydrogen burner is converted to rectangular flow by positioning the CR-transition duct in the subsonic region, then a two dimensional nozzle of mach number of 2.5 is used to deliver supersonic flow to the two dimensional combustor. The facility photograph is shown in [1].

Combustor configuration. The mixing of fuel stream injected into the supersonic combustor is poor because of the high speed cross flow, due to the suppression of turbulence and prevention of fuel penetration. Thus it is essential to design mixing devices like ramps, which produce axial vortices [5—8]. Also, ignition and sustained combustion of hydrocarbon fuels are difficult because of the low pressure in the supersonic combustor and low residence time of the fuel. Thus, flame holding devices like cavities are used for sustained combustion [9—14]. Here, the combustor has been designed with ramps for mixing and cavities for flame holding. Fig. 4 shows the ramp-cavity based SCRAMJET combustor. It has an entry cross-section of 85×28 mm^2. The first constant area section is called isolator, followed by a 6 mm step on both sides. Ramps and cavities are used for fuel injection and flame holding. The length of the ramps is 200 mm and angle is 4°. The ramps are staggered in the width of 85 mm. There are two ramps at the top. There are one full width ramp and two half-width ramps at the bottom surface. There are cavities at both top and bottom surface following the ramps. The cavities run from one sidewall to other, for a width of 85 mm. The cavities are stable cavities with an *L/D* ratio of 6.75, with a slanting rear wall of 45°. A constant area section of 85×40 mm^2 follows the cavities for length of 142 mm. A diverging area combustor of 410 mm length is provided to avoid thermal choking. The semi-cone angle of the diverging combustor is 3.2°. Static pressure transducers are mounted on the top wall of the combustor. Inside wall flushed thermocouples are mounted in the bottom walls and skin thermocouples are mounted on the outer wall of the combustor. Pressure and temperature measurements are used to evaluate the performance of the combustor.

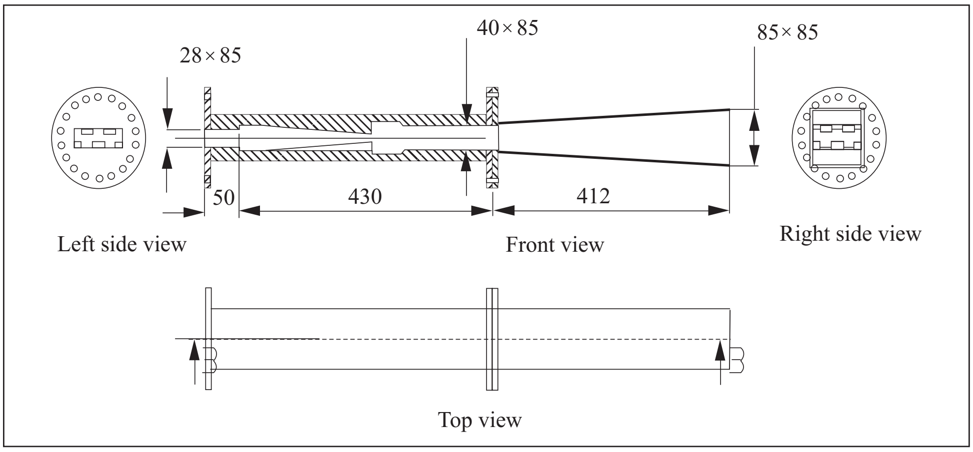

Fig. 4. Ramp-cavity SCRAMJET combustor

Fuel injection scheme. The fuel injection locations in the combustor are shown in the Fig. 5 (see color inset). There are normal injections ahead of each ramp. The fuel injection scheme is designed based on fuel penetration heights in supersonic flows [15]. The second stage fuel has been injected from the base of all ramps, over the cavities. The injection hole diameter is 0.4 mm. The combustion studies on the ramp-cavity SCRAMJET combustor has been carried out with kerosene fuel barbotaged with hydrogen. The internal flow path of each injector is shown in [1]. The barbotaging of kerosene with hydrogen helps ignition of kerosene [16, 17]. The solid model of the combustor integrated with barbotage injection system is shown in [1].

3. Computational analysis (non-reacting case). Computational analysis for the two configurations (transition duct in the supersonic and subsonic flow) has been carried out, for «no fuel injection» case using the computational fluid dynamics code «CFX Tascflow». This is a full three dimensional Navier— Stokes, unstructured solver. Fig. 6 (see color inset) shows the solid models of the geometry. Fig. 7 (see color inset) shows the computational domain. The simulation has been carried out from the throat of the nozzle for both the cases.

Fig. 8, *a*, *b* (see color inset) shows the mach number plots from the nozzle throat to the combustor entry, for supersonic transition and subsonic transition respectively, in the vertical plane. It is clear that the flow is uniform across the plate at the combustor entry, for the subsonic transition case. For the supersonic transition duct case, due to the duct geometry and change in the nature of the flow, shock waves are present in this plane and at the core the flow again accelerates.

Fig. 9, *a* (see color inser) shows the mach number plots at different cross sections, for the supersonic transition case. Planes 1,2 and 3 are in the nozzle, where the mach number distribution is uniform. The planes 4 and 5 are in the middle of the transition duct and at the entry to the combustor. In the plane 4, near the side there is an expansion zone and in the corresponding zone in plane 5, the separated flow exists. Fig. 9, *b* (see color inset) shows the mach number plots at different cross sections for the subsonic transition case. The flow is uniform in all the planes and uniformly expands to the designed mach number.

Fig. 10, *a*, *b* (see color inset) shows the mach number plots at the symmetry plane of the combustor, for supersonic transition case and subsonic transition case, respectively. In the subsonic transition case, the flow expands at the step of the combustor. For the supersonic transition case expansion is suppressed. It is clear that the separated zone is larger and extends almost to the end of the ramps at the top wall. Also, in the bottom plate the separated zone extends up the ramp leading edge.

It should be noted that the first stage fuel injection locations are ahead of the ramps, which is now in the large separated zone for the supersonic transition case. And this feature is absent for the subsonic transition case.

Fig. 11, *a*, *b* (see color inset) shows mach number plots for the plane parallel and shifted 20 mm to the symmetry plane for supersonic transition and subsonic transition case, respectively. For the subsonic transition case the flow field is similar to that of the symmetry plane. But in the supersonic transition case large separation zone is absent.

Fig. 12 (see color inset) shows the top wall pressure distribution at the symmetry plane for both the cases. The pressure plot indicates the flow field discussed above. In the case of subsonic transition, the pressure decreases suddenly due to the step after the constant area section. In the supersonic transition case, the pressure increases near the step region. This indicates that the effect of step is suppressed in the separated zone. The pressure increases in the ramp region, in both cases. But the magnitudes of the pressure are very high for the case of supersonic transition. The expansion at the base of the ramps and downstream

Table 1. **Gas flows (supersonic transition duct)**

No	Parameter	Hot reacting flow	
		Expected	Actual
1	Air , kg/s	0.750	0.625
2	Hydrogen, kg/s	0.014	0.012
3	Oxygen, kg/s	0.180	0.181
4	Kerosene fuel, gm/s	24.0	26.00
5	Equivalence ratio combustor (kerosene)	0.40	0.467
6	Equivalence ratio combustor (barbotage hydrogen)	0.01	0.01

pressure distribution is similar in both the cases, with different magnitudes. Thus it is clear from the CFD analysis that, there exists a strong three dimensional effect caused by the transition duct in the supersonic flow.

4. Experimental studies. It is clear from the CFD analysis that the flow has a strong three-dimensional flow field. The supersonic flows exhibit a strong suppression characteristic over the turbulence and disturbances, thus uniform fuel distribution is difficult. These three — dimensional disturbances generated by the transition duct in the supersonic flow is augmenting the ignition and combustion of kerosene. Combustion studies are carried out for two facility configurations of supersonic and subsonic transition, to evaluate the pros and cons of non-uniform flow conditions at the scramjet combustor entry.

Combustion study for supersonic transition. Tab. 1 shows the flow conditions achieved in the combustion experiment.

Fig. 13, *a* (see color inset) shows test results — the wall pressure distribution along the combustor. The pressure curve for no fuel injection case indicates that the pressure levels are higher than those predicted by the CFD analysis. Ignition of kerosene with hydrogen barbotage is achieved. There is a substantial pressure rise between the no fuel injection and fuel injection cases. Also the pressure rise is observed ahead of ramp fuel injectors. This indicates the presence of the separated zone.

Combustion study for subsonic transition. The Tab. 2 shows the flow conditions achieved in the combustion experiment.

Fig. 13, *b* (see color inset) shows test results — the wall pressure distribution along the combustor. The pressure curve for no fuel injection case indicates that the pressure levels are lesser compared to the previous experiment as predicted by the CFD analysis. The pressure decrease near the step, which was absent in the supersonic transition case, could be observed in the wall pressure plot. In this experiment ignition of kerosene with hydrogen barbotage could not be achieved in the combustor, so the pressure rise between the no fuel injection case and fuel injection case could not be observed.

Table 2. **Gas flows subsonic transition duct**

No	Parameter	Hot reacting flow	
		Expected	Actual
1	Air , kg/s	0.750	0.645
2	Hydrogen, kg/s	0.014	0.012
3	Oxygen, kg/s	0.180	0.192
4	Kerosene fuel, gm/s	24.0	23.2
5	Equivalence ratio combustor (kerosene)	0.40	0.40
6	Equivalence ratio-combustor (barbotage hydrogen)	0.01	0.01

Fig. 14 (see color inset) shows the comparison of pressure distribution for supersonic transition and subsonic transition case for «no fuel injection» case. The pressure rise was small for the subsonic transition case, as the flow is clean and shock free at the entry to the combustor.

5. Conclusions. Experimental and computational investigations have been carried out on the effect of uniform and non-uniform flow conditions on the ignition and sustained combustion of kerosene in the supersonic flow. Investigations have been carried out with positioning CR-transition duct in supersonic and subsonic flow to achieve rectangular flow field at the two dimensional combustor. CFD analyses have been carried out on both supersonic and subsonic transition configurations in a non-reacting (without fuel injection) flow conditions. Experimental studies have been conducted on similar geometries with kerosene injection. The following conclusions are drawn from the above studies. The presence of the transition duct in the supersonic flow produces dominant three dimensional flow fields with complex shock and expansion waves. The presence of these waves help in better mixing of fuel and air. The presence of the transition duct produces higher pressure in the combustor, providing favourable flow conditions for ignition and sustained combustion of kerosene, which could not be achieved in clean supersonic flow. This is an important input for the vehicle designer, while carrying out end to — end simulation of the vehicle. Non-uniform flow at entry of the combustor is a positive indication for better combustor performance. But the stagnation pressure losses should be minimized, by designing the transition flow path to produce similar non-uniform flow at the entry on the combustor with minimum pressure losses. The combustor stagnation losses should also be minimized by using aerodynamic shape of the ramps, angled fuel injection and by controlling the blockage.

Умови однорідного та неоднорідного потоків створюються уведенням круглого та прямокутного з’єднувальних повітропроводів у дозвуковий та надзвуковий потоки перед камерою згоряння. Чисельні та експериментальні дослідження проводились на основі ramp-cavity з надзвуковими та дозвуковими з’єднаннями. Визначено, що неоднорідний тривимірний потік, викликаний наявністю з’єднувального повітропроводу, сприяє запалюванню керосину, чого не можна досягти за однорідного потоку на вході камери згоряння.

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