Ablative Thermal Protection,
Thermal Response Characterization,
and Integrated Simulation Technique
Thermal Responses of Ablator for Reentry Capsules with Superorbital Velocity

By

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Abstract: This paper summarizes thermal responses of ablator for reentry capsules with superorbital velocity together with analysis method and the experimental data acquisition in the designing and development process of the MUSES-C reentry capsule. On the prediction of thermo-chemical response of the carbon-phenolic ablator heat shield for the MUSES-C reentry capsule, an ablation analysis code has been developed taking account of the effect of the pyrolysis gas on the surface recession rate. A validation study of the mathematical model for the ablation has been conducted through the comparison between the numerical results and experimental ones which have been carried out in the arcjet facilities of ISAS and NASA/ARC.

1. INTRODUCTION

MUSES-C spacecraft is planned to be launched by M-V rocket in 2003 to make in-situ observations of an asteroid and also to obtain samples of the surface materials (ISAS, 2001). In the final phase of the mission, a small capsule with asteroid sample conducts reentry flight directly from a interplanetary-transfer orbit. The capsule entering with the velocity of 12 km/s is expected to encounter severe aerodynamic heating up to 15 MW/m² due to the high reentry speed. The interior of the capsule needs to be thermally protected against the severe aerodynamic heating within a given temperature by the time of landing. The designing and development of the thermal protection system is an important issue for the success of the reentry mission.

It is desired to establish the thermal protection system (TPS) materials necessary and currently available to protect the vehicle in the environments. Based on the preliminary screening tests, carbon phenolic ablator was adopted for the heatshield material of the MUSES-C reentry capsule. A cloth-layered carbon phenolic ablator, which is typical material for solid rocket motor nozzles, is refined and developed especially for the heat shield as 'MC-CFRP'. Because the cloth-layered carbon phenolic has especially large allowable stress in the layer-plane, it

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can function not only thermal protection material but also load-carrying material, which is desirable feature for the heatshield material for small capsules with limited weight budget.

The thermal behavior of the ablator is strongly related to the flight environment such as the flow enthalpy, the impact pressure, and the heat flux. The ground heating facility cannot completely duplicate the flight environment. In order to predict the thermal behavior the ablator in the flight environment based on the ground heating data, it is desired to develop a computational analysis code for the ablation phenomena. Also, for designing the TPS at minimum weight, it is indispensable to develop the code with high accuracy in the prediction of the thermo-chemical behavior of the ablator. Thus, we have conducted a series of arcjet experimental studies to verify the ongoing computational design code (Hirai, et al., 1998) for the TPS.

The simulation environment of the ground heating test and flight environment is schematically shown in Fig. 1. The heating test environment of ISAS arc heater (Yamada and Inatani, 2002) is limited both in terms of flow enthalpy and in the size of the test pieces. It is desirable to conduct the tests in other facility for correlating and validating the results. Then a series of heating test in 60 MW IHF (Interaction Heating Facility) at NASA ARC was also conducted at the final stage of the qualification process. The major objective is to confirm the ISAS-test-derived results both from view point of the modeling of thermal responses and mechanical durability such as delamination by use of the higher enthalpy flow of 40 MJ/kg and maximum heating of 20 MW/m² with model size of 70 mm in diameter.

As shown in Fig. 2, by observing the MC-CFRP test piece (T/P) after the arcjet tests, we can easily find that there exist 4 distinct zones in the T/P, that is to say, lost, charred, decomposed and virgin zone, respectively. Thus, the computational code should be able to predict with high accuracy the positions of these boundaries and the temperature response in the virgin zone.

![Fig. 1: Simulation Environment of the Ground Heating Test and the Flight Environment.](image-url)
2. THERMAL ANALYSIS OF ABLATION

2.1 Construction of the Mathematical Model

The procedure of constructing the mathematical model for the MC-CFRP ablation phenomena is shown in Fig. 3.

![Fig. 3: Procedure of Construction of the Mathematical Model for the MC-CFRP Ablation.](image)

Because of the difficulties in measuring the thermal conductivity of the char at high temper-
ature and heat of pyrolysis, we have made use of laser heating tests under the inert atmosphere, where these parameters are determined by one-dimensional calculations so as to reproduce the surface temperature and char thickness obtained at experiments.

After the thermo-chemical parameters are estimated, arc heating tests are conducted to confirm the mathematical model for the ablation including the surface reaction. The schematic of the MC-CFRP T/P configuration is shown in Fig. 4, where the dual sleeve T/P is adopted to prevent the side wall heating and then to assure the precision measurements of the in-depth temperature (Yamada and Inatani, 2002).

![Fig. 4: Schematic of the MC-CFRP T/P Configuration at Arc Heating Experiments.](image)

It is well known that MC-CFRP is thermally expanded when heated. Thus, after each heating test, MC-CFRP T/P is cut in half and the cross section is observed through microscope to estimate the thermal expansion by counting the number of layers. The surface recession due to the surface chemical reaction is determined by subtracting the quantity from the total length change of the T/P.

### 2.2 Assumptions

The carbon phenolic ablator consists of the resin and carbon cloth. (C-H-O system). As the ablator is heated, the resin begins to decompose at sufficiently elevated temperatures, yielding a pyrolysis gas and a carbon residue (namely, 'char'). As the surface temperature increases, surface reaction between the residue and air becomes activated, and surface recession due to the oxidation reaction or sublimation prevails.

The pyrolysis gas products from the in-depth reaction zone transpires through the heated surface and the venting of the gas products into the boundary layer blocks some of the incident convective heat load (namely, 'blocking effect'). The resin decomposes mainly by endothermic chemical reactions, and the resulting pyrolysis gas products absorb further heat as they percolate to the surface through porous, already charred material.
To simulate the above noted ablation phenomena, one-dimensional energy and mass conservation equations are formulated in a finite difference form implicitly in time. In this paper, the following assumptions are made for simplicity to model the ablation phenomena for the carbon phenolic ablator.

1. Chemical composition of the char is pure solid carbon.
2. Chemical reaction between the external flow boundary layer gas and the char occurs only on the surface and this thermo-chemical process causes the surface recession.
3. Pyrolysis gas product of the carbon phenolic resin is in thermal-equilibrium with the char layer as it percolate through the layer, and also it is inert with respect to the boundary layer gas.
4. Lewis number of the boundary layer gas is unity so that the mass transfer coefficient equals the heat transfer coefficient.

2.3 Governing Equations

The in-depth governing equations for one-dimensional ablator thermal response are given in the standard expressions (Moyer, et. al., 1968 & Potts, 1995) as follows. In the present scheme, momentum equations are not included.

\[
\frac{dC_k}{dt} = -A_k \exp \left( -\frac{\Delta E_k}{RT} \right) (1 - C_k)^{n_k}
\]

\[
\rho = \rho_v - (\rho_v - \rho_c) \sum_{k=1}^{N} f_k C_k
\]

\[
\frac{\partial \dot{n}_G}{\partial x} = \frac{\partial \rho}{\partial \tau}
\]

where \(\Delta h_G\) means heat of pyrolysis per unit mass of gas produced and \(\rho, C, k, h\) are the density, specific heat, thermal conductivity and enthalpy of the ablator, respectively, and subscript C, V and G means fully charred state, virgin state, and pyrolysis gas, respectively. Also, \(\tau, x\) are time, in-depth distance from the surface, and the mass flux of the pyrolysis gas.

The momentum equation of the pyrolysis gas should be included for study of the effect of the internal pressure of the pyrolysis gas on the ablator performance such as the delamination, and the attempt is actually under going (Yamada, et. al., 2000). From the standpoint of the accurate estimation of the internal temperature distribution, the present governing equations are enough.

\[
\dot{n}_W = \dot{n}_C + \dot{n}_{G,W}
\]

\[
\dot{q}_{\text{net}} = \dot{q}_{\text{conv}} - \dot{q}_{\text{rad}} - \dot{m}_C(h_w - h_C)
\]

where \(\dot{q}_{\text{conv}}\) means hot wall convective heat rate, which includes the blocking effect.

\[
\dot{q}_{\text{conv}} = g_h(h_{\text{rec}} - h_w)
\]

\[
\dot{q}_{\text{rad}} = \varepsilon \sigma T_w^4
\]
where $g_h$ is the heat transfer coefficient, which is the same value as the mass transfer coefficient on the assumption (4). This $g_h$ value is related to the hypothetical cold wall heat transfer value $g_{ho}$ by the following equations.

$$q_{cw} = g_{ho} h_{rec}$$  \hspace{1cm} (9)

$$g_h = g_{ho} \phi_{blow}$$  \hspace{1cm} (10)

where $\phi_{blow}$ means blocking correction factor. A standard form of this blocking correction factor is shown as follows.

$$\phi_{blow} = \frac{aB_0}{\exp(aB_0) - 1}$$  \hspace{1cm} (11)

$$B_0 = \frac{\dot{m}_W}{g_{ho}} = \frac{\dot{m}_C}{g_{ho}} + \frac{\dot{m}_{GW}}{g_{ho}} = B_{C0} + B_{GW}$$  \hspace{1cm} (12)

where the blowing parameter ($B_0$) can be calculated by using the table ($B_c \sim T_W, P_W$) as will be shown in the next section, and by estimating $B_G$ value timewisely.

2.4 Surface Reaction of the Residue

In the case of carbonaceous ablator, the residue material is assumed pure solid carbon. So, the surface reaction consists in the solid carbon reaction in the air and pyrolysis gas as shown in Fig. 6.

As for the surface recession rate estimation, a bridging formula (Scala’s second order transition scheme (Scala, 1965) is applied to smoothly interpolate the two limiting values listed below.

1)surface mass loss rate due to rate-controlled reaction (chemically non-equilibrium)
2)surface mass loss rate due to diffusion rate controlled reaction (chemically equilibrium)

The graphite, air and the pyrolysis gas reaction kinetics expression is used for the former, while, the chemical equilibrium theory is used for the latter. Especially in the higher enthalpy, the effect of the pyrolysis gas on the recession rate turns out not to be negligible. The chemical composition of the MC-CFRP is C$_{2.85}$N$_{7.54}$O as shown in Fig. 6. As the result of the
surface equilibrium thermochemistry, the equilibrium composition, enthalpy of the gas mixture adjacent to the surface, and the non-dimensional mass loss rate is obtained and tabulated for prescribed values of surface pressure and temperature. To compute recession rate, these resultant tables are interpolated.

Figure 7 shows a typical analytical result of nondimensional recession rate at the impact pressure of 1 atm. It is easily recognized from the figure that
1) the char recession rate in the diffusion-controlled region decreases as the nondimensional pyrolysis gas rate decreases (below 3000°C),
2) the char recession rate in the sublimation region increases as the nondimensional pyrolysis gas rate decreases (below 3000°C).

For the pure solid carbon, the non-dimensional char mass loss rate \( (B_C) \) in air under the assumption (2) is shown in Fig. 10, where the mass loss rate is non-dimensionalized by
the incoming mass flow rate of air. From this figure, we notice that the oxidation reaction prevails in air stream up to the surface temperature around 3000°C, the plateau portion of $B_C$ corresponds to the reaction ($C + O \rightarrow CO$), and sublimation reaction prevails above 3000°C.

2.5 Thermal Decomposition of the Resin

The thermal-decomposition reaction rate of the resin is assumed to follow the Arrhenius kinetic reaction equation and is determined by applying the least square method to reproduce the thermograms of the Thermo-Gravimetric (TG) data. Fig. 8 shows examples of the TG data.

![Figure 8: Examples of Thermogravimetry Curves of MC-CFRP.](image)

The enthalpy and the specific heat of the pyrolysis gas mixture is computed by the equilibrium thermo-chemistry with condensed phase carbon excluded from possible equilibrium products. The chemical formula of the products is assumed to be equal to the elemental composition of the resin minus the percentage of the solid carbon residue.

3. ABLATION ANALYSIS FOR ARCJET MODEL

So far, a series of experimental studies have been carried out in the arcjet facilities of ISAS. The objective is to validate the above noted mathematical model for ablation. This section shows the comparison between the simulation results and experimental results.

3.1 Test Conditions

Due to the test facility limitation, small material samples are tested, which are fabricated in cylinders with 25 or 50mm-diameter. MC-CFRP samples are made of resin-impregnated carbon cloth that is molded in parallel lamina. Test conditions are summarized in Table 1.
Table 1: Arc Heating Test Conditions.

<table>
<thead>
<tr>
<th>Test Piece Material</th>
<th>MC-CFRP or C/C or Graphite</th>
</tr>
</thead>
<tbody>
<tr>
<td>Convective Cold Wall Heat Flux</td>
<td>1 - 12 MW/m²</td>
</tr>
<tr>
<td>Heating Duration</td>
<td>30 sec</td>
</tr>
<tr>
<td>Working Gas</td>
<td>Air</td>
</tr>
<tr>
<td>Flow Bulk Enthalpy</td>
<td>10 - 15 MJ/kg @ISAS</td>
</tr>
<tr>
<td></td>
<td>- 40 MJ/kg @NASA/ARC</td>
</tr>
<tr>
<td>Impact Pressure</td>
<td>60 - 5 kPa</td>
</tr>
</tbody>
</table>

3.2 Measurements

Measurements of model response are made both during the run and after its completion. Internal temperature measurements are continued for 5 minutes after the heating time. These include:

1. Surface Recession
2. Char Thickness
3. Total Mass Loss
4. In-depth and Surface Temperature

Some thermocouples are furnished at various locations from the initial top surface. The positions are determined in the preliminary studies so as to record less than 300°C temperature rise during the heat soak period. The heating rate is measured by calorimeter and the surface temperature by one-color pyrometer. After the heating test, MC-CFRP T/P is observed to be slightly thermally expanded. Thus, by eliminating the thermal expansion effect from the measured surface recession, we extract the surface recession due to the surface reaction.

3.3 Thermal Response of the Carbon Phenolic Ablator

Surface Recession

As described in the previous section the nondimensional char recession rate is strongly correlated to the nondimensional pyrolysis gas rate, which is expressed as

\[ B'_g \equiv \frac{\dot{m}_g}{q_{CW}/h_c} \]

\[ \dot{m}_g = \dot{m}_{Total} - \rho_c \cdot V_{recess} \]

where \( \rho_c \) is the char density and \( V_{recess} \) is the recession volume. The time-averaged values of the above equation can be experimentally obtained by measuring the total weight loss \( \dot{m}_{Total} \) and the recession volume under the assumption of constant recession during the heating tests. A series of arc heating tests have been conducted in ISAS, and the data obtained by utilizing NASA ARC have been added into them. Here it needs to be pointed out that the enthalpy level of IHF in ARC ranges up to 40 MJ/kg as was schematically shown in Fig. 1 or Table. 1, then the nondimensional pyrolysis gas rate in ARC is predicted higher than that in ISAS. Figure 9 shows a categorization of \( B'_g \) data obtained at ISAS and ARC.

Fig. 10 shows the comparison between the B-prime curve (computed non-dimensionalized recession rate by carbon ablation model, see again Fig. 7) and measured surface recession for
several kinds of carbonaceous ablators. Due to the experimental difficulties, the experiments have been carried out under the condition of surface temperature below 3000°C. Although the obtained data show some scatter, the surface recession of the carbonaceous ablators may be approximated by carbon ablation model fairly well.

Figure 11 shows the correlation of the char layer recession rate with parameters of the nondimensional pyrolysis gas blowing rate. The result shows a degree of difference that may be caused by the effects out of the recession modeling. The recession is suggested to be multiplied by 1.6 as an augmentation factor from the standpoint of designing of the thermal protection system. Each reaction rates are calculated under the assumption of chemical equilibrium with respect to the flow impact pressure and enthalpy. The mass flow rate of the ablation products is expressed as non-dimensional gas blowing parameter as a ratio to the mass flow rate of incoming air. They were correlated as a function of the wall temperature on the basis of arc heating test results as shown in Fig 10. The recession data obtained in the heating test are correlated under the assumption that the pyrolysis gas makes chemical reaction with high temperature air in the shock layer.

**Char Penetration Depth**

Fig. 12 shows the comparison of the measured and computed char penetration depth. From this figure, we find that our numerical code can predict the char penetration depth fairly well.

**Blocking Parameters**

From the measured value of total mass loss ($\dot{m}_W$), we have estimated the blowing factor ($B_0$) defined below and then blocking correction factor ($\phi_{blow}$).

$$B_0 = \frac{\dot{m}_W}{\dot{q}_{CW}/h_{rec}}$$  \hspace{1cm} (15)

Fig. 13 shows these two factors. From this figure, it seems that $B_0$ ranges 0.15 ~ 0.35 and 0.8 ~ 0.9 under the arc heating test conditions.
Thermal Responses of Ablator for Reentry Capsule

![Graph showing Bc vs. Wall Temperature Characteristics.](image)

**Fig. 10:** Bc vs. Wall Temperature Characteristics.

![Graph showing Comparison of Bc between Experimentally-obtained and Theory.](image)

**Fig. 11:** Comparison of Bc between Experimentally-obtained and Theory.

**Reaction Front of Pyrolysis**

Fig. 14 shows the measured spatial distribution of peak temperature, where the origin of lateral axis is set equal to the in-depth location of apparent reaction front of pyrolysis, which is obtained from the observation of the post-test T/P. From this figure, the peak temperature at the location is estimated around 300°C, which approximately corresponds to the initiation temperature of pyrolysis from Fig. 8.
In-depth Temperature Response

Fig. 15 shows the comparison of the measured and computed in-depth temperature response. Due to the experimental difficulties, precision temperature measurements have been carried out only at heating rate of 1 MW/m². The shadowed region in the figures represents the uncertainties of T/C locations, which are assumed to be ±0.5mm. From these figures, we can
find that our numerical code can predict the in-depth temperature response fairly well.

Now, precision temperature measurements are planned at heating rate 6 MW/m², which corresponds to the maximum heating condition at 50mm-diameter T/P.

4. DISCUSSIONS

So far, the experimental data for the ablation behavior of MC-CFRP have been obtained in the arcjet heating tests at ISAS and NASA/ARC in the heat flux range of 1 to 12 MW/m².

As for the carbon phenolic ablator used for the MUSES-C reentry capsule (MC-CFRP) surface recession, the carbon ablation model has been confirmed by the arcjet experiments of the non-dimensionalized recession rate versus surface temperature curve in the temperature range up to 3000°C, which corresponds to the diffusion rate controlled region.

As for the in-depth temperature response of the MC-CFRP, it needs approximately 50mm-diameter T/P to prevent the heating from the side walls of the T/P and to make precision measurements of the in-depth temperature. From above written comparison studies, we have found that the experimental results of the in-depth temperature can be reproduced fairly well by our numerical code under the condition of cold wall heating rate of 1 MW/m². The further experimental studies are needed to make the numerical code more reliable.

As for the blocking effect of the MC-CFRP, we have not yet obtained the direct verification of the numerical model. For an indirect approach, the measured surface temperature was compared with the computed radiative equilibrium surface temperature, which is derived from the measured mass loss of the T/P and a tentative mathematical model for the blocking effect from a literature. The experimental results of arcjet tests show that measured surface temperature is approximately equal to the computed radiative equilibrium temperature.

These results are qualitatively equivalent to the results which are obtained in the previous laser heating tests. So far, we have concluded that the numerical model for the blocking is not contradictory to the experimental results of the arcjet heating tests. The blocking effect is supposed to vary with ejected gas species, flow conditions (stagnation point, laminar, or
Another important technical issue is the effect of pyrolysis gas on the ply-lifting of the MC-CFRP. In the present study, the pyrolysis gas product of the MC-CFRP resin is assumed to transpire through the charred zone without any hydrodynamic resistance. However, this pyrolysis gas may participate in the surface reaction. Thus, further investigations are needed for this issue.

5. CONCLUSION

So far, the experimental data for the ablation behavior of the carbon phenolic ablator used for the MUSES-C reentry capsule (MC-CFRP) have been obtained in the arcjet heating tests at ISAS and NASA/ARC at the heat flux range of 1 to 12 MW/m².

The charring ablation computer code for the prediction of the thermal and recessional response of the carbonaceous ablators has been developed and applied to the numerical simulation studies for arcjet heating tests taking account of the effect of the pyrolysis gas on the total recession rate.

REFERENCES


Trajectory-based Coupled Analysis of Viscous Shock Layer and Charring Material Ablation at Stagnation Point

By

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Abstract: To predict the characteristics of the ablation thermal protection for the MUSES-C superorbital re-entry capsule, the trajectory-based analysis of the hypersonic shock layer flow and the ablator interior has been made on the stagnation line in a coupled manner. For the flow analysis, the viscous shock-layer (VSL) equations with the nonequilibrium chemistry of C-O-H-N 26 species are solved. For the ablator, the in-depth thermal response is solved by the charring material ablation (CMA) model. The boundary condition at the surface, which is necessary for both solutions, is determined by the mass and energy balance. Various surface reactions and the surface injection of the pyrolysis gas are considered. The results are compared with the experimental data obtained at the arc-heated wind tunnel. Analysis by using the empirical relations all along the re-entry trajectory is also made and the results are compared with those by the VSL/CMA coupled method. It is shown that such engineering prediction method provides reasonable prediction for the surface temperature and the mass loss of the ablator with sufficient safety margin. The effects of the initial ablator thickness on the ablator temperature are also discussed.

1. INTRODUCTION

Recently, our interest in asteroids have been intensively rising because of its their potential hazard of impact at the earth and the expectation that materials of early solar system still remains in their bodies. From the latter viewpoint, exploration of the asteroids is expected to provide us lots of knowledge about the history of our solar system. One of the most exciting missions to asteroids is the sample return mission, in which a spacecraft flies to an asteroid, takes some sample of the materials of the asteroid body and returns a small capsule carrying the sample container to the earth. For simplicity of the spacecraft system and reduction in the mission cost, it is necessary to establish the technology of the superorbital re-entry, where a re-entry capsule enters the Earth's atmosphere not from the circular orbit but directly from the interplanetary trajectory at superorbital velocity to save the spacecraft fuel for orbit insertion.

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At the Institute of Space and Astronautical Science (ISAS), Japan, the asteroid sample return mission called "MUSES-C" (ISAS 1995) is in development aiming for launch in 2003. In this mission, a 20 kg small blunt cone capsule is used and its nominal entry velocity is 12.5 km/s at altitude 100 km.

For the thermal protection of the spacecraft that does not need to be reusable, the ablator is suitable from a viewpoint of reliability and mass budget. To protect the payload from the severe aerodynamic heating of superorbital re-entry, the ablator of the carbon-phenolic type is planned to be used in the MUSES-C mission. Under severe aerodynamic heating of re-entry flight, the temperature rises inside the ablator and decomposition of the resin creates the char layer under the surface. At the ablator surface, both the gas products of the surface reactions of the char layer and the pyrolysis gas from the ablator interior are injected into the shock layer flow to reduce the aerodynamic heating. In the case of the MUSES-C capsule, the peak stagnation aerodynamic heating is expected to occur around 60 km altitude, where the shock layer flow is in fully chemical nonequilibrium. Consequently, for evaluation of the heat shield performance we must consider the chemical nonequilibrium of the freestream air and the ablation gas in the shock layer with an appropriate model for the finite-rate chemical reactions at the surface and the injection of the pyrolysis gas. In this study, we use the viscous shock-layer flow with the nonequilibrium chemistry of the 26 carbon-oxygen-nitrogen-hydrogen species to consider the mixture of the air-species, products of the surface reactions and the pyrolysis gas.

Prediction of the aerodynamic heating environment with ablation in the stagnation region is made by the viscous shock-layer (VSL) analysis with small computation time (Suzuki et al. 1998). However, the surface temperature and the injection rate of the pyrolysis gas must be calculated by other method. For the ablator interior, the charring material ablation (CMA) analysis is known to be an useful prediction method. In the existing CMA analysis method, however, the wall heating rate and the rate of surface mass loss by the chemical reactions at the char layer surface must be assumed using some empirical relations (Hirai et al. 1998). For the char removal, the surface mass loss rate is known to be successfully described by the Metzger's relation (Metzger et al. 1967). This relation is, however, an empirical one based on the experimental data and the shock layer flow should be numerically solved with an appropriate surface model for more accurate prediction considering the chemical nonequilibrium effects. Consequently, to analyze self-consistently the aero-thermodynamic environment of the re-entry capsule with ablation, we need to use the trajectory-based coupled method, in which the VSL analysis and the CMA analysis are simultaneously made by matching both solutions at the ablator surface.

The objectives of the present study are: 1) to make the trajectory-based analysis of the ablation thermal protection for the superorbital re-entry capsule by the coupled analysis of the shock layer flow and ablator interior, 2) to assess the accuracy of the present model by comparisons with the experimental data and results of the empirical relations, and 3) to predict the time history of the ablator temperature and ablator mass loss along the re-entry trajectory of the MUSES-C capsule. The analyses are made only on the stagnation line, where the most severe aerodynamic heating occurs.

2. METHOD OF ANALYSIS

The concept of the coupled analysis is illustrated in Fig. 1. In the present study, the flow analysis is made by solving the VSL equations. The flow is assumed to be steady at any time,
since the time scale for the flow is much smaller than that for the thermal response of the ablator and for the change in the freestream conditions along the entry trajectory. For the unsteady thermal response of the ablator interior, the CMA analysis is made. An appropriate ablator surface model is necessary to couple the VSL analysis and the CMA analysis.

Fig. 1: Concept of Coupled Analysis of Shock Layer Flow and Ablator Interior.

2.1 VSL Analysis

The axi-symmetric viscous shock-layer (VSL) equations with chemical nonequilibrium and thermal equilibrium are solved on the stagnation line by the finite difference method. The flow is assumed to be laminar. The formulation and the solution method are the same as those by Moss (1974). In the present study, we consider the chemical nonequilibrium model of the 26 species (N₂, O₂, N, O, NO, NO⁺, e⁻, N⁺, O⁺, N₂⁺, O₂⁺, C, C₂, C₃, CO₂, CO, CN, CO⁺, C⁺, H, H₂, HCN, HCO, C₂H₂, C₂HandCH). The detail of the gas model is given by Suzuki et al. (1998). The effects of radiation of high temperature gas are not considered in the flow analysis. The radiative heating, however, is predicted by the empirical relation (Tauber et al., 1991) and included in the aerodynamic heating on the ablator surface. In the VSL analysis, the computational domain is bounded by the shock wave surface and the wall surface. The conditions just behind the shock wave are determined from the entry trajectory and atmospheric model by using the Rankine-Hugoniot relations with frozen chemistry. The wall conditions are determined by the ablator surface model discussed later. The computational domain is divided into 101 grid points. The grid is clustered to the shock wave and to the wall with the minimum spacing 3 x 10⁻⁴ of the shock standoff distance.

2.2 CMA Analysis

In the present study, we consider the CFRP ablator, which is composed of the C, H and O elements. When the surface of the ablator is heated and the temperature of the ablator interior
rises, the resin begins to decompose into the pyrolysis gas and the char layer of pure carbon. Hence, the ablator interior is divided into three zones, that is, the char layer, reaction zone and the virgin zone, as shown in Fig. 1. As the heating continues, the reaction zone penetrates deep in the ablator. The pyrolysis gas transpires through the porous char layer and is injected into the shock layer. At the surface of the char layer, the temperature becomes very high and the recession occurs due to the char removal by oxidation and sublimation of solid carbon.

The governing equations for the in-depth ablator thermal response with pyrolysis effects are given on the coordinate moving with the recessed surface as follows (Clever et al. 1975; Hirai et al. 1998):

\[
\rho \cdot C_p \frac{DT}{Dt} = \frac{\partial}{\partial x} \left( k \frac{\partial T}{\partial x} \right) + \Delta h \frac{\partial \rho}{\partial t} + (S \rho C_p + m_p C_{p_p}) \frac{\partial T}{\partial x},
\]

\[
\frac{D}{Dt} = \frac{\partial}{\partial t} + S \frac{\partial}{\partial x},
\]

\[
m_p(x) = - \int_x^{base} \frac{\partial \rho}{\partial t} \, dx,
\]

where \( x, m_p \) and \( S \) are the depth measured from the surface, the mass flux of the pyrolysis gas and the speed of the surface recession, respectively. The temperature of the pyrolysis gas is assumed to be equilibrated with the char layer temperature (\( T \)), since the transpiration velocity of the pyrolysis gas is quite small. In the equation (3), the mass flux of the pyrolysis gas is equated with the rate of change in the ablator density (\( \rho \)), since we assume no transit time of the pyrolysis gas in the char layer. The chemical reactions of the pyrolysis gas with the char layer are not considered. At the base plate (see Fig. 1), the adiabatic wall condition is assumed. The surface temperature is given by the ablator surface model. The ablator interior is divided into 101 grid points. The equation (1) is solved by the Crank-Nicholson method with the second order accuracy in time.

As the typical properties of the CFRP ablator, we set the model after Clever et al. (1975) and Potts (1995). The density of the virgin material (\( \rho_v \)) and char (\( \rho_c \)) are 1448 kg/m\(^3\) and 1185 kg/m\(^3\), respectively. The thermal conductivity (\( k \)) and specific heat (\( C_p \)) are given in the functions of the temperature and density. For simplicity of analysis, the heat of resin decomposition (\( \Delta h \)) and the specific heat of pyrolysis gas (\( C_{p_p} \)) are set to be constant at \( 2.16 \times 10^7 \) J/kg and \( 1670 \) J/kg-s, respectively.

The rate of the thermal decomposition of the resin is given by the Arrhenius-type equation (Clever et al. 1975):

\[
\frac{\partial (\rho/\rho_v)}{\partial t} = - \left( \frac{\rho - \rho_c}{\rho_v} \right)^2 \cdot \sum_{i=1}^{3} A_i \exp \left( -E_i / RT \right).
\]

By assuming the temperature rise at constant rate, the thermogravimetry curve (variation of degree of decomposition with temperature) is obtained as shown in Fig. 2, which seems similar to the experimental data (Hirai et al. 1998).

### 2.3 Model of Ablator Surface

The ablator surface model is introduced to determine the composition of the gas at the wall, surface recession rate, ablation injection velocity and the wall temperature.
The mass fraction of the $i$-th species at the wall is determined by solving the surface mass balance equation written in the generalized form:

$$- \rho D_i \cdot \frac{dC_i}{dy} + \rho \cdot C_i \cdot v_w = J_i^{\text{pyro}} + J_i^{\text{sub}} + J_i^{\text{cat}},$$

where $\rho$, $C_i$ and $D_i$ are the density of gas mixture, the mass fraction and the diffusion coefficient of the $i$-th species, respectively. The first and the second terms of the left hand side represent the normal ($\gamma$-direction) fluxes due to diffusion and ablation injection velocity ($v_w$), respectively. On the right hand side, we consider the injection of the pyrolysis gas, the sublimation and oxidation of carbon, and the catalytic recombination reactions.

The total injection rate of the pyrolysis gas is determined from the CMA analysis. The pyrolysis gas is expected to be in chemical equilibrium, since the velocity of transpiration in the char layer is quite low. Then the composition of the pyrolysis gas is calculated from the equilibrium constant (JANAF Thermochemical Tables 1985). The CFRP pyrolysis gas is assumed to be composed of 59% C, 13% H and 28% O elements (Hirai et al. 1998).

The surface mass flux due to sublimation is given by the Hertz-Knudsen-Langmuir relation (Blottner 1970):

$$J_i^{\text{sub}} = \alpha_i \cdot (p_{e,i} - p_i) / \sqrt{2 \pi (R/M_i) T_w}, \quad (i = C_3)$$

$$J_i^{\text{sub}} = 0, \quad (i = \text{others})$$

where $R$, $T_w$, $M_i$, $p_{e,i}$ and $p_i$ are the universal gas constant, wall temperature, molecular weight, equilibrium vapor pressure and partial pressure of $i$-th species, respectively. The major species in the product of sublimation of carbon is $C_3$ molecule (Blottner 1970; Suzuki et al. 1998). Other species are ignored.

The following oxidation reactions are considered:

$$C(\text{solid}) + O \rightarrow CO,$$

$$C(\text{solid}) + O_2 \rightarrow CO + O.$$
The mass flux due to the surface oxidation $J_{i}^{\text{oxi}}$ is given after Blottner (1970):

$$
J_{i}^{\text{oxi}} = -\rho \cdot C_i \cdot \gamma_{i}^{\text{oxi}} \sqrt{RT_w / 2\pi M_i}, \quad (i = O, O_2)
$$

(10)

$$
J_{i}^{\text{CO}} = -J_{O}^{\text{oxi}} \frac{M_{CO}}{M_{O}} - J_{O_2}^{\text{oxi}} \frac{M_{CO}}{M_{O_2}}
$$

(11)

$$
J_{i}^{\text{oxi}} = 0, \quad (i = \text{others})
$$

(12)

where $\gamma_{i}^{\text{oxi}}$ represents the reaction probability with respect to the mass flux of the $i$-th species gas striking a unit area of the wall per unit time.

The mass fluxes in relation to the recombination of $O_2$ are given by the finite catalytic wall model (Suzuki et al. 1998):

$$
J_{i}^{\text{cat}(O_2)} = -\rho \cdot C_i \cdot \gamma_{i}^{\text{cat}(O_2)} \sqrt{RT_w / 2\pi M_i}, \quad (i = O)
$$

(13)

$$
J_{O_2}^{\text{cat}(O_2)} = -J_{O}^{\text{cat}(O_2)}
$$

(14)

The reaction probability for the oxidation and $O_2$ recombination are given by Park’s model (Park 1976) as a function of the wall temperature.

The ablation injection velocity at the wall ($v_w$) is determined by the surface mass loss rate ($m_w$), the pyrolysis rate ($m_p$) and the gas density at the wall ($\rho_w$):

$$
m_w + m_p = J_{C_3}^{\text{sub}} - J_{O}^{\text{oxi}} \frac{M_{C}}{M_{O}} - J_{O_2}^{\text{oxi}} \frac{M_{C}}{M_{O_2}} + \sum_{r=\text{pyro}} J_{r}^{\text{pyro}} = \rho_w v_w
$$

(15)

To obtain the nonequilibrium composition and injection velocity of ablation gas at the surface, the equations (5) and (15) are solved simultaneously by iterative procedure.

The net convective heating to ablator surface is defined by considering the energy transport by ablation injection as:

$$
q_{\text{conv,net}} = k \left( \frac{\partial T}{\partial y} \right)_w + \sum_{i} \left( \rho_i \cdot C_i \cdot \frac{\partial T}{\partial y} \right)_w
$$

$$
- \{ (m_w + m_p) \cdot h_w - (J_{i}^{\text{oxi}} \cdot h_{CO} + J_{O}^{\text{oxi}} \cdot h_{O} + J_{O_2}^{\text{oxi}} \cdot h_{O_2}) - J_{C_3}^{\text{sub}} \cdot h_{C_3} - m_p h_{\text{pyro}} \}
$$

(16)

The wall temperature is calculated by considering the energy balance at the surface considering the heat sink capability of the ablator, aerodynamic heating (convective and radiative), heat of phase change at the surface and radiative cooling as:

$$
-k \frac{\partial T}{\partial x} = q_{\text{conv,net}} + q_{\text{rad}} - (m_{\text{oxi}} \Delta h_{\text{oxi}} + m_{\text{sub}} \Delta h_{\text{sub}}) - q\epsilon T_w^4
$$

(17)

where the emissivity of the ablator surface $\epsilon$ is set as 0.725 (Suzuki et al. 1998). The radiative heating rate ($q_{\text{rad}}$) is considered by using the empirical relation (Tauber et al., 1991). In the present case, the radiative heating is much smaller than the convective heating.

### 2.4 Coupled Analysis along Re-Entry Trajectory

Figure 3 shows the method of the VSL/CMA coupled analysis. When the computation proceeds to a new time step, the freestream condition is determined by the trajectory data and the atmospheric model. First, we guess the wall temperature, pressure and the injection rate of
the pyrolysis gas and then the composition of the pyrolysis gas injected into the shock layer is computed by assuming the chemical equilibrium. By using these quantities, both the shock layer flow and the surface equations are iteratively solved in a coupled manner. After the wall condition is calculated, the thermal response of the ablator interior is solved. This procedure is repeated until convergence in the injection rate of the pyrolysis gas is obtained. After that, computation proceeds to the next time step.

Figure 4 shows the nominal reentry trajectory of the MUSES-C sample-return capsule. The horizontal axis represents the time from the flight altitude at 200 km. The MUSES-C capsule has an axi-symmetric blunt cone configuration with nose radius $R_n$ of 0.2 m, semiapex angle of 45 deg and base diameter of 0.4 m, as shown in the figure. The computation is done along this trajectory until time 160 sec, at which the flight velocity decreases to 1.4 km/s, with the step size of 0.5 sec. The computation time for each case is about 20 minutes on Fujitsu VPP-800 supercomputer. Note that the present VSL analysis is valid at altitude lower than 75 km, since we assume the continuum flow and the thermal equilibrium. This means that the aerodynamic heating and the surface mass loss at higher altitude should be predicted by other methods. For computation in the early stage of the re-entry flight (before time 80 sec in Fig. 4), we use Tauber's relation (Tauber et al., 1990) for the convective heating at the stagnation point, Metzger's relation (Metzger et al. 1967) for the mass loss rate of the ablator surface and Marvin's relation (Marvin et al. 1967) for heating reduction effect by the ablation injection. Note that discontinuity in the convective heating rate might occur at the flight time when we start the VSL analysis, since the heating rate by the empirical relation is close to that at the fully catalytic wall and it is expected to be significantly larger than the result by the VSL analysis considering the finite-rate surface reactions. To avoid such discrepancy, the convective heating rate by Tauber's relation is multiplied by a factor 0.5 in the present study. However, this modification is not made when these empirical relations are used all along the re-entry trajectory.

3. RESULTS AND DISCUSSION

3.1 Comparison with Experimental Data of Arc-Heated Wind Tunnel

To check the validity of the present analysis, we compare the computed results with the experimental data of heating test of the ablator in the arc-heated wind tunnel (Hirai et al. 1998). The experiments were conducted by using the 500 kW arc-heated test facility at ISAS (Inatani et al. 1995). The stagnation pressure and the maximum total enthalpy of the flow are 0.05 MPa and 15 MJ/kg, respectively. The test piece of the ablator having circular cylinder shape with 50 mm diameter and 40 mm thickness is put in the flow. For the numerical analysis of the flow around the test piece, the freestream condition must be exactly specified. In such high enthalpy test facilities, however, it is quite difficult to measure all the quantities of the freestream accurately. In the present study, we estimate these values by using the experimental data of the LIF velocimetry (Inatani et al. 1995) and assuming the chemical equilibrium as shown in Table 1.

Figure 5 shows the variation of the surface mass loss rate with the wall temperature for graphite. In this case, the pyrolysis gas is not produced and the mass loss occurs only at the surface. The extent of the mass loss rate is evaluated by the non-dimensional blowing parameter defined as:

$$B = \frac{m_{\text{char}} H_i}{q_{cw}},$$  \hspace{1cm} (18)
where $m_{\text{char}}$, $H_t$ and $q_{\text{cw}}$ are the surface mass loss rate, total enthalpy of the flow and the cold wall heat flux, respectively. The cold wall heat flux is defined as the stagnation-point convective heating rate on highly cooled surface and is almost the same as the heating rate on the fully catalytic wall without ablation. The computational results are compared with the experimental data and the empirical prediction by Metzger’s relation (Metzger et al. 1967) in the figure. They are qualitatively and quantitatively in good agreement. It should be noted that the computational results show that the surface mass loss rate depends not only on the wall temperature but also on the total enthalpy of the flow. However, such tendency is not described in the empirical relation. The process of the surface mass loss is divided into three regimes with respect to the wall temperature, that is, the reaction-controlled regime, diffusion-controlled regime and sublimation regime. At the wall temperature less than 3000 K, the surface oxidation reactions (see equations (8) and (9)) are dominant. When the wall temperature is less than 1500 K, the mass loss is in the reaction-controlled regime and its rate increases with the wall temperature because the reaction probability of the surface oxidation increases as shown in Fig. 6. The effect of surface oxidation by oxygen molecule is expected to be negligibly small, since the probability of this reaction is quite small. When the wall temperature is between 1500 K and 3000 K, the mass loss is in the diffusion-controlled regime and the rate is almost constant, since the extent of the surface oxidation is limited by the amount of the mass flux of the atomic oxygen transported to the surface by diffusion. At the wall temperature higher
than 3000 K, the sublimation becomes significant and the mass loss rate rapidly increases with the wall temperature.

Figure 7 shows the variation of the total mass loss per unit area with the cold wall heat flux. In this case, the material is the CFRP ablator and the pyrolysis gas is considered. The density of the virgin material and char of the ablator are set as 1330 kg/m³ and 1070 kg/m³, respectively. The duration of heating is 30 sec. The computational results are slightly smaller than the experimental data. The major possible reasons for this discrepancy are 1) inaccuracy in the estimation of the freestream condition of the arc-heated wind tunnel, and 2) occurrence of spallation in the experiments which enhances the mass loss of the ablator by injecting spalled particles into the flow.
3.2 Trajectory-base Analysis for MUSES-C Capsule

Figure 8 shows the time history of the surface and base temperature of the ablator. Three cases for the initial ablator thickness (10 mm, 20 mm and 30 mm) are calculated. The maximum surface temperature is predicted as 3000 K by the VSL/CMA coupled analysis. This indicates that sublimation is not significant even at the peak surface temperature. The result of the surface temperature obtained by using the empirical relations all along the trajectory for the
initial ablator thickness 30 mm is also plotted as a dashed line in the figure. In the latter case, the maximum surface temperature is about 3500 K and higher than the VSL/CMA result, since the cold wall heat flux is assumed in the latter case. However, they are qualitatively in good agreement. Comparison between the results by the VSL/CMA analysis and the empirical relations indicates that the latter gives us reasonable prediction in the sense that the results have some amount of safety margin. The temperature of the ablator surface hardly depends on the initial ablator thickness, while the temperature at the ablator base strongly depends on the ablator thickness. In the case of 10 mm initial ablator thickness, the bottom temperature gradually rises up to 800 K due to insufficient thermal insulation. In the present case, the ablator thickness of 30 mm is enough to keep the temperature rise at the base less than 50 K. The time for the peak of the surface temperature almost coincides with that of the wall heating rate. However, rise in the bottom temperature is significantly delayed due to poor thermal conductivity of the resin material and it reaches the peak after the hypersonic flight. The temperature distribution both in the shock layer flow and in the ablator interior at time 100 sec is shown in Fig. 9. The density distribution inside the ablator is also plotted. The temperature significantly decreases from the surface to the bottom of the ablator. The reaction zone of the ablator is clearly recognized as the region where the rapid change in its density occurs.

Figure 10 shows the time history of the ablator mass loss rate per unit area and the surface recession. The initial ablator thickness is set as 30 mm. The maximum mass loss rate due to the pyrolysis gas is almost the same as that by the removal of the char layer at the ablator surface. This fact indicates that the surface injection of the pyrolysis gas should not be neglected for the ablation analysis and that the coupled analysis is necessary for such atmospheric entry simulation, since the injection rate of the pyrolysis cannot be determined by the flow analysis only. The peak of the pyrolysis is observed before the peak of the char removal, which occurs around the peak aerodynamic heating. After the peak, the char removal rapidly decreases, while the mass loss by pyrolysis gradually decreases and still exists after the recession of the surface.
has stopped. The temporal variation of the composition of the pyrolysis gas at the surface is shown in Fig. 11. The chemical equilibrium is assumed. During the severe aerodynamic heating, the major species are CO, C2H, H, C2H2, H2 and C3. The mass fraction of H2O is negligibly small. Consequently, H2O is not considered in the present study.

Penetration of the reaction zone and the char layer in the ablator with respect to time is illustrated in Fig. 12. In this figure, the locations, at which the ablator density is \( \rho_a + 0.05 \times (\rho_0 - \rho_a) \) and \( \rho_a + 0.95 \times (\rho_0 - \rho_a) \), are defined as the boundary between the char layer and the reaction zone and the boundary between the reaction zone and the virgin layer, respectively. In the present analysis, the total recession depth and the final thickness of the char layer are
predicted as about 1 mm and 4 mm, respectively. Consequently, the initial ablator thickness of 30 mm is sufficient to survive through the re-entry flight of the MUSES-C capsule. Note that the reaction zone continues to penetrate into the ablator even after the penetration of the char layer has stopped, since the temperature rise inside the ablator is significantly delayed as discussed in Fig. 8. To predict the final depth of the reaction zone in the ablator and the maximum temperature at the ablator bottom, computation must be continued for very long
time after the flight regime of significant aerodynamic heating.

Figure 13 shows the time history of the surface recession by the VSL/CMA analysis and the empirical relations. Before time 120 sec, their difference is negligibly small. After that, however, the result by the empirical relations indicates that the recession still continues. In the VSL/CMA analysis, the surface oxidation is assumed to be caused mainly by atomic oxygen and it disappears when the flight velocity decreases and the temperature in the shock layer is not high enough to excite the dissociation of oxygen molecule.

![Graph showing surface recession](image)

Fig. 12: Penetration of Reaction Zone and Char Layer in Ablator.

![Graph showing time history of surface recession](image)

Fig. 13: Time History of Surface Recession of Ablator.
4. CONCLUDING REMARKS

(1) The trajectory-based analysis of the ablation thermal protection in the stagnation region for the MUSES-C superorbital re-entry capsule has been made self-consistently by the coupled analysis of the chemical nonequilibrium viscous shock-layer (VSL) flow and the thermal response of the charring material ablator (CMA).

(2) Computation is made to simulate the heating test in the arc-heated wind tunnel. The results are in good agreement with the experimental data.

(3) The trajectory-based ablation analysis is made for the MUSES-C capsule both by using the VSL/CMA coupled method and by using the empirical relations all along the trajectory. Their results are qualitatively in good agreement. However, the latter gives higher peak temperature and mass loss rate. Consequently, the empirical prediction seems reasonable in the sense that the results have some amount of safety margin.

(4) The surface temperature of the ablator is not sensitive to the ablator thickness, while the bottom temperature strongly depends on the ablator thickness. In the case of the MUSES-C capsule, the initial thickness of the ablator should be as large as 30 mm.

(5) The rise in the bottom temperature and the penetration of the reaction zone into the ablator still continue after the peak of the aerodynamic heating and the surface temperature.

(6) In the case of MUSES-C capsule, the maximum mass loss rate due to the pyrolysis gas is almost the same as that by the removal of the char layer. The surface injection of the pyrolysis gas should not be neglected for the ablation analysis and the coupled analysis is necessary for trajectory-based ablation analysis to determine the injection rate of the pyrolysis gas.

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An Integrated Numerical Method for Predicting Aerodynamic Heating Environment for MUSES-C Entry Capsule with Ablation

By

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Abstract: An integrated numerical method to obtain trajectory-based aerodynamic heating environment for entry probe vehicles with ablation is briefly described. In this method, a thermochemical nonequilibrium CFD code is loosely coupled with an ablation module along the entry trajectory. Radiative heating is estimated by the use of multi-band radiation module. The effect of earlier turbulent transition of the boundary layer due to ablation product gas is included. Calculated results for MUSES-C sample return capsule are shown and that are compared with the results of existing analysis.

1. INTRODUCTION

Ablation of heatshield during entry flight is essentially a time dependent phenomenon because the amount of ablation is affected by the net heating rate that varies along the entry trajectory. Furthermore, the net heating rate itself is critically affected by ablation because ablation reduces the convective heating rate by so-called convective blockage effect. Therefore, the accurate prediction of heating environment with ablation is only available when a coupled analysis of flowfield and thermal response of ablator is accomplished along the entry trajectory.

The situation can be further complicated if we consider strongly radiating flowfield such as that occurred in the entry flight of Galileo probe vehicle. Emission and absorption of radiation change temperature of the flowfield that alters the property of radiation, and thus constitutes a coupling problem. Shape change of heatshield due to surface recession can have influence over heating rate. This in turn changes the recession rate. These are typical examples that show how various issues are closely related to each other, and why a trajectory-based aerodynamic heating problem should be solved iteratively by an integrated numerical method.

The flow features that should be considered in the integrated numerical method are (i) thermochemical nonequilibrium of high temperature flowfield with ablation products, (ii) thermal response of ablative heatshield, (iii) radiative heat transfer coupled with the flowfield, (iv) shape

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change of heatshield due to surface recession, (v) effect of spalled particles, and (vi) injection induced turbulence due to ablation product gas (Suzuki T. et al. 2002). For example, the analysis of Galileo entry flowfield requires all of these flow features to be fully considered. In case of the Pioneer-Venus probe vehicles, those (iv) and (v) and coupling effect of the flowfield and radiation in (iii) may be neglected. However, our previous study has indicated that (vi) should be taken into account for accurate reproduction of flight data (Takahashi & Sawada 2002). The required flow features for an analysis of MUSES-C reentry flight may be the same as those for Pioneer-Venus case, because the entry velocity and entry angle are similar.

Validation of analysis code is very important for future design of ablative heatshield. In order to accomplish this, the flight data obtained in the past entry flight, such as Apollo entry capsule, Pioneer-Venus entry probes, and Galileo probe provide unique opportunities for validation (Park 1999). Some of the results in our previous studies for developing an integrated analysis method for entry flight heating environment and its validation can be found in references (Izawa & Sawada 2000a; Suzuki T. et al. 2002; Takahashi & Sawada 2002). In this work, we first give a brief description of numerical methods that constitute the integrated analysis code. We then show the calculated results for MUSES-C entry flight, and give the related discussions.

2. NUMERICAL METHODS

2.1 Outline of CFD Module

In the computational fluid dynamics (CFD) module, we solve the Navier-Stokes equations that account for thermochemical nonequilibrium reactions. Because we must deal with two different gas flows, i.e., the freestream gas and the ablation product gas, that have different chemical compositions, we cannot assume a constant elemental ratio of freestream gas (Park 1989) to the entire computational domain. Therefore, we instead employ the concept of elemental density conservation equations proposed by Hassan et al. (1992).

In the analysis of MUSES-C entry flight, we assume 11 chemical species for high temperature air (N, O, N₂, O₂, NO, N⁺, O⁺, N₂⁺, O₂⁺, NO⁺, e⁻), and 10 chemical species for ablation product gas (C, C₂, C₃, CO, CN, C⁺, H, H₂, C₂H, H⁺). Therefore, we have 21 species conservation equations in total. Instead of solving all of these species conservation equations, we introduce conservation equations for those elemental densities of \( \rho_N, \rho_O, \rho_C, \rho_H \), and properly remove the same number of species conservation equations. These elemental densities are related to the densities of chemical species by the following equations:

\[
\frac{\dot{\rho}_N}{M_N} = \frac{\rho_N}{M_N} + 2 \frac{\rho_{N_2}}{M_{N_2}} + \frac{\rho_{NO}}{M_{NO}} + \frac{\rho_{N^+}}{M_{N^+}} + \frac{\rho_{N_{2^+}}}{M_{N_{2^+}}} + \frac{\rho_{NO^+}}{M_{NO^+}} + \frac{\rho_{CN}}{M_{CN}},
\]

\[
\frac{\dot{\rho}_O}{M_O} = \frac{\rho_O}{M_O} + 2 \frac{\rho_{O_2}}{M_{O_2}} + \frac{\rho_{NO}}{M_{NO}} + \frac{\rho_{O^+}}{M_{O^+}} + \frac{\rho_{O_{2^+}}}{M_{O_{2^+}}} + \frac{\rho_{NO^+}}{M_{NO^+}} + \frac{\rho_{CO}}{M_{CO}},
\]

\[
\frac{\dot{\rho}_C}{M_C} = \frac{\rho_C}{M_C} + 2 \frac{\rho_{C_2}}{M_{C_2}} + \frac{\rho_{CN}}{M_{CN}} + \frac{\rho_{CO}}{M_{CO}} + \frac{\rho_{C_3}}{M_{C_3}} + \frac{\rho_{C^+}}{M_{C^+}} + \frac{\rho_{C_2H}}{M_{C_2H}},
\]

\[
\frac{\dot{\rho}_H}{M_H} = \frac{\rho_H}{M_H} + 2 \frac{\rho_{H_2}}{M_{H_2}} + \frac{\rho_{C_2H}}{M_{C_2H}} + \frac{\rho_{H^+}}{M_{H^+}}.
\]

Note that these are simple statements that number density of each element is the same. The total density \( \rho \) is given by a sum of these elemental densities as

\[
\rho = \dot{\rho}_N + \dot{\rho}_O + \dot{\rho}_C + \dot{\rho}_H.
\]
One can show that the governing equations for these elemental densities can be written as

\[
\frac{\partial \tilde{\rho}_s}{\partial t} + \frac{\partial}{\partial x_j} \left( \tilde{\rho}_s u_j \right) = \frac{\partial}{\partial x_j} \left( \rho D \frac{\partial \tilde{c}_s}{\partial x_j} \right),
\]

where \( s \) denotes the elemental species, \( u_j \) the mass averaged velocity component, \( D \) the diffusion coefficient, and \( \tilde{c}_s \) the mass fraction of elemental species \( s \).

We now have 21 species conservation equations, 4 elemental density conservation equations, and a total density conservation equation. We remove 4 species density conservation equations for \( N_2, O_2, C_2, H_2 \), and retain 4 conservation equations for elemental densities. These densities of removed chemical species conservation equations can be found by solving Eq. (1). Moreover, we remove the total density conservation equation because we have Eq. (2). Therefore, we solve 17 species conservation equations and 4 elemental density conservation equations for 21 chemical species. We also solve two momentum equations, a total energy equation, and a vibrational-electronic energy conservation equation. All of these conservation equations are independent to each other.

Park’s two-temperature model (Park 1989) is employed to describe the nonequilibrium thermochemical states. The molecular viscosity for each chemical species is either given by Blottner’s model (Blottner et al. 1971) or by a curve-fit. The thermal conductivity is given by Eucken’s relation (Vincenti & Kruger 1967). The total viscosity and conductivity are calculated using Wilke’s empirical mixing formula (Wilke 1950). We assume the diffusion coefficient to be constant for all species with a constant Schmidt number of 0.5.

The governing equations are integrated by the cell-centered finite volume scheme utilizing AUSM-DV numerical flux (Wada & Liou 1994) and MUSCL approach for attaining higher order spatial accuracy. In the time integration, the LU-SGS algorithm is employed for obtaining a faster convergence, which is combined with the diagonal point implicit method (Eberhardt & Imlay 1992) for stability maintenance of reaction terms.

2.2 Outline of Material Thermal Response Code

Ahn et al. (2002) developed a computer code named Super Charring Materials Ablation (SCMA) in order to study the thermal response of ablator for Pioneer-Venus entry capsules. In the SCMA code, four conservation equations, i.e., conservation of solid density, pyrolysis gas density, momentum of pyrolysis gas, and total energy are solved one-dimensionally.

A unique feature of the present SCMA code is that it solves the momentum equation of pyrolysis gas. Therefore, one can include various phenomena related to gas motions, such as friction and inertial forces. This is in contrast with the existing CMA code or HBI method where steady flow assumption was made to eliminate this equation. Details of the code and results of the code validation study can be available in the reference.

2.3 Surface Reactions

In the MUSES-C entry flight calculation, we consider the following four types of chemical reactions that occur at the ablator surface: (i) surface oxidation that produces CO, (ii) surface nitridation that produces CN, (iii) surface sublimation that produces C\(_3\), and (iv) recombination of ions and electrons. The concentration of surface reacting species can be found by solving the relation that the production rate at the wall balances with diffusion (Suzuki T. et al. 2002).
2.4 Radiation Module

The radiative heat flux at the wall surface is computed from the obtained flowfield by solving the radiative transfer equation with the tangent-slab approximation. The emission and absorption coefficients are first calculated using a multiband model that considers $O(10^3)$ wavelength points. For example, in the analysis of MUSES-C entry capsule, 2294 wavelength points are chosen for air and 7610 wavelength points for ablation product gas in the wavelength range from 750 to 15000 Å. The absorption coefficient of the gas mixture is expressed as a sum of those for individual species in the form of (Park & Milos 1990)

$$\kappa_\lambda = \sum_s n_s \sigma_{\lambda_s},$$  \hspace{1cm} (4)

in which $n_s$ denotes the number density and $\sigma_{\lambda_s}$ the absorption cross section at given wavelength value for chemical species $s$. The cross section value is curve fitted using five parameters in the form of

$$\sigma_{\lambda_s} = \exp \left( \frac{A_{\lambda_1}}{z} + A_{\lambda_2} + A_{\lambda_3} \ln z + A_{\lambda_4} z + A_{\lambda_5} z^2 \right),$$  \hspace{1cm} (5)

where $z = 10000/T_v$ with $T_v$ denoting the vibrational temperature.

2.5 Turbulence Model

Ablation product gas that is injected into the boundary layer from the wall surface is assumed as inherently turbulent according to the concept of injection-induced turbulence that was proposed by Park (1984). The employed turbulence model can be written as a sum of eddy viscosity given by the Park’s model and also the turbulence model of Baldwin & Lomax (1978) as

$$\mu_t = (\mu_t)_{inj} + (\mu_t)_{BL}.$$  \hspace{1cm} (6)

The first term is given by

$$(\mu_t)_{inj} = \rho d^2 |\omega|,$$  \hspace{1cm} (7)

where $\omega$ denotes vorticity. The mixing length $d$ is defined by

$$d = \max (0, d_w - 0.4y) \exp \left( -y^+ / A^+ \right),$$  \hspace{1cm} (8)

in which $d_w$ gives the wall mixing length that takes the maximum value at the wall, and decays exponentially in the boundary layer according to Van Driest theory (Van Driest 1956). The subscript $w$ indicates the wall value. The wall mixing length $d_w$ is given by the relation

$$(\mu_t)_{Park} = \rho_w d_w^2 |\omega|_w$$  \hspace{1cm} (9)

2.6 Coupling Method

First, let us describe how the wall boundary conditions are given in the present calculations. Note that the wall properties are iteratively determined through coupling between CFD and SCMA modules. The mass injection rate of pyrolysis gas from the wall is given by the SCMA code as

$$m_p = \varepsilon \rho_p u_p,$$  \hspace{1cm} (10)
where subscript $p$ denotes pyrolysis gas. The void fraction $\varepsilon$ represents the degree of porosity for ablator. Because the SCMA code gives the void fraction one-dimensionally to the wall surface, we assume the porosity at the wall is identical with the void fraction. Therefore, at the wall, the ratio of void portion and catalytic wall mostly consisting of char is given by $\varepsilon : 1 - \varepsilon$.

The chemical composition of ablation product gas at the wall surface is calculated by assuming thermochemical equilibrium condition. The surface pressure is given by CFD solution while the surface temperature by the SCMA code. The SCMA code requires surface heat flux along the entry trajectory as the boundary condition. Such heat flux is provided by CFD solution at the chosen trajectory points. At intermediate time between two trajectory points, heat flux value is interpolated either by a linear or a quadratic function. Note that the heat flux value at the new trajectory point depends not only on the heat flux value at the previous trajectory point, but also on the intermediate heat flux value that is interpolated using the heat flux at the new trajectory point. Therefore, the heat flux value at the new trajectory point needs to be given by CFD solution iteratively.

The coupling of radiative heat transfer in this calculation is indirectly made. Once the radiative heat flux is obtained from CFD solution, it is fed into the SCMA code as a part of the net heating rate at the wall surface. The SCMA code returns the surface temperature and mass injection rate that influence the entire flowfield and radiation. Therefore, the flowfield and radiation is coupled through the boundary condition. A direct coupling between the flowfield and radiation needs to be considered in the case of Galileo, but can be neglected in the case of MUSES-C.

As to the surface recession, the SCMA code itself does not take account of surface recession explicitly. It is implicitly accounted in the code by changing the position of the outer wall surface and redraws the numerical mesh inside of the ablator. In the CFD calculation, one can include the effect of surface recession by changing the shape of capsule at each trajectory point. However, such effect has not been considered in this study simply because the amount of surface recession is negligibly small.

### 2.7 Code Validation History

The results of validation studies for each component of computer codes that constitute the present integrated analysis method are summarized here. The thermochemical package used in the CFD code was originally developed for airflows accounting for five neutral air species. Using this package, Niizuma & Sawada (1997) attempted to reproduce Fay-Riddell correlation of the blunt body convective heating rate, and obtained a good agreement with theoretical values. They also obtained a fair agreement of heat flux at the nose of space shuttle with the corresponding flight data. The package has also been checked extensively in a series of calculation of shock layer thickness over a sphere and a cone in intermediate hypersonic range (Furudate et al. 1999a, 1999b). The package was also extended to include carbonaceous species and was applied to solve the entry flowfield of Pioneer-Venus probes (Izawa & Sawada 2000a; Takahashi & Sawada 2002).

The SCMA code developed by Ahn et al. (2002) was validated by the comparison with the experimental data obtained in arc heated airflow and was applied to solve the thermal response of ablative heatshield for Pioneer-Venus probe vehicles. The SCMA code was also examined in detail of the ablator material of MUSES-C (Ahn 1998).

The injection-induced turbulence model was first incorporated into a CFD code by combining with one-equation turbulence model. The code was applied to obtain the velocity profile
in the boundary layer over a spherical blunt body with foreign gas injection through a porous wall. A peculiar velocity profile seen in the experimental data was qualitatively reproduced (Dendou & Sawada 1998). Izawa & Sawada (2000b) studied an alternative method for combining the injection-induced turbulence model with a CFD code by employing a zero equation turbulence model. This new scheme successfully reproduced the surface heat transfer rate at the stagnation point of a blunt body with a foreign gas injection in the hypersonic flowfield. Though the heating rate at the downstream side was overestimated by a factor of 2, the overall agreement with the experiment was judged fair. With the use of this new scheme, the entry flowfield of Pioneer-Venus probe vehicles with ablation was calculated. A reasonable agreement with the flight data was obtained for the temperature history not only at the stagnation point but also at the frustum region (Izawa & Sawada 2000a; Takahashi & Sawada 2002).

3. MUSES-C ENTRY FLIGHT SIMULATION

Figure 1 shows the chosen trajectory points for CFD calculations along the entry trajectory of MUSES-C. The entry angle is assumed as -12 deg at an altitude of 200 km. Figure 2 shows a typical example of computational mesh system with 59 × 77 grid points. It covers the forebody part of MUSES-C capsule.

![Image](image_url)

Fig. 1: Entry trajectory of MUSES-C. Entry angle is -12 deg at 200 km. Trajectory points for CFD calculations are indicated.

Calculation starts from obtaining an initial CFD solution at the first trajectory point at 55 sec of flight time. We assume that ablation is absent and that the boundary layer is laminar. A constant wall temperature of 3000 K is arbitrarily chosen. From this initial solution, we then obtain thermal response of ablator from the beginning of entry flight to this first trajectory point. We assume the wall heat flux is zero before 30 sec of flight time (density is below $1.0 \times 10^{-7}$ kg/m$^3$) and interpolate it by a quadratic function up to the first trajectory point. Putting the obtained heat flux profile along the entry trajectory into the SCMA code, we can find the temperature and injection rate at the wall surface at the first trajectory point. These values are used as the new boundary conditions for CFD calculation. We then obtain the new CFD solution that accounts for ablation and turbulence effect at the same trajectory point.
The new CFD solution in turn gives the new wall heat flux value. We specify the new heat flux profile along the trajectory and put it again into the SCMA code. These iterative procedures are continued until the wall heat flux profile is converged.

Figure 3 shows the convergence history of convective heat flux along the body surface at 55 sec from 200 km. One can see that the convergence is virtually attained within 5 iterations. The wall heat flux given by the turbulent solution becomes larger than that given by the initial laminar solution without ablation even at the stagnation point. The convective blockage effect of ablation product gas is totally cancelled by the effect of assumed turbulence. The convergence history of wall temperature for this altitude is plotted in Fig. 4. The wall temperature substantially decreases from the initially assumed wall temperature due to ablation.
Next, we try to obtain the CFD solution at 60 sec of flight time. As before, we assume the initial flowfield to be laminar and without ablation. We then linearly interpolate the heat flux value from 55 sec to 60 sec, which is fed into the SCMA code. The solution from the SCMA code gives the wall temperature and injection rate at 60 sec that are put into the CFD code as the boundary conditions. The converged and consistent solution accounting for ablation at 60 sec is finally obtained iteratively. The converged solutions at later flight time are similarly obtained. The surface heating profile accounting for ablation and turbulence at the chosen trajectory points are summarized in Fig. 5. Because of the turbulent heating, the heating rate in the downstream region becomes larger particularly at 70 and 75 sec of flight time.

Figure 6 shows the stagnation point heat transfer rates with ablation along the entry trajectory. The total heat flux is defined as a sum of the convective heat flux and the wallward...
radiative heat flux minus the shockward radiative heat flux from the wall. A maximum convective heat flux value of about 700 W/cm² is obtained at 70 sec. The wallward radiative heat flux is about 15% of the convective heat flux at this peak heating point.

The obtained temperature histories at the ablative surface are shown in Fig. 7. At the stagnation point, the present temperature history agrees fairly well with that given by Suzuki K. et al. (1998). They studied the stagnation heating and ablation process of the MUSES-C capsule by using the laminar viscous shock layer equations. A fair agreement of heat flux value is attained because Suzuki K. et al. (1998) considered the catalytic recombination of O and N atoms at the wall surface but not the effect of turbulence, while the present study considers the effect of turbulence that significantly enhances the heating rate but not the catalytic recombination. One can see that the temperature at the frustum edge is elevated and becomes
The computed surface recession at the stagnation point is shown in Fig. 8. Although the present solution is slightly shifted from that given by Suzuki K. et al. (1998), the agreement between these solutions is fairly good. The calculated recession profile along the wall surface at the last trajectory point is shown in Fig. 9. Because of the significant turbulent heating, the surface recession becomes larger at $s/R = 0.3$ than that at the stagnation point, and it then gradually decreases toward the juncture point but slightly increases again at the frustum edge.

4. DISCUSSIONS

In this work, we first describe a trajectory-based integrated analysis code briefly. This integrated analysis code consists mainly of CFD module, SCMA module, and radiation module.
The coupling method of these modules is also explained. A short validation history of these modules and the integrated code is given. The calculated results of heating environment for MUSES-C sample return capsule by this integrated analysis code are shown. It is shown that the converged solution along the entry trajectory can be obtained by loosely coupling approach between the CFD and SCMA codes. The convergence at each trajectory point is reached within a few iterations.

The maximum convective heat flux at the stagnation point is found to be about 700 W/cm\(^2\) at the peak heating point, whereas the wallward radiative heat flux becomes only about 15% of the convective heat flux value. The wall temperature as well as the surface recession rate at the stagnation point along the entry trajectory duplicates well with that given by the existing analysis. This agreement is attained because the catalytic recombination of atoms at the wall surface is included in the existing analysis, whereas the present calculation assumes a noncatalytic wall for those recombination reactions but includes the turbulence effect of the ablation product gas.

It is found that the wall temperature at the downstream frustum region is significantly elevated by the effect of turbulence. This phenomenon was also observed in the numerical analysis of aerodynamic heating environment for the Pioneer-Venus entry probes. The actual flight data also indicated such phenomenon was really the case. Therefore it will be interesting to see whether the actual flight data of MUSES-C will confirm the present prediction of higher heating rate that will occur in the downstream frustum region.

It is indeed important to assess the integrated analysis code in terms of how well the available flight data into a planetary atmosphere can be reproduced. Our continuous efforts have been focused on this issue. So far, we have succeeded in obtaining a reasonable agreement of temperature profiles with those taken in the entry flights of the Pioneer-Venus probe vehicles (Takahashi & Sawada 2002). Our next target is to reproduce the peculiar recession profile of Galileo probe ablator that was reconstructed from the actual flight data (Milos 1996). For this purpose, we are now trying to develop a numerical method for computing strongly radiating flowfield for Jovian entry flight. Preliminary results are already shown in the work of Matsuyama et al. (2002).

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Thermal Protection System of the Reentry Capsule with Superorbital Velocity

By

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Abstract: In the final phase of the MUSES-C mission, a small capsule with asteroid sample conducts reentry flight directly from the interplanetary transfer orbit at the velocity over 12 km/s. The severe heat flux, the complicated functional requirements, and small weight budget impose several engineering challenges on the designing of the thermal protection system of the capsule. The heat shield is required to function not only as ablator but also as a structural component. The cloth-layered carbon-phenolic ablator, which has higher allowable stress, is developed in newly devised fabric method for avoiding delamination due to the high aerodynamic heating. The ablation analysis code, which takes into account of the effect of pyrolysis gas on the surface recession rate, has been developed and verified in the arc-heating tests in the facility environment of broad range of enthalpy level. The capsule was designed to be ventilated during the reentry flight up to about atmospheric pressure by the time of parachute deployment by being sealed with porous flow-restrict material. The designing of the thermal protection system, the hardware specifications, and the ground-based test programs of both MUSES-C capsule are summarized and discussed here in this paper.

1. Introduction

The purpose of the MUSES-C mission is to deliver a spacecraft to an asteroid and then to contribute scientific measurements and observations of the asteroid and finally to collect the asteroid samples and to return them to the earth. In the final phase of MUSES-C a small capsule with asteroid sample will conduct reentry flight directly from the interplanetary transfer orbit to the earth at the velocity over 12 km/s. MUSES-C is scheduled to be launched in 2003 and the capsule to return in 2007.

The severe aerodynamic heating environment at the reentry, complicated functional requirements, and small weight budget impose several engineering challenges on the designing of the thermal protection system (TPS) of the MUSES-C reentry capsule. A major technical issues comes from its hypervelocity atmospheric reentry flight in which the entry velocity is super-orbital (12 km/s). The estimation of the reentry flow environment and the development

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of thermal protection system considering material and structures are essential to the design of the capsule. A number of additional issues have been addressed to assess their effects on the capsule TPS. These include capsule shape, targeting initial reentry conditions, because this reentry severely tests our thermal protection design capability, a significant research base, both experimental and analytical, has been formed in support of the capsule heat-shield design effort. Design work is initiated in 1996 based on the ISAS’ experiences in the related technical background for both the aerothermodynamics, thermal protection materials and its responses under heating, descent and recovery subsystems, and several flight lessons in the past.

In this paper, the mission scenario and the subsystem requirements for the thermal protection system (TPS) are described first, then the research and development work of the thermal protection system of the MUSES-C reentry capsule are followed with design result.

Fig. 1: MUSES-C Reentry Capsule.

2. Mission and TPS Requirements

Mission Scenario and Flight Sequence

On the surface of the asteroid in the MUSES-C mission, the sampler container with asteroid samples is inserted into the small capsule from backward and latched to maintain the inside in vacuum condition. Then the mother spacecraft leaves the asteroid for the earth. In the final phase of the mission, when the mother spacecraft approaches the earth, the capsule is separated and conducts reentry flight to the earth's atmosphere directly from interplanetary transfer orbit and recovered on the ground as shown in Fig. 2. After having experienced severe reentry aerodynamic heating, the capsule is decelerated down to subsonic velocity. Then the parachute is released and deployed synchronized to the aft cover jettison at the altitude of about 10 km. Simultaneously the forebody heatshield is also jettisoned in order to prevent soak-back heating from the highly-heated forebody heatshield. On the heatshield jettison, the antenna is also expanded to begin to send the beacon signal for the sake of rapid recovery. The capsule
Fig. 2: MUSES-C Sample Return Capsule: Reentry Scenario.

Fig. 3: Exploded View of the Reentry Capsule.
configuration which realizes the mission scenario described above are schematically shown in Fig. 3.

**TPS requirement**

The required function throughout the orbital flight, reentry flight, and descent/recovery phase are accommodated into a small capsule. The capsule is designed so as to have aerodynamic stability and perform ballistic reentry flight. Since the lightweight design and maximum internal capacity envelope is required, a slightly fat disk with the truncated cone shape was selected from various geometries. In the preliminary aerodynamic geometry trade-off phase, relation among the entry trajectory condition, nose radius, which affects stagnation heat transfer rates, and aerodynamic stability, was investigated. Then baseline shape of the capsule is 45° half-angle sphere cone with the nose radius of 0.20 m and the diameter of 0.4m with the weight of about 16.5 kg.

Preliminary study pointed out the extremely high heat flux and high impact pressure on the capsule. Due to the small weight budget allocated to the capsule, the heat shield is required to function not only as thermal protection material but also as a structural component. Thus, the charring ablation by carbon-phenolic was adapted for the thermal protection among several trade-off. The main requirements for the capsule TPS are as follows;
1) Heatshield needs to thermally protect the inside under requested value of temperature; generally 60 °C at the time of parachute deployment to protect interior instruments.
2) Heatshield needs to stand the aerodynamic load throughout the reentry flight (including the time at the maximum dynamic pressure). 3) Heatshield needs to seal the inside against the pervasion of the external hot airflow during the reentry within an enough degree to satisfy the above requirement: Since the capsule needs to jettison parachute cover against atmospheric pressure, it is desirable to equilibrate inner pressure to outer pressure level for the sake of minimizing pressure drag. Thus ventilation or pressure-equilibration is required during the reentry.
4) Total recession of the heatshield needs to be below 1 % of the diameter for minimizing the uncertainty of the aerodynamic coefficient of the capsule.
5) Aftbody heatshield needs to be light enough to be released over 10 m/s at the jettison.

In order to design the thermal protection system that satisfies the above requirements, following items are considered and investigated during the design phase; 1) Estimation of reentry environment including aerothermodynamic special issues such as the boundary layer transition, radiative heating. 2) Development of heatshield material and the prediction of its behavior under highly heating environment. 3) Designing of special configurations such as gap between forebody/aftbody heatshield, sealing mechanism, and method of ground-based test of them. These are described in the following sections.

### 3. Reentry Flight Environment

#### 3.1 Stagnation Point Heat Flux

Initial conditions of the capsule reentry are mainly determined by interplanetary transfer orbit planning of the mother spacecraft. Especially, reentry flight path angle is one of the alternative options of the capsule side since it affects on the maximum heat flux and total heat input. Thus reentry flight environment for designing TPS was set up taking account of the dispersion
of atmosphere, aerodynamic coefficients, and orbit. From the standpoint of the heatshield designing, behavior of the ablator in the aerodynamic heating along the reentry trajectory needs to be considered of prime importance. The flight path angle was so selected so that the estimated heat flux do not exceed experimentally tested value: $15 \text{ MW/m}^2$. The aerodynamic heating rate on the stagnation point of the forebody is estimated by using Detra-Kemp-Riddel model for the convective heating and Tauber's model for the radiative heating along the reentry trajectory described previously. Time-profile of the aerodynamic heating are shown in Fig. 4 and also summarized in Table 1. This estimated heat flux is endorsed to contain safety margin to a certain extent by the further estimations; Nonequilibrium viscous shock layer (VSL) analysis of the stagnation point convective heat transfer rate and ablation blocking effect was conducted including major 7 species of ablation products in addition to the 11-species of the air (Suzuki et. al.,1997).

![Fig. 4: Time Profile of the Heat Flux on the Capsule.](image)

Although the reentry velocity is determined in advance from interplanetary-transfer trajectory, reentry flight path angle still have some degree of freedom. As a result of analytical estimations of the total recession along the reentry trajectory, there proves to be optimal trajectory with a certain flight path angle, which minimizes the total recession of the forebody heat shield. The reentry path angle needs to be fixed also taking account of the internal temperature, which is affected by the uncertainty of the thermal data and the capability of the ground heating facility as will be described in the following sections.

### 3.2 Heat Flux Distribution on the Capsule

On estimating heat flux distribution especially on the aftbody of the capsule, the radiation database of the 11-species-air and 6-species carbonaceous ablation products is developed with attention to UV and VUV radiation (Fujita, 1997). Then the heat flux distribution has been
Table 1: Capsule Flight Environment Summary.

| Inertial Velocity                    | 12 [km/s] |
| Flight Path Angle                   | 12-14 [deg] |
| Max Convective Heat Flux            | 13-15 [MW/m²] |
| Max Radiative Heat Flux             | 3-4 [MW/m²] |
| Total Heat Flux                     | 16-18 [MW/m²] |
| Convective Total Heat Input         | 250-280 [MJ/m²] |
| Radiative Total Heat Input          | 24-26 [MJ/m²] |
| Total Heat Input                    | 270-300 [MJ/m²] |
| Total Heat Input                    | 270-300 [MJ/m²] |
| Dynamic Pressure                    | 0.6-0.7 (0.8-1.1) [kg/m²] |
| Max Deceleration                    | 40-50 (60-80) [G] |

the values in the parenthesis are taken from DASH mission and are also used for the design target for TPS.

analyzed through full Navier-Stokes equation coupled with radiation heat flux (Ohtsu, 1999). These results were compared with the existing and simpler modeling technique, and results show that the radiation heating at the stagnation region is about 10 to 15% of the cold wall convective heating.

Boundary layer transition from laminar to turbulent is of great interest because it can be induced by the ablation gas injection and causes the heat flux enhancement. Komurasaki et al. (1998) conducted numerical analysis based on the k-ε turbulent model. The occurrence of the boundary layer transition exhibits recognizable enhancement of the heat flux especially at the skirt part of the capsule. In the regime of Reynolds number of 2.5 x 10⁴, which corresponds to that at maximum heat flux rate, the onset of the boundary layer transition can not completely denied at the ratio of injection mas flow to in-coming mass flow rate of below 10%. Taking into account the results out of these aerothermodynamic analyses, the cold wall heat input to the capsule both on the forward and backward surface is specified from designing point of view as shown in Fig. 5, where presented are the summation of both convective and radiative heat fluxes derived from the numerical analyses described here. Due to the uncertainty in the transition estimates described above, the stagnation heating load is applied to all the part of the foreside surface. If it occurs in the region of the later period of the reentry flight, the total mass loss of the heatshield at the skirt part is considered below that in the stagnation part.

4. Ablative Thermal Protection

4.1 Ground Test and Analysis Scenario

The simulation environment of the ground heating test and flight environment is schematically shown in Fig. 6. The heating test environment of ISAS arc heater (Yamada et al., 1999) is limited both in terms of flow enthalpy and the size of the test pieces. It is desirable to conduct the tests in other facility for correlating and validating the results. Then a series of heating test in 60 MW IHF (Interaction Heating Facility) at NASA ARC was conducted at the final stage of the qualification process. The major objective is to confirm the ISAS-test-derived results both from view point of the modeling of thermal responses and mechanical durability such as delamination by use of the higher enthalpy flow of 40 MJ/kg and maximum heating of 20
$MW/m^2$ with model size of 70 mm in diameter. The result from the heating test at IHF was satisfactorily where the data reproducibility and the estimates by the modeling were certified as well as the thermal and mechanical durability of the heat shield presented here.

4.2 Carbon-Phenolic Ablator and LCCP: Delamination Countermeasure

The heat shield material's selection and evaluation are ones of the major issues in the design process. The present mission offers a considerably high heating load up to $15MW/m^2$ for 30 seconds, then a charring ablator is to be a unique solution. Based on the existing material both from view point of heat resistant characteristics and matured manufacturing and process control, a carbon-phenolic material is chosen among many possibilities in fabric arrangement,
such as cloth lay-up, tape-wrapped, chop-molded, and so on. Taking account that the heat shield is required to function load-carrying component, the cloth-layered ablator has great merit that it has high allowable stress in layer plane. Nevertheless, it has demerit that the delamination tends to occur in high heating environment. Exactly describing, it occurs when the decomposed pyrolysis gas pressure rises up higher than the inter-layer allowable stress of the ablator. The pyrolysis gas must be released properly in order to prevent the delamination. In the present application to the capsule, tilted lay-up of carbon cloth with respect to the surface were selected. Especially for the nose stagnation, the lattice layered carbon phenolic (LCCP) was used. (Fig. 7) LCCP belongs to carbon phenolic cured by using prepregs that have cut-slit every 10 mm interval. Figure 8 shows the surface of the lattice ablator during heating test taken by use of ND filter (left) and surface photo after the test (right). The lattice pattern on the surface shows the slits of cloth for releasing the pyrolysis out gassing, and it maintains stable surface condition under heating. A qualitative simulation of the effect of pyrolysis gas pressure on the delamination numerically tried recently.

![Fig. 7: Tilt-layered and Lattice Layered Carbon Phenolic (LCCP).](image)

### 4.3 Ablation Analysis

It is easily imagined that a massive ablation products out of the body surface will affect the shock layer flow greatly. For the charring ablator such as a carbon-phenolic material that is often used for the thermal protection in relatively high heating rate, the reaction at the surface itself is divided into oxidation dominant regime below 3000K of surface temperature and sublimation dominant regime for higher temperature. In the latter regime, the surface recession rate increases rapidly with the temperature rise, a massive flow of ablation product must be taken into consideration in addition to the pyrolysis gassing. The concern from the design view point is whether existing stagnation heating estimates such as simple Fay and Riddel relation
is still valid in such a high enthalpy flow. In addition to that, since the outer flow enthalpy is higher than that of normal reentry vehicle, careful estimates is required in giving a realistic heating condition to the ablative materials. Then, a fully thermal and chemical nonequilibrium shock layer flow analysis was made under the condition that there is a carbonaceous ablation products with concerned surface temperature regime. The reaction at the surface is modeled, where the surface catalytic recombination, ion recombination, oxidation and sublimation, and pyrolysis gas out of the surface are taken into account. A series of numerical analyses has been carried and it confirmed that both the simple stagnation convective heating estimates and the existing blocking effects which correlates the cold wall heating and heating under ablative injection gives the safer estimates of the stagnation heating in such a high enthalpy and strong blowing conditions in the present reentry environment. This confirmation helped reduce the design task of reentry trajectory and made bridging between the heating test results of the ablator presented later and the real flight condition possible and realistic. The ablation analysis code, which takes into account of the effect of pyrolysis gas on the surface recession rate, has been developed, and verified in the arc-heating tests in the facility environment of broad range of enthalpy level.

In the basic evaluation studies, data acquisition for refining the simulation model were done in ISAS segmented-type arc heater of 1 MW maximum input power, which simulates heat flux environment ranging from 1 to 12 MW/m² with flow enthalpy up to 20 MJ/kg for the flat-faced test model of 25 mm in diameter for the highest heating condition. The flow enthalpy is far less than that in the flight, however, the numerical-simulation-based correlating studies made it possible to bridge from the test results to the flight condition, as presented in the previous section. In the heating test, the surface temperature and internal temperature distributions are measured as well as the heating test conditions such as heat flux, flow enthalpy, impact pressure, which is needed for characterizing the thermal response of the ablator. Modeling of thermal responses of the ablator is essential to predict the ablator behavior in the flight environment. In the flight capsule, one of the important design constraints is the inner face temperature after heating. The thickness of the heat shield must be determined from the standpoint of the total
amount of recession and the temperature characteristics.

**One-Dimensional Ablation Analysis**

An one-dimensional charring ablation analysis code is refined so as to simulate correctly for the present application. The code is to solve the thermal conduction including temperature-dependent thermal properties, pyrolysis gassing, and a set of ablation models as a boundary condition. At the boundary following effects are taken into account; air mixture of incoming flow, composition of pyrolysis gas, and products of the surface reaction by oxidation and sublimation (Potts, 1994). Each reaction rates are calculated under the assumption of chemical equilibrium with respect to the flow impact pressure and enthalpy. The mass flow rate of the ablation products is expressed as non-dimensional gas blowing parameter as a ratio to the mass flow rate of incoming air. They were correlated as a function of the wall temperature on the basis of arc heating test results as shown in Fig 9. The recession data obtained in the heating test are correlated under the assumption that the pyrolysis gas makes chemical reaction with high temperature air in the shock layer. Figure 10 shows the correlation of the char layer recession rate with the pyrolysis gas blowing rate. The result shows a degree of difference that may be caused by the effects out of the recession modeling. However, the estimated total recession of the present application is about 2 to 3 mm, and it seems satisfactorily small to be compared with the total thickness. Thermal properties of ablutor such as thermal diffusivity, specific heat, heat of reactions are not only measured by the conventional technique where the temperature and its rate are low and small, but also tuned by using data obtained in laser heating and arc heating test especially in high temperature beyond 400° C. The model refinement is made primarily based on the ISAS' arc-heating test results.

![Graph showing correlation between Bc and wall temperature.](image)

**Fig. 9: Bc vs. Wall Temperature Characteristics.**

**Thermal Analysis of Some Special Parts**

In addition to these studies for the model refinement and qualification of the basic characterization of the ablative heat shield, the flight capsule is composed of various parts on the heat
shield such as the ablator gaps between two parts, venting holes, electrical harness penetration, fixture-insertion, etc. as presented later. Due to the difficulty in modeling such a complicated parts, these parts are accommodated in the heating test pieces, then they were evaluated by arc-heating tests. At the edge and/or corner part of the heat shield, the simple one-dimensional in-depth analysis presented above is not enough to give an accurate temperature distribution. Therefore, for these special and design-critical parts such as the outer edge of the foreside heat shield is modeled and 2-D thermal conductivity analysis was made in which a modified ablation code is incorporated.

5. Design Integration to Heatshield

TPS of the capsule includes ablator heat shields and the sealing mechanisms. High-speed reentry leads to capsule forebody heating rates of over $15 \text{ MW/m}^2$. In such an extremely high heating environment, a carbon-phenolic was of primary interest as the heat shield material as described because of its relatively low recession rate in comparison with other ablators (e.g. silica-phenolic, or alumina-phenolic etc. The forebody heat shield is an integrated forebody ablator of carbon phenolic. The aftbody heatshield is a generic name referring to 3 ablator components called parachute-cover ablator, sampler ablator, and support ablator, respectively. A schematic view of the reentry capsule is shown in Fig. 11.

5.1 Front Heat shield

Delamination Counterpart

Forebody ablator is required to protect internal instruments against extremely high heat flux without being crushed even in the high impact pressure. The problem of the delamination can be avoidable by adapting tilted lay-up ablator or LCCP. Since the thermal conductivity in in-depth direction becomes larger for the tilted lay-up, an allowable tilt angle is carefully investigated by heating test and numerical analysis both from the standpoint of the delamina-
Fig. 11: Design Result of the Heatshield of the Reentry Capsule.

- Heatshield Thickness

The front ablator is required to protect internal instruments against extremely high heat flux without being crushed even in the high impact pressure. Total recession of the front heatshield is estimated based on a worst combination of thermal properties. On investigating the structural strength against the aerodynamic load all throughout the reentry, we have adopted the assumption that the only virgin layer of the ablator can behave as load carrying component, namely charred and pyrolysis layers have zero allowable stress. Then the total thickness of the ablator needs to be determined to satisfy enough virgin layer thickness requirement. Moreover the ablator thickness simultaneously needs to satisfy the internal temperature re-
requirement. Baseplate temperature estimation error margin of 20 °C is allotted taking account of measurement errors and dispersions of the thermal properties of the ablator material.

In order to reduce the heatshield weight satisfying the above requirements, the inside of the forebody heatshield was scraped and insulator was installed there except the portion where the pyrotechnique devices for the parachute-cover separation were mounted. The insulator is made of silica fiber, and averaged density is light (0.1 g/cm³).

5.2 Parachute Cover

Since the heat flux on the aftbody is relatively small in comparison with that on the forebody, the total thickness of the parachute cover ablator can be reduced to 20 mm, but it is still too heavy because it must be jettisoned from the capsule at proper speed by the pyrotechnique with limited energy at time of parachute deployment. The parachute cover ablator is scraped to the thickness of 7 mm from the inside except the bridge girder annularly left to fix the cover itself to the aluminum structure. The 7 mm part of the ablator is so designed that the virgin layer left until parachute deployment is 1 mm. Design philosophy and proper margin needs to be investigated at once. The internal space between the ablator and the aluminum structure is stuffed with an insulator, which is designed to make the temperature of the structure be same as that of 20 mm 'girder bridge'. Otherwise heat conduction would concentrate on the foot of the girder bridge, which causes local temperature rise of the inner instruments.

5.3 Sampler Ablator, Support Ablator

The sample container is inserted into the capsule by a helical spring whose point of action is sampler ablator itself. The support ablator is a kind of guide rail not only for the sampler ablator but also for the parachute cover. Obviously, both of the sampler and the support ablator need to have some degree of mechanical strength and proper hardness. Thus they are constituted by carbon phenolic with 20 mm thickness. Although joint bolts, cable mounts are installed on the ablators, the thickness is designed to be 20 mm throughout the aftbody.

5.4 Sealing Mechanism and Structure

The capsule needs to be ventilated during the reentry flight up to about atmospheric pressure by the time of parachute deployment, it is sealed with porous insulator, which allows gradual ventilation, by the outer hot air. The parachute cover needs to be separable from the forebody heatshield and the support ablator respectively for the sake of the parachute deployment. The sampler ablator also needs to be separable from the support ablator. From the subsystem requirement, at least 3 clearances of the heatshields need to be sealed against in-coming outer hot gas during the reentry heating below subscribed level. On the other hand, the inner pressure is desirable to be almost equilibrated to the outer atmospheric pressure at the time of the parachute deployment because it reduces pyrotechnique energy required for the parachute-cover jettison and leads to total weight reduction. Thus sealing is required to be not complete sealing nor no sealing; namely to be 'loose' sealing to make gradual ventilation of the inside of the capsule.

An axisymmetric FEM analysis estimated the extent of deformation of the heatshield. Not a little deformation due to the large linear expansion coefficient of the carbon phenolic makes it difficult to seal directly gaps between ablators even at the deepest bottom surface. Slight dependence of the linear expansion coefficient on the temperature-profile during the
measurement also makes the deformation analysis further complicated. Thus a kind of flow resister was put in the annular gap between the forebody ablator and the parachute cover, which was the same material as the insulator used for the back of ablator. The insulator made of silica looks like 'cotton puff', and function as appropriate flow resister against the incoming hot airflow. Mass flow rate through the flow resister during the reentry flight was estimated experimentally as shown in Fig. 13 and the enthalpy or heat input due the airflow turned to be below 700J, which is acceptable heat input in comparison with temperature rise due to aerodynamic heating. The point that the flow insulator is not destroyed nor burned was confirmed by arc heating test as long as it was installed 30mm from the surface as shown in Fig. 14.

![Fig. 12: A Schematic Concept of Ventilation and Sealing of the Capsule.](image)

![Fig. 13: Time Profile of the Air Mass Flow Rate entering the Capsule.](image)
5.5 Functional Verification of TPS Components

The functional safety of each component or special portion of the TPS has been also tested in the arc heating test.

Cable Penetration

For health-check of the capsule, the umbilical cable needs to penetrate the aftbody ablator as previously shown in Fig. 11. The burn-out and melt-down of the cable causes severe damage on the capsule due to the invasion of the hot gas during the reentry phase. Even if the cable survive the reentry, the heat conduction through the cable needs to be small enough. The survivability of the cable was tested and confirmed in the arc heating facility by means of the dummy model test piece as shown in Fig. 15. In these series of experiments, the measured total heat conduction through the cable was below the value estimated under the assumption that the heat conducts through the copper pipe with the equivalent diameter and the length as the cable and with the temperature gradient between the melting point and the room temperature. The total heat input estimated though the cable is below 1000 J, which can be absorbed safe by the structures.

Gap Clearance

There are several gaps ranging submillimeter to 1.5 mm between heatshield components on the capsule surface. These gaps must not increase the internal temperature excessively beyond the predicted value by the ablator thermal analysis. Thus, the functional safety during the reentry phase were tested and confirmed by means of appropriate test piece as shown in Fig. 16. As long as the gaps within submillimeter to 2.0 mm in the no flow environment, the temperature of the bottom (depth to gap ratio of about 10) of the gaps is equilibrated to the temperature of the bottom of the ablator, which means the baseplate temperature is not increased by the existence of the gaps. In case that the flow needs to be considered, the thermal analysis was conducted taking account of the pressure gradient as described in the previous section.
6. Concluding Remarks

The designing of the thermal protection system of small capsules such as MUSES-C mission, the hardware specifications, and the ground-based test programs of both MUSES-C capsule were summarized and discussed in this paper. The severe heat flux, the complicated functional requirements, and small weight budget impose several engineering challenges on the design of the thermal protection system of the capsule. The heat shield is required to function not only as ablator but also as a structural component. The cloth-layered carbon-phenolic ablator, which has higher allowable stress, is developed with newly devised fabric method for avoiding
delamination due to the high aerodynamic heating. The ablation analysis code, which takes into account of the effect of pyrolysis gas on the surface recession rate, has been developed and verified in the arc-heating tests in the facility environment of broad range of enthalpy level. The capsule was designed to be ventilated during the reentry flight up to about atmospheric pressure by the time of parachute deployment by sealing it with porous flow restrictor, which realize gradual pressure increase up to atmospheric pressure against the outer hot air.

REFERENCES