Hypersonic Ground-Based Experimentation and Test Technique
Arc Heating Facility and Test Technique for Planetary Entry Missions

By

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Abstract: A 1-MW segmented-type arc heater has been designed and installed in the ISAS high enthalpy flow facility for the purpose of basic study of aerothermophysics and the development of thermal protection materials for the atmospheric hypersonic vehicles. The aerothermophysical flight environment for the vehicles, generally speaking, can not be duplicated in the ground facility; In most cases of vehicles reentering with super-orbital velocity, the flow enthalpy of the ground facility submits to be far below the requirement from the flight environment. In this meaning, the simulated flow environment needs to be well-characterized for heating tests in addition to understanding of its effect on the thermal behavior of the thermal protection materials. The facility has successively conducted heating tests of the thermal protection materials used for reentry vehicles. This paper describes the performance and flow characterization of the ISAS high enthalpy flow facility and its activities in relation to the thermal protection material tests and testing technique.

1. INTRODUCTION

The Institute of Space and Astronautical Science (ISAS) has an asteroid sample return mission named MUSES-C, which is planned to be launched in 2003. In the final phase of the MUSES-C, a small capsule with asteroid sample will conduct reentry flight directly from interplanetary transfer orbit (ISAS, 2001). High enthalpy flow facility is indispensable not only to development of thermal protection material but also to functional verification of components of the thermal protection systems.

The aerodynamic heating load the capsules will encounter in these high velocity reentry missions is estimated to be beyond 10 MW/m². Due to the high heat load environment, the carbon phenolic ablator was adapted for the thermal protection material of the reentry capsules described above. Thermal behavior of the ablator is strongly related to the flow enthalpy; Because the ablation effect, the recession rate of the ablator is a function of the flow enthalpy as described in the literature (Yamada et. al., 2002), duplication of the enthalpy with enough duration up to several minutes is of prime importance for the specification of the high enthalpy flow simulator. Although no other high enthalpy flow generator except arc

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heaters can offer enough heating duration, the enthalpy level of the arc heater is generally below flight environment. The numerical analysis is an important tool for bridging the discrepancy between the flight environment and the simulated one. The role of the arc heater as a ground high enthalpy flow generator contributes to the basic data acquisition for code-tuning and verification of the numerical analysis code. The simulation environment and the heating tests thus need to be carefully designed and conducted taking account of the thermal behavior of the thermal protection material.

Formerly, ISAS had a high enthalpy flow facility where the Huels-type arc heater was installed mainly for the development of the thermal protection materials for the reentry vehicles from low earth orbit (Hinada et. al., 1996). Though Huels type arc heater is simple in its structure and easy for maintenance, the enthalpy level is up to 10 MJ/kg in realistic operational condition, which is far low in comparison with the flight environment of the reentry vehicles with superorbital velocity. Thus, 1-MW segmented-type arc heater was newly designed and installed in the ISAS high enthalpy flow facility enlarging the operational enthalpy envelope for reentry with superorbital velocity. The facility has continually conducted heating tests for the purpose of the development and data acquisition of thermal protection materials and of the verification of the functional safety of the TPS components. The facility was also used for basic study of aerothermophysics by means of optical diagnostics (Yamada et. al., 2002). This paper describes the performance and flow characterization of the ISAS high enthalpy flow facility in relation to the thermal protection material tests and measurement technique of parameters such as heat flux, internal temperature of the test piece.

2. ARC WIND TUNNEL FACILITY

The arc wind tunnel facility has been built for the purpose of developing the thermal protection material and basic study of the aerodynamic feature of the reentering bodies. Fig. 1 shows a block diagram of ISAS arc wind tunnel facility subsystems; The facility is divided into 6 major subsystems; 1) arc heater, 2) power supply system, 3) gas supply system, 4) exhaust system, 5) cooling water system, and 6) measurement system. The facility has two sets of arc heaters; A segmented-disk type arc heater was newly installed in 1996 in addition to the Huels-type arc heater. Although common subsystems are so designed that heaters are exchangeable in accordance with the required enthalpy and pressure level, the segmented-disk type is now primarily in use. It is because enthalpy range of the segmented-disk type covers that of Huels by changing its disk configuration. The specification of the Huels type heater is documented in the literature (Hinada et. al., 1996).

**Segmented-type Arc Heater**

The specification of the segmented-disk type heater is shown in Table 1. The independent operation parameters are the plenum pressure and the discharge current, while the discharge voltage, accordingly the total input power, and the mass flow rate are automatically determined with fixed hardware configuration such as the number of disk packs, and the nozzle throat diameter. Because the Mach number for example should fall into of secondary priority in comparison with the heat flux from the standpoint of the issue to be tested, the smallest nozzle with the exit diameter of 25 mm is often used to obtain higher heat flux.

ISAS segmented-disk type arc heater consists of the rear anode part, the converger part, the constrictor disk parts, the diverger part, and the front cathode part as shown in Fig. 2.
Each disk is made of oxygen-free copper, and is electrically insulated each other by ceramic heatshield insulators made of boron nitride. The working gas, normally air, is supplied into each segmented disk through gas supply tubing connected to the mixing bottle which damps pulsatory supply. And the gas enters the boa inside through 4 swirl injection ports connected to the slot-gap along the ceramic heat shield. The arc discharge is stabilized between the two electrodes located at the end of converger/diverger part respectively where the flow decelerated by area expansion effect takes not a small part in stabilization of the arc. In order to minimize the electrode erosion due to the current concentration, an axial magnetic field is applied to the arcfoot by a magnet coil built inside each electrode segment. The arcfoot rotates in radial plane under the influence of the Lorenz force. Each disk is independently water-cooled by demineralized water with conductivity below 20 \( \mu \text{S} \). Constrictor disks are arranged together into 4 constrictor disk packs with 14 constrictor disks per pack. The number of installed constrictor packs can be changed in accordance with the required enthalpy level as will be described in the following section. Since the segmented-disk type arc heater is fixed-arc length type heater, the variable range of discharge current is limited by the arc stability in a given disk configuration.

**Radiative Heating Simulation**

In case of reentry with hyperbolic or superorbital velocity, namely beyond 10 km/s, such as the MUSES-C asteroid sample return capsule, radiative heat flux is estimated to be 3 MW/m\(^2\).
while the maximum convective heat flux is 15 MW/m². The new ISAS arc wind tunnel can simulate the radiative heat flux up to 5 MW/m² on the area of 1 cm² by means of 500W CW (continuous wave) YAG laser system with wavelength of 1064nm. Intensity distribution in the irradiation area is homogenized by an optical rectifier about 30 cm from the test piece surface. The laser radiation is lead to the rectifier by the optical fiber and directly inserted into the test chamber passing through quartz window.
Facility Subsystems

The power supply is switching regulator type that converts 2500 VAC max. current to DC. Three rectifiers are connected in series to this circuit to diminish the current ripple.

The working gas heated and expanded into the test section is first pressure-recovered by diffuser and is evacuated through the heat exchanger by vacuum pumps. After slightly pressurized by the exhaust fan, the exhaust gas is lead to the air cleaner to remove toxic nitric oxides, and finally ejected to the atmosphere. Vacuum pump system is composed of 3 mechanical booster pumps and 2 oil rotary pumps. This evacuation system enables to dump out the working gas flow rate of 20 g/s keeping the test cabin pressure as low as 0.3 Torr.

There are two water-cooling loops in the facility: One is a closed-loop demineralized cooling water (DCW) line and the other is open-looped low pressure (L/P) cooling water line. The closed-loop DCW line is used for primary cooling of the arc-heater. In order to keep the desired electrical resistance between the electrodes of the heater, the conductivity of the cooling water is usually maintained below 10 μS. When the conductivity exceeds the upper limit of 20 μS, the demineralizer supply new DCW into the line.

The arcjet is expanded through the nozzle into the test chamber. The test chamber is 1.5 m in diameter by 1.5 m long. Entire wall of the cabin is water-cooled by L/P cooling water line. For the optical measurement purposes, seven major windows toward the test section are installed by taking into account of various application for diagnostics purposes. Since 300 mm side windows can cover all the expanded plume, absorption spectroscopy with Abel’s inversion can be easily conducted. In order to measure the surface temperature of the test piece by a pyrometer or to radiatively heat up the surface by YAG laser, front surface can be seen directly though the front window at the sight angle of 53 deg.

Model Injection System

Duration required for heating tests of thermal protection materials such as heat shield especially used for the ballistic entry probes is 30 to 60 sec. Mainly due to degradation of the electrical resistance between each segmented-disk, the diverger and the converger, maintenance labor is required for the next test run after several operational runs. For efficient operation of the arc heater, it is desirable to conduct several tests in one operation run. Reciprocating motion table is installed in the test cabin that enables heating test of five test pieces including heat flux sensor etc. in one operation. Test piece can be inserted into the arc jet within 1 second of transient motion. While a test piece is under heated in the arcjet, temperature rise of the next piece in the stand-by position is about 10°C. In most case, this temperature rise can be negligible as long as it is recognized as initial bias, otherwise, in a case where delicate measurement is required, one test piece for one operation run is conducted.

Heater Operation and Control

The segmented-type arc heater is ignited by high-voltage breakdown with argon gas filled in the arc chamber. Within one second after the arc break-down, the argon gas supply is terminated and followed by the air toward main discharge establishment. The discharge current and the plenum pressure are set to the target operational condition that realizes required heating environment manually in several seconds. When the heater reaches thermally equilibrium state, normally 30 sec is required judging from temperature rise of the cooling water, the heat flux sensor and impact pressure tubes are injected into the arc jet to confirm the heating
environment. After having accomplished the target condition, the test model is injected into the test section.

3. OPERATION ENVELOPE

Operational Characteristics

Independent operational control parameters are the discharge current and plenum pressure; the discharge voltage is determined spontaneously on the basis of conductivity of the high enthalpy gas and the arc length. The operation envelope is shown in Fig. 3. The envelope is not maximum envelope in the exact literal meaning of it, but is realistic normal operation envelope in which loads on all systems is relatively light. The lower current boundary is determined by the arc instability due to power supply system and the boundary in the higher current is due to electrode erosion limit.

Figure 4 shows thermal efficiency, defined as ratio of the total flow enthalpy to the input power, vs. gas enthalpy characteristics with parameter of the plenum pressure or arc chamber pressure. It is recognized that the thermal efficiency is higher with higher plenum pressure. At a given plenum pressure, the thermal efficiency decreases with increasing the specific enthalpy. Since the convective heat transfer to the wall is proportional to the temperature gradient at the boundary, the increase of the enthalpy enhance the thermal dissipation at the wall.

The bulk enthalpy of the arc jet is measured by the energy balance method.

\[
h = \frac{I \cdot V}{\bar{m}} \eta \approx \frac{I \cdot V - \sum C \cdot F_i \cdot \Delta T_i}{\bar{m}}
\]

where \( h \) is the bulk enthalpy, \( \eta \): thermal efficiency, \( \bar{m} \): mass flow rate, \( I \): discharge current, \( V \): discharge voltage, \( C \): specific heat of the water, \( \Delta T_i \): temperature rise due to operation of \( i \)-th cooling water supply tubing, and \( F_i \): flow rate. Although temperature rise due to the flow
Fig. 4: Thermal Efficiency vs. Discharge Current Characteristics.

Fig. 5: A schematic View of 3 operational configuration of the arc heater.
friction in the supply tubing is not negligible, this effect can be eliminated by measuring the temperatures soon before and after the corresponding portion of the heater.

*Optional 1-Pack Operation*

The enthalpy for simulating rocket flight environment is generally much lower than that for reentry vehicles. Heating tests with excessive high enthalpy flow make the surface phenomena of the thermal protection material different from that in real flight. Simulating enthalpy level is of great importance for appropriate heating tests.

ISAS arc heater can be operated with various number of segment disk packs in accordance with the required enthalpy and the heat flux, impact pressure as shown in Fig. 5. The configuration of 1-pack operation consists of 1 segment pack installed between the diverger and converger. The flow enthalpy in 1 or 2-pack operation ranges 7 to 10 MJ/kg; This is half of the enthalpy in normal 4-packs operation, because the arc heating length is reduced. In order to obtain lower enthalpy without arc instability, it is good to install the cooling extension pack downstream of the front (cathode) electrode, which reduces the gas enthalpy down to 3 MJ/kg. The configuration of 1 constrictor pack with the cooling extension is appropriate for heating test of TPS material for sounding rocket; It is true that the configuration fails to simulate maximum impact pressure but can simulate the impact pressure at maximum heat load.

4. **SIMULATION ENVIRONMENT**

For designing of the thermal protection system (TPS) of the hypersonic vehicles, TPS material data and its thermal behavior need to be obtained in the ground high enthalpy facility; Carbon phenolic ablator is used for the heat shield of the MUSES-C capsule. For example, ablation blocking effect is empirically expressed as a function of the flow enthalpy, cold wall heat flux, surface temperature, and pyrolysis and reacting gas mass flow rate from the ablator surface. Thus, for the development work of the thermal protection material these parameters need to be measured in addition to the flow enthalpy for each heating test. The measurement of the bulk flow enthalpy is described in the previous section.

![Diagram](attachment:diagram.png)

*Fig. 6: A schematic View of Heat φ50 mm diam. Flux Probe (Gardon Gauge Installed).*
Fig. 7: Impact Pressure vs. Distance from the Nozzle Exit Characteristics.

Fig. 8: Heat Flux vs. Distance from the Nozzle Exit Characteristics.
Convective Heat Flux

Cold wall heat flux is measured by circular foil water-cooled heat flux sensor (Gardon gauge) as shown in Fig. 6. The sensing foil is connected at its perimeter to a heat sink having a thermoelectric potential different from that of the foil material, thus forming thermocouple junction. When the sensor is exposed to a heat source, thermal equilibrium is rapidly established, and the equilibrium thermoelectric potential between center and edge of the foil changes in proportion to the heat flux. The duration of the injection of the heat flux probe in the arc jet flow is 1 second which is enough for thermoelectric equilibrium of the constantan center foil. Cold wall convective heat flux is expressed as

\[ q = C \cdot \frac{h \cdot \sqrt{P_{02}}}{\sqrt{R_n}} \]

where \( P_{02} \) is the impact pressure, \( R_n \) the effective nose radius of the test model and \( h \) the specific enthalpy of the flow. In order to attain high heat flux, small diameter of the test piece, higher enthalpy, and higher impact pressure is required. Thermal analysis of the heat shield material is usually conducted under the assumption of one-dimensional heat conduction problem. The minimum diameter of the test piece is determined comparing the heat penetration depth or decomposition depth of the material with the test piece radius. About \( 25 \) mm diameter of test piece is minimum from the standpoint of the assumption of one-dimensional heat conduction. Since the variation of the enthalpy as for the discharge current is relatively small as was already shown in Fig. 3. The variation of the impact pressure as for the distance

Fig. 9: Simulation Environment Summary.
from the nozzle exit is large as shown in Fig. 7. Thus the heat flux varies along distance from the nozzle exit. As shown in Fig. 8, the heat flux on φ 50 mm diam. flat face test piece ranges from 1.0 MW/m² at 200 mm from the nozzle exit to 6 MW/m² at 80 mm in 4-pack configuration. Heat flux of 12 MW/m² can be obtained with φ 25 mm diam test piece at 25 mm from the nozzle. The simulation environment is summarized in Table 2. By installing the gordon gauge on the flat plate, heat flux characteristics on the flat plate is also obtained respectively in 1-pack and 4-pack configuration as shown in Table 3.

Operation envelope from the stand point of the enthalpy level and dynamic pressure is summarized and shown in relation to the mission environmental requirement in Fig. 9.

Table 2: Heating Environment of 4-pack Configuration.

<table>
<thead>
<tr>
<th>Flow Enthalpy (MJ/kg)</th>
<th>14 - 20 MJ/kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Heat Flux on φ 25 flat face</td>
<td>-13 MW/m²</td>
</tr>
<tr>
<td>on φ 50 flat face</td>
<td>1 - 6 MW/m²</td>
</tr>
<tr>
<td>Impact Pressure</td>
<td>0.05 - 0.7 kg/cm²</td>
</tr>
</tbody>
</table>

Table 3: Heating Environment for Flat Plate.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>4-Pack</th>
<th>1-Pack</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flow Velocity (km/s)</td>
<td>5.5</td>
<td>2.6</td>
</tr>
<tr>
<td>Flow Enthalpy (MJ/kg)</td>
<td>15</td>
<td>3.5</td>
</tr>
<tr>
<td>Recovery Temp (K)</td>
<td>8000</td>
<td>3000</td>
</tr>
<tr>
<td>Heat Flux (kW/m²)</td>
<td>300</td>
<td>23</td>
</tr>
<tr>
<td>Dynamic Pressure (kg/cm²)</td>
<td>0.4</td>
<td>0.06</td>
</tr>
</tbody>
</table>

Optical Diagnostics

Conventional sensors are of great use, it is true, but more sophisticated diagnostics, non intrusive optical diagnostics is advantageous for further understanding of high enthalpy flow. Laser diagnostics system has been established in ISAS high enthalpy flow facility as shown in Table 4.

By using the previously described LIF (laser induced fluorescence) spectroscopy instrumentation, a Doppler shift measurement of these LIF signal gives and ideal non intrusive velocimetry. Figure 10 is a typical example of 2D image of the expanded flow out of the arc heater in which fluorescence form copper atoms of 578 nm was detected and processed. The measurement was made by laser wave length of 327 nm by excimer-pumped dye laser and second harmonic generator. This result demonstrates the velocity distribution and the uniformity of the expanded flow out of the nozzle. The line which crosses the vector is virtual extension
line of the nozzle wall. It is recognized that the flow direction near the boundary is along this virtual nozzle extension line. The velocity along the radial distance in the flow as shown as hatched line in the upper side of Fig. 10. These results show that the difference of the velocity at the center and at the edge of the arc jet is about 85%. Then the flow uniformity is satisfied as long as the diameter of the model is as large as 30 to 50 mm. The velocity measured by LIF and those calculated by corresponding non equilibrium nozzle flow was compared with good agreement. Roughly 80 to 90% of the bulk enthalpy turned out to be converted into the flow kinetic energy.

Nonequilibrium temperatures of nitric oxide has been measured experimentally by multiline LIF thermometry. NO molecules in a given rotational and vibrational state in γ-band has been selectively excited by laser irradiation. Since the relative intensity of LIF signals are so selected that LIF intensity is proportional to the population distribution of selectively-excited molecules in a given different energy state, nonequilibrium temperatures (rotational and vibrational) can be deduced independently by comparing the intensity under the assumption that Boltzmann distribution is applicable to each energy state. Figure 11 shows a typical temperature measurement result in an operation condition with enthalpy of 10 MJ/kg, where the measured vibrational temperature is about 1000 K higher than rotational temperature by 300 K.

Table 4: Laser Diagnostics System Specification.

<table>
<thead>
<tr>
<th>Excimer Laser (Lumonics PM-882)</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Laser Medium</td>
<td>XeCl 308nm</td>
</tr>
<tr>
<td>Band Width</td>
<td>1 nm @300nm</td>
</tr>
<tr>
<td>Pulse Width</td>
<td>20 ns</td>
</tr>
<tr>
<td>Laser Energy</td>
<td>600 mJ /Pulse</td>
</tr>
<tr>
<td>Repetition Frequency</td>
<td>10 Hz</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Dye Laser (Lumonics HD-500S)</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Wavelength</td>
<td>300 - 800 nm</td>
</tr>
<tr>
<td>Band Width</td>
<td>0.04 cm⁻¹ @300nm</td>
</tr>
<tr>
<td>Conversion Efficiency</td>
<td>10 - 20 %</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>ICCD Camera (Hamamatsu C-4077)</th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Gate Width</td>
<td>7ns - 14 ms</td>
</tr>
<tr>
<td>Wavelength Range</td>
<td>160 - 850 nm</td>
</tr>
</tbody>
</table>

5. HEATING TEST TECHNIQUE

Test Piece Attachment

The ablator heating tests as shown in Fig. 12 have mainly two aspects; One is to obtain time profile and distribution of the internal temperature for use of tuning or verification of numerical ablation analysis codes. The other is to demonstrate or verify functional safety of thermal protection components.

Because the size of the test piece is limited less than 50 mm in diameter, temperature data acquisition needs to be quasi-one dimensional for use as data for numerical analysis. In
order to make the quasi one-dimensional assumption valid, the heating on the side wall needs to be eliminated or be decreased down to trivial level in comparison with the heat quantity conducted inside. In most cases, the heat flux on the side is relatively small enough that the above assumption is valid; The side-wall heating rate is about 100 to 200 kW/m² when the stagnation heat flux is about 6 MW/m² for example. The bakelite side-wall heating protection
is attached to the test piece by means of the epoxy bond (AREMCO 805) as shown in Fig. 13. Though the bond is charred during the heating tests, it still keep the test piece attached to the bakelite holder. The bakelite holder except the bottom is normally not reusable and is replaced every test.

A water-cooled side-wall heat protection attachment is installed for the most delicate heating tests of materials with small heat capacity such as light weight insulator of the MUSES-C heatshield (MC-Lite). In this case even slight heat input makes the data acquisition inaccurate. The water-cooled attachment is attached to the test piece by a couple of fixing screws and the gap between them is sealed with epoxy bond as shown in Fig. 14. Though the bond will be charred during the heating tests, it still fill up the gap because of low dynamic pressure at the side.

**Temperature Measurement**

The necessity of the side-wall heating protection by means of appropriate holders is described in the previous section. In addition to the side-wall heating protection, we also need to pay attention to the thermocouples used and their insertion into the test piece. The electric resistance of the carbon phenolic ablator ranges 0.2 to 50 $\Omega \cdot cm$, which differs in its magnitude depending on the direction (perpendicular or parallel to the layer-plane) and the thermal phase status (virgin or charred). Because carbon phenolic ablators do not have good electrical conductivity, the electrically sheathed-type thermocouples necessarily need not to be adapted as long as dc amplifiers are electrically floated from the ground. K-type (chromel-alumel) thermocouples are useful for the measurement of the internal temperature distribution; Though the measurement limit is relatively low, at most 1200 deg C, the electromotive force is so large that the signal to noise ration is high in comparison with other types of thermocouples such as B or R.

The important point to be emphasized here is the diameter of thermocouples and the insertion direction into the test pieces. Normally the heat conductivity of the thermocouples as
a metal cable is in the order of several tens to a hundred W/m/K, which is several tens to a hundred times larger than that of the carbon phenolic ablator (0.3 to 0.9 W/m/K depending on the temperature). When the thermocouple is inserted into the test piece in the same direction as the temperature gradient, the heat conduction through the thermocouple is not negligible in comparison with the heat along the ablator, which makes the measured temperature small. For
accurate temperature measurement the thermocouples need to be inserted in radial direction, namely parallel to the heating surface as was previously shown in Fig. 13. As a typical example, the temperature profiles measured with radially and axially-inserted thermocouples are shown in Fig. 15. In the case of heating with 6 MW/m² – 30 sec, the maximum temperature discrepancy at the surface depth of mm is about 40°C around the center value about 200°C.

6. CONCLUDING REMARKS

A 1-MW segmented-type arc heater has been designed and installed in the ISAS high enthalpy flow facility for the purpose of the basic study of aerothermophysics and the development of thermal protection materials for the atmospheric hypersonic vehicles. The operation characteristics of the facility and the simulated heating environment was briefly described in this report. Although the aerothermophysical flight environment for the vehicles can not be duplicated even in the facility, the facility has successively conducted heating tests by taking account of the behavior of thermal protection material in the high enthalpy flow, and it contributed much to the designing of the thermal protection system of the superorbital reentry bodies.

REFERENCES


Utilization of Expansion Tube for MUSES-C Reentry Simulation

By

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Abstract: An expansion tube has a unique capability of generating a hypervelocity flow without experiencing a stagnation condition of the test gas. Related to MUSES-C superorbital reentry flow condition, experimental and numerical simulations were conducted in domestic institutions as well as in University of Queensland, Australia. This paper summarizes the characteristics of the facility and results of their investigations.

1. INTRODUCTION

The MUSES-C capsule during its reentry phase will experience severe convective and radiative heat transfer at superorbital velocities of about 11 km/s. A considerable amount of high quality experimental heat transfer data in the high-enthalpy environment is needed for validating computational codes used in the design of thermal protection system for the reentry capsule. However there are very few impulsive test facilities capable of generating reliable hypervelocity flow around bodies in the laboratory conditions. NASA Ames Research Center built in 1960s a ballistic range which was combined with a shock tunnel for generating a counter flow against the projectile motion to carry out reentry aerothermodynamic experiments (Carros and DeRose 1970). Although this facility would be suitable for superorbital reentry aerodynamics study, it is very expensive and synchronization of operation is rather difficult. On the other hand an arc driven wind tunnel capable of generating high enthalpy flows has rather limited particle velocity simulation capability (Park 1990). In free-piston driven shock tunnel, because of the complex hypersonic nozzle expansion flow problems coupled with

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melting of nozzle wall material, the stagnation enthalpy practically attainable is limited to about 25 MJ/kg (Ito et al. 1998, Hanemann et al. 2000). An expansion tube (Resler & Blosson 1952, Trimpi 1962) is capable of generating high enthalpies over 25 MJ/kg. Recently experiments in superorbital enthalpy conditions using an expansion tube have been reported (Morgan 1998, Palmer et al. 1997, Neely et al. 1991, Paull & Stalker 1992). Both convective and radiative heat transfer over bodies can be simulated in an expansion tube, with comparable operational costs to that of a shock tunnel. Recently a free-piston-driven expansion tube facility has been commissioned in Shock Wave Research Center, Institute of Fluid Science, Tohoku University, Japan. Related numerical simulations were conducted in Institute of Space and Astronautical Science, Shizuaka University and Tohoku University. In this paper the results of the non-equilibrium hypervelocity flow simulation, both experimental and numerical, over the MUSES-C reentry capsule are reported.

![Fig. 1: x-t diagram of free-piston-driven expansion tube operation.](image)

2. PRINCIPLE AND FEATURES

Figure 1 shows the typical x-t (distance vs. time) diagram of a free-piston-driven expansion tube operation. The driver gas (DG) is compressed by the free-piston in the compression tube (CT). Upon rupture of the primary diaphragm, a strong shock wave (SW1) propagates through the test gas (TG) filled in the shock tube (ST), subsequently impinging on the secondary diaphragm. Following the rupture of the secondary diaphragm the shock wave (SW2) as a transmitted wave travels through the acceleration tube (AT), compressing the acceleration gas (AG) initially filled at low pressure. Reflected expansion waves in turn propagate through the TG, which further expands freely to an ultra-high velocity of the interest. The test flow is
observed at the exit of the acceleration tube during the period from the arrival of the contact surface and the leading expansion wave.

3. EXPERIMENT

3.1 International

Experimental study on the MUSES-C reentry simulation was initiated at Center for Hypersonics, University of Queensland (Morgan 1998). Using their expansion tubes, X-1 and X-2, flow visualization experiments using two-wavelength holographic interferometer (McIntyre et al. 1998, McIntyre et al. 2000), the measurement of heat transfer onto the stagnation region (Palmer et al. 1997), and simulation of ablating heat shield wall with hydrogen injection from the wall near the stagnation point (Sasoh et al. 2000) were conducted. Currently, their research efforts have been successfully continued using a new facility, X-3, of increased facility dimensions.

3.2 Domestic

Figure 2 schematically illustrates the expansion tube, JX1, which is installed at Shock Wave Research Center, Institute of Fluid Science, Tohoku University (Sasoh et al. 2001). It consists of a free-piston driver, shock tube (ST), acceleration tube (AT) and test/dump chamber. The reservoir has an inner diameter of 250 mm and length of 1 m. The inner diameter and length of the compression tube (CT) are 150 mm and 3.0 m, respectively. The 7.0-kg free piston is made of aluminum alloy. Between CT and ST, a layer of 1.6-mm-thick, mild steel primary diaphragm is inserted. The rupture pressure of the diaphragm is controlled by the depth of the cross grooves machined on its surface. The static rupture pressure is calibrated using a hydraulic test device. The inner diameter of both ST and AT is 50 mm. Before filling the acceleration gas (AG), AT and the test/dump chamber are evacuated using a turbo-molecular vacuum pump down to $3 \times 10^{-2}$ Pa.

Nine piezoelectric pressure transducers (PCB 113A03 or 112A21), four in ST and five in AT, are flush mounted on their inner surface. The rise time of all the pressure transducers is 1 $\mu$s. The shock speed is determined by the method of time of flight over the separation distance between adjacent transducers.

![Schematic of expansion tube, JX1.](image-url)
Table 1: Expansion tube operation conditions. Pressures are of an absolute value.

<table>
<thead>
<tr>
<th>Component</th>
<th>Parameter</th>
<th>Value</th>
<th>Material/Species</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reservoir</td>
<td>Fill pressure</td>
<td>2.6 MPa</td>
<td>Dry air</td>
</tr>
<tr>
<td>Compression tube (CT)</td>
<td>Fill pressure</td>
<td>118 kPa</td>
<td>He (DG)</td>
</tr>
<tr>
<td>Primary diaphragm</td>
<td>Static rupture pressure</td>
<td>20 MPa</td>
<td>Mild steel</td>
</tr>
<tr>
<td>Shock Tube (ST)</td>
<td>Fill pressure</td>
<td>3.5 kPa</td>
<td>Dry air (TG)</td>
</tr>
<tr>
<td>Secondary diaphragm</td>
<td>Thickness</td>
<td>17 μm</td>
<td>Cellophane</td>
</tr>
<tr>
<td>Acceleration tube (AT)</td>
<td>Fill pressure</td>
<td>Case A: 16 Pa</td>
<td>Dry air (AG)</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Case B: 3.5 Pa</td>
<td>Dry air (AG)</td>
</tr>
</tbody>
</table>

Table 2: Measured free stream characteristics of test flows.

<table>
<thead>
<tr>
<th>Variable (Unit)</th>
<th>Case A</th>
<th>Case B</th>
</tr>
</thead>
<tbody>
<tr>
<td>SW2 speed (km/s)</td>
<td>7.9±0.1</td>
<td>8.9±0.1</td>
</tr>
<tr>
<td>Static pressure (kPa)</td>
<td>9.7±1.2</td>
<td>2.9±0.5</td>
</tr>
<tr>
<td>Pitot pressure (kPa)</td>
<td>800±50</td>
<td>670±13</td>
</tr>
<tr>
<td>Mass-averaged particle velocity (km/s)</td>
<td>7.5 ±0.1</td>
<td>8.5±0.1</td>
</tr>
<tr>
<td>Temperature (K)</td>
<td>2300±400</td>
<td>1050±200</td>
</tr>
<tr>
<td>Density (kg/m³)</td>
<td>1.2×10⁻²</td>
<td>8.2 × 10⁻³</td>
</tr>
<tr>
<td>Flow Mach number</td>
<td>8.4±0.6</td>
<td>13.5±1.2</td>
</tr>
<tr>
<td>Stagnation enthalpy (MJ/kg)</td>
<td>31±0.9</td>
<td>37±0.2</td>
</tr>
</tbody>
</table>

Typical operation conditions of the expansion tube are tabulated in Table 1 (Sasoh et al. 2001). The stagnation enthalpy of the test flow is varied by varying the fill pressure of AG. Dry air is used as the test gas in all the experiments. In Table 2, measured and estimated test flow parameters are tabulated. The shock speed is determined by monitoring the static pressures measured at 0.680 m and 0.015 m from the AT exit, respectively.

The test flow parameters tabulated in Table 2 are calculated using a computer program assuming for equilibrium flow. The velocity of CS2 is assumed to be equal to the particle velocity behind SW2. The computation is iteratively carried out until all the above-mentioned relations are satisfied for a given temperature. The stagnation enthalpies in the present experiments are estimated to be 31 MJ/kg and 37 MJ/kg cases A and B, respectively.

The radiating shock layer surrounding the forebody is visualized using a high-speed image converter camera (IMACON 468, Headland Photonics). Six frames are sequentially recorded in a single shot. Each frame has 576 × 385 pixel resolution. The exposure time is adjusted to 200 ns in each frame. The framing interval is varied from 5 to 32. A Xe flush light with 500 μs pulse duration is used in order to back-light the model. The system is triggered by the pressure signal at 2 m upstream of the acceleration tube exit.

Figure 4 show photographs of the radiating shock layer observed around the model. Considering that the images are the side view of the axi-symmetric flow, the intensity of the radiation near the stagnation point where the physical thickness is small, is strongest. As will be seen in the next section, good agreements are obtained between the experiment and the numerical simulation both on the shock stand-off distance and the bow shock wave shape.
4. NUMERICAL SIMULATION

4.1 Test flow simulation

Related to the MUSES-C program, a numerical simulation code for the superorbital reentry flow has been developed and applied for the flow field prediction around the reentry capsule of MUSES-C (Otsu et al. 1998, Otsu et al. 1999). Numerical simulation of the test flows around the MUSES-C reentry capsule model which are described in the previous section are conducted here.

The governing conservation equations are associated with axi-symmetric Navier-Stokes equation. Park's two-temperature model (Park 1987a) is employed to consider thermodynamic nonequilibrium situation between the translational-rotational energy and the vibrational-electron translational energy. Eleven species consisting of N, O, N₂, O₂, NO, N⁺, O⁺, N₂⁺, O₂⁺, NO⁺, and e⁻ are considered. The vibrational energy is calculated using a harmonic oscillator model. On the electronic excitation energy, the first two terms of partition function are taken into account. The relaxation time for a translational-vibrational energy exchange is estimated using Millikan and White's model (Millikan & White 1963) combined with Park's correction term (Park 1987b). The viscosity of each species is evaluated by curve-fitting method based on the tabulated data (Gupta et al. 1990). The heat conductivity of the translational, vibrational and electron temperature are calculated using an Eucken's relation (Vincenti & Kruger 1965, Candler & MacMormack 1988). The total viscosity and conductivity are calculated using Wilke's semi-empirical mixing rule (Wilke 1950). For further detail on the method of the numerical simulation, the reader should refer to Otsu et al. 2002.

Figures 4a and 4b show isobars around the MUSES-C reentry capsule model under the
condition of Case A and B, respectively. A sharp pressure rise from the free stream value corresponds to the bow shock location. The high-pressure is distributed up to the body surface. The agreement in the bow shock shape with the experimental result is reasonable not only in Case A but also in Case B where the flow speed is higher.

To compare the result more concisely, the radiation intensity is calculated based on the numerical result. For this purpose, the radiation estimation database, SPRADIAN (Fujita et al. 1997), is employed. After calculating the emission and absorption coefficients at every grid point from numerically-simulated flow field data, the radiation intensity is obtained by solving the radiation transfer equation along the line of sight. Figure 5 shows numerical results of radiation intensity distribution from the shock layer. The radiation intensity is normalized by the maximum value.
in the shock layer. The numerical results show the same tendency as the experimental image of Fig. 3: The radiation intensity around the stagnation region is stronger than that in the leeward region as easily expected. Near the stagnation region, the thickness and shape of the radiating shock layer agrees with the experiments very well. On the other hand, in the leeward region, the numerical radiating layer becomes slightly thinner than the experimental one. Further discussions on the flow characteristics in the shock layer and the sensitivity analysis in the numerical simulation, the reader should refer to Otsu et al. 2002.

Here, the numerical conditions are determined from the free stream condition experimentally determined. The thickness of the radiating shock layer shows a good agreement between the numerical simulation and the experiment. It can be concluded that the evaluation of the test time and the free stream condition in the experiment is reasonable. The result of the present study has a significant contribution to establish credibility of the expansion tube facility as a reliable testing facility for hypersonic flows. Further investigations are necessary, such as comparison of surface heat transfer and radiation intensity between the experimental and numerical results, and estimation of influence of thermo-chemical model on such comparisons.

4.2 Tube operation simulation

Computational Fluid Dynamics (CFD) code that considered nonequilibrium thermochemical reactions is developed for obtaining the whole transient flowfield in the JX-1 expansion tube operation (Mizuno et al. 2001, Mizuno et al. 2002). In the free-piston driver section, Euler equations in an axi-symmetric form are chosen as the governing equations. The piston motion is included in the present simulation by solving the equation of motion where friction between piston surface and sidewall of compression tube is accounted for. The details of the numerical treatment of the moving piston are described in Mizuno et al. 2001. Heat loss from the sidewall is neglected. An ideal gas assumption is employed.

The governing equations in the shock and acceleration tubes are the Navier-Stokes equa-
tions in an axi-symmetric form. The set of equations consists of global mass, species mass, momentum, total energy, and vibrational-electronic energy conservation equations. Six neutral species, i.e., O, N, O₂, N₂, NO and He are assumed. The reaction rate coefficients proposed in the two-temperature model of Park 1990 are employed. The governing equations are integrated by the cell-centered finite volume method. The convective numerical flux is calculated by AUSM-DV upwind scheme. (Wada & Liu 1995) The MUSCL approach is employed for the higher spatial accuracy. A two-level second order explicit Runge-Kutta method is used for time integration. The diagonal point implicit method (Eberhardt & Imaly) is employed for improving stability in the integration of the source terms. The CFL number is assumed as 0.4 in all calculations.

Figure 6 shows the time evolution of the flowfield in the shock tube section shown as isopy-
Fig. 7: Isopycnics in the acceleration tube, $t$ indicates the time from rupture of the primary diaphragm.

Complicated flow structures are indicated. In particular, the contact surface dividing the test gas from the driver gas (CS1) is significantly perturbed by the interaction with the boundary layer as it propagates downstream. A relatively smooth flow begins to appear at the primary diaphragm location after $t=254$ $\mu$s. In this model, the primary diaphragm is assumed to gradually open in a duration time of 200 $\mu$s. Hence, a smoothly expanding flow can appear only after this period. A smoothly expanding flow from the throat begins to separate at the throat and a slip line is formed. The time evolution of the flowfield in the acceleration tube is shown in Fig. 7. The transmitted shock wave (SW2) first propagates into the acceleration tube section, and then a contact surface dividing the test gas from the acceleration gas (CS2) comes in. Behind CS2, there appears the test gas region that is terminated by CS1. One can notice that these two contact surfaces are further perturbed as they propagate downstream, and make
the test gas region also significantly perturbed. Note that the test gas region becomes longer as time elapses due to the unsteady expansion.

Figure 8 shows the mole fraction profiles in the acceleration tube at the time when SW2 reaches the end of the acceleration tube section. One can see the mole fraction of O atoms is kept relatively low and the fraction of nitrogen atoms is almost zero in the test gas region behind CS2. Note that a significant amount of He that is mixed with the test gas appears in front of CS1.

As to the primary diaphragm, it is found that the gradually opening diaphragm model is appropriate with a 200 \( \mu \)s of the opening time, if the diaphragm is assumed to be located slightly shifted downstream. As to the secondary diaphragm, it is shown that an ideal rupture model can give a better agreement with experiment than those of the holding time model or diaphragm inertia model. The duration of test time estimated from the Mach number profile at the end of acceleration tube is found to give a fairly good agreement with that determined in the experiment.

5. CONCLUDING REMARKS

As is reported here, an expansion tube has a potential of simulating superorbital flows. Diagnostics of the generated flow combined with numerical simulation are critical issue to evaluate the quality of the test flow. Efforts for improving the performance of the facility and determination of experimental conditions are still continued.

![Fig. 8: Mole fraction along the axis of symmetry at the moment SW2 comes to the end of acceleration tube.](image-url)
ACKNOWLEDGMENT

The first author would like to thank valuable suggestions from Professor K. Takayama of Shock Wave Research Center, and appreciate various technical helps from Messrs. H. Ojima, T. Ogawa, K. Takahashi, T. Watanabe, K. Asano and M. Kato of Institute of Fluid Science.

REFERENCES


Experimental Study of Nonequilibrium Phenomena behind Strong Shock Waves Generated in Super-orbital Reentry Flight

By

Takashi ABE*†, Atsushi MATSUDA†, Kazuhisa FUJITA*, and Shunichi SATO*

(1 February 2003)

Abstract: The flow behavior in a thin air behind the shock wave with a shock velocity around 12 km/sec was investigated experimentally. For the experiment, a free-piston-driven shock tube was used and a radiation from behind the shock front was investigated spectroscopically. Being based on the radiation spectrum emitted from the region behind the shock wave, the distributions for rotational and vibrational temperatures of N₂ and N₂⁺, and electron density were determined. These results were compared with those for a pure nitrogen gas which was investigated previously. It was found that there is remarkable resemblance between the distributions in an air and in a pure nitrogen gas. This suggests that the existence of molecular oxygen in an air has only a little influence on the behavior of temperatures of N₂ or N₂⁺, and the behavior of electron density behind the shock wave, compared with those observed in a pure nitrogen gas.

1. INTRODUCTION

The prediction of the flow field is an important issue for designing the reentry body which conduct an atmospheric flight with a hyper speed. The prediction, however, is not simple because, in the shock layer, various real gas phenomena occurs depending on the flight speed. Generally speaking, the faster the speed is, the more complex the real gas phenomenon is. The existing method is based on the experience gathered for the flight speed range smaller than that of the MUSES-C reentry capsule. Therefore, for the prediction of the flow field for the MUSES-C capsule, the top priority task is to gather the experience or the experimental data about the flow field. Unfortunately, such data is scarce currently. This is the reason for us to start to gather such data. For this purpose, we examine the flow behavior behind the shock wave with a shock velocity similar to the flight speed of the MUSES-C capsule; around 12 km/sec. First of all, a nitrogen gas was investigated as a test gas through which the shock wave propagates. This is because, in the nitrogen gas, the real gas effect such as chemical reaction is limited in compare to the air which comprises both nitrogen and oxygen. In this
investigation, we measured the distribution of (1) the vibrational rotational temperatures, (2) electron number density, by means of the spectroscopic technique1, 2. From this investigation, the phenomena which can not be predicted by the existing flow model. One is the abnormally lower rotational temperature, and another is the abnormally large electron number density just behind the shock wave. The theoretical understanding of these phenomena is underway. In parallel to the theoretical understanding, and based on the investigation of the phenomena observed in a nitrogen gas, we attempt to investigate the behaviour of the flow generated in an air. To this aim, the investigation by the shock tube was carried out by using an air as a test gas. In this report, we focus on this topic. To investigate the behavior of rotational and vibrational temperatures of N₂ or N₂⁺, electron excitation temperature, and electron density behind the shock wave in an air, and to clarify the difference in the nonequilibrium behavior behind the shock wave between for an air and for a pure nitrogen gas.

2. EXPERIMENTAL SETUP

In the experiment, was employed the double diaphragm free-piston-driven shock tube equipped at the Institute of Space and Astronautical Science in Japan. In this shock tube, the fastest shock propagation velocity of approximately 15 km/sec in an ambient pressure of 0.1 torr could be attained. To avoid any impurity in a test gas, the simulated air composed of oxygen 22% and nitrogen 78% was employed as for the test gas. In the present experiment, the test conditions for the initial ambient pressure 2.1 and 0.3 torr were selected, which produces a shock wave propagating at the velocity of 8 and 12 km/sec, respectively. Needless to say, the shock wave propagation velocity of 8 and 12 km/sec corresponds to the reentry velocity from the lower earth orbit and of the direct reentry from an interplanetary orbit, respectively.

In order to investigate the nonequilibrium phenomena behind the shock wave, radiation emitted from the highly-heated region behind the shock wave was investigated.

Figure 1 shows the optical setup at the measurement section of the shock tube. In this shock tube, the flow property behind the shock wave is considered to be one-dimensional. As a matter of the fact, since the shock front is almost perpendicular to the wall of the tube and the investigated region behind the shock front is comparatively near the shock front, we can
assume that the non-uniformity for the flow behind the shock wave, which might be caused by the thick boundary layer, is small and the radiative characteristic is almost uniform in the direction parallel to the shock front. As shown in the Fig.1, a radiation emitted from a location apart from the shock front by a selected distance is focused on the entrance slit of the spectrometer (ORIEL MS127i) by means of an ellipsoidal mirror and is decomposed into a spectrum by the spectrometer. The spatial resolution of the measurement in this system is less than 1.2 mm.

The spectrum image of the radiation decomposed by the spectrometer was recorded by means of an image-intensified CCD camera (ANDOR IN-STA SPECV IS510 ICCD). The ICCD camera was driven by the delayed signal generated by the pressure sensor mounted upstream of the measurement section. In each run of the shock tube, was recorded one ICCD image for the radiation spectrum emitted from a location with a selected distance apart from the shock front. Accumulating the radiation spectrum data for various locations, which was obtained by driving the ICCD camera with a various delayed time in repeated runs of the shock tube operations, an entire distribution of the radiation spectrum was obtained for the regions behind the shock front.

For this method to make sense, the reproducibility of each run of the shock tube is inevitable. To monitor the reproducibility, not only the pressure signals and the shock propagation velocity measured from them but also the distribution of the total radiation intensity distribution was monitored. For this monitoring, the pin photodiode sensor was installed on the opposite side of the spectrometer. Throughout this experiment, the reproducibility was found to be fairly good.

The He-Ne laser in this figure was used as a light source for the laser schlieren technique to detect the density rise of the shock front, which gives an origin of the distance of the location starting from the shock front.

3. SPECTRUM RESULTS

3.1 Spectrum for the case of 12 km/sec

Typical radiation spectrum at the distance 1.4 mm apart from the shock front is shown in Fig. 2 where the shock propagation velocity is 12.4 km/sec.

As shown in this figure, in the wavelength range from 250nm to 500nm, the contribution of $N_2$ and $N_2^+$ molecules is dominant. Therefore, the behavior of the spectrum in this wavelength range is dominated by rotational and vibrational temperatures of $N_2$ and $N_2^+$. This means that those temperatures can be determined from the measured spectrum. The emission observed at 485nm is the Balmer line of atomic hydrogen which originates from the water vapor evaporated from the shock tube wall. The broadening of Balmer line is primarily dominated by the Stark broadening effect. Since this effect is mainly related to the existence of the electron, the electron number density can be determined from the measured spectrum. Additionally, in the case of the shock propagation velocity of 12 km/sec, the emission from an atomic nitrogen is dominant in the region far from the shock front more than 4 mm. Those emissions enable us to determine an electron excitation temperature of atomic nitrogen by using the relative spectrum intensity method.

When we compare, in the case of shock propagation velocity of approximately 12 km/sec, a radiation spectrum behind the shock wave in an air with the one in a pure nitrogen gas, the radiation spectrum in the case of an air is found to be very similar to that in the case of a pure
3.2 Spectrum for the case of 8 km/sec

In the case of the shock propagation velocity of 8 km/sec, we focus on the wave length region ranging from 280 to 340 nm. In this shock propagation velocity case, however, the radiation spectrum in air shows a slight discrepancy with that in a pure nitrogen gas, as shown in Figs. 4(a) and 4(b). This discrepancy between in the case of an air and in a pure nitrogen gas is considered to be due to the contribution from NO molecule whose band spectrum peak exists in an ultra violet region. The reason why the radiation from NO is remarkable in the case of the shock propagation velocity of 8.3 km/sec is that NO does not dissociate completely even in the equilibrium state, in contrast to the higher shock propagation velocity of 12 km/sec. This suggests that, in this shock propagation velocity, contribution of not only N\(_2\) and N\(_2^+\) but also NO should be taken into account, unlike in the higher shock propagation velocity, to determine the temperatures being based on the radiation spectrum behind the shock wave in air.

4. ROTATIONAL AND VIBRATIONAL TEMPERATURES DISTRIBUTION

4.1 Temperature determination method

In order to determine rotational and vibrational temperatures of N\(_2\) or N\(_2^+\) from the measured spectrum, the spectrum fitting method was applied. This is because the spectrum resolution obtained in this experiment is not high enough to determine rotational temperature, from the ratio of each rotational spectrum, independently from the vibrational temperature determination. To calculate the spectrum database theoretically, was employed the radiation calculation code SPRADIAN 3, which enables us to predict a radiation spectrum for a number density and temperature of each species. The detail of this method is described in Ref. 1 and Ref. 4. The typical spectrum fitting result is shown in Fig. 5. In this figure, both numerical and ex-
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Experimental spectrum in the wave length region shorter than 400nm agree well with each other. The discrepancy observed in the wave length region longer than 400nm is considered to be due to the existence of N and H\(_2\) lines appearing in the spectrum obtained experimentally. As a conclusion, from this result, rotational and vibrational temperatures of N\(_2\) are determined to be 3500 K and 5000 K, respectively, and those temperatures of N\(_2^+\) are determined to be 17500 K and 9000 K, respectively. The distribution of temperatures can be obtained by applying this method to each spectrum measured behind the shock wave.

4.2 Temperature distribution for case of 8 km/sec

Figure 6 a) shows temperatures distribution behind the shock wave in an air in the case of shock propagation velocity of 8.3 km/sec. As shown in this figure, in the region just behind the shock front, the measured rotational temperature is highly nonequilibrium with the numerical
Fig. 4: Radiation spectrum, a) shock propagation velocity of 8.3 km/sec in the case of air, and b) shock propagation velocity of 8.3 km/sec in the case of pure nitrogen.

prediction of the translational temperature. Here, the numerical prediction is based on Park’s two-temperature-model 7. Since, in the Park model, it is assumes that rotational temperature is fully equilibrium with translational temperature, it can be concluded that the experimental and numerical rotational temperatures significantly disagree at least just behind the shock wave. In the region far from the shock front, both rotational and vibrational temperatures are close to the equilibrium temperature as predicted by Park model. These behaviors are very similar to those observed for a pure nitrogen gas as shown in Fig. 6 b).

4.3 Temperature distribution for the case of 12 km/sec

Figure 7 a) shows temperature distributions behind the shock wave in an air in the case of the shock propagation velocity of 12.4 km/sec. This figure shows that the measured rotational temperature is nonequilibrium with the numerical rotational temperature which is assumed
to be equal to the translation temperature, and that there is, again, discrepancy between experimental result and numerical prediction based on Park model. As for the vibrational temperature, it is nearly close to the numerical prediction.

In Fig. 7 b), the previous result for a pure nitrogen gas in the case of the shock propagation velocity of 11.9 km/sec is depicted for comparison. It should be noted that the temperature distribution in an air resembles the one in a pure nitrogen gas very well.

In the numerical prediction, however, the difference between an air and a pure nitrogen gas appears in the behavior of vibrational temperature and especially in the difference of the distance by which the temperature distribution reaches the equilibrium state, as shown in Figs. 7 a) and b). The distance to reach the equilibrium state in an air is shorter than that in a pure nitrogen gas. The vibrational temperature behind the shock wave in an air doesn’t rise, in a process of the relaxation, as rapidly as that in a pure nitrogen gas.

However, in the experimental result, especially in the case of the shock propagation velocity of 12 km/sec, rotational and vibrational temperatures can’t be determined based on the molecular spectrum, because emission from an atomic nitrogen becomes stronger in the region far behind the shock wave more than 4 mm. Therefore, it is difficult to determine experimentally the difference, between in an air and in a pure nitrogen gas, of the distance to reach the equilibrium state. As for the behavior of the vibrational temperature, the difference between in an air and in a pure nitrogen gas is, according to the numerical calculation, at most a several hundred degree which is within an uncertainty unavoidable in the measurement and, therefore, it is difficult to distinguish those differences between in an air and in a pure nitrogen gas in this experiment, even though they appears in the numerical prediction.

5. ELECTRON DENSITY DISTRIBUTION

The local electron density strongly influences the Stark broadening of the Balmer line of hydrogen, besides the weak influence of the electron temperature. Therefore, the electron density can be determined by applying the theoretical spectrum fitting to the measured Balmer
Fig. 6: Temperature distribution, a) shock propagation velocity of 8.3 km/sec in the case of air, and b) shock propagation velocity of 8.3 km/sec in the case of pure nitrogen.

The theoretical spectrum of the Balmer line was calculated based on Ref. 5. Figure 8 shows an example of spectrum fitting for the Balmer line. In this figure, the numerical spectrum agrees with the experimental one fairly well. Therefore, it is concluded that the electron density is $4.8 \times 10^{22} \text{ m}^{-3}$, for the electron temperature $3 \times 10^4 \text{ K}$.

Figure 9 a) shows an electron density distribution obtained by applying this method at each location behind the shock wave in an air. As shown in this figure, according to the numerical prediction, the electron density increases gradually just behind the shock wave, and then at the location approximately 7 mm behind the shock wave, it increases rapidly and reaches the equilibrium state. On the other hand, the measured distribution is almost flat and its value agrees with that of the plateau region of the numerical simulation. That is, there is a big discrepancy, just behind the shock wave, between experimental result and numerical prediction. This discrepancy between the experimental result and the numerical prediction was already observed behind the shock wave in a pure nitrogen gas as shown in Fig. 9 b).
Electron density behavior experimentally observed behind the shock wave in an air well resembles that previously observed in a pure nitrogen gas. In the numerical result, however, the difference in an electron density between an air and a pure nitrogen gas appears slightly. According to the numerical prediction, the electron density in the equilibrium state in an air is higher than in a pure nitrogen gas. But this difference can’t be observed experimentally since the difference is small enough to be within the uncertainty unavoidable in the experiment.

6. ELECTRON EXCITATION TEMPERATURE DISTRIBUTION

In the region far behind the shock front more than 4 mm, the emission from the atomic nitrogen becomes prominent in the case of the shock propagation velocity of 12 km/sec. Therefore, the electron excitation temperature of atomic nitrogen can be determined by making use of the relative line intensity method. For the temperature determination, atomic lines in the
Fig. 8: Spectrum fitting at Hβ line; $V_s=12.4$ km/sec, $L=4.85$ mm, $N_e=4.8 \times 10^{22}$ m$^{-3}$, $T_e=30000$K.

Fig. 9: Electron density distribution, a) shock propagation velocity of 12.4 km/sec in the case of air, and b) shock propagation velocity of 12.6 km/sec in the case of pure nitrogen.

Wave length regions of 380-440 nm and of 480-540 nm were selected. Figures 10 a) and b) show the radiation spectrum at the location of 14.87 mm behind the shock wave. The atomic spectrum line which were used for temperature determination is indicated by the wave length and transition array as shown in these figures. By using the transition probability $A_{ij}$, the statistical weight $g_i$ based on Ref. 6, and the relative radiation intensity of energy transition $I_{ij}$, the quantity $\ln(I_{ij}/A_{ij}g_i)$ is plotted against upper energy level as shown in Fig. 11. From this figure, the electron population is observed to be close to the Boltzmann distribution, which is assumed for the present temperature determination method. The electron excitation tem-
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Fig. 10: Radiation spectrum of N\textsubscript{2} shock propagation velocity of 12.4 km/sec, 14.87 mm distance behind the shock wave in air, a) wave length region 380nm-440nm, b) wave length region 480nm-540nm.

Fig. 11: Boltzmann plot; shock propagation velocity 12.4 km/sec, distance from the shock front 7.9 mm.

Temperature can be determined to be 6600K at the distance of 7.9 mm behind the shock front.

Figure 12a) shows the electron excitation temperature distribution behind the shock wave in an air in the case of the shock propagation velocity of 12.4 km/sec. From this figure, it is observed that an electron excitation temperature obtained from the experiment is distributed around 6000 - 7000 K, and is lower than that predicted by the numerical prediction based on Park’s two-temperature-model.
Fig. 12: Electron excitation temperature distribution, a) shock propagation velocity of 12.4 km/sec in the case of air, and b) shock propagation velocity of 11.9 km/sec in the case of pure nitrogen.

It should be noted that the electron excitation temperature in the figure equals to the vibrational temperature shown in Figs. 12a) and b), since the equilibrium between them is assumed in the Park's model. Being compared to the previous results for the pure nitrogen gas shown in Fig. 12b), the behavior of the present electron excitation temperature for an air is found to be very similar to that for a pure nitrogen gas.

7. CONCLUSION

The vibrational and rotational temperatures of \( N_2, N_2^+ \) molecules, the electron excitation temperature of atomic nitrogen, and the electron number density were determined based on the radiation spectrum, for an air behind the shock wave propagation with 8 km/sec and with 12 km/sec. In the shock propagation velocity of 12.4 km/sec, the radiation spectrum in an air, in the wave length region for the spectrum contributed by nitrogen originated molecules (i.e., \( N_2, N_2^+ \)), well resembles that in a pure nitrogen gas examined previously. As a result, the rotational and vibrational temperature distribution tendencies of \( N_2 \) and \( N_2^+ \) are similar to that in a pure nitrogen gas. Therefore, the following conclusion which is similar to that for a pure nitrogen gas is reached; 1) rotational temperatures of \( N_2 \) and \( N_2^+ \) are nonequilibrium with the translational temperature and show disagreement with the Park's two-temperature-model, 2) vibrational temperatures of \( N_2 \) and \( N_2^+ \) are also nonequilibrium with the translational temperature, but almost agree with the numerical prediction.

The behavior of the electron density distribution behind the shock wave in an air shows disagreement with the numerical prediction, especially at the region just behind the shock wave. This is, again, similar to the result obtained previously for a pure nitrogen gas.

As for the electron excitation temperature behind the shock wave in an air, its behavior is also almost similar to that in a pure nitrogen gas.

Although the difference between an air and a pure nitrogen gas appears theoretically, such difference is too small to be detected in the present experimental setup.
REFERENCES

Pressure- and Temperature-Sensitive Paint Measurements in Shock Tunnel

By

Yoshiaki Nakamura* and Mitsuru Kurita*

(1 February 2003)

Abstract: Pressure- and temperature-sensitive paint measurements can make possible the measurement of global surface pressure and temperature distributions on a body at a time, which are different from conventional sensors’ point measurements. Shock tunnels have been widely used to make hypersonic flows at low cost and to examine the flow field around high-speed vehicles. However, they have a limitation of duration. The present study applies pressure- and temperature-sensitive paints to investigate an interaction flow field due to two bodies, i.e., a hemisphere-cylinder and a delta wing, in a hypersonic flow with $M_{\infty} = 8.1$. As a result, the present methods were found to be useful and effective to analyze the hypersonic flow field.

1. INTRODUCTION

The concept of two stage to orbit, TSTO, has been studied as one of choices for future space transportation system (Tanatsugu 2002). Although many aerodynamic subjects related with this concept have been dealt with so far, we focus on the flow field with aerodynamic interaction between two bodies shown in Fig. 1. In the flow field treated here in this study, there are many experimental parameters such as the relative position between two bodies and angle of attack. Therefore, a highly efficient experimental method is required to reduce the cost of implementing the experiment.

The methods of pressure sensitive paint, PSP, and temperature sensitive paint, TSP, are good candidates for those efficient measurements because they can perform two-dimensional measurements regarding pressure and temperature; i.e., the distributions of pressure and temperature can be obtained at a time. Moreover, we can conduct experiment with more information at a low cost.

On the other hand, shock tunnels are a very useful experimental facility to develop spacecraft because they can produce hypersonic flows rather easily. However, they have a drawback of very short duration (Nakakita et al. 2000; Kurita et al. 2001; Nakakita et al. 2002).

Then, in this study we make attempt to develop a new experimental method used in shock tunnels and make the experiment more efficient. More specifically, we apply the PSP and TSP
methods to measure the pressure and temperature fields of aerodynamically interacted flow field between two bodies in the regime of hypersonic speed. Furthermore, key problems will be made clear in these measurements and the method to improve them will be discussed.

2. EXPERIMENTAL METHODS

2.1 Painting Methods of PSP and TSP

We have used Bath-Ru (Bathophenanthroline Ruthenium Chloride) as a luminophore, or a pressure sensitive substance in the PSP method. This has the advantage that the experiment can be conducted by using only visible light; that is, ultraviolet light is not required here. Consequently excited light does not greatly dissipate while passing through an observation window. Thus, the light path can be easily and safely designed.

As for the painting method, we have developed a new method where a sticky material, or a kind of silicone (Kurita et al. 2001) is first painted on the model surface, on which the luminophore melted into an organic solvent, the dichloromethane, CH$_2$Cl$_2$, is painted.

This makes the response time fast, because an uncovered layer with the luminophore can be made as a result of vaporization of the organic solvent, which is open to the ambient gas, or oxygen (see Fig. 2a).

From measurements by a small-sized shock tube, it is judged that the response time lies between sub-millimeters and millimeters. This is sufficient for the shock tunnel at Nagoya University with a freestream Mach number of 8.1 (see Table 1), because it has an effective duration of tens of milliseconds.

Furthermore, the painting method can be applicable to most of common metals. In addition,
Pressure- and Temperature-Sensitive Paint Measurements

Fig. 2: Schematic of painting methods: (a) PSP, (b) TSP.

this painted substance can be eliminated easily with organic solvent. However, it is necessary to take care that foreign objects such as dust do not attach on the surface. In addition, in the present study, an attempt to strengthen the luminescence intensity was made by thinly painting the model surface white as a ground.

Since TSP weakens the contact between oxygen and temperature-sensitive substance, Rhodamine B, which conveniently works with visible light, was mixed with a binder, PMMA (Methyl Methacrylate Polymer), which was painted on the model surface (see Fig. 2b). It has been reported that the mixture employed here does not show the characteristics of quenching due to oxygen, or pressure (Iijima et al.). In this study as a method to reduce change in luminescence intensity after running the wind tunnel, transparent acrylic paint is thinly painted on the surface of TSP to protect the TSP surface, the details of which will be described later. However, it is conjectured that a temperature gradient will occur in the layer of the acrylic paint. In addition, in each measurement method, to reduce the interaction in luminescence between bodies, either one of the two bodies was painted black.

2.2 Model

The model employed here is a combination of a delta wing and hemisphere cylinder, as shown in Fig. 1. In this case, there is a possibility that the surface temperature of the model changes due to aerodynamic interaction. It is generally considered that in the PSP measurement method the material of the model that can reduce change in temperature even slightly is preferred, because PSP generally depends on temperature (Nakakita et al. 2000; Asai et al. 1999). To do that, in the present experiment the model was made of aluminum alloy. Incidentally, according to our measurements, the dependence of PSP used in the present system on temperature is about -1.0%/K under the conditions of the atmospheric pressure and a room temperature of 297.8K.

In the case of TSP measurements, temperature change of the model surface was made as
large as possible in order to simplify the experiment. In the present experiment, as a ground of TSP, the surface of the delta wing was coated with urethane resin so that insulation of heat was enhanced.

2.3 Measurement System

In this study, the measurement system mainly consists of instruments for general use (see Fig. 3). Thus, the measurement method by PSP and TSP was made easy to use for researchers who would like to use it. The camera employed here is a single-lens reflex camera (Canon EOS D30), which has a red color filter of $\lambda = 580$ nm, and IR-cut filter put on the lens.

As a light source for excitation, a stroboscope was used with a blue color interaction filter ($\lambda = 472 \pm 39$ nm) in PSP measurements and a bluish green color filter ($\lambda = 490\pm50$ nm) in TSP measurements. This stroboscope is irradiated about 24 msec after the wind tunnel starts to run.

2.4 Calibration Method

To express the relation between luminescence intensity and pressure, the relation of Stern-Volmer was applied (Nakakita et al. 2000; Asai et al. 1999).

\[
\frac{I_{\text{ref}}(P_{\text{ref}}, T)}{I(P, T)} = A + B(P, P_{\text{ref}}) \quad (1)
\]

\[
\frac{P}{P_{\text{ref}}} = C_1 \left(\frac{I_{\text{ref}}}{I}\right) + C_2 \quad (2)
\]

where reference values for $P_{\text{ref}}$ and $I_{\text{ref}}$ are those before the wind tunnel starts to run, i.e., those at a pressure of about 1 Torr. $P$ and $I$ are data while the wind tunnel is running.

In addition, in this study we do not take into account the change in surface temperature of the model. This temperature change is assumed to be small because of short duration of the shock tunnel and the model material employed here, so that the effect can be included into the coefficients of calibration expressions.

Thus, the in-situ method was employed as calibration, where around each pressure tap, the average value of PSP data within 5 pixels in the spanwise direction and 20 pixels in the chordwise direction was made to agree with the data from a pressure transducer.
On the other hand, in the TSP measurement method, the relation between luminescence intensity and temperature can be expressed by the following empirical equation, if temperature-sensitive paint does not have the characteristics of quenching due to oxygen, or the binder does not make oxygen penetrate into it (Asai et al. 1999).

\[ \frac{I(T)}{I(T_{\text{ref}})} = f\left(\frac{T}{T_{\text{ref}}}\right) \]  

where reference data, \( I(T_{\text{ref}}) \), are those before running the wind tunnel, i.e., those at a pressure of nearly 1 Torr, and \( I(T) \) is data while the wind tunnel is running.

3. EXPERIMENTAL RESULTS

3.1 Visualization due to Schlieren Photograph

Figure 4 shows a schlieren photograph. The main characteristics of the present flow field with aerodynamic interaction are shock wave/boundary layer interaction. The effect of a pressure increase due to a strong bow shock generated before the hemisphere cylinder propagates upstream in the subsonic region of the boundary layer of the delta wing, so that the boundary layer separates. A series of compression waves generated upstream of the separated boundary layer lead to an oblique shock wave, or separation shock wave. These two shock waves: the bow shock and the separation shock cause rather weak shock/shock interaction. In addition, the separation of the boundary layer observed in this study is relatively large. As a result, the bow shock takes an vertically asymmetrical shape. Moreover, the reflection of the bow shock was not observed between the hemisphere cylinder and the delta wing.

![Fig. 4: Schlieren photograph.](image)

3.2 PSP Measurement Method

Results by the present PSP measurement method are shown in Fig. 5. We can see a relatively large pressure increase region due to the effects of the bow shock along the symmetrical line of the delta wing surface below the hemisphere cylinder. In addition, the dark, round part is the effects of putty used to make smooth the body surface around a screw hole.

To make the phenomenon easy to understand, qualitative pressure distribution on one wing-half that has a threshold for selected values is shown in Fig. 6a. We can see a region of relatively small pressure increase enclosing the large pressure increase region below the hemisphere cylinder, which is likely be due to a separation shock wave. These flow characteristics agree with those of the schlieren photograph mentioned earlier.
Then, we will show pressure distribution on the hemisphere cylinder combined with schlieren photograph for comparison in Fig. 7. Because of the shape of the bow shock, the pressure distribution shows a vertically asymmetrical distribution, which is different from the symmetric distribution in the case of a hemisphere cylinder alone. From the shape of the pressure distribution, it is judged that boundary layer separation shock wave is close to a half cone. However, we need to make clear more detailed flow pattern by comparing with results of numerical simulation.

### 3.3 TSP Measurement Method

Results of application of the temperature sensitive measurement method to the upper surface of the delta wing are shown in Fig. 8. We can observe a region with relatively large temperature increase due to the bow shock below the hemisphere cylinder. This is quite similar to the results of the pressure measurement mentioned earlier. There also exists a region with small temperature increase due to the separation shock wave, enclosing the region with the large temperature increase. In passing, Fig. 6b shows the temperature distribution on one wing half.

In addition, one problem of the present experiment is that the intensity of local luminescence at the location with the strongest aerodynamic interaction has been reduced after running the wind tunnel compared with that before it runs. At present this does not affect the characteristics of the overall flow field because the region affected by that is small. From now on, the reason for that, the amount of the effects, and proper countermeasures against it are going to be examined, leading to the improvement of painting methods for PSP and TSP.
4. SUMMARY

We devised a new painting method for PSP and TSP, and measured the aerodynamically interacted flow field for a model composing of two bodies by applying PSP and TSP methods to shock tunnel testing at a hypersonic speed. The results well agreed with the characteristics of the flow field obtained from schlieren photography. Moreover, it was found that pressure and temperature fields in the interacted region show similar distributions. Thus, it may be asserted that PSP and TSP are an effective and efficient method in shock tunnel testing. These methods will be more often used from now on.

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