Theories and technologies for duplicating hypersonic flight conditions for ground testing

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INTRODUCTION
The development of aeronautics and astronautics, from its practical beginning with the Wright brothers’ first airplane on 17 December 1903, has been driven by one primary aim to fly faster, higher and cheaper. There is an exponential increase in both aircraft speed and altitude over the past 100 years, and now the era of the hypersonic flight is approaching [1]. With practical hypersonic airplanes, a two-hour flight to anywhere in the world will be possible and the space-access expense will be cut by 99% with reusable Two-Stage-To-Orbit (TSTO) techniques. The impact of hypersonic flight on the world’s modern society could be revolutionary. Hypersonic flight is, and in the foreseeable future will be, the driver of national security, and civilian transportation and space access. After 50 years of hypersonic research, both manned and unmanned hypersonic flights have been successfully achieved; however, practical hypersonic flight with air-breathing propulsion is still far ahead of us [2–4].

Hypersonic gas dynamics is fundamentally different from subsonic and supersonic ones. Hypersonic flows are usually characterized by the presence of strong shocks and non-equilibrium gas dynamics and chemistry where gas molecular vibrations, O2-dissociations, N2-dissociations, and electronic excitation and ionizations take place as the flight speed increases from 1.5 km/s to 10 km/s. The accurate prediction of non-equilibrium chemically reacting gas flows is a critical issue for the design of any hypersonic vehicle. The ground-testing facility has been the main means for hypersonic vehicle development for decades because of limitation in physically modeling non-equilibrium chemically reacting gas flow, especially, for air-breathing airframe-integrated cruisers. The most difficult problem arising from the hypersonic testing is that the model scaling laws widely applied in subsonic and supersonic gas dynamics are not valid for hypersonic flows in which thermochemical reactions dominate—that is, the reaction scale remains unchanged when the test model is reduced in its size for wind-tunnel tests.

Several types of hypersonic test facilities have been developed so far, such as shock tunnels, air-heated wind tunnels, arc-heated wind tunnels and combustion-based propulsion test facilities. These different facilities are used to address various aspects of the design problems associated with hypersonic flight, and there is no single ground-based facility capable of duplicating the hypersonic flight environment [5]. Nowadays, critical designs of hypersonic vehicles are usually validated with hypersonic flight tests [6]. The flight test is a more reliable tool than others on the ground, but quite expensive and time-consuming for new vehicle development [7]. If one looks back at the last century, supersonic flights had been achieved within 60 years from 1903 to Concorde beginning its commercial service across the Atlantic Ocean in 1967. However, practical hypersonic flight is still under exploration after 50 years of work since 22 September 1963, when Robert M. White flew the North American X-15 at a Mach number of 6.7. There are several technical challenges for hypersonic vehicle development, such as hypersonic propulsion, integrated aerothermal structure and advance ground-testing facilities. The lack of advanced test facilities that can reproduce true hypersonic flight conditions is considered to be the fundamental one. The existing test facilities are inadequate for required technology development, such as propulsion higher than Mach 8, thermal environment identification and large-scale, integrated thermal-structural testing [3–5].

To carry out a reliable ground test at hypersonic Mach numbers, there are several technical challenges for hypersonic vehicle development, such as propulsion higher than Mach 8, thermal environment identification and large-scale, integrated thermal-structural testing [3–5].

To carry out a reliable ground test at hypersonic Mach numbers, there are four key parameters required for hypersonic test facilities. The first parameter is the test gas and it is the key issue to ensure correct thermochemistry in test flows. The second is the total flow pressure and temperature with which the altitude pressure and temperature, and the flight speed can be reproduced with the proper nozzle expansion. Consequently, the chemical-reaction process can be simulated correctly, as well as the aerodynamic forces and momentum. The third is the nozzle size. In other words, the nozzle exit must be big enough to accommodate a large vehicle model so that the chemical-reaction time scale can be much smaller than that of the test gas flowing past the model. The last one is the test duration. For air-breathing hypersonic vehicles, the airframe is also part of the scramjet engine; therefore, the test...
duration has to be long enough to sustain stable supersonic combustion in the engine. Qian estimated the requirement and suggested that a 100-ms test time is sufficient for supersonic combustion research. Actually, there is a balance point for creating the test duration as long as the facility is not seriously damaged because of extremely high total temperature that causes a severe thermal barrier on the hypersonic wind tunnel. It is even more challenging that one has to simulate all these four parameters simultaneously in a single test facility, and it is the reason why there is still no single ground-based facility capable of duplicating the hypersonic flight environment after the 50 years of development of advanced hypersonic test facilities. Over recent decades, different facilities have been developed to address various aspects of the design problems associated with hypersonic flight [5]. This idea works and a lot of experimental data are generated, but uncertainty and low accuracy are imposed on these wind-tunnel test data, resulting in slow development of hypersonic vehicles.

There are several kinds of hypersonic test facilities in the world and these facilities are listed in following, with brief comments. Conventional wind tunnels are operated on the blow-down mode where a high-pressure supply gas is separated from a low-pressure vacuum chamber by a valve. The test gas is expanded through a nozzle into the test section, ending in the damp tank. The test-gas supply must be heated to prevent condensation in the test chamber after expansion. The static temperature of test flows is about 50 K, and these wind tunnels operate under cold at low sonic speeds and do not replicate the correct velocity or enthalpy and, of course, no chemistry either. Electric arc and combustion-heated facilities operate similarly to the conventional wind tunnels except for the way in which the test gas is heated. Duplicating the re-entry heating is possible in arc jets, but this type of facility is limited by the pressure and mass flows because of the need to heat the large volume of gas as it moves through the arc. Its flow contamination and non-uniformity result in its application to ablation and material testing. In combustion-heated facilities, the total temperature is limited to Mach 7 and even lower if a hydrocarbon-based fuel is accepted. These facilities are capable of producing a large volume of hot gases with minutes of test time, but the combustion products such as H2O and CO2 remain in the test flows. Therefore, the chemistry is very different from that in the flight test. Shock tunnels use a moving shock to generate high-pressure test gases. The stronger the moving shock, the higher the test-gas enthalpy. The benefits to shock tunnels are that flight velocities up to Mach 25 can be simulated, but these facilities are limited to a few milliseconds of test time. Fully duplicating hypersonic flight conditions for propulsion testing has remained a top challenge for 60 years since Stalker published his study of the free-piston shock tunnel [8,9]. However, if the true hypersonic flight conditions could be realized with a single ground-test facility, it would be an invaluable tool for hypersonic research, not only for evaluating critical hypersonic technologies, but also for exploring aero-thermal-physics in hypersonic and high-temperature gas flows.

Aiming at the challenge of the large-scale hypersonic test facility, 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 laboratory [10–12]. This shock tunnel is 265 m long and equipped with a 2.5-m-diameter nozzle. The calibration tests demonstrated that the facility is capable of reproducing pure air-flow with Mach numbers from 5 ~ 9 at an altitude of 25 ~ 50 km with at least 100-ms test duration. The AIAA Award Committee comments that the JF12 hypervelocity shock tunnel is the world’s largest shock tunnel capable of true hypersonic flight simulation and the work has advanced the state of the art in large-scale hypersonic test facilities.

THEORIES FOR HIGH-POWER DETONATION DRIVERS

In order to reproduce true hypersonic flight conditions, a high-power driver for shock tunnels is the first critical technology. Considering the test flow for full-scale tests of X-51a or X-43A with air-breathing propulsion at Mach number 9, the flow power required is as high as 600 MW if the nozzle diameter is chosen to be 2.5 m. The driver power must be up to 3000 MW if the wind-tunnel efficiency is assumed to be 20%. By comparing with the free-piston driver, there is 60 times higher than the piston power of 50 MW when the one-ton piston is accelerated to about 300 m/s. This gap is too big for mechanical-energy-supported drivers to surmount.

In 1992, a new detonation driver proposed by Yu was fitted successfully into the high-enthalpy shock tunnel, named TH2-D. This work was largely performed as a joint research project at the Shock Wave Laboratory, RWTH Aachen, Germany, with the Institute of Mechanics, CAS [13,14]. This detonation driver is operated in the backward-detonation mode with a special damptank that works to accommodate the high-energy gases from detonation fronts and reduce the high mechanical loading on the detonation driver after the shock reflection. According to Taylor rarefaction wave theory [15], there is a column of detonated gases with uniform high pressure and temperature generated from the closed end of the detonation tube if the detonation is initiated at the end. The length of this column is approximately half the distance along which the detonation front propagates. The new detonation driver consisting of the driver section and the damptank is a perfect driver for shock tunnels if the main diaphragm is installed at the closed end. In the operation mode, the incident shock wave generated in the driven section propagates in the opposite direction of the detonation front, and it is called the backward-detonation mode. The detonation is the most rapid and violent combustion in the nature, in which the shock heats and compresses gas mixtures in the driver section, thus triggering autoignition, and a balance is attained such that the autoignition supports the shock propagation. In such a process, the detonable fuel is consumed 10^3 to 10^8 times faster than that by a flame. For perspective, this can be

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compared with the solar energy intercepted by the Earth, about \(4 \times 10^{16}\) watts. A 20-square-meter detonation wave, operating at a power level, is equal to all the power the Earth receives from the Sun [16].

To make a high-power detonation driver realizable, the second critical technology is the direct detonation initiation. Experimentally, it is found that, for a given detonable mixture under certain initial conditions, a definite quantity of energy must be used to initiate a detonation instantaneously [17]. From the detonation theory, this means that the strong blast wave generated by a powerful igniter upon the rapid deposition of its energy decays asymptotically to the Chapman-Jouguet (CJ) detonation. If the igniter energy is less than the critical value, the reaction zone progressively decouples from the blast as it decays and a deflagration results. Because the transition from deflagration to detonation under unconfined conditions is extremely difficult, spherical detonation is almost initiated in practice via the blast-initiation mode if even possible. It is the case for the large-diameter detonation driver because the direct ignition is the base for achieving the perfect incident shock wave in the driven section for a backward-detonation driver. To solve this problem, a special initiation tube was developed in the Institute of Mechanics, CAS, based on the concept of multi-energy-amplification. For this concept, an electric spark plug is used to initiate a direct detonation in a small diameter tube, since this is easy to achieve with the blast wave reflection from its tube wall. Then, the detonation in the small tube is used to initiate the second tube with a bigger diameter than the first one, and then the third one. After these multi-ignitions, a certain amount of high-temperature gases is accumulated in the multi ignition detonator and it is powerful enough to trigger a direct detonation within a large-diameter detonation tube. With this new detonator, a high-power shock-tunnel driver is developed based on the backward-detonation mode with a damp tank. On the condition of the hydrogen–oxygen mixture at the initial pressure of 35 MPa and ambient temperature, the 45 GW power output is achieved by the high-power detonation driver that is of a 400 mm diameter and 80 m in length. Equipped with the critical diaphragm technology, the high-power detonation driver is ready for the large shock tunnel to reproduce hypersonic flows at required flight conditions.

**NEW CONCEPTS FOR LONG TEST DURATION OF SHOCK TUNNELS**

After more than 60 years, high-enthalpy facilities suitable for aero-thermal-dynamics are still mostly based on the shock-tunnel principle [5]. Shock tunnels create very high temperature and pressure in the nozzle plenum and flight velocities up to Mach 20 can be simulated for aerodynamic testing of chemically reacting flows. However, this application is limited due to milliseconds of its test duration. Holden commented that typically shock tunnels are used for cold-wall aero-thermal measurements under hypersonic conditions but, with the increased test time, it is possible to test air-breathing propulsion systems, rocket and plume interactions, aerodynamic-force and moment testing, and even dynamic testing [18]. The test duration is about two to three milliseconds for the HIEST, which is the largest free-piston-driven high-enthalpy shock tunnel in the world [19], and about 30 ms for LENS II, which is the best hypervelocity wind tunnel operated with a heat-light gas driver, the so-called Large Energy National Shock Tunnel [20]. LENS II has a 1.55-m nozzle for Mach numbers from 4 to 7 and is inadequate for required M > 8 propulsion development. Actually, for testing of full-scale X-51a with a scramjet engine, the required test time must be much longer and the required nozzle diameter must be even bigger.

There are several physical issues responsible for the short test time of shock tunnels and at least the following three issues are critical. The first physical issue is the wave reflection from the interface that separates the driver gas from the driven gas, as shown in Fig. 1. If the reflected incident shock reflects again, the total pressure and temperature of the test gas in the region 5 will be changed during the nozzle-operating process, resulting in low quality of the test flows and early termination of the test duration. The shock reflection on the gas interface depends on the sound-speed change across the interface. In order to avoid the interface wave reflection, the sound speed of the detonated gas products must meet a certain requirement to match the sound speed of the twice-compressed test gases. This is the so-called tailored-interface condition for reflected-type shock tunnels. For the backward-detonation driver, the tailored-interface condition was derived by Li et al. [21] and shows the way to control the interface wave reflection, for both shock and rarefaction waves.

**Figure 1.** Physical issues for developing a long-test-duration shock tunnel.
For most of aerodynamic experiments, air is used as the test gas at the ambient temperature and the incident shock Mach number is determined with the required total temperature from the normal shock relations. The initial pressure in the driven section is determined with the required total pressure at the corresponding flight condition. Therefore, the sound-speed ratio can be adjusted only by selecting different detonable gas mixtures because of the given state of the test gas in the front of the interface. To produce a flow at Mach 5, the oxygen–hydrogen mixture is too powerful to use and nitrogen has to be added to reduce its energy release per unit mass. Doing so results in the sound-speed drop of the detonated products, and it agrees well with the low-sound-speed requirement and leads to the tailored-interface condition for detonation-driven shock tunnels. When simulating the flow at high Mach numbers, the sound speed of the compressed test gas increases because of the high total temperature, and the required sound-speed increase of the detonated gases could be achieved by reducing the nitrogen fraction. Adjusting the nitrogen fraction in the oxygen–hydrogen mixture can lead to two effects. The first is to drive out a stronger incident shock wave to meet a higher Mach number requirement from flow generation. The other is to generate a higher temperature in the detonation products to match the faster-sound-speed requirement from the tailored-interface operation condition. Generally speaking, from our experimental data, the acetylene–oxygen gas mixture is suitable for the low Mach number of the hypersonic regime and the hydrogen–oxygen gas mixture is good for the high-Mach-number range.

The second physical issue is the shock–boundary interaction that takes place near the end of the driven section. The boundary layer develops while the incident shock propagates in the driven section to its end where the nozzle is attached. There is a thin diaphragm between the driven section and the nozzle so that the second shock compression is achieved to generate the required total pressure and temperature. The shock–boundary interaction occurs after the incident shock reflects back from the diaphragm. The test gas in the boundary layer flows back and the other part in the central area remains stationary. This flow-field structure will induce the driver gas to go along the boundary layer to the end of the driven section, as shown in Fig. 2. The flow motion makes the driver gas enter the nozzle throat much earlier than expected and finally leave a relatively large percentage of the test gas behind. The physical issue is called test-gas contamination and can reduce greatly the effective test duration of shock tunnels. To avoid fast-entering of the driver gas into the nozzle throat, a cylindrical steel ring is suggested to add around the nozzle throat and its outer diameter reaches to the boundary of the shock-induced boundary layer to block the way of the driver-gas motion. Numerical simulations have demonstrated that the test duration could be increased by 60% due to the application of the new concept [22].

The last issue related to shock-tunnel test time is precursor shock damping in the vacuum tank. The shock wave comes from the nozzle flow starting after nozzle-diaphragm ruptures and propagates towards the test section after it reflects back from the end wall of the vacuum tank that is used to ensure a sufficiently low back pressure for the hypersonic nozzle flow and maintain the static pressure at the given flight altitude. The shock speed is higher than the sound speed in the ambient air and reaches to about 400–500 m/s or more. To avoid its effect on the test-flow region, the length of the vacuum tank must be longer than the distance along which the shock wave propagates during the test duration. The shock-tunnel test duration is usually proportional to its length because the upstream running wave will come back quickly after it reflects from the other end of the driver section, and this is the reason why the test time for shock tunnels is short—usually a few milliseconds. In such a case, the several-meter vacuum tank is long enough for the precursor shock to go forward and backward during testing for conventional shock tunnels and it is not a problem at all. However, if one wants to develop a shock tunnel with a 100-ms test duration, the length of the vacuum tank will become a serious problem for its engineering design. For the JF12 hypervelocity shock tunnel, it is required to provide the distance for the precursor shock to propagate for about 250 ms to ensure 100-ms test time. Assume the precursor shock speed is 500 m/s; the length of the vacuum tank is as long as 62.5 m.
It is not realizable in engineering design for a vacuum tank of a 3.5-m diameter because the tank stiffness will get weaker and weaker as its length becomes longer and longer. Therefore, the length reduction of the vacuum tank becomes a meaningful work to be considered for large-shock-tunnel development.

Several different configurations based on the shock dissipation of the vacuum tank were numerically simulated, and two of these configurations are presented in Fig. 3. The figure in the upper shows the pressure distribution in the E-shaped vacuum tank and the escaped time is 172 ms calculated from the beginning of the nozzle starting process. The low half of the figure shows the pressure distribution in the T-shaped vacuum tank and the escaped time is 159 ms. The E-shaped vacuum tank demonstrates better performance. For such a aerodynamic design, a 250-ms delay time is achieved for a 30-m-long vacuum tank. The physical process occurs as follows: the precursor shock reflects from the wedge installed at the end of the E-shaped vacuum tank and propagates in the lateral direction to additional tanks. This will lead to the gas flow behind the shock wave moving into the additional tanks that are part of the E-shaped tank and results in strong shock dissipation. Therefore, the precursor shock wave gets weaker and weaker as it propagates in the vacuum tank and its disturbances on the test-flow region are delayed. And also the tank design becomes realizable in engineering and a lot of laboratory space is saved.

CHALLENGES FROM HIGH-ENTHALPY FLOW TESTING

Typically, shock tunnels are mainly used for cold-wall aero-thermal measurement in hypersonic testing; however, it becomes a formidable task to carry out a true hypersonic flight simulation for long testing duration. There are at least three physical issues that have to be considered for thermal-sensor designs. The first issue is the high-temperature environment; the stagnation temperature is about 2250 K for Mach 7 and 4500 K for Mach 10, and this environment is very tough for thermal sensors to survive. The second one is the strong shear flow at 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 resulting from the heat flux multiplying by the test time can melt the sensor surface towards the shear flow. The last is the surface oxidation of the heat-flux sensors and the large error in experimental data may be induced for repetitive usage.

According to the total temperature and the test duration of the JF12 hypervelocity shock tunnel, a new type of thermal couples is proposed based on three physical concepts. The first concept is to make the sensor diameter as small as possible to reduce heat transfer to thermal sensors. The second is to make the isolation layer between two electrodes as thin as possible to increase sensor sensitivity. The last is to choose carefully two electrode metals with similar thermal properties, and this is found to be very helpful in reducing the systematic errors of the thermocouple sensor. Calibration tests were carried out for the new type of thermocouples with a 7° 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 7, the experimental data are presented in Fig. 4. From the figure, it is observable that all the measurement data fall within the error band of ±7%. This is the best result from shock-tunnel tests in the world because the errors are about 10%–20% for conventional shock-tunnel measurements. Regarding the significant progress in China, Kremeyer, in the annual review of Aerospace America in 2015 from the Hypersonic Technologies and Aerospace Planes Program Committee, evaluated that the Chinese Academy of Sciences’ Institute of Mechanics finished calibrating its 3.5-m-diameter J12 hypervelocity wind tunnel up to Mach 9, with heat-transfer measurements and simulations yielding accurate results for external aerodynamics and supersonic.
combustion trains. After multiple Wu-14 hypersonic glide vehicle re-entry flights, China revealed its new Mach 10–12 DF-21D missile [23].

The shock tunnel can be used to generate hypersonic flows with a Mach number as high as required, but its application in aerodynamic experiments is limited due to its short test duration. Especially, 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 with experimental data accuracy. The first one is the pulsed impact on the test models due to the sudden starting of nozzle flows and the compelling vibration of the force-measurement system is induced and damps quickly as the test flow becomes steady. The other is the inertia vibration that depends on self-vibration frequencies of the aerodynamic-force measurement system consisting of the test model, balance and model support. For typical shock tunnels, the periods of these kinds of vibrations are as long as the test time and this results in a difficult problem for accurately extracting the aerodynamic signals from the signal output of the measurement system. One way to solve the problem is to increase the rigidity of the measurement system, but its signal output can be too weak to be detected and the data accuracy also is low.

Based on the 100-ms test duration of the JF12 hypervelocity shock tunnel, a new concept is proposed for developing the aerodynamic-force measurement system. The idea goes as follows: first, the critical frequency is chosen to be the basic one for 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 make sure that the thermophysical process 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 equal to the critical one. In this way, the aerodynamic-force measurement with the JF12 hypervelocity shock tunnel demonstrated that the concept works perfectly, and the data accuracy reaches that of the blown-down type of conventional hypersonic wind tunnels. The experimental results are presented in Fig. 5 for reference. From the voltage signal of the pitching moment, the period of the compelling vibration lasts for about 40 ms and the inertia vibrations follow; three entire cycles are obtained within about 100 ms [24]. Therefore, it becomes easy to get high-accuracy data from the inertia vibration periods with simple data processing, but very difficult from the compelling vibration period. Unfortunately, the test time for all the conventional shock tunnels falls in about 2–40 ms.

### CONCLUDING REMARKS

Successful development of the Long-test-duration Hypervelocity Detonation-driven Shock Tunnel was demonstrated with its performance tests and applications. Theories for developing the advanced ground-test facility being capable of duplicating hypersonic flight conditions were validated by the application of several critical techniques proposed in this paper, such as the high-power detonation driver, the tailored-interface condition for detonation-driven shock tunnels, the shock–boundary interaction control device at the end of the driven section and the E-shaped aerodynamic configuration of the vacuum tank. The duplicated flight conditions for Mach 5–9 at altitudes of 25–50 km is the most important part of the trajectories for various spacecraft because the air-breathing hypersonic vehicle flying within is the most important and very challenging research area. The 2.5-m-diameter nozzle flow with more than 100-ms test duration can be generated at a reasonable operation expense and makes the JF12 hypervelocity shock tunnel a powerful platform for testing full-scale hypersonic propulsion and integrated vehicles/engines under duplicated hypersonic flight conditions. Theories and technology for JF12 hypervelocity shock tunnels have advanced the state of the art in large-scale hypersonic facilities and, as such, the advanced wind tunnels could be in the class of national instruments and equipment for scientists to push further the frontier of aeronautics and astronautics. Technologies developed for aero-thermal heat flux, and aerodynamic-force and moment measurements enable carrying out conventional experiments with the JF12 hypervelocity shock tunnel and this is the typical aerodynamic research area. The development of the optical and intrusive flow diagnostic techniques will make the shock tunnel a non-substitutable tool for investigating fundamental physics in hypersonic and high-temperature gas flows. Any progress in this discipline is a primary driver for advancing hypersonic vehicles.

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