Assessment of CFD Estimation of Aerodynamic Characteristics of Basic Reusable Rocket Configurations

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(Received May 24th, 2004)

Flow-fields around the basic SSTO-rocket configurations are numerically simulated by the Reynolds-averaged Navier-Stokes (RANS) computations. Simulations of the Apollo-like configuration is first carried out, where the results are compared with NASA experiments and the prediction ability of the RANS simulation is discussed. The angle of attack of the freestream ranges from 0° to 180° and the freestream Mach number ranges from 0.7 to 2.0. Computed aerodynamic coefficients for the Apollo-like configuration agree well with the experiments under a wide range of flow conditions. The flow simulations around the slender Apollo-type configuration are carried out next and the results are compared with the experiments. Computed aerodynamic coefficients also agree well with the experiments. Flow-fields are dominated by the three-dimensional massively separated flow, which should be captured for accurate aerodynamic prediction. Grid refinement effects on the computed aerodynamic coefficients are investigated comprehensively.

Key Words: Compressible Flows, Simulation, Aerodynamic Characteristics

Nomenclature

x, y, z : Cartesian body axes
 $D_{\max}$: maximum diameter of body
 S_{ref} : reference area, $D_{\max}^2 \pi / 4$
 M : Mach number
 Re : Reynolds number based on $D_{\max}$
 Pr : Prandtl number
 μ : viscosity
 α : angle of attack, deg
 C_A : axial force coefficient
 C_N : normal force coefficient
 C_{ma} : pitching-moment coefficient
 computed about theoretical apex

Subscripts

$\text{inf}(\infty)$: freestream condition
max: maximum value
lam: laminar value
tur: turbulent value

1. Introduction

A wide variety of space access concepts have been considered and studied over the last two decades to reduce launch costs and improve the reliability and flexibility of space transportation systems. In 1994, NASA concluded that one option recommended for access to space was development of a fully reusable, all rocket propelled, single-stage-to-orbit (SSTO) launch vehicle.¹⁾ In 1991, the Strategic Defence Initiative Office (SDIO) initiated studies to develop an SSTO rocket that resulted in development of a vertical takeoff and vertical landing (VTVL) concept by

McDonnell Douglas Aerospace Co. (MDAC) called the Delta Clipper.²⁾ The Delta Clipper has extensively demonstrated the reusability of the SSTO rocket and the flight capability of VTVL vehicles. At Institute of Space and Astronautical Science (ISAS/JAXA) in Japan, ground and flight tests of VTVL-SSTO rocket system are being conducted to establish the basic technologies.³⁾

There are two types of entry flight concepts: nose, and base entry, and each has its advantages. The configuration of such a VTVL-SSTO rocket is totally different from conventional rocket configurations and its flight trajectory includes a wide range of flow conditions; subsonic through hypersonic speeds and a variety of attack angles. In addition, high L/D and high maneuverability is one of the key issues for design of the vehicle configuration because general VTVL-SSTO rockets have no wings.

To achieve atmospheric flight, various aerodynamic problems must be solved. In the preliminary design stage of the vehicle configuration, CFD analysis has strong advantages because it supplies a wide range of aerodynamic data, whereas only limited data may be supplied by the wind-tunnel experiments. CFD is already an established essential tool in the design process of aircraft based on the fact that certain level of reliability has been established by accumulation of the flow simulations. However, in the other fields CFD reliability has not been established. This is true for space launch vehicles. Aerodynamic analysis of the VTVL-SSTO rocket includes a wide range of speed regime and attack angles where the flowfield over the body is dominated by three-dimensional separated flows. In recent years, state-of-the-art computations, such as the LES/RANS hybrid method and detached eddy simulations^{4–6)} have become available due to rapid progress in computer power. A large number of sim-

ulations is required for the preliminary design stages, and simulations using RANS are still useful for preliminary stages. The reliability and trend of the RANS simulations should be evaluated in advance because RANS simulation has a limited ability to capture massively separated flows.

An accumulated database for a wide range of flow conditions of RANS simulations would be useful even though it may have limited accuracy. The purpose of this paper is not to pursue accurate simulations but is to evaluate typical RANS simulations.

The study starts with simulations of the Apollo configuration at subsonic to supersonic flow regimes and a wide range of attack angles. The Apollo configuration was chosen since a large amount of experimental data is available⁷⁾ that can be used for comparison with computed results, although the configuration was designed only for entry-flight and is inadequate as an SSTO rocket configuration. The slender Apollo-type configuration is also considered and wind-tunnel experiments have been conducted by the authors to compare the aerodynamic data. As described in the following sections, flowfields over the body are dominated by large separated flows; leeside longitudinal vortexes, and massive separated recirculation. The flow structures of such separated flows should be captured for accurate aerodynamic prediction. A grid refinement study is performed for such separated flows.

2. Numerical Method

2.1. Geometries

The body configurations considered in the present study are shown in Fig. 1. All geometries are non-dimensionalized by the maximum diameter of the body $D_{\max}$. Although flow simulations are performed for various Apollo-like configurations, two configurations are considered in this paper to discuss the ability of RANS simulations. Apollo configuration parameters are set to be the same as the parameters described in Ref. 7. Case 3b has a cone half-angle of 21° , which is the same configuration considered in the previous studies.^{8,9)} Other parameters are the same as the Apollo configuration.

2.2. Body axes

As shown in Fig. 2, the angle of attack $0^\circ \leq \alpha \leq 90^\circ$ is the nose-entry conditions and $90^\circ \leq \alpha \leq 180^\circ$ is the base-

Fig. 2. Sketch showing body system of axes.

entry conditions. The pitching-moment coefficient C_{ma} is computed about the theoretical apex. The definition of the aerodynamic forces is shown in Fig. 2.

2.3. Computational domain and grid

The flow-fields are assumed to be symmetric about the pitch-plane ($y = 0$). Therefore, the computational domain covers only half the body. The effect of the support sting is not considered. Details of the computational grids are listed in Table 1. Grid 01 consists of 53 equispaced circumferential planes ($\Delta\phi = 3.6^\circ$) extending around half the body. In each circumferential plane, the grid contains 61 radial points between the body and the computational outer boundary and 91 axial points between the nose and the rear end of the body. This small grid size is intentionally used since a main purpose of this study is to assess the ability of RANS simulation where large numbers of cases are simulated in a standard computer environment. The minimum grid spacing near the wall is 3.0×10^{-5} and the computational domain

Table 1. Details of computational grids.

Grid name	Minimum grid spacing	(ξ, η, ζ -dir)
Grid 01	3.0×10^{-5}	(91 $\times$ 53 $\times$ 61)
Grid 02	3.0×10^{-5}	(91 $\times$ 85 $\times$ 61)
Grid 03	3.0×10^{-5}	(91 $\times$ 85 $\times$ 117)
Grid 04	3.0×10^{-5}	(163 $\times$ 53 $\times$ 171)

Fig. 1. Configuration parameters.

Fig. 3. Definition of computational grid (Grid 01).

Fig. 4. Entire view of computational grid (Grid 01).

ranges from -10.0 to $+10.0$ in all directions as shown in Fig. 4.

An additional three computational grids, Grid 02, 03 and 04, are also used for grid refinement studies. Grid 02 has the same grid distributions in the circumferential planes as Grid 01 and consists of 85 equi-spaced circumferential planes. Grid 03 consists of 91 axial points, which is the same as Grid 01, 117 radial points and 85 equi-spaced circumferential planes.

2.4. Flow solver

The basic equations are the three-dimensional compressible Navier-Stokes equations.

$$\frac{\partial \hat{Q}}{\partial t} + \frac{\partial \hat{E}}{\partial \xi} + \frac{\partial \hat{F}}{\partial \eta} + \frac{\partial \hat{G}}{\partial \zeta} = 0 \quad (1)$$

Length, density ρ , and velocity components u , v , w are normalized by $D_{\max}$, ρ_{∞} and the freestream velocity of sound c_{∞} , respectively. The pressure, density and velocity components are related to the energy for an ideal gas by the following equation:

$$p = (\gamma - 1) \left\{ e - \frac{1}{2} \rho (u^2 + v^2 + w^2) \right\} \quad (2)$$

The metrics are evaluated using second-order central-difference formula for interior points. Numerical fluxes for convective terms are evaluated by the SHUS scheme (Simple High-resolution Upwind Scheme),¹⁰⁾ extended to high-order space accuracy by the 3rd-order upwind-biased MUSCL interpolation¹¹⁾ based on primitive variables. The viscous terms are evaluated by the 2nd-order central differencing. The flow-field is considered to be fully turbulent, and the Baldwin-Lomax turbulence model¹²⁾ (BL-model) is applied. Note that this popular turbulence model is also used intentionally because the objective of the present paper includes the assessment of the prediction ability of typical RANS simulations. The LU-ADI factorized implicit algorithm¹³⁾ is used for the time integration with a globally constant time step with no inner iterations. The non-dimension-

Table 2. Flow conditions for calculations for Apollo.

M_{∞}	0.7, 0.9, 1.2, 2.0
Re_{∞}	1.0×10^6
α	0, 15, 24, 36, 45, 60, 70, 80, 90, 100, 110, 120, 130, 144, 156, 165, 180°

al time step used here is $dt^* = \frac{dt}{D_{\max}/c_{\infty}} = 0.001$.

2.5. Flow parameters

The flow conditions for the calculations for Apollo are listed in Table 2. Since the flight dynamic pressure is significantly large in the subsonic to supersonic conditions, consideration of the subsonic to supersonic aerodynamic characteristics is important for the preliminary design stage. Therefore, aerodynamic characteristics in this speed regime are focused on. Pr_{lam} , Pr_{tur} and μ_{lam} are fixed at 0.72, 0.90 and 1.0 in all calculations.

2.6. Data processing

The aerodynamic forces on the body are strongly influenced by flow unsteadiness behind the body. At subsonic and transonic conditions, the leeside pressure varies significantly with time due to unsteady re-circulating wake flow. Since the unsteady wake flow has temporal periodicity, the aerodynamic coefficients in the following discussions are all time-averaged values. The time average is taken over a sufficiently long time span of about $\delta t^* = 60$. The computed largest variations from the mean are $\Delta C_A = 0.671$ at $M_{\infty} = 0.7$, $\alpha = 70^\circ$ and $\Delta C_N = 0.382$ at $M_{\infty} = 0.7$, $\alpha = 0^\circ$.

3. Wind-tunnel Experiments for Case 3b

The wind-tunnel experiments were carried out in the ISAS/JAXA transonic 600×600 mm wind-tunnel. It is a blow-down type with a measuring time of about 30 s and freestream Mach number from 0.3 to 1.3. The flow conditions of the present experiments and the experiments by William et al. are listed in Tables 3 and 4, respectively.

The present experiments are carried out at nose-entry condition, $\alpha = 0^\circ \sim 36^\circ$. Aerodynamic forces and moments were measured using a sting-mounted 6-component strain-gauge balance. The dimensions and details of the model are shown in Fig. 5. The blockage ratio at $\alpha = 0^\circ$ is about 1.39%. The diameter of the model is only about four times larger than the diameter of the support sting. Therefore, the support sting may significantly influence the pressure behind the body. Since pressure change in the base region has little effect on C_N for this configuration, computed C_N is only compared with the experimental results for Case 3b. It is well known that leeside longitudinal vortical flows over slender bodies of revolution become asymmetric at

Table 3. Flow conditions for experiments and calculations for Case 3b.

M_{∞}	0.7, 0.9, 1.2
Re_{∞}	1.53, 1.75, 1.89×10^6
$\alpha(\text{exp.})$	0 $\sim$ 36°
$\alpha(\text{CFD})$	0, 15, 24, 35°

Table 4. Flow conditions for William et al.'s experiment.⁷⁾

M_{∞}	0.7, 0.9, 1.2, 2.0
Re_{∞}	3.9, 3.3, 2.8, 3.6×10^6

Fig. 5. Schematic view of wind-tunnel test model for Case 3b.

high angles of attack,¹⁴⁾ inducing a side force even at zero side-slip angle. If so, the aerodynamic characteristics are significantly influenced. Neither side force nor flow asymmetry was observed in the present experiments. The result justifies the assumption of symmetrical flow about the pitch-plane in the computations.

4. Results and Discussions

4.1. Comparison with experiments for Apollo configuration

Comparison of the computed and experimental⁷⁾ aerodynamic coefficients of the Apollo configuration is shown in Figs. 6(a)–(c). The computed results are obtained by computations on Grid 01. The linear and non-linear nature of the aerodynamic characteristics against the angle of attack is predicted successfully by the present computations.

Flow structures over the body change dramatically with angle of attack. At low and medium angles ($\alpha = 0^\circ \sim 45^\circ$), the aerodynamic characteristics change linearly with α ; the flow over the body is attached or is separated and shed into leeside longitudinal vortex flow as shown in Fig. 7(a). At high angles ($\alpha > 45^\circ$), the aerodynamic characteristics show non-linear nature; the cross-flow becomes dominant and cross-flow separation occurs in the wide portion of the body surface. At the base entry conditions near $\alpha = 180^\circ$, the flow is highly decelerated in the base region and separates at the sharp shoulder of the body to produce massive separated re-circulating flow behind the body as shown in Fig. 7(b).

Generally, computed results are in good agreement with experiments. However, some discrepancies can be seen in C_N at $M_\infty = 0.7$, $\alpha \approx 90^\circ$ and C_A at $\alpha \approx 0^\circ$ and 180° in all Mach numbers. Grid refinement studies are performed for these conditions and discussed in the following sections. It must be noted that despite computations with the comparatively small grid size, typical RANS simulation can predict the aerodynamic characteristics in a wide range of flow conditions. Except for the conditions mentioned above, the prediction uncertainty is estimated to be 5%. This level is sufficient to the preliminary design stages.

Grid refinement study for $M_\infty = 0.7$, $\alpha \approx 90^\circ$

A disagreement was seen in C_N and C_{ma} especially at $M_\infty = 0.7$, $\alpha \approx 90^\circ$. In such high-angle-of-attack conditions, large cross-flow separation occurs over the entire body surface and recirculating wake flow is generated behind the body as shown in Fig. 8. Since the computed C_N

(a) Axial force coefficient, C_A .

(b) Normal force coefficient, C_N .

(c) Pitching moment coefficient, C_{ma} .

Fig. 6. Computed aerodynamic coefficients for Apollo with experimental data.⁷⁾

is larger than the experimental data, the disagreement is thought to be due to underestimation of leeside pressure. Therefore, the computational grids are refined especially in the circumferential and radial directions (Grid 02 and 03 in the Table 1). Grid 02 is obtained by refinement of Grid 01 in the circumferential direction and Grid 03 is ob-

(a) $M_\infty = 0.7$ $\alpha = 45^\circ$. (b) $M_\infty = 0.7$ $\alpha = 144^\circ$.

Fig. 7. Time-averaged computed Mach number distributions in symmetric plane and near-surface stream pattern for Apollo (Grid 01).

Fig. 8. Time-averaged computed local Mach number distributions in symmetric plane and near-surface flow pattern for Apollo at $M_\infty = 0.70$, $\alpha = 90^\circ$ (Grid 03).

tained by refinement in both the axial and circumferential directions. The grid refinement effect is shown in Figs. 9(a)–(c). The computed results with Grid 02 show little improvement but the computed results with Grid 03 are remarkably improved and agree well with the experiment. Computed surface pressure distributions in the circumferential direction at $x = 0.50$ are shown in Fig. 10 for each grid, where $\phi = 0^\circ$ corresponds to windside.

(a) Axial force coefficient, C_A . (b) Normal force coefficient, C_N . (c) Pitching moment coefficient, C_{ma} . Fig. 9. Grid refinement study on aerodynamic coefficients for Apollo for $M_\infty = 0.70$.

There is little difference in the separation location. The pressure distribution in the attached flow region, $\phi = 0^\circ \sim 90^\circ$, is almost the same, large difference can be seen in the leeside pressure distributions, $\phi = 90^\circ \sim 180^\circ$. Leeside

Fig. 10. Time-averaged computed surface pressure distributions at $x = 0.50$ for Apollo at $M_\infty = 0.70$, $\alpha = 90^\circ$.

Fig. 11. Time-averaged computed local Mach number distributions at $x = 0.50$ for Apollo at $M_\infty = 0.70$, $\alpha = 90^\circ$.

pressure predicted by Grid 03 is higher than that predicted by the other two grids.

In addition, the leeside pressure distribution with Grid 03 is flat but there is a slight gradient in the results with Grid 01 and 02. This pressure gradient is due to poor grid resolution. Comparison of the computed local Mach number distributions in the $x = 0.50$ axial plane is shown in Fig. 11. An apparent grid refinement effect can be seen in flow structures in the re-circulation region. In the computations with Grid 03, the shear layers are captured sharply and inclined outward compared to those with Grid 01. This leads to comparative weak recirculating flow, which yields the higher leeside pressure distributions. As a result, C_N became smaller and showed good agreement with the experiments. These results indicate that sufficient grid resolutions are required in the region of the recirculating wake flow especially at high-angle-of-attack conditions around $\alpha = 90^\circ$.

Grid refinement study for $\alpha = 0^\circ$

and 180° at all the Mach numbers

(a) $M_\infty = 0.7$ $\alpha = 0^\circ$.

(b) $M_\infty = 0.7$ $\alpha = 180^\circ$.

Fig. 12. Time-averaged computed Mach number distributions in symmetric plane for Apollo (Grid 01).

Table 5. C_A for Apollo at $\alpha = 0^\circ$.

M_∞	Experiment	CFD(Grid 01)	CFD(Grid 04)
0.70	0.5706	0.6811	0.6306
0.90	0.7267	0.8371	0.7887
1.20	1.0227	1.1997	1.0816
2.00	0.9526	1.0015	0.9495

The C_A is predicted to be larger than experiments at $\alpha \approx 0^\circ$ and 180° . The separation line is essentially fixed at the shoulder due to its small shoulder radius and massive re-circulation occurs behind the body under these conditions as shown in Fig. 12. Therefore, the discrepancy comes from the disagreement in the base pressure. Therefore, computations with grid density highly concentrated in the recirculation region were carried out as a grid refinement study (Grid 04 in Table 1). The flow conditions are $M_\infty = 0.7, 0.9, 1.2, 2.0$ and $\alpha = 0^\circ$. Comparison of the computed and measured C_A is shown in Table 5. Under all the conditions, computed C_A decreases as the number of the grid points increases, showing better agreement with the experiment. The effect of the grid density on the computed pressure distributions in the symmetric plane at $M_\infty = 1.20$, $\alpha = 0^\circ$ is shown in Fig. 13. Little difference is observed in the windside pres-

Fig. 13. Computed pressure distributions in symmetric plane for Apollo at $M_\infty = 1.2$, $\alpha = 0^\circ$.

Table 6. C_A for Apollo at $M_\infty = 1.2$.

α	Experiment	CFD(Grid 01)	CFD(Grid 04)
180	-1.3296	-1.4604	-1.3133
165	-1.3180	-1.4894	-1.3610
156	-1.2787	-1.4408	-1.3620
144	-1.2072	-1.3149	-1.2633

sure distributions but the base pressure of Grid 04 is larger than that of Grid 01, resulting in the decrease of C_A for Grid 04.

A grid refinement study was also carried out under the conditions of $M_\infty = 1.2$, $\alpha = 180^\circ$, 165° , 156° and 144° , and the results are shown in Table 6. Under all conditions, the magnitude the C_A decreases. Increase of leeside pressure contributes to improvement of computations. The prediction uncertainty is estimated to be about 8% of the magnitude. The remaining discrepancy would be reduced by recent simulation techniques, such as the LES/RANS hybrid method⁵⁾ and DES,¹⁵⁾ which predict a base pressure higher than the conventional RANS simulation and is validated by comparison with experiments. Note that the purpose of the present paper is not to pursue better agreement but to show that the RANS simulation is useful and sufficient at the preliminary design stage.

4.2. Comparison with experiment for Case 3b

Comparisons of the aerodynamic coefficients obtained by our wind-tunnel experiment for Case 3b are shown in Fig. 14. All the results are obtained in computations with Grid 01. Measured C_Y is also shown in Fig. 14, indicating that there is no side-force so the leeside flow structure is symmetric. C_N variations are almost linear with respect to α because the leeside longitudinal vortex becomes strong with increase in α . Such aerodynamic characteristics are predicted successfully by the present computation. The gradient of C_N with respect to α , $\partial C_N / \partial \alpha$, is predicted accurately, which is important in aerodynamic considerations. Also in the Apollo results, the computed $\partial C_N / \partial \alpha$ agrees well with measured data at $\alpha = 0^\circ \sim 45^\circ$ as shown in Fig. 6(b). How-

Fig. 14. Computed normal force coefficient for Case 3b.

Experiment

(a) $M_\infty = 0.9$, $\alpha = 24^\circ$.

Experiment

(b) $M_\infty = 0.9$, $\alpha = 36^\circ$.

Fig. 15. Comparison of computed and experimental near surface stream pattern (time-averaged in computation).

ever small discrepancies can be seen between the computed and measured results at $M_\infty = 0.90$, 1.20 , $\alpha = 35^\circ$. Computed and experimental near surface stream patterns at $M_\infty = 0.9$ are presented in Fig. 15. As can be seen in the experimental results at $\alpha = 24^\circ$, the leeside separated region is relatively narrow and the computed separation location agrees well with the experiment. Whereas at $\alpha = 35^\circ$, the leeside separated region is significantly large covering almost the entire leeside surface of the body, and the computed near-surface flow pattern is different from the experiment to some extent. At high angles of attack and transonic conditions, the side wall effect of the wind-tunnel is considered to be large so a further grid-refinement study is not conducted. At

$\alpha = 35^\circ$, the gap between the model nose and top wall of the test section is about 100 mm.

Note that the CFD C_N prediction is sufficiently reliable at low angles of attack, $\alpha = 0^\circ \sim 24^\circ$.

5. Conclusions

The reliability and trend of RANS computations for the aerodynamic prediction of a basic reusable rocket configurations have been evaluated. Generally, computed aerodynamic coefficients for Apollo are in excellent agreement with the experiment. The prediction uncertainty is estimated to be less than 5% of the magnitude of the aerodynamic coefficients and the level is sufficient for the preliminary design stage. The computed values of C_N for the slender Apollo-type configuration are also in good agreement with the experiment. The conventional RANS computation is sufficiently reliable and useful for the preliminary design stage.

Grid resolution studies have been performed at conditions where discrepancy is observed. At $M_\infty = 0.7$ and $\alpha \approx 90^\circ$, a sufficient grid density is required, especially for the re-circulating flow for accurate estimation of C_N . At $\alpha \approx 0^\circ$ and 180° , computed values of C_A tend to be slightly larger than the experimental values, which is thought to be due to the underestimation of base pressure. Some improvement is observed as the number of the grid points increases in the re-circulating region.

Acknowledgments

This research was partially supported by a Grant-in-Aid from the Ministry of Education, Culture, Sports, Science and Technology of Japan (No. 13450400).

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