Dynamic Inversion Controller Design for Balloon-Launched Supersonic Aircraft

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While balloon launch is one of the most cost-efficient ways to realize a supersonic flight experiment, it presents some issues for the controller. D-SEND#2 is one flight experiment of this kind which was carried out in 2015. An unpowered test vehicle was lifted to an altitude of 30 km by a balloon and then released. After separation, the vehicle’s onboard flight control computer selected a target Boom Measurement System (BMS) according to the separation point. The vehicle then autonomously flew to the BMS selected and established the prescribed sonic boom measurement flight conditions. The design of the guidance, navigation, and control system for the D-SEND#2 flight test was exceptionally challenging, and it was solved by a combination of a sophisticated guidance law and a control law based on dynamic inversion and timescale separation. This paper describes a controller design method for a balloon-launch flight experiment and applied it to a D-SEND#2 control system design. Flight results are presented to show the effectiveness of the controller design method.

Key Words: Guidance and Control, Dynamic Inversion, Timescale Separation, D-SEND

Nomenclature

| Symbol | Description |
|--------|-------------|
| $A_2, A_3, A_4$: | system matrices for linearized model |
| $A_{HPP}, B_{HPP}, C_{HPP}, D_{HPP}$: | matrices for high-pass filter |
| $C_L, C_D, C_m$: | aerodynamic coefficients |
| $C_{LC}, T_i$: | constants for a lead compensator |
| $DCM$: | direction cosine matrix |
| $K_i$: | control gain in $i$th layer |
| $K_{iI}, K_{iF}$: | controller gain for an integral of ( ) |
| $N_s$: | lead factor |
| $X, V, \Theta, \Omega$: | vectors of position, velocity, attitude, angular velocity |
| $f_i$: | equations of motion in $i$th layer |
| $u$: | control vector |
| $x$: | state variables |
| $\tilde{x}_i$: | controlled variables in $i$th layer |
| $\dot{x}_i$: | variables other than those controlled in $i$th layer |
| $a_i$: | acceleration of $Y$ axis |
| $p, q, r$: | angular velocity |
| $q_{HPP}, s$: | pitch rate after passing high-pass filter |
| $q_{LC}$: | pitch rate after passing lead compensator |
| $u, v, w$: | velocity in body axis frame |
| $x_{HPP_n}$: | state variable of high-pass filter at $n$ step |
| $\alpha, \beta$: | angle of attack, and sideslip angle |
| $\gamma$: | flight path angle |
| $\delta$: | control variables |
| $\delta_c, \delta_a, \delta_r$: | angles of elevator, aileron, and rudder |
| $\sigma$: | standard deviation |
| $\phi, \theta, \psi$: | Euler angle of the vehicle |

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1. Introduction

Recently, a supersonic transport (SST) has come to be considered as an achievable target for the aerospace industry, and some companies have presented actual plans. However, developing a SST requires many problems to be solved, one of the most important of which is how to reduce sonic boom. Various approaches to solve this issue have been proposed, but any solution should be evaluated by flight experiment to demonstrate its effectiveness, and supersonic flight tests are very costly. Among the various ways of carrying out supersonic flight experiments, the balloon launch method is one of the most cost-efficient.

While balloon launch is very cost-efficient, it has several limitations. To address them, this paper presents a systematic controller design procedure and a controller design scheme of using the Hierarchy-Structured Dynamic Inversion (HSDI) method, which is based on dynamic inversion (DI) and timescale separation. This solves the above problems while implementing a sophisticated guidance law using a real-time optimization technique.

Dynamic inversion has been widely applied to flight control problems. Meyer et al. applied it to the design of a helicopter autopilot, where a nonlinear model was transformed into canonical form. Lane and Stengel applied DI to the control of aircraft at high angles of attack, and Enns et
al. summarized how to select controlled variables and how to design control laws using DI.11 There are many other related works.12–15

On the other hand, timescale separation of the rotational dynamics combined with the singular perturbation theory has been proposed, whereby attitude angles are taken as slow variables and angular velocities are taken as fast variables.16,17 In this method, the slow variables are controlled by the fast variables, which are in turn controlled by aerodynamic control surfaces. This method takes advantage of the characteristics of the equations of motion of a typical aircraft. A general survey of the timescale separation technique applied to aerospace is summarized in Naidu and Calise.18

In this paper, we present a design procedure for a controller using the HSDI method, which combines the advantages of DI and the timescale separation technique. Its application to a balloon-launched vehicle is then precisely described and its flight results are shown, followed by concluding remarks. The major contribution of this paper is the presentation of a design procedure applying the HSDI method and proof of its effectiveness through an actual flight test.

2. Overview of Balloon Launch Flight Experiment

This section provides an overview of a general balloon launch flight experiment and problems to be solved by guidance, navigation, and control (GNC) systems. Then, D-SEND#2 (the second phase of the Drop test for Simplified Evaluation of Non-symmetrically Distributed sonic boom) is described, to which the controller design procedure proposed is applied.

2.1. Balloon launch flight experiment

Balloon launch is a cost-efficient way to realize a supersonic flight test since it is not necessary to equip the test vehicle with an engine or booster. Instead, a stratospheric balloon is used to lift the vehicle to a very high altitude of 18–37 km. Then the vehicle is released and accelerates to supersonic speed by gravity. The nature of this method has attendant problems to be solved.

In typical aircraft development, an aircraft flies under its own power and its flight envelope is expanded step-by-step from low speed, low altitude through to high speed, high altitude. However, such incremental flight envelope expansion requires much time and cost. A balloon drop test with an unpowered sub-scale test vehicle was fabricated, and the strength of its sonic boom in flight was measured in the D-SEND#2 project.19 The vehicle’s physical characteristics are shown in Table 1. For flight control, the vehicle has two stabilators and a single rudder. The stabilators move differentially for roll control (one-half of the difference between the deflection angles of the left and right stabilators is treated as aileron angle) and collectively for pitch control (the average surface deflection angle is treated as elevator angle).

Figure 1 shows the plan of the D-SEND#2 flight test. The test was conducted at the Esrange Space Center in Sweden, which is supervised by the Swedish Space Corporation (SSC). Three Boom Measurement Systems (BMSS) were set up in the Zone-B flight test area (Fig. 6(c)). A BMS comprises a series of microphones attached to the mooring cable of a blimp to record the sound of the sonic boom generated by the test vehicle at various heights above the ground, and associated ground support facilities. The vehicle is lifted

Another problem is that it is almost impossible to control the horizontal position of the balloon. Although the balloon’s ascent rate can be controlled to some extent by exhausting buoyancy gas, the balloon is carried by the wind and so the only lateral control is to wait for a specific wind direction before launching and to allow drift for a certain time. Consequently, the horizontal position control capability is very poor.

Finally, since the vehicle is unpowered, its flight trajectory must be carefully designed to realize the experiment’s target flight conditions. This directly demands the GNC systems to be highly versatile.

2.2. Overview of D-SEND#2

The Japan Aerospace Exploration Agency (JAXA) has been developing an aircraft design concept to reduce sonic boom. To validate this concept, an unpowered sub-scale test vehicle was fabricated, and the strength of its sonic boom in flight was measured in the D-SEND#2 project.19 The vehicle’s physical characteristics are shown in Table 1. For flight control, the vehicle has two stabilators and a single rudder. The stabilators move differentially for roll control (one-half of the difference between the deflection angles of the left and right stabilators is treated as aileron angle) and collectively for pitch control (the average surface deflection angle is treated as elevator angle).

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to an altitude of 30 km suspended vertically from a balloon, and separates when it drifts to within a given distance from any one of the BMSs. After separation, the vehicle’s onboard flight control computer selects one of the BMSs as a target according to the separation point. The vehicle then autonomously flies to the selected BMS and establishes prescribed sonic boom measurement flight conditions in its vicinity. Once the vehicle has passed by the BMS, its flight is promptly terminated to avoid leaving the Zone-B area. Since the vehicle was not equipped with a recovery system, such as parachutes or an air bag, flight termination results in its destruction and so the experiment is a ‘single shot’ with only one chance of achieving its objectives.

2.3. GNC requirements

Table 2 shows the GNC system requirements of D-SEND#2, which include mission and flight requirements along with controller design criteria.

The conditions for sonic boom measurement were geometrically checked during the controller design process. The sonic boom is modeled as propagating along the surface of a cone. Multiple sonic booms emanating at different points along the flight path may overlap at a BMS due to variations of the flight path angle. This phenomenon is called sonic boom focusing, and it is necessary to avoid it at a BMS. This constraint is included in the vehicle’s mission requirements and is required for both direct sonic boom measurement and the measurement of sonic boom reflections from the ground. During the GNC system evaluation process, aerodynamics experts analyzed more precisely whether or not the sonic boom would be measured correctly and or not sonic boom focusing would occur at a BMS. The results of the precise analysis coincided well with those of the simplified geometrical analysis.

2.4. GNC design challenges

As described above, the test vehicle separates from the carrier balloon at an arbitrary location within the separation area and is then required to attain specific flight conditions around a selected BMS. These conditions are related not only to the state variables against the air, such as Mach number and $\alpha$, but also to those against the ground, such as position, altitude and flight path angle. The following are the major challenges in satisfying these demands.

1. There is no nominal (i.e., predetermined) trajectory. A reference flight trajectory is determined by the relationship between the separation point and the BMS selected. These parameters can be determined only after separation.
2. The vehicle has only three control surfaces, but many state variables must be controlled to achieve the required measurement conditions.
3. The required measurement conditions must be sustained for the prescribed duration without the aid of thrust or speed brakes.
4. The vehicle’s flight envelope ranges from zero velocity at an altitude of 30 km to supersonic flight at less than 10 km (Fig. 2).

These factors make the design of D-SEND#2’s GNC system more difficult than for any of JAXA’s previous flight experiments. Accordingly, the mission success rate requirement was relaxed to a less stringent 90%. In contrast, the success rate required for a former balloon-launched flight test, the High Speed Flight Demonstration Phase II flight test, was 98%.

2.5. GNC system of D-SEND#2

This section presents an outline of the D-SEND#2 GNC system.

2.5.1. System overview

The GNC system comprises a navigation interface, reference trajectory generation function, model linearizing func-
are decomposed into initial conditions are obtained beforehand, and each of these generated during the vehicle is separated from the balloon, a reference trajectory is to prevent structural coupling with control commands. The structural coupling motions and is necessary for dynamic inversion control. The model linearizing function calculates linearized equations of motion and is necessary for dynamic inversion control. The structural coupling filter is a low-pass filter designed to prevent structural coupling with control commands.

2.5.2. Reference trajectory

Since initial conditions are not determined until the vehicle is separated from the balloon, a reference trajectory is generated during the flight. Optimum trajectories for various initial conditions are obtained beforehand, and each of these are decomposed into five flight phases as listed in Table 3. Within each phase, except for the dive phase, the reference trajectory is generated by approximating the optimal trajectory segment using simple polynomials. The reference trajectory is precisely described in Kawaguchi et al. and Suzuki and Ninomiya.

During a flight, the reference trajectory generation function is invoked at 10 Hz to generate guidance commands to the control law from the current state variables using the approximated reference trajectory. In the control law, a lateral control is invoked at 10 Hz to generate guidance commands to enhance control performance. The guidance commands for each flight phase and the lateral control phase are shown in Table 4.

2.5.3. Model linearizing function

As is mentioned earlier, the hierarchy-structured dynamic inversion method uses linearized equations of motion to approximate the flight dynamics. In the model linearizing function, state variables measured by sensors and estimated using a mathematical model are used to calculate differential matrices. Since this calculation has a heavy computational load, this function is implemented as a low-cycle 10 Hz task.

3. Control Law Design

In this section, the general procedure of HSDI based on the controller design method is first described, and then the control law design of the D-SEND#2 flight test is explained in detail. Then, the evaluation results of the controller designed using the three methods are presented. Finally, the design method is compared to a conventional PID control law.

3.1. Design procedure

The control law using HSDI and the timescale separation method is designed using the following procedure.

1. Determination of control law structure (selection of feedback variables)
2. Single error analysis
3. Linear analysis
4. Monte-Carlo simulation
5. Linear analysis

In the first step, the controller structure is determined and controller gains are tuned for a representative case. In the second step, each of the error parameters is set to either a ±3σ value for errors with a normal distribution, or to a minimum/maximum value for errors with a uniform distribution, and the simulation results are evaluated. The controller gains are tuned to accommodate all of these error cases and the controller structure is revised if necessary. In the third step, linear analysis is applied to ensure the stability of the controller and the controller structure is again revised to satisfy controller design criteria. Then, Monte-Carlo simulation (MCS) is carried out and the controller is revised until it satisfies the system requirements. Finally, linear analysis is again applied to confirm that the controller has sufficient stability since MCS can evaluate mission success rate but not controller stability. Single error analysis is not necessarily repeated because MCS covers single error cases.

3.2. General description of HSDI

HSDI exploits the characteristics of a vehicle’s flight dynamics. Figure 4 shows the concept of HSDI. The rigid-body motions of an aircraft—angular acceleration, angular velocity, angle and acceleration, velocity, and position—have different timescales. Therefore, the entire controller structure can be arranged in a hierarchy and DI can be applied separately to each layer. State variable $x$ can be separated into four groups according to the timescale.

Table 3. Flight phases.

| No. | Phase         | Description                          |
|-----|---------------|--------------------------------------|
| 1   | Acceleration  | Acceleration and heading adjustment   |
| 2   | Pull-up       | Pull-up maneuver for gliding flight   |
| 3   | Glide         | Gliding flight for range adjustment   |
| 4   | Dive          | Dive maneuver for re-acceleration     |
| 5   | Measurement   | Sonic boom measurement at target conditions |

Table 4. Guidance commands.

| Phase         | Flight | Lateral control | Commands |
|---------------|--------|-----------------|----------|
| 1             | 1      | 1               | $\psi_f, \theta, a_f$ |
| 2             | 1      | 1               | $\psi_f, a, a_f$   |
| 2             | 2      | 2               | $\psi, \alpha, \beta$ |
| 3             | 3, 4   |                  | $\psi, \gamma, \beta$ |
| 4             | 4      | 4               | $N_1, \psi, \beta$ |
| 5             | 4      | 4               | $N_1, \phi, \beta$ |
causal relationship between control surface deflection and plant using state feedback. For typical flight dynamics, the causal relationship between control surface deflections and state variables is as follows: Control surfaces generate moments that excite angular accelerations, and angular accelerations change angular velocities and attitude angles. Then, attitude angles affect aerodynamic forces, which in turn, alter the vehicle’s acceleration, velocity, and position. DI reverses this causal chain to calculate control commands from a commanded vehicle position.

The procedure of HSDI is as follows. In the ith layer, let a controlled variable be \( x_i \), a control variable be \( x_{i+1} \), and other variables and control variables be \( \hat{x}_i \) and \( \delta \), respectively. Then, the equation of motion is

\[
\hat{x}_i = f_i(\hat{x}_i, x_i, x_{i+1}, \delta). \tag{6}
\]

A simple proportional controller can be designed as

\[
\hat{x}_i = -K_i(x_i - x_i). \tag{7}
\]

Using Taylor series expansion of Eq. (6) and Eq. (7), \( x_{i+1} \), can be determined by solving

\[
\hat{x} \approx \hat{x}_{\text{ref}} + \left( \frac{\partial f_i}{\partial x_{i+1}} \right) (x_{i+1} - x_{i+1}) = -K_i(x_i - x_i), \tag{8}
\]

and it is solved as

\[
x_{i+1} = x_{i+1} - \left( \frac{\partial f_i}{\partial x_{i+1}} \right)^{-1} \left[ x_{\text{ref}} + K_i(x_i - x_i) \right], \tag{9}
\]

where \( x_{\text{ref}} \) is the estimated differential value of \( x_i \) calculated from the current state variables. The right-hand side of Eq. (7) is a very simple example of a controller definition and can be modified by adding terms such as an integral term or a feed-forward term. In such cases, Eq. (9) should also be modified. This method can handle nonlinear dynamics and is therefore able to deal with a wide flight envelope range without gain scheduling.

In general, a DI controller is susceptible to model errors, but timescale separation helps improve robustness of the DI controller.\(^{17}\) The hierarchical structure simplifies the dependencies between state variables. The state variables in the ith layer act as integral terms of state variables in the \((i + 1)\)th layer. For example, model errors cause angular velocity errors in the fourth layer, and errors in the fourth layer cause angle errors in the third layer. However, angular velocity commands are generated to reduce angle errors, and these commands prompt angular velocity change, which reduces angle error.

### 3.3. HSDI applied to D-SEND#2

#### 3.3.1. Selected variables for HSDI

Table 5 shows specific control command conversions applied in the D-SEND#2 control law. HSDI is applied to the underlined conversions, while conventional PID control is used for the remainder. Concerning the third layer lateral control conversion, the directional command \( \psi_c \) is converted to \( \phi_c \) and \( \beta_c \), and then \( \psi_c \) and \( \beta_c \) are used to generate \( p_c \) and \( r_c \) commands. To control vehicle direction properly, directional angle is replaced by \( \psi_f \), which is the direction of a projection of the vertical tail onto the horizontal plane. The direction of the vertical tail for the body axis is expressed as \([0 \ 0 \ -1]^T\), and this can be written for the local horizontal axis as

\[
DCM^{-1}[0 \ 0 \ -1]^T = \begin{bmatrix}
    \cos \psi \sin \theta \cos \phi + \sin \psi \sin \phi \\
    \sin \psi \sin \theta \cos \phi - \cos \psi \sin \phi \\
    \cos \theta \cos \phi
\end{bmatrix}.
\]

\( \tag{10} \)
Therefore, $\psi_F$ is defined using its projection to the local horizontal plane as follows:

$$\psi_F = \tan^{-1}\left(\frac{\sin \psi \sin \theta \cos \phi - \cos \psi \sin \phi}{\cos \sin \theta \cos \phi + \sin \psi \sin \phi}\right).$$

(11)

There are two reasons for using PID controllers for some variables. The first is to reduce computational load. For example, during flight phase 2, $a_c$ is converted to $q_c$, and this requires $\dot{a}_c$. To calculate $\dot{a}_c$, it is necessary to calculate aerodynamic coefficients many times, which requires rare tables with up to five dimensions and then interpolation. Such a heavy computational load overloads the flight control computer (FCC). The second is to avoid the singularity points of Euler angles. The equations of motion implemented in the FCC are expressed by Euler angles because quadratures do not clearly express the relationship between controller gains in the fourth layer and control variables, which is quite important for tuning controller gains. During flight phases 1 and 2, the vehicle is orientated almost vertically downward, therefore the lateral and directional motion are coupled.

### 3.3.2. HSDI control law example

The general formulation of a HSDI-based control law was described in sections 3.2 and 3.3.1. Although page limitations do not allow us to present the formulas for each flight and control phase, the specific formulas for flight phase 3 are given here as an example.

In the second layer, $y_t$ is converted to $\theta_t$ using

$$A_2 = \left[\begin{array}{c} \frac{\partial y_t}{\partial \theta} \\
\end{array}\right]$$

(12)

$$\theta_t = \theta - A_2^{-1} \cdot \left\{ y_t + K_{y} \cdot (y_t - y_c) + K_{y_c} \cdot y_c \right\}.$$  

(13)

The flight control software is equipped with the aircraft model, and $\dot{y}_c$ can be obtained using the equation

$$\dot{y}_c = \frac{\partial y}{\partial \theta} \cdot \dot{\theta} + \frac{\partial y}{\partial v} \cdot \dot{v} + \frac{\partial y}{\partial \omega} \cdot w + \frac{\partial y}{\partial \phi} \cdot \dot{\phi} + \frac{\partial y}{\partial \theta} \cdot \dot{\theta} + \frac{\partial y}{\partial \psi} \cdot \dot{\psi};$$

(15)

where the derivatives of $y$ with respect to state variables are known and the time derivatives of state variables can be calculated from the aircraft model and state variables measured. Once $\dot{y}_c$ is obtained, $\dot{\theta}_c$ can be calculated by numerical differentiation of $\psi_c$. Then, $\dot{\theta}_c$ is used in the third layer accompanied by $\dot{\psi}_c$ and $\dot{\beta}_c$, and angular velocity commands are calculated.

$$A_{3,1} = \left[\begin{array}{c} \frac{\partial \theta}{\partial \psi} \\
\end{array}\right]$$

(16)

$$\dot{q}_c = q = A_{3,1}^{-1} \cdot \left\{ \dot{\theta}_c + K_{\theta} \cdot (\theta - \theta_c) \right\}$$

(17)

$$\phi_c = -K_{\psi} \cdot (\psi - \psi_{\text{BMS}})$$

(18)

$$A_{3,2} = \left[\begin{array}{c} \frac{\partial \psi}{\partial \phi} \cdot \partial \phi \\
\end{array}\right]$$

(19)

$$\left[\begin{array}{c} p_c \\
\end{array}\right] = \left[\begin{array}{c} p \\
\end{array}\right] - A_{3,2}^{-1} \cdot \left[\begin{array}{c} \dot{\phi}_c + K_{\phi} \cdot (\phi - \phi_c) \cdot K_{\phi} \cdot (\psi - \psi_{\text{BMS}}) \\
\end{array}\right].$$

(20)

In the fourth layer, longitudinal motion, and lateral and directional motion are merged into a single equation.

$$A_4 = \left[\begin{array}{c} \frac{\partial p}{\partial \delta_e} \cdot \partial \delta_e \\
\frac{\partial q}{\partial \delta_a} \cdot \partial \delta_a \\
\frac{\partial r}{\partial \delta_r} \cdot \partial \delta_r \\
\end{array}\right]$$

(21)

$$q_{LC} = C_{LC} \left(1 + T_{1s} \right) \cdot q$$

(22)

$$x_4 = \left[\begin{array}{c} q \\
\end{array}\right]$$

(23)

$$u_c = \left[\begin{array}{c} \delta_e \\
\delta_a \\
\delta_r \\
\end{array}\right]$$

(24)

where $q_{LC}$ is the output of a lead compensator for $q$, and $\delta_{e,0}$, $\delta_{a,0}$, and $\delta_{r,0}$ are the control commands from the previous step. In these equations, $A_{3,1}$, $A_{3,2}$, and $A_4$ can be obtained using numerical differentiating equations of motion.

### 3.3.3. Replacement of HSDI

As has been mentioned, HSDI is replaced by a PID con-
controller to reduce computational load in two parts: the transformations of $\alpha_c$ to $q_c$, in flight phase 2, and of $N_{zc}$ to $q_c$ in flight phases 4 and 5.

In flight phase 2, the following is applied.

$$q_c = -K_\alpha(\alpha - \alpha_c).$$

In flight phases 4 and 5, a high-pass filter is applied to $q_c$, and $q_c$ is obtained as

$$q_c = -K_{N_c}(N_c - N_{zc}) - K_{q_{HPF}} q_{HPF},$$

where the high pass filter is defined as

$$q_{HPF,n} = C_{HPF} x_{HPF,n} + D_{HPF} q_n$$

$$x_{HPF,n+1} = A_{HPF} x_{HPF,n} + B_{HPF} q_n.$$

3.4. Evaluation

As mentioned in section 3.1, the controller designed was evaluated using linear analysis and MCS. In these evaluations, the result of each flight simulation case was classified as a flight success, mission failure, violation, or flight failure according to the criteria summarized in Table 6. Simulation results were compared with the criteria in the order presented in the table, so each case was classified as a single category.

3.4.1. Single error analysis

For the single error analysis, 217 error cases were carried out, and the results are shown in Table 7. As is shown in this table, there were three mission failure cases. Each of these cases was precisely investigated, and Table 8 shows the summary. The cases were accepted because the vehicle had limited control devices.

3.4.2. Linear analysis

For the linear analysis, the vehicle’s equations of motion were linearized at each one-second interval along the representative trajectory. A nonlinear closed-loop system consisting of aircraft motion, actuators, sensors, transport delay, and control system was built and linearized using Matlab Simulink, and then safety margins (gain margin, phase margin and delay margin) were evaluated for open-loop systems cut at each control surface.

Figure 5 shows the results of gain margin and phase margin. This analysis showed that there was one point that failed to meet the criteria. Although the equations of motion have an unstable mode at this point, the time constant is much longer than the flight time, so its effect can be neglected. Other than this point, it was confirmed that all criteria were satisfied.

3.4.3. Monte-Carlo simulation

The total performance of the GNC system was evaluated using MCS. The error models in each simulation run were initialized with different values according to their distributions and the vehicle release points were normally distributed within the designated separation area. The MCS evaluation results are shown in Table 9. This table shows an estimated success rate with a confidence level of 95%, and this confirms that the mission success rate is greater than 90%, which meets the requirement.

3.5. Discussion regarding the approach proposed

The D-SEND#2 controller design approach proposed here is based mainly on timescale separation and a dynamic inversion method. This approach has several merits compared to a
conventional PID control law with gain scheduling.

A PID controller for such a wide flight envelope should be divided into many segments, and all gain sets must be tuned and smoothly interpolated. If the vehicle’s mathematical model changes, as often happens during aircraft development, all gains must be retuned and confirmed again. On the other hand, HSDI requires far fewer gains to cover a wide flight envelope. Moreover, it is very easy to deal with model changes because all that is required is to replace the mathematical model in the controller. These features reduce the controller design workload.

Concerning controller robustness, a PID controller directly reduces errors independent of the plant model so that control performance is robust against plant model errors. In contrast to this, the control performance of a DI-based controller depends deeply on the plant model, so plant model error affects controller performance. However, the HSDI method combines DI and the timescale separation method, and the time-scale separation method enhances controller performance as reported in Menon et al.16)

4. Results of the Flight Experiment

The actual flight test of D-SEND#2 was conducted on July 24, 2015. Flight results are shown in Fig. 6. These figures show the following. The vehicle was released from the balloon within a separation area and a target BMS was selected. After separation, it accelerated in free-fall to a supersonic speed and flew stably over the BMS. During the measurement phase, it achieved and held the objective Mach number condition, and then terminated its flight within Zone-B. The flight data shows that the cross-range error at the BMS was 18.4 m and the sonic boom was successfully measured. As shown in Table 10, the flight data satisfied all the requirements for the design conditions.

Tracking errors are shown in Fig. 7, in which state variables and guidance commands are represented by black solid and gray dash-dotted lines, respectively. In these figures, guidance commands are plotted with flight statuses, and each figure shows errors in the specific flight or lateral control phases in which each variable is used as a control command (Table 5). From these figures, each status generally converges to its command except for some exceptions. In Fig. 7(b), \( \alpha \) oscillation grows as time evolves. This is caused by an increase in dynamic pressure, and \( \alpha \) is controlled well. In Fig. 7(c), \( \theta \) shows relatively large errors at 10–20 s. This is because the dynamic pressure is too low to pull the vehicle up quickly. In glide phase, \( \gamma \) shows poor tracking performance (Fig. 7(g)). Since \( \gamma \) is not controlled in the pull-up phase, \( \gamma \) error is large at the beginning of the glide phase. This caused a large overshoot in the glide phase, but it has a very limited influence on mission success. The controller performed quite well in general, and as a result, all of the mission requirements were satisfied.

Aerodynamic coefficients were estimated from flight data, and Fig. 8 shows the time evolution of \( C_D \). At the start of the flight, low dynamic pressure results in significant measurement errors to \( \alpha \) or \( \beta \), so the estimation of aerodynamic coefficients is carried out only after 20 s. Other coefficients, such as \( C_L \), or \( C_m \), are relatively very close to their model values, but \( C_D \) has a large error; its value reaching more than the 3 \( \sigma \) value of the error model. This led to excessive speed compared to nominal simulation cases, but the mission requirements were still satisfied. As single error analysis showed that a \( C_D \) error with a value of –3\( \sigma \) for the entire flight had a significant effect on the mission success, this actual er-

Table 10. Flight conditions.

| Items       | Requirements | Results   |
|-------------|--------------|-----------|
| Mach number | [1.2, 1.4]   | 1.386     |
| \( C_L \)   | [0.10, 0.13] | 0.123     |
| Flight alt. | \( H \leq 11 \) km | 8.03 km |
| Sonic boom  | Measured at BMS | Measured |
| Focusing    | No focusing at BMS | No |
error could have had a great impact. However, these results show that the maximum Mach number during the measurement phase was just below the upper limit of 1.4, and the controller proposed is sufficiently robust to handle such a large error.

These results demonstrate that the controller designed has good performance under significant aerodynamic model error, and imply that the controller design method presented here is practically effective for such a challenging problem as a balloon launch flight experiment.

5. Conclusions

This paper describes a controller design method for a balloon launch flight test, and its application for an actual flight test. Such flight tests present the guidance, navigation, and control system with a number of challenges due to the nature of the balloon launch method, and the guidance and control law are designed to accommodate these problems using the hierarchy-structured dynamic inversion (HSDI) method. HSDI combines dynamic inversion and timescale separation in a clearly organized design procedure. Single error analysis, linear analysis, and Monte-Carlo simulation were carried out to evaluate stability margins and the mission success rate of the controller before the flight test. Then, the flight test was executed and its results showed that the controller had good performance under significant aerodynamic model error. Accordingly, this implies that the controller design method presented is practical and effective for such challenging problems as balloon launch flight experiments.

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