Pre-Compression Flow on Fore-Body of Space-plane at Mach 10

A. Murakami and M. Maita
National Aerospace Laboratory
Tokyo, Japan
PRE-COMPRESSION FLOW ON FORE-BODY OF SPACE-PLANE AT MACH 10

Akira MURAKAMI and Masataka MAITA
National Aerospace Laboratory
Tokyo, JAPAN

Abstract
Pre-compression on a fore-body of a space-plane with air-breathing engines is effective on an improvement of the propulsion performance at super/hypersonic speeds. On the other hand, the inlet distortion that is caused by flow expansion at the vicinity of the fore-body side-edge and boundary layer growth can cancel the pre-compression effect on the propulsion performance. In the present investigation, pre-compression flow on the fore-body of the space-plane at a Mach number of 10 was experimentally investigated at the NAL’s hypersonic wind tunnel. The test results show that longitudinal vortices due to inboard pressure gradient in the spanwise direction could occur along the center-line of the lower surface of the fore-body in spite of the windward side. The vortices could cause serious pressure loss at the engine inlet location. In the vicinity of the side-edge of the fore-body, also, the flow distortion at the engine inlet location could be observed due to flow spillage (i.e. flow expansion).

Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>B</td>
<td>maximum body width of 1.5% space plane</td>
</tr>
<tr>
<td>L</td>
<td>total length of 1.5% space plane</td>
</tr>
<tr>
<td>Lpw_knk</td>
<td>location of the kink point in the spanwise wall pressure distribution</td>
</tr>
<tr>
<td>Lsp_y</td>
<td>location of the separation line at the 2nd ramp leading edge</td>
</tr>
<tr>
<td>H_{inlet}</td>
<td>engine inlet capture height</td>
</tr>
<tr>
<td>M</td>
<td>freestream Mach number</td>
</tr>
<tr>
<td>Po</td>
<td>freestream total pressure</td>
</tr>
<tr>
<td>Pinf,cal</td>
<td>calculated freestream static pressure</td>
</tr>
<tr>
<td>Pp</td>
<td>Pitot pressure</td>
</tr>
<tr>
<td>Pp,cal</td>
<td>calculated freestream Pitot pressure</td>
</tr>
<tr>
<td>Pw</td>
<td>wall pressure</td>
</tr>
<tr>
<td>Re</td>
<td>freestream unit Reynolds number</td>
</tr>
<tr>
<td>To</td>
<td>freestream total temperature</td>
</tr>
<tr>
<td>Tinf,cal</td>
<td>calculated freestream static temperature</td>
</tr>
<tr>
<td>x</td>
<td>distance from the nose of the model</td>
</tr>
<tr>
<td>y</td>
<td>distance from the center-line of the model</td>
</tr>
<tr>
<td>z</td>
<td>distance from the surface of the model</td>
</tr>
<tr>
<td>α</td>
<td>angle of attack</td>
</tr>
</tbody>
</table>

Introduction

Space-plane, i.e. fully reusable space transportation system with air-breathing engine will be expected as the future space transportation system so that it has the capability to realize lower operation cost, higher operability as compared with the current expendable space transportation system. To realize such space transportation system, however, a lot of advanced technologies must be applied, especially, the air-breathing propulsion system is the most important key technology. The design of the air-breathing propulsion system for the space-plane cannot be made without the propulsion/airframe integration because the compression by the airframe can play an important role in the air compression process at super/hypersonic speeds. The pre-compression of the fore-body is effective on the improvement of the propulsion performance but the flow distortion at the engine inlet location due to flow expansion around the fore-body side-edge and boundary layer growth on the fore-body possibly aggravates the propulsion performance. Therefore, the research on not only the propulsion system itself but also the pre-compression flow on the fore-body is important from a point of view of the propulsion/airframe integration design.

The “cone” type configuration is proposed as the fore-body configuration of space plane from a point...
of view of low airframe drag and high volume efficiency, but not always from a point of view of the pre-compression effect. Because high three-dimensionality of the pre-compression flow can cause distortion at the engine inlet.

In the present investigation, the pre-compression flow on the "cone"-type fore-body of the space plane is experimentally investigated at the Mach10 hypersonic wind tunnel as a part of the research of propulsion/airframe integration design.

### Description of Wind Tunnel Tests

**Test Facility**

Tests were carried out at the Mach10 hypersonic wind tunnel system of National Aerospace Laboratory of Japan. This facility has 1.27m diameter test section and its duration is approximately 40seconds. Test condition in the present tests is shown in Table1. Freestream Pitot pressure (Ppo,cal) , freestream static pressure (Pinf,cal) and freestream temperature (Tinf,cal) are calculated by the real gas correction1-3 in the facility nozzle flow.

### Table 1 Wind Tunnel Flow Conditions

<table>
<thead>
<tr>
<th>Mo</th>
<th>Po (Mpa)</th>
<th>To (K)</th>
<th>Ppo,cal (kPa)</th>
<th>Pinf,cal (kPa)</th>
<th>Tinf,cal (K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>9.55</td>
<td>2.446</td>
<td>1044</td>
<td>8.73</td>
<td>0.0737</td>
<td>53.77</td>
</tr>
<tr>
<td></td>
<td>-2.451</td>
<td>-1093</td>
<td>-8.81</td>
<td>-0.0747</td>
<td>-59.05</td>
</tr>
<tr>
<td>9.68</td>
<td>5.868</td>
<td>1055</td>
<td>19.93</td>
<td>0.1639</td>
<td>54.60</td>
</tr>
<tr>
<td></td>
<td>-5.882</td>
<td>-1104</td>
<td>-20.13</td>
<td>-0.1656</td>
<td>-57.38</td>
</tr>
</tbody>
</table>

**Model**

Fig.1 shows a schematic view of the wind tunnel test model used in the present tests. The broken line in Fig.1 indicates an 1.5% scaled model(L=1313.7mm) of the space plane which is designed as the reference configuration by National Aerospace Laboratory of Japan. The wind tunnel test model simulates just the fore-body of the space plane. The length of the model is 1,133mm and the span is 340mm. The lower surface of the model has 2ramps besides the nose part ramp(the 1st ramp) with 5degree to the airframe axis, of which the deflection angle is 4degree for the 2nd ramp and 2degree for the 3rd ramp, respectively. On the lower surface of the model 38 pressure taps along the center-line, and 22 taps in the spanwise direction at the "survey plane-A" (x=983mm: x/L=0.748) where is the location of engine inlet, 19 taps at the "survey plane-B" (x=838mm :x/L=0.638) and 17taps at the "survey plane-C" (x=705mm: x/L=0.537).

### Pitot Pressure Survey

In the present tests, Pitot pressures were surveyed at the location of the "survey plane-A", "-B" and "-C" as well as wall pressure measurements. Fig.2 shows the photograph of the Pitot rake with 24 probes used for the Pitot pressure survey. During the wind tunnel running, the rake can be traversed in the direction normal to the freestream.

**Pressure Measurement System**

Electronically scanned pressure measurement system manufactured by Pressure Systems Inc. (model SP-8400) was used in wall pressure and Pitot pressure measurements. Two pressure scanner modules with 48ports and the range of 34kPa(5psi) which were installed inside the model were used for wall pressure measurements, and 32ports pressure scanner module with the range of 103kPa(15psi) for Pitot pressure measurements.

### Results and Discussion

**Oil Flow Visualization**

Results of oil flow visualization on the lower surface for angles of attack of α=0deg and 5deg are shown in Fig.3(a) and (b), respectively. Flow condition is of a freestream Mach number of M=9.68 and a freestream unit Reynolds number Re=4x10^6.

In the case of α=0deg(Fig.3(a)), the flow on the 1st ramp is observed toward the center-line from the body side-edge, and the large separation is also observed due to the second ramp shock/boundary layer interaction upstream of the 2nd ramp leading edge. In the present test condition, the boundary layer on the 1st ramp would be so laminar that the flow separation would be very large due to the shock/boundary layer interaction. It was also found that the separation became larger as the Reynolds number decreased. Downstream of the 2nd ramp leading edge, however, the separation lines are steeply curved toward the center-line, and they are along the center-line on the 3rd ramp. The feather pattern that indicates a generation of a pair of longitudinal vortices could be also observed outside of the separation lines. As described later, wall pressure in the vicinity of the center-line is lower due to the 2nd shock/boundary layer interaction than that at the outside(near the body side edge). This inboard pressure gradient in the spanwise direction
could induce the flow toward the center-line and result in the generation of the longitudinal vortices in spite of the windward side. Near the body side edge on the 3rd ramp, the flow is toward the body side edge due to flow expansion.

For $\alpha=5^\circ$, the flow pattern on the 1st ramp is similar to that in the case of $\alpha=0^\circ$. The separation due to the 2nd ramp shock/boundary layer interaction is, however, relatively small because of the thinner boundary layer. Also, the separation lines to indicate a generation of longitudinal vortices disappears downstream of the 2nd ramp so that the flow pattern is almost two dimensional in the vicinity of the center-line. This would be because the spanwise pressure gradients and the boundary layer thickness would be relatively small as compared with the case of $\alpha=0^\circ$. On the other hand, the flow deflection near the body side edge is larger due to the larger flow expansion.

**Center-Line Wall Pressure Distribution**

Fig. 4 shows the wall pressure distributions along the center-line. Symbols and lines in the figure indicate the measured and the designed values, respectively.

The measured wall pressure on the 1st ramp is in fairly good agreement with the designed value except at the nose part ($x/L<0.1$) and just upstream of the 2nd ramp. The pressure at the nose part is due to the strong shock generated by the blunt nose which could induce higher pressure for $\alpha=0^\circ$, but for $\alpha=5^\circ$ due to flow expansion at the body side edge. The pressure rise just upstream of the 2nd ramp leading edge is due to the 2nd ramp shock/boundary layer interaction because the size of the separation region observed in the oil flow visualization is smaller as the angle of attack increases. Downstream of the 2nd ramp leading edge, the designed compression system cannot be achieved, especially for $\alpha=0^\circ$ although the wall pressure at the engine inlet location ($x/L=0.748$) reaches to the designed value. The large separation on the 1st ramp due to the 2nd ramp shock/boundary layer interaction deforms the shock system around the center-line on the 2nd and 3rd ramps so that the wall pressure distribution on the center-line is different from the designed one.

**Spanwise Wall Pressure Distribution**

The spanwise wall pressure distributions at $x/L=0.537$, 0.638 and 0.748 are shown in Fig. 5(a), (b) and (c), respectively. The span location is normalized by the body width ($B=240\text{mm}$).

In the case of $\alpha=0^\circ$, the wall pressure distribution at any location shows the distribution in which the wall pressure around the center-line is lower. As described before, the low pressure around the center-line is due to the shock system deformation involved with the flow separation by the 2nd ramp shock/boundary layer interaction, thereby the steep pressure gradient in the spanwise direction arises around $y/B=0.1-0.2$. This spanwise pressure gradient induces the "inflow" toward the center-line within the boundary layer. The longitudinal vortices could be consequently generated in spite of the windward side. The location of the steep pressure gradient is closer to the center-line as increasing the freestream Reynolds number and the angle of attack. Fig. 6 shows plots of the kink point location of spanwise pressure distribution against the location of the separation line obtained in the oil flow visualization. It shows that the location of the steep spanwise pressure gradient depends on the size of the separation on the 1st ramp due to the 2nd ramp shock/boundary layer interaction. It indicates that the longitudinal vortices induced by the spanwise pressure gradient is associated with the 2nd ramp shock/boundary layer interaction.

On the other hand, the wall pressure at the outer location ($y/B>0.2$) in the case of $\alpha=0^\circ$ is higher than the designed value, which is due to the displacement effect by the boundary layer development. Its distribution at the outside is nearly constant in spanwise direction, and the influence of flow expansion at the body side edge on the wall pressure is relatively small for $\alpha=0^\circ$. In the case of $\alpha=5^\circ$, however, the influence reaches around the center-line at $x/L=0.748$ corresponding to the engine inlet location. The wall pressure even at the outer location is not constant in the spanwise direction, and also it falls below the designed value at $y/B>0.3$.

**Pitot Pressure Distribution**

Pitot pressure contours in the case of $\alpha=0^\circ$ and $5^\circ$ are shown in Fig. 7 and Fig. 8, respectively. In each figure, the height and the span are normalized by the engine inlet capture height ($H_{\text{inlet}}$) and the body width ($B$), respectively. Flow condition is of $M=9.68$ and $Re=4\times10^6$.

In the case of $\alpha=0^\circ$, the large pressure loss is caused in the vicinity of the center-line. At the "survey plane-C" ($x/L=0.573$) just downstream of the separation on the 1st ramp, the pressure loss region is
larger. Although the pressure loss region is reduced in the downstream direction, it cannot be diminished by the induced longitudinal vortices. The pressure loss region at the “survey plane-A”(x/L=0.748) corresponding to the engine inlet location occupies approximately 75% in height and 30% in width of the inlet capture area. The large separation on the 1st ramp, inducing such large pressure loss, is because of the laminar boundary layer on the 1st ramp in the present test condition. In the actual flight condition, the boundary layer on the 1st ramp would be so turbulent that the separation due to the shock/boundary layer interaction would be small. The pressure loss region in the actual flight condition would be, therefore, much smaller than that of the present test results. In the case of \( \alpha = 5 \text{deg} \) as shown in Fig.8, the pressure loss region is relatively small as compared with the \( \alpha = 0 \text{deg} \) case.

On the other hand, the influence of the curved shock and the flow expansion near the body side-edge on the Pitot pressure appears at the outside. The area in which it extends is approximately 15% of the engine inlet capture area. The difference between \( \alpha = 0 \text{deg} \) and \( 5 \text{deg} \) is observed in the influence on the wall pressure distribution but not clearly in the influence on the Pitot pressure survey.

**Concluding Remarks**

From the Mach10 wind tunnel tests of the 1.5% space plane fore-body model, the following remarks on the pre-compression flow can be made:

1. The separation on the fore-body due to the ramp shock/boundary layer interaction induces the low pressure region around the center-line, which generates a pair of longitudinal vortices downstream of the separation region in spite of the windward side.
2. The size of the longitudinal vortices depends on that of the separation upstream.
3. By the longitudinal vortices, the pressure loss around the center-line cannot be diminished downstream of the separation region.
4. The area in which the influence of the curved shock and the flow expansion at the body side-edge extend is approximately 15% of the engine inlet capture area in the present test condition.

In the present tests, the boundary layer on the 1st ramp would be so laminar for \( \alpha = 0 \text{deg} \) that the separation due to the 2nd ramp shock/boundary layer interaction is quite large. Consequently the pressure loss region around the center-line at the engine inlet location occupied 75% in height of the engine inlet capture area. In the actual flight condition, the boundary layer would be so turbulent that further investigation will be needed on the influence of the nature of the boundary layer on the fore-body.

**Acknowledgement**

The authors with to thank Dr. Yasushi Watanabe, Dr. Koichi Hozumi and his staff of NAL’s hypersonic wind tunnel for their technical assistance in the wind tunnel tests.

**References**

Fig. 1 Schematic View of Wind Tunnel Model

Fig. 2 Test Set-up of Pitot Pressure Survey
Fig. 3 Oil Flow Pattern on Lower Surface (M=9.68, Re=4x10^6)

(a) \( \alpha = 0 \text{deg} \)

(b) \( \alpha = 5 \text{deg} \)

American Institute of Aeronautics and Astronautics
Fig. 4 Center-Line Wall Pressure Distributions

Fig. 6 Variation of the Kink-point in Pressure Distribution with the Location of Separation-line on the 1st Ramp

Fig. 5 Spanwise Wall Pressure Distributions
Fig. 7 Pitot Pressure Contours ($\alpha=0^\circ, M=9.68, Re=4\times10^6$)

Fig. 8 Pitot Pressure Contours ($\alpha=5^\circ, M=9.68, Re=4\times10^6$)