Aerodynamic Testing at Duplicating Hypersonic Flight Conditions with Hyper-Dragon

Z. Jiang and H. Yu

Abstract In this paper, aerodynamic testing carried out with the JF12 hypervelocity shock tunnel is reported, and three experiments are discussed. The first experiment is aerodynamic heat test in hypersonic boundary layer, and a new thermal sensor for heat flux measurement at the extreme high-temperature condition is described, and the hypersonic boundary physics is explored. The second one is the aerodynamic force testing, and a new concept is proposed for optimizing the force and moment measurement system in shock tunnels. With this technology, the real gas effect is investigated, and its dominating mechanism is considered to be the gas molecular vibration. The last is the scramjet test from which two engine operation models are observed, that is, the continuous and pulsed combustion. Because most of the scramjet tests were conducted in combustion-based test facilities, the experimental data at the duplicated hypersonic flight condition would be of fundamental importance for exploring coupling of combustion, supersonic flow, and shock dynamics.

1 Introduction

Experimental data on aerodynamic performances of flight vehicles are of primary importance for developing new vehicles. Various types of wind tunnels for simulating flight conditions from subsonic to hypersonic speeds have been built up for aerodynamic testing for more than a century. Aerodynamic testing at duplicating hypersonic flight condition still is one of the most challenging research topics that have not been solved so far. It is not only because that hypersonic flight conditions

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are very difficult to duplicate but also because the measurement environment in hypervelocity flows is extremely harsh for both the thermo-sensors and the aerodynamic balances. Theories and technologies on the wind tunnel for duplicating hypersonic flight conditions were reported by Jiang and Yu [1, 2], including how to carry out the reliable aerodynamic testing that was considered as a critical problem in hypersonic research [3, 4], and will be reported and discussed in this paper.

The thermal sensor is the most important device for the aerodynamic heat flux measurement. At hypersonic flight conditions, there are at least three physical issues that have to be considered for thermal sensor designs. The first issue is the high-temperature environment, and the stagnation temperature is about 2250 K for Mach 7 and 4500 K for Mach 10. This environment is very tough for the thermal sensor to survive because the temperatures are very above the melt point for almost all the metals. The second one is the strong shear flow along wall surfaces due to hypersonic flow velocity, which results in an extremely high heat flux that is proportional to the velocity gradient, and the difference between the stagnation and wall temperatures. The large amount of aero-thermal heat that can be calculated from the heat flux multiplying by the test time can melt the sensor surface toward the shear flow in a minute. The last is the surface oxidization of the heat flux sensors, and the big error arising experimental data will be induced for its repetitive usage.

The shock tunnel can be used to generate hypersonic flows with Mach numbers as high as required ones [5], but its application in aerodynamic experiments is limited due to its short test duration. Especially, the aerodynamic force and moment testing has been a challenge for shock tunnel experiments for decades. Several kinds of balances have been proposed so far, but the data accuracy is difficult to improve. There are two physical issues that interfere experimental data accuracy. The first one is the pulsed impact on the test models due to suddenly starting of nozzle flows, and the compelling vibration of the force measurement system is induced and damps as the test flow becomes steady. The other is the inertia vibration that depends on the self-vibration frequencies of the aerodynamic force measurement system that consists of the test model, balance, and model support. For typical shock tunnels, the periods of these kinds of vibrations are as long as the shock tunnel test time, and this results in a difficult problem for accurately extracting the aerodynamic signals from the signal output of the force measurement system. One way to solve the problem is to increase the rigidity of the system, but its signal output can be too weak to be detected and the data accuracy also is low. Therefore, the testing of aerodynamic force and moment at hypersonic flight conditions has been a frontier research topic for decades.

Another important issue for aerodynamic testing is the scramjet test. To simulate the correct combustion process, it is necessary to ensure the combustion to take place at the required aerodynamic environment with proper reaction rates at correct locations. The requirements indicate that the testing should be carried out at true flight conditions with a full-scale model in a proper test duration during which the stable combustion can be reached. So far as we known, most of the scramjet tests were conducted in combustion-based test facilities, and the obtained experimental data lead to better understanding on supersonic combustion, but the contamination effects are difficult to evaluate and important for flight test planning. For exploring coupling of chemical reactions, supersonic flows, and shock wave dynamics, the experimental data at duplicated hypersonic flight condition would be of fundamental importance, and its testing is also a challenging issue.

In this paper, the recent progresses of the experiments with JF12 Hyper-Dragon are summarized, and the relevant experimental data are presented with brief discussions and remarks.

2 JF12 Hyper-Dragon

Aiming at the challenges to evaluate critical hypersonic technologies and explore the aero-thermal physics in hypersonic and high-temperature gas flows, the Institute of Mechanics, CAS, announced their success in developing the long-test-duration hypervelocity detonation-driven shock tunnel named as the JF12 hypervelocity shock tunnel under the serial number in the state dynamics, as shown in Fig. 1. An endearment name was suggested to this facility, and it is the hypervelocity detonation-driven real gas shock tunnel (short for Hyper-Dragon).

This shock tunnel, as shown in Fig. 1, is about 265 m long in total and equipped with a 2.5 m nozzle. The driven section is 85 m in length and 720 mm in diameter. The detonation driver is 120 m in length and 400 mm in diameter. The damping section locates at the left end of the facility and is 19 m in length and 1000 mm in diameter. The vacuum tank at the right end has a 600 m^3 volume and is 50 m in length. The Hyper-Dragon is operated with the backward-running detonation mode. The calibration tests demonstrated that the facility is capable of reproducing the pure airflow with Mach numbers from 5 to 9 at the altitude of 25–50 km with at least 100 ms test duration.

3 Aerodynamic Heating in Hypersonic Boundary Layer

According to the total temperature and test duration of the JF12 hypervelocity shock tunnel, a new thermocouple is proposed based on three physical concepts. The first concept is to make the sensor diameter as small as possible to reduce

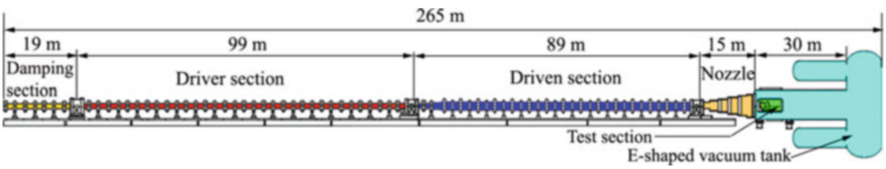

Fig. 1 Schematic of the long-test-duration hypervelocity detonation-driven shock tunnel

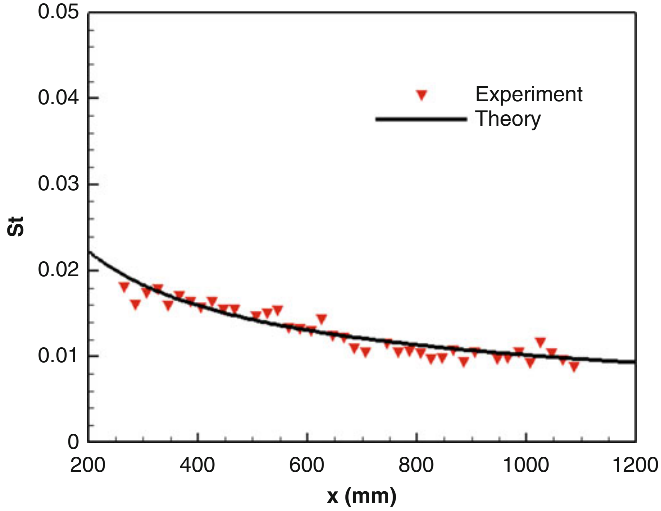

Fig. 2 Comparison between theoretical and experimental results along generatrix of the sharp cone model for Mach 7 with total temperature of 2250 K

the heat transfer to thermal sensors. The second one is to make the isolation layer between two electrodes as thin as possible to increase sensor sensitivity. The last is to carefully choose two electrode metals with similar thermal properties, and this is found very helpful to reduce systematic errors of the thermocouple sensors.

Calibration tests for the new thermocouples were carried out with a 70 sharp cone test model. The new thermocouple is 1.4 mm in diameter, and the test model is 1000 mm in length and 528 mm in diameter. For the total temperature of 2250 K at Mach number 7, the experimental data are presented in Fig. 2. It is observable that all the measurement data fall within the error band of ±7%. This is the very good result from shock tunnel tests in the world because the errors are about 10–20% for conventional shock tunnel measurements.

Another test case is a hypersonic vehicle with complete configuration from engineering. The flight test data demonstrated that there is a special area with very high heat flux, and the severe ablation has taken place. However, before the flight test, both the experimental date from conventional hypersonic wind tunnel and the numerical results from CFD simulations showed there is nothing special in aerodynamic heating.

A full-scale model being 6 m long is accepted for our experiment, and 500 new thermal-sensors are installed in the model. The experiment was carried out at March 7 with total temperature of 2250 K. It is very exciting that the experimental data verified well the finding from flight tests. Further investigation demonstrates a very important phenomenon, that is, the hypersonic boundary layer physics. For a hypersonic vehicle, there is an entropy layer generated from the vehicle nose. The body shock at high March numbers is overlapped together with the boundary layer.

Moreover, the high temperature resulting from viscosity friction induces chemical reactions with energy transfer. Therefore, the hypersonic boundary layer is not scalable in wind tunnel experiments and cannot be predicted with supersonic linear theory. Therefore, the thermal-aerodynamic testing at duplicating hypersonic flight conditions is really necessary, and the hypersonic boundary layer physics needs more attentions.

4 Aerodynamic Forces with Real Gas Effects

In order to improve the technology for force measurements with shock tunnels, a new concept is proposed for developing the aerodynamic force measurement system based on the 100 ms test duration of JF12 hypervelocity shock tunnel. The idea goes as follows: First, the critical frequency is chosen to be the basic one for developing the force measurement system so that two or three perfect periods can occur during the test duration. Then, a test model is designed as large as possible to ensure that the physical process on the model surface is true. And also, the test model must be made as light as possible to reduce the load of balances. And then, the model support rigidity can be increased so that its lowest self-vibration frequencies could be much higher than the critical frequency. Finally, the balance is carefully designed so that its lowest frequency can be two or three times more than the critical one.

The six-component stress balance was specially designed with the new concept for JF12 hypervelocity shock tunnel to insure that the low-frequency vibration of the balance output signs has enough full periods within the 100 ms test duration. This homemade stress balance is named SWPD-S01 under laboratory serial number. The test model for the experiment is selected to be a sharp cone with 10° angle because of its wide acceptance for hypersonic wind tunnel calibrations. The model is measured 1500 mm in length and 57 kg in weight and may be the largest model for shock tunnel tests. By its large, the force signal of the stress balance would be much larger than nozzle flow perturbations, and the data reliability would be improved significantly. The test condition is for Mach 7 at about 35 km altitude with the total temperature of 2200 K. Experiments was carried out with the optimized aerodynamic force and moment measurement system, and the obtained data look quite promising.

The experimental data collected from the SWPD-S01 stress balance at a 5° attack angle are presented in Fig. 3. Voltage signals indicate that the compelling vibration damps very rapidly and the lowest frequency oscillation observable from the pitching moment presents at least three full periodical cycles within the 100 ms test duration. It means that high accuracy of the pitching moment could be reached with this stress balance by applying simple data processing techniques. As to the axial force and normal force, their low-frequency oscillations have more than enough periodical cycles to achieve high accurate data. The periodicity of the force measurement data from the JF12 hypervelocity shock tunnel indicates that its accuracy could be as high as that from blow-down hypersonic wind tunnels.

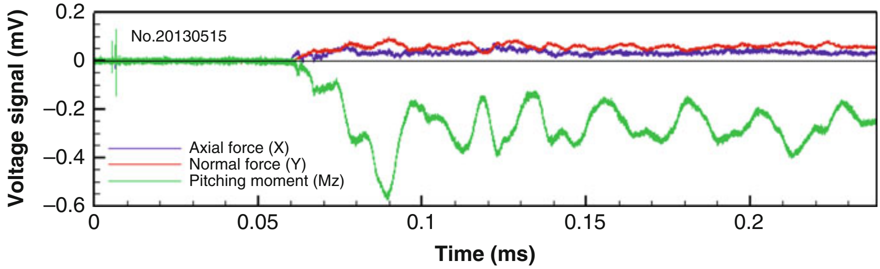

Fig. 3 Balance signals from aerodynamic force and moment tests for Mach number 7

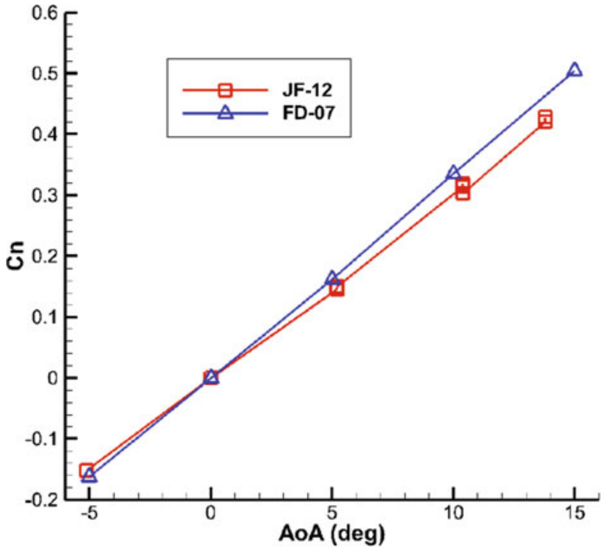

Fig. 4 Experimental data of the normal force coefficient for Mach number 7

A series of the aerodynamic force measurements for Mach 7 were carried out at different attack angles, and the experimental data of the normal force coefficient are plotted in Fig. 4 with the corresponding results from a conventional hypersonic wind tunnel named DF-07. By examining the discrepancy between the data from two wind tunnel tests, it could be concluded that the normal force coefficient of the JF12 hypervelocity shock tunnel is smaller than that from DF-07 hypersonic wind tunnels. The discrepancy is increased as the attack angle is set to be large and large and reaches to about 10% at the 15° attack angle. The axial force coefficient is plotted in Fig. 5, and the curve from the JF12 hypervelocity shock tunnel is located below the one from the DF-07 hypersonic wind tunnel. By considering differences in wind tunnel test conditions, several main physical issues are discussed below for future investigation.

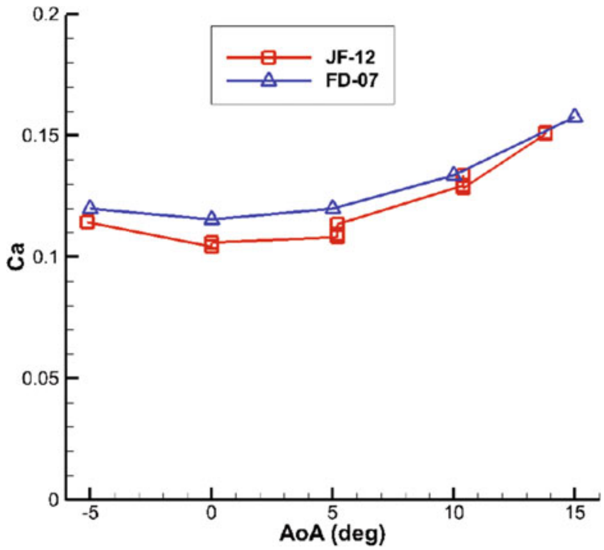

Fig. 5 Experimental data of the axial force coefficient for Mach number 7

The big difference for two wind tunnel test conditions is the total temperature. It is 2200 K for the JF12 hypervelocity shock tunnel and about 800 K for the DF-07 hypersonic wind tunnel. This temperature difference can induce two main physical issues that will affect more or less the aerodynamic forces acted on the sharp cone model. The first issue is the high-temperature boundary development. The boundary layer will get thicker when the temperature is higher. The thicker boundary will reduce the drag force, and this effect is demonstrated in Fig. 5. The other is the pressure variation distributed along the cone model surface, and it will cause differences on both the normal and axial forces. There are two important physical processes that are related with the pressure variations. One of these two is the temperature-induced pressure change in boundary layers, but it could be very small according to the boundary layer theory that says the pressure gradient normal to the solid surface could be assumed to be zero. Furthermore, there is no stagnation area for the sharp cone model, and the stagnation pressure change related to chemical reactions can be ignored. Another process is the post-shock pressure change. The post-shock temperature at the windward side is different from the leeward side if the attack angle is not zero. For the JF12 hypervelocity shock tunnel tests, the post-shock temperature at the windward side can vary from 350 to 700 K when the attack angle varies form 5–15°. Meantime, the post-shock temperature variations at the leeward side are below 400 K. It is known that gas molecular vibrations are excited significantly when the static temperature is raised to above 500 K, and this will result in the constant pressure-specific heat change and the energy transfer from mechanical to internal energy. For the JF12 hypervelocity shock tunnel tests, the gas molecular vibrations exist at the windward side, but not at the leeward side. This physical phenomenon will lead to the pressure unbalance behind the conic shock wave around the cone model and result in the normal force coefficient difference as shown in Fig. 5. The difference appears to be larger and larger as the attack angle increases because the conic shock wave is getting stronger and stronger. This is one kind of real gas effects and was not classified clearly before.

5 Integrated Vehicle Frame/Engine Testing

One of the most challenging tests in hypersonic research is the full-scale scramjet experiment. It is also a difficult problem for the JF12 hypervelocity shock tunnel. During 100 ms test time, one has to complete not only several physical processes such as fuel injection, fuel/air mixing, gas mixture ignition, and stable combustion but also to time the shock tunnel starting, the fuel injection device, and the igniter and measurement system in a correct order. This problem is solved successfully and verified with the experiment in the JF12 hypersonic shock tunnel.

Scramjet tests were conducted with the Performance Test Engine (PTE) model installed into a wave-rider-typed vehicle to receive proper inlet flows, and the installation is shown in Fig. 6. The PTE model consisting of an inlet, a combustor, and an expansion nozzle is 2.2 m long. The inlet is two-dimensional and has two compression surfaces. The combustor is made with two cavities for flame stabilization, and fuel injections are distributed between the cavities for heat release control. The PTE nozzle is two-dimensional and 1.5 m in length. The test flow is for Mach 7 with the total flow temperature of 2200 K. Forty pressure transducers are distributed from the inlet to the nozzle to monitor pressure variations during

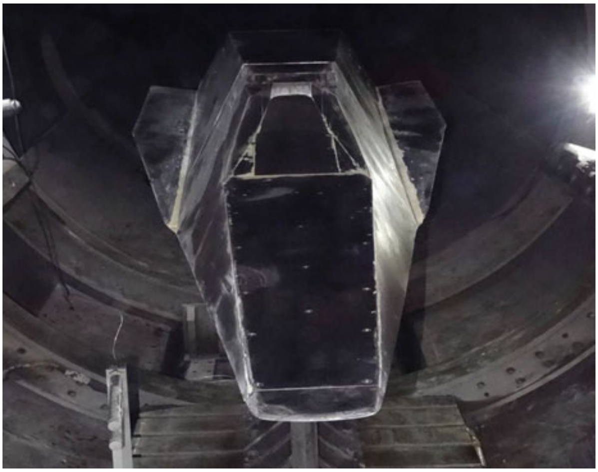

Fig. 6 Performance Test Engine mounted in the JF12 hypervelocity shock tunnel

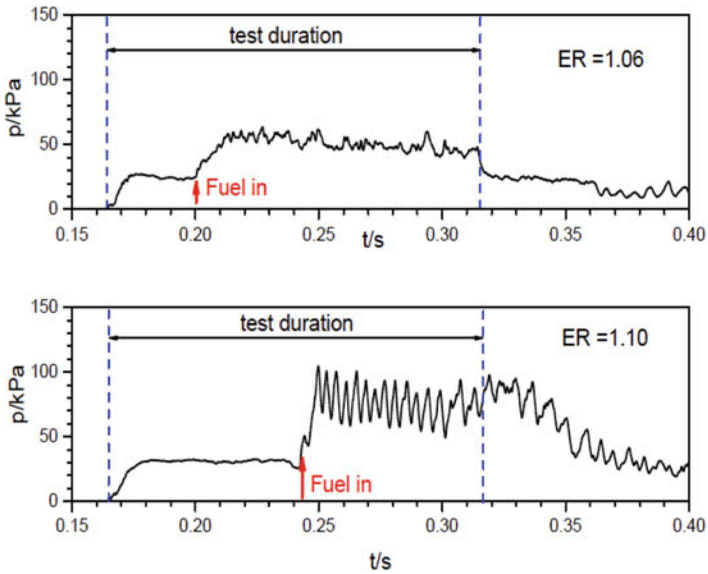

Fig. 7 Pressure variations recorded in combustor: the up half showing stable combustion at ER = 1.06 and the low half showing unstable combustion at ER = 1.10

combustion. Two high-speed cameras are mounted toward test windows to observe high-temperature combustion products from the nozzle and the shock wave structure around the inlet.

Two operation modes for the PTE model are observed from experimental results. One mode is continuous combustion, and the other is pulsed combustion. Typical pressure variations in the PTE mode are plotted in Fig. 7 to demonstrate the two operation modes. From the upper half of the figure, it is observable that the pressure profile is almost flat with pressure perturbations. This experimental data indicate that the continuous combustion maintains in the PTE model even if the flame may be not stable. The periodical pressure oscillations are observed from the low half of the figure, and the oscillatory frequency of the combustion pressure is about 200 Hz. The equivalence ratio (ER) is only the difference between these two test cases. This pulsed combustion phenomenon could be defined as the engine surging and is a critical issue for scramjet engine operation control.

Generally speaking, for a designed scramjet engine, there will be a critical operation point. Once the equivalence ratio is increased over the point, the local temperature downstream will increase and so is the local sound speed. This can induce the upstream-travelling compression waves, and a shock will be generated quickly due to the sharp temperature gradient ahead. The thermal choking takes place in the engine, and the inlet goes unstart, and combustion flame dies out. Photos from two cameras show that the high-temperature gas is not observable when a shock wave is digested from the inlet. After the combustion flame dies out, the inlet restarts and the combustion in the engine ignites again.

6 Concluding Remarks

Theories for developing the advanced ground test facility being capable of duplicating hypersonic flight conditions were validated to be useful by experimental applications reported in the paper. The experimental data at duplicated hypersonic flight conditions show obvious discrepancy from these with cold hypersonic wind tunnels. Three physical issues are shown to be important from this experimental research. The first issue is gas molecular vibrations, one of the key phenomena belonging to real gas effects. The cold wind tunnel tests can be numerically simulated by CFD techniques with acceptable accuracy, but modeling correct thermochemistry in hypervelocity flows is still a challenge. The second one is the hypersonic boundary physics. The entropy layer, shock layer, high-temperature boundary layer development, and its transition interact with each other and result in a nonlinear and multi-physics process that is out of supersonic linear theory. The last is the supersonic combustion. At different aerodynamic environments, the combustion process may be different, especially for flame igniting and extinguishing. These are the key factors for developing scramjet engines; therefore, it is worth further investigations.

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