Technical Note

New Structure for an Aerodynamic Fin Control System for Tail Fin-Controlled STT Missiles

Seung-Hwan Kim[1]; Yong-In Lee[2]; and Min-Jea Tahk[3]

Abstract: This paper presents a new design method for an aerodynamic fin-control system for tail-fin controlled skid to turn (STT) missiles. The performance of the actuation system plays a decisive role in determining the performance of the flight control system for a highly maneuverable missile. To control the missiles by aerodynamics, control surfaces, sometimes called fins, are used. Deflection angles of these fins are the control variables of aerodynamics, but aerodynamicists prefer to use analytic variables called aileron, elevator, and rudder instead of these physical variables because these three analytic variables dominantly influence the roll, pitch, and yaw motion of the missile, respectively; and each can be considered a linear combination of four fin deflection angles. On that basis, roll, pitch, and yaw autopilots for controlling the attitudes or lateral acceleration of the missile are designed, and aileron, elevator, and rudder commands, respectively, are generated as consequence outputs of each autopilot. In the existing fin-actuation control scheme for the typical tail-fin controlled cruciform missiles, these outputs are distributed to four fin defection commands. After that, the four fins are actuated by fin controllers so that their deflections follow the commands. This paper shows that such control schemes can cause a significant deterioration in flight control system performance when fin-actuators have certain physical constraints such as slew rate, voltage, or current limit, or have an uncertainty of actuator dynamics, and proposes a new control scheme that alleviates such problems. This scheme can be widely applied to various fin-actuation control systems. But in this paper, for convenience, a tail-fin controlled cruciform missile is used as the example, and the proposed control scheme is shown to give better performance than the existing one. **DOI:** 10.1061/(ASCE)AS.1943-5525.0000088.

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Author keywords: Fin actuation control system; Missile autopilot; Non-linearities; Conversion logic.

Introduction

The performance degradation attributable to actuator nonlinearities is one of the major issues of flight control in the high maneuver region. Actuator nonlinearities degrade the performance of the flight control system and even can lead to instability (Bernstein and Michel 1995; Lyshevski 1999). For this reason, various methods of preventing instability because of saturation have been examined. Most previous studies focused on the design method of a flight control system to come up with actuator saturation (Hess and Snee 1997; Lu et al. 2003). This paper presents a new control scheme for an actuation system instead of a design method for a flight control system. When the actuation system includes saturation (e.g., voltage, current, and slew rate), the proposed scheme shows a very effective control method. To control the missiles by aerodynamics, control surfaces, sometimes called fins, are used. Many different types of fins exist but are largely categorized into three types: the canard type located in the front part; the wing type located in the middle part; and the tail-fin type located in the rear part of the missile. Regardless of their types, these fins need suitable actuation systems. Most of the skid to turn (STT) missiles have a four fin-actuation control system that is located in the rear of missile.

The control variables of the aerodynamics are fin deflection angles, but aerodynamicists prefer to use analytic variables called aileron, elevator, and rudder instead of the physical variables of fin deflection angles because these three analytic variables dominantly influence the roll, pitch, and yaw motion of the missile, respectively; and each can be considered a linear combination of four fin deflection angles. On that basis, roll, pitch, and yaw autopilots for controlling the attitudes or lateral acceleration of the missile are designed, and aileron, elevator, and rudder commands, respectively, are generated as consequence outputs of each autopilot.

In the existing fin-actuation control scheme for the typical tail-fin controlled cruciform missiles, three actuation commands computed by an autopilot algorithm are distributed to four fin defection commands. After that, the four fins are actuated by fin controllers so that their deflections follow their commands. This paper shows that the performance of such control schemes can be degraded significantly when fin actuators have certain physical constraints, such as slew rate, voltage or current limit, or uncertainty of actuator dynamics, and propose a new control scheme that alleviates such problems. The proposed scheme can be applied widely to various fin-actuation systems. In particular, by using a tail-fin controlled cruciform missile, the proposed new actuation control scheme is shown to have a better performance than the existing scheme.

This paper is organized as follows. The "Conversion Logic" section is devoted to the derivation of an optimal conversion logic between four fin deflection angles and the three control deflection

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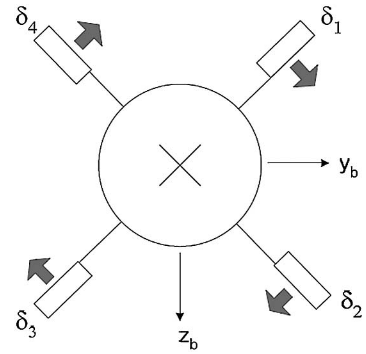

Fig. 1. Definition of cruciform actuator fins

angles of aileron, elevator, and rudder. The "New Structure of Fin-Actuation Control System" section introduces a new control structure of fin actuators that have severe nonlinearities such as voltage saturation and a limitation of angular rate. To confirm the performance of the new fin-control structure, roll, pitch, and yaw autopilot for a generic STT missile are described in the "Simulation Results" section. The new actuator control scheme is applied to a designed flight control system. This section also includes the performance results comparing the new method to the existing one. Finally, concluding remarks about the new structure of the actuation system are presented, and further studies are described.

Conversion Logic

In this section, the conversion logic between four fin deflection angles and the three control deflection angles of aileron, elevator, and rudder are considered. The cruciform tail fins of STT missiles with a rear view are depicted in Fig. 1. In it, δ_1, δ_2, δ_3, and δ_4 denote fin deflection angles.

As shown in Fig. 1, a positive rolling moment is generated by a positive δ_1, δ_2, δ_3, and δ_4. Furthermore, a positive pitching moment is generated by a positive δ_1 and δ_2 and a negative δ_3 and δ_4; and a positive yawing moment is generated by a positive δ_2 and δ_3 and a negative δ_1 and δ_4. Thus, the roll control deflection angle, δ_r, pitch control deflection angle, δ_p, and yaw control deflection angle, δ_y, can be expressed as Eqs. (1)–(3). These equations will be called mixing logic.

$$\delta_r = \frac{1}{4}(\delta_1 + \delta_2 + \delta_3 + \delta_4) \tag{1}$$

$$\delta_p = \frac{1}{4}(\delta_1 + \delta_2 - \delta_3 - \delta_4) \tag{2}$$

$$\delta_y = \frac{1}{4}(-\delta_1 + \delta_2 + \delta_3 - \delta_4) \tag{3}$$

Control deflection angle commands can also be written in a form similar to the described mixing logic. By introducing a new control deflection angle, δ_{x_c}, which can be defined arbitrarily by the control designer, one gets Eq. (4):

$$\begin{bmatrix} \delta_{r_c} \\ \delta_{p_c} \\ \delta_{y_c} \\ \delta_{x_c} \end{bmatrix} = \frac{1}{4}\begin{bmatrix} 1 & 1 & 1 & 1 \\ 1 & 1 & -1 & -1 \\ -1 & 1 & 1 & -1 \\ k_1 & k_2 & k_3 & k_4 \end{bmatrix}\begin{bmatrix} \delta_{1_c} \\ \delta_{2_c} \\ \delta_{3_c} \\ \delta_{4_c} \end{bmatrix} \tag{4}$$

where k_1, k_2, k_3, and k_4 = constants to be determined. Manipulating the matrix inversion results in the following equation of four fin deflection commands:

$$\begin{bmatrix} \delta_{1_c} \\ \delta_{2_c} \\ \delta_{3_c} \\ \delta_{4_c} \end{bmatrix} = \begin{bmatrix} a_1 & b_1 & -c_1 & d_1 \\ a_2 & b_2 & c_1 & -d_1 \\ a_1 & -b_2 & c_2 & d_1 \\ a_2 & -b_1 & -c_2 & -d_1 \end{bmatrix}\begin{bmatrix} \delta_{r_c} \\ \delta_{p_c} \\ \delta_{y_c} \\ \delta_{x_c} \end{bmatrix} \tag{5}$$

where

$$\Delta = k_1 - k_2 + k_3 - k_4$$

$$a_1 = \frac{-2(k_2 + k_4)}{\Delta}, \qquad a_2 = \frac{2(k_1 + k_3)}{\Delta}$$

$$b_1 = \frac{-2(k_2 - k_3)}{\Delta}, \qquad b_2 = \frac{2(k_1 - k_4)}{\Delta}$$

$$c_1 = \frac{2(k_3 - k_4)}{\Delta}, \qquad c_2 = \frac{2(k_1 - k_2)}{\Delta}$$

$$d_1 = \frac{4}{\Delta}$$

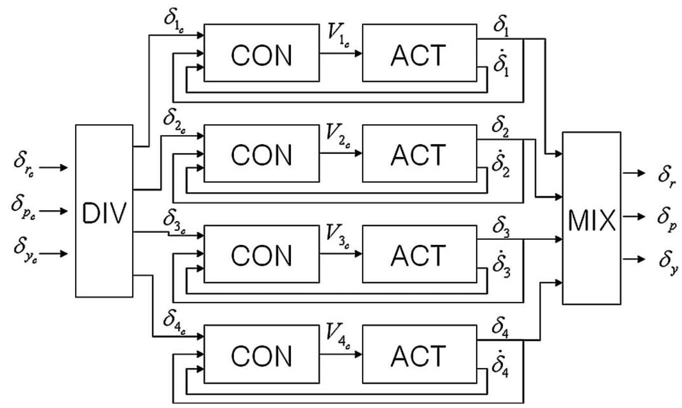

Fig. 2. Existent actuating system

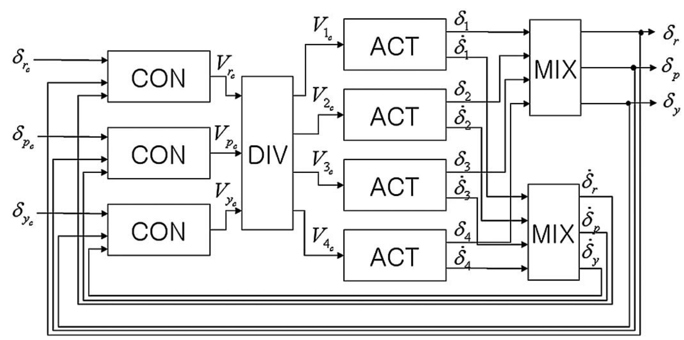

Fig. 3. Proposed actuating system

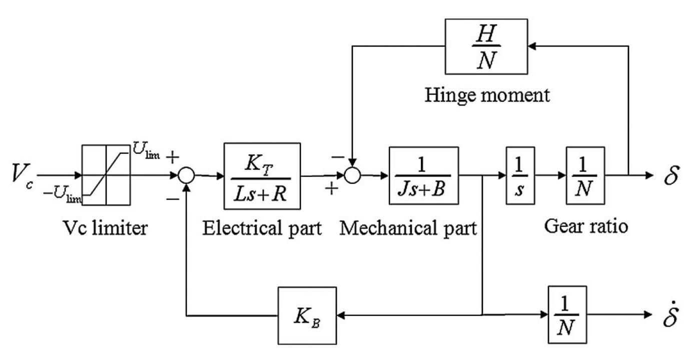

Fig. 4. Electromechanical actuator

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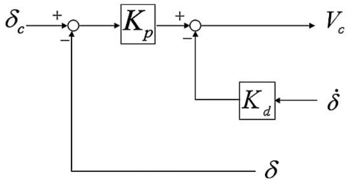

Fig. 5. Simple controller

Table 1. Numerical Values

Parameters	Values	Parameters	Values
K_T	0.303125	K_B	5.7333×10^{-4}
L	0.35×10^{-3}	H	0
R	0.933	K_P	6
J	8.5354×10^{-7}	K_d	0.02
B	2.0835×10^{-6}		
N	274	$U_{\lim}$	28

From Eq. (5), the conversion logic from the three control deflection commands to the four fin deflection commands is not unique because δ_{x_c} is composed of the arbitrary constants k_1, k_2, k_3, and k_4.

The unique conversion logic from the three control deflection commands to the four fin deflection commands is obtained by considering the following minimization problem:

$$\min J = \sum_{i=1}^{4} (\delta_{i_c})^2 \tag{6}$$

If a solution δ_{x_c} is obtained, which minimizes the cost function, it will be an optimal conversion logic that makes the smallest fin deflection commands for given control deflection commands. Manipulating the following calculation

$$\frac{dJ}{d\delta_{x_c}} = 0 \tag{7}$$

$$\frac{d^2 J}{d\delta_{x_c}^2} > 0 \tag{8}$$

one obtains the solution given in Eq. (9):

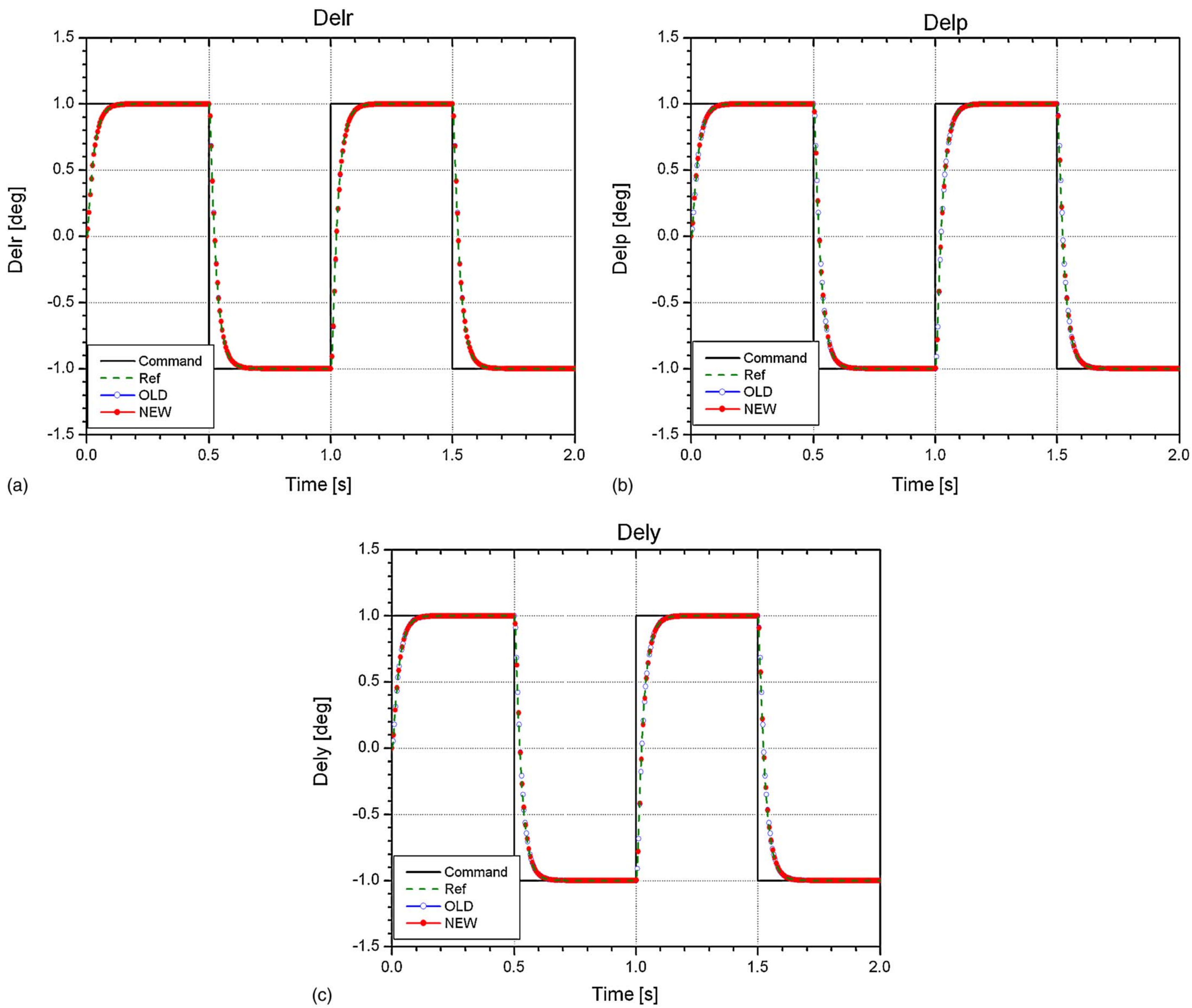

Fig. 6. Comparison of actuator responses when $\delta_{r_c} = \delta_{p_c} = \delta_{y_c} = 1°$: (a) roll deflection; (b) pitch deflection; (c) yaw deflection

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$$\delta_{x_c} = \frac{1}{4}\begin{bmatrix} (k_1 + k_2 + k_3 + k_4)\delta_{r_c} + (k_1 + k_2 - k_3 - k_4)\delta_{p_c} \\ +(-k_1 + k_2 + k_3 - k_4)\delta_{y_c} \end{bmatrix} \quad (9)$$

Inserting Eq. (9) into Eq. (5), one obtains

$$\delta_{1_c} = \delta_{r_c} + \delta_{p_c} - \delta_{y_c} \quad (10)$$

$$\delta_{2_c} = \delta_{r_c} + \delta_{p_c} + \delta_{y_c} \quad (11)$$

$$\delta_{3_c} = \delta_{r_c} - \delta_{p_c} + \delta_{y_c} \quad (12)$$

$$\delta_{4_c} = \delta_{r_c} - \delta_{p_c} - \delta_{y_c} \quad (13)$$

This implies an optimal conversion logic from the three control deflection commands to the four fin deflection commands in a sense of the smallest fin deflection. Eqs. (10)–(13) will be called division logic.

New Structure of Fin-Actuation Control System

Assume an STT missile with four tail-fins are driven by electromechanical actuators. As previously mentioned, autopilots for the missiles generate three desired control deflection commands: roll control deflection angle, δ_r; pitch control deflection angle, δ_p; and yaw control deflection angle, δ_y. Autopilots are designed on the basis of the assumption that fin actuators are linear systems, but real electromechanical actuators have nonlinearities such as a voltage limit in the electrical part and a limitation of an angular rate in the mechanical part. The performance degradation of an actuator because of these nonlinearities has been an important factor adversely affecting the stability and performance of a missile autopilot. In particular, even if just one of the four fin actuators is saturated by large voltage or a large angular rate, the saturation can bring forth bad control performances of all control deflection angles, δ_r, δ_p, and δ_y because one fin deflection is connected to all control deflection angles by conversion logic.

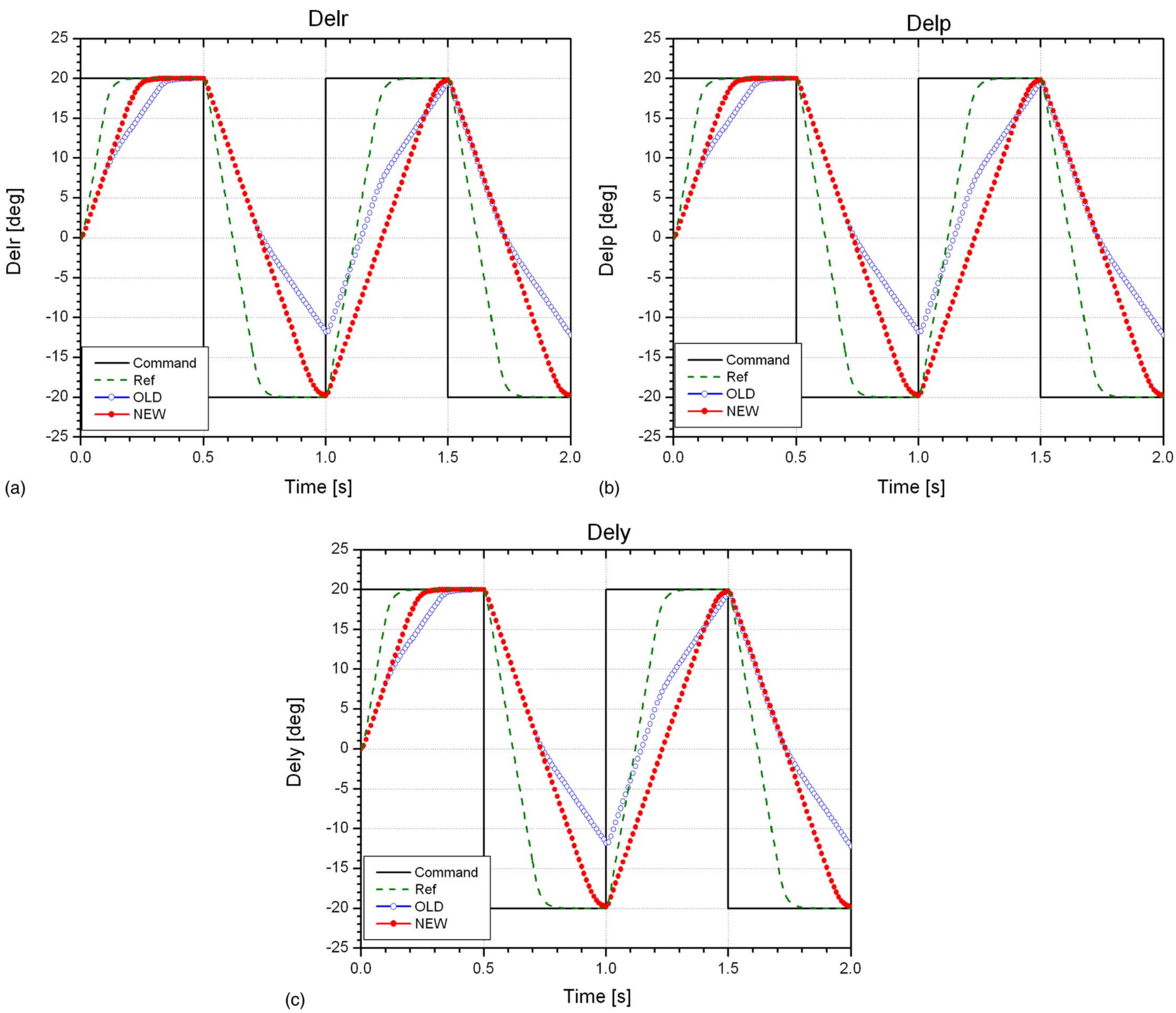

Fig. 7. Comparison of actuator responses when $\delta_{r_c} = \delta_{p_c} = \delta_{y_c} = 20°$: (a) roll deflection; (b) pitch deflection; (c) yaw deflection

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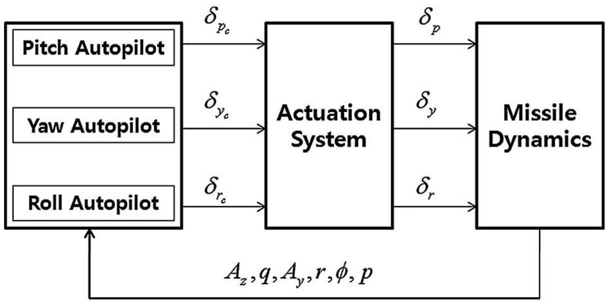

Fig. 8. Missile autopilot scheme with actuation system block

Now, consider a new control structure of fin actuators that have severe nonlinearities such as voltage saturation and a limitation of angular rate. In Fig. 2, a block diagram of an existing fin-actuation system for STT missiles is shown. A fin actuator and its controller are denoted by ACT and CON, respectively; and the division logic and mixing logic are denoted as DIV and MIX, respectively. Each fin controller produces a voltage command, V_c, by using a fin command and feedback variables such as angle and angular rate.

As shown in Fig. 2, an existing actuation system is simply composed of four fin actuators with four corresponding controllers that control δ_1, δ_2, δ_3, and δ_4. If one fin actuator is saturated, all control deflections are affected resulting in a very slow response to all autopilots.

Fig. 3 shows the new structure of a fin-actuation system for STT missiles. In this structure, three controllers correspond to autopilot commands instead of four fin controllers. These three controllers are designed to control analytic inputs directly, that is, aileron, δ_r; elevator, δ_p; and rudder, δ_y. These variables are calculated in a linear combination of fin deflections [see Eqs. (10)–(13)]. The performance of the new structure is exactly same as the existing one when four fin actuators and controllers have the same dynamics and are all linear, but the new structure will give better performance than the existing one when the aforementioned nonlinearities exist. Such an expectation comes from the fact that roll, pitch, and yaw deflection angles are directly feedback.

Simulation Results

To investigate the performance of the new fin-control structure, a typical electromechanical actuator model was assumed, as shown in Fig. 4. For its control, a classical proportional derivative (PD) controller was used, as shown in Fig. 5. In Fig. 4, a saturation block implies a voltage limiter.

The numerical values of parameters in Figs. 4 and 5 are given in Table 1. Figs. 6 and 7 show actuator responses when all aileron and elevator rudder commands are applied to the actuation system by 1° and 20°, respectively. In these figures, NEW denotes the simulation results from the new structure of actuation system, as shown in Fig. 3, and OLD denotes the results from the existing one, as shown in Fig. 2. Ref denotes the results obtained only when each channel's actuation command is applied without the other channel's commands. According to Fig. 6, all responses coincide with one another when the 1° command is applied. In the other case, shown in Fig. 7, the performance of the reference is better than that of the new structure and the existing one, but the performance of the new structure is superior to the existing one.

To confirm the performance of the proposed new scheme to the actuator-controller design method, simulations were performed. For the simulation, the general missile autopilot of three channels was selected, as shown in Fig. 8. The structure of the selected autopilot consisted of a three-loop pitch/yaw acceleration control loop and a roll stabilization control loop.

Simulation Cases

The simulation cases were as follows :
Case 1: The pitch/yaw acceleration command was set to 10 g and the roll attitude command was set to 10° (i.e., a low maneuver case).
Case 2: The pitch/yaw acceleration command was set to 35 g/15 g and the roll attitude command was set to 10° (i.e., a high maneuver case).

Fig. 9 shows the response of the pitch, yaw, and roll autopilot for Case 1, the low maneuver case. It shows the good tracking performance for both methods. On the other hand, Fig. 10 shows the results for Case 2. For the proposed new scheme, the results indicated good tracking performance even in the case of a large command.

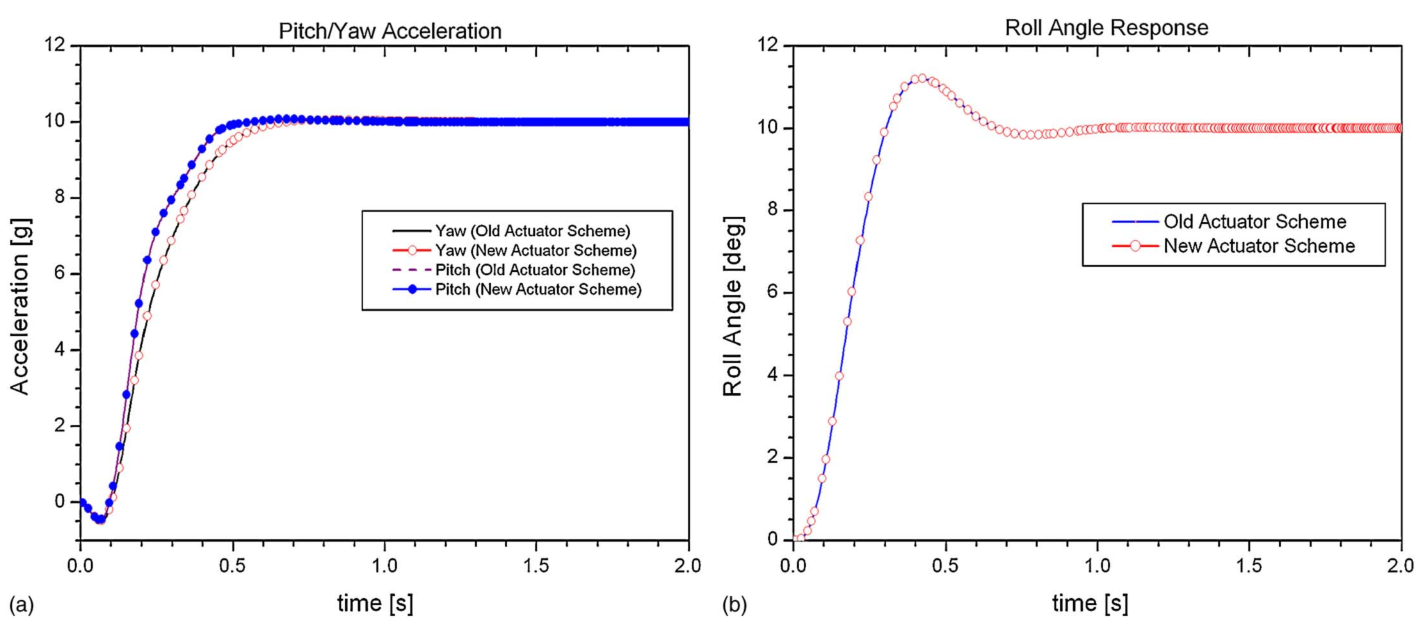

Fig. 9. Responses of missile autopilot in low maneuvering case: (a) pitch/yaw response; (b) roll response

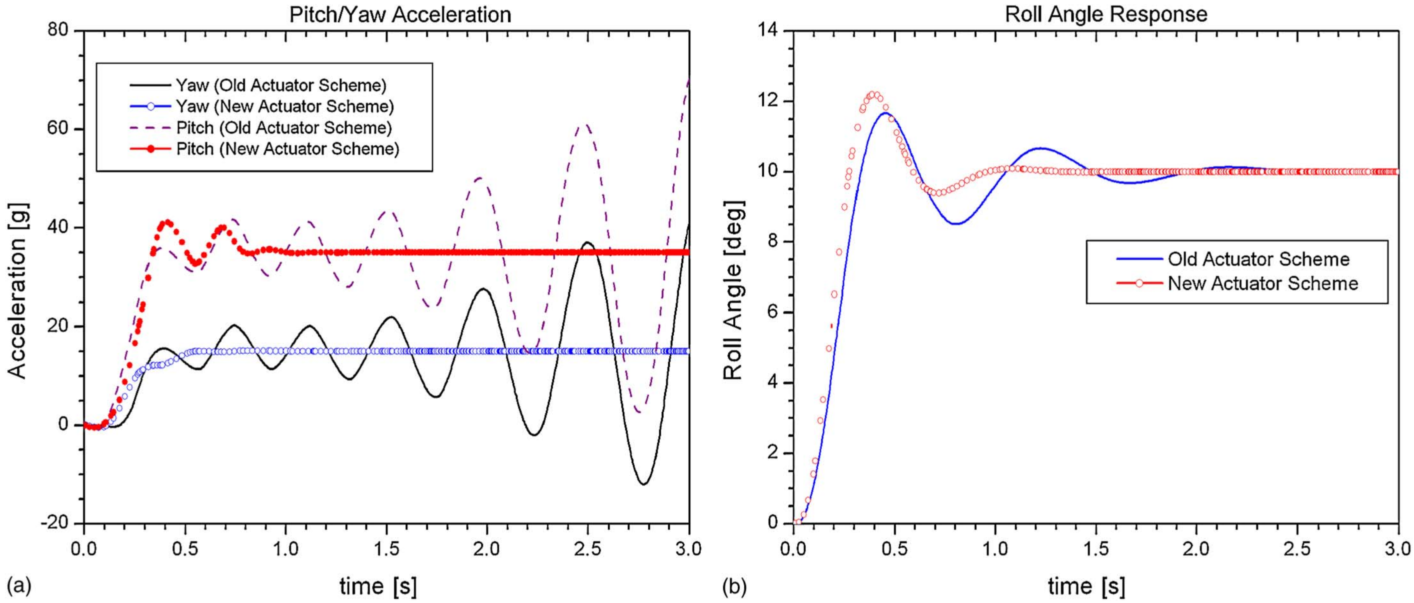

Fig. 10. Responses of missile autopilot in high maneuvering case: (a) pitch/yaw response; (b) roll response

On the other hand, the old method showed that the performance of the actuator worsens according to the increase of command.

Conclusions

In this paper, performance of the existing fin-actuation system for the typical tail-fin controlled cruciform STT missiles were shown to be degraded significantly when fin actuators had certain physical constraints such as slew rate, voltage or current limit, or uncertainty of actuator dynamics. A new scheme of an aerodynamic fin-control system to alleviate such problems was proposed. Finally, with computer simulations including an autopilot loop, the proposed control scheme was shown to have a better performance than the existing scheme when certain nonlinearities existed in actuator dynamics.

Certain issues were not covered in this paper. For example, how to utilize an additional analytic control fin δ_x in other purposes such as for the alleviation of body coupling or fin blanket effect, or how to suitably limit the fin deflection angle commands in advance so that they are not mechanically saturated in any case, among other issues were not addressed and have been left for further study.

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