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Flight testing of a reconfigurable control system on an unmanned aircraft

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Abstract— A radio-controlled aircraft was built and equipped with air-data and inertial sensors. A radio frequency link was added to transmit data and receive commands from a ground station. Data from several flight tests were used to characterize the dynamic response of the aircraft. Despite the high level of noise associated with the low-cost sensor suite, consistent identification of critical aircraft parameters was obtained. Flight tests were also conducted with actuator failures induced on one elevator, one aileron, and one engine. Recursive parameter identification produced parameters tracking the effects of the failures, such as reduced effectiveness of pitch commands due to a locked elevator, or roll and sideslip due to engine failure. The identified parameters were also used in reconfigurable control experiments, where knowledge of the aircraft parameters was used to compensate for the effect of failures, reducing the pilot's workload. Overall, the paper demonstrates that recursive identification and reconfigurable control algorithms are implementable in real-time, even in low-cost platforms. They can be designed to effectively compensate for actuator failures and aircraft damage.

***Index Terms*—Flight control, fault tolerance, flight testing, actuator failure, control reconfiguration, UAV.**

I. INTRODUCTION

Recent advances in computing power and electronic hardware have enabled the development of very sophisticated programmable flight control systems, and a trend has been set to build such systems into modern aircraft. These aircraft can be designed with a much wider range of functional characteristics than was previously possible, and at the same time with customized control responses giving specific flying qualities. With the trend toward programmable flight control comes an obligation to use the new capabilities of these systems to enhance the autonomy of flight control and the safety of flight [1]-[3].

Hardware needed for this project was obtained in part from grant F49620-97-1-0405 (Air Force Office of Scientific Research, Air Force Materiel Command, USAF), and in part from development funds at the University of Utah.

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Small radio-controlled (R/C) model aircraft have been used for a limited number of experimental projects in flight control [4]-[12]. These aircraft offer several benefits to the researcher over full-size aircraft. The radio link removes the test pilot from the hazardous environment of the experimental aircraft, permitting tests that would pose unacceptable risks for an on-board pilot. The time-to-build and cost of a small unmanned aircraft are also far below those for manned aircraft, so that flight tests can be performed with limited resources.

This paper presents the results of experimental research in real-time parameter estimation and reconfigurable flight control conducted during 2001 and 2002 at the University of Utah. The algorithms were implemented on the model aircraft shown in Fig. 1. A platform was intentionally chosen which kept the on-board equipment to a minimum in order to have the freedom to perform risky tests. Other projects have been concerned with developing sophisticated testbeds for a wide range of future work typically focused on autonomous flight, while the goal here was limited to demonstrating identification and reconfiguration algorithms with actual failures.

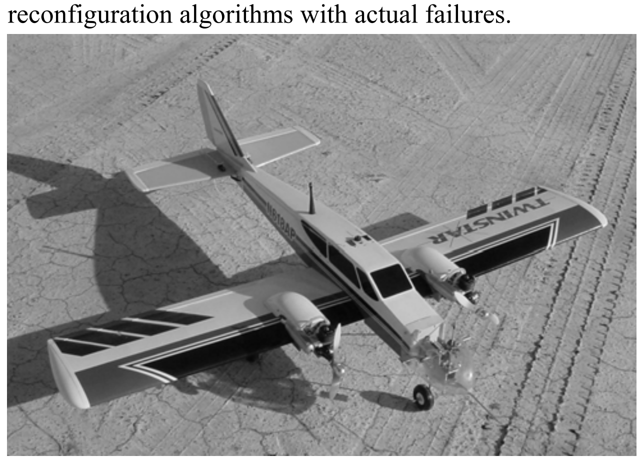

Fig. 1. Radio-controlled model aircraft used for flight tests.

II. FAILURE IMPLEMENTATION

Four in-flight failures were planned, and several modifications were necessary to the airframe and servo system that allowed the failures to be implemented. To implement the failures, the pilot operated a "fault switch" during the flights.

- Frozen left elevator. The airframe was modified by

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splitting the single full-width elevator into two independent half-width elevators. For an elevator failure, the left elevator was commanded to its neutral position and the right elevator received the usual elevator command.

- Frozen left aileron. Each aileron was fitted with a separate servo and R/C channel. Aileron failures were implemented the same way as elevator failures.
- Left or right engine failure. The separate throttle servos were given separate R/C channels. For an engine failure, one engine was commanded to a lower throttle setting than the other.
- Separation of left stabilizer/elevator. An R/C channel and servo were connected to a release mechanism that could, on a command from the pilot, release the entire left half of the horizontal tail.

A more thorough discussion of all aspects of the project is given in [13].

III. Parameter Identification

A. Linear Parameterization

The aircraft model linear parameterization discussed in [1] was used as a starting point for parameter identification. Significant difficulties were encountered in the estimation of some of the channels. Indeed, the rotational accelerations $\dot{q}$, $\dot{p}$, and $\dot{r}$, could not be obtained reliably.

However, it became apparent that the rotational accelerations of the R/C aircraft were small ($\dot{q} \approx 0$, $\dot{p} \approx 0$, and $\dot{r} \approx 0$). This property was attributed to the high level of inherent stability of the R/C aircraft, which has large stabilizing surfaces on the rear of the aircraft, and a wide wing span.

An alternative approach was therefore pursued for the linear parameterization. Specifically, the rotational accelerations were assumed to be zero, and the aircraft model was rearranged so that q, p, and β appeared on the left-hand side of the equations. The resulting equations were obtained

$$\begin{aligned}
a_n &= \theta_{a_n,\alpha}(q_n\alpha) + \theta_{a_n,bias}(10) \\
q &= \theta_{q,el}(v_n\delta_e) + \theta_{q,bias}(10v_n) \\
a_y &= \theta_{a_y,\beta}(q_n\beta) + \theta_{a_y,bias}(10) \\
p &= \theta_{p,ail}(v_n\delta_a) + \theta_{p,bias}(10v_n) + \theta_{p,el}(v_n\delta_e) + \theta_{p,rud}(v_n\delta_r) \\
\beta &= \theta_{\beta,rud}(\delta_r) + \theta_{\beta,ail}(\delta_a) + \theta_{\beta,r}(r/v_n) + \theta_{\beta,bias}
\end{aligned} \tag{1}$$

The variable definitions are:

- a_n body-axis vertical accelerometer signal at the CG
- q_n dynamic pressure/5
- α angle-of-attack
- q the pitch rate
- v_n velocity/50
- δ_e symmetric elevator command
- a_y body-axis lateral accelerometer signal at the CG
- β angle of sideslip
- p roll rate
- r yaw rate
- δ_a anti-symmetric aileron command
- δ_r rudder command

Scaling factors were applied to some variables so that the dynamic range of the various signals would be similar.

The model parameters are:

- $\theta_{an,\alpha}$ lift coefficient
- $\theta_{an,bias}$ zero-lift angle of attack
- $\theta_{q,el}$ elevator pitch rate effectiveness
- $\theta_{q,bias}$ trim pitch rate
- $\theta_{ay,\beta}$ sideslip coefficient
- $\theta_{ay,bias}$ trim side force
- $\theta_{p,ail}$ aileron roll rate effectiveness
- $\theta_{p,bias}$ trim roll rate
- $\theta_{p,el}$ elevator roll rate effectiveness
- $\theta_{p,rud}$ rudder roll rate effectiveness
- $\theta_{\beta,rud}$ rudder sideslip effectiveness
- $\theta_{\beta,ail}$ aileron sideslip effectiveness
- $\theta_{\beta,r}$ roll rate sideslip effectiveness
- $\theta_{\beta,bias}$ sideslip bias

Each of the equations of (1) is an independent relation that is a special case of the linear parameterization

$$y = \theta^{*T} w \tag{2}$$

where y is a scalar output, θ^* is a vector of unknown parameters, and w is a vector of input signals. Taking, as an example, the second equation of (1), and putting it into the form of (2), the three variables are defined as

$$y = q \qquad \theta^{*T} = \begin{bmatrix}\theta_{q,el} & \theta_{q,bias}\end{bmatrix} \qquad w = \begin{bmatrix}v_n\delta_e & 10v_n\end{bmatrix}^T \tag{3}$$

B. Least-Squares Algorithm

The problem of off-line parameter identification is to find $\theta(n)$ such that the relation (2) is satisfied. Because the signals y and w contain the contributions of unmodeled dynamics, nonlinearities, and measurement noise, a best fit must be obtained for the n measurements $y(k)$ and $w(k)$. For identification of a batch of data, define

$$W = \begin{bmatrix}w(1) & w(2) & ,\ldots, & w(n)\end{bmatrix}^T \tag{4}$$

W is a matrix of n measurements of the vector w^T. The scalar y is now redefined as a vector of n measurements. To obtain this best fit, a least-squares optimization criterion may be used [1], [2].

$$J(\theta) = \|y - W\theta\|^2, \tag{5}$$

In order to minimize J, the partial derivative of J is set equal to zero and solved for θ, which gives the batch form of the least-squares algorithm,

$$\theta = \left[W^T W\right]^{-1} \cdot \left[W^T y\right] \tag{6}$$

For example, in (3), θ is a 2 × 1 vector, w becomes W, which is a $n \times 2$ matrix, and y is a $n \times 1$ vector.

IV. Reconfigurable Control

A. Recursive Least-Squares Identification

In Section III, data sets were treated as batches, and parameters were computed that gave the best fit over each set. This approach is useful for model building and for validation, but it assumes that the parameters remain constant over the length of the data set. To track variations associated with systemic changes such as mechanical damage and component failures, the parameters must be continuously updated during regular operation. The desired result can be achieved by using a modified version of the least-squares optimization criterion (5) that is suitable for adaptation. For example, a useful criterion is [1]

$$J(\theta(n)) = \sum_{k=1}^{n} \lambda^{n-k} [y(k) - \theta^T(n) w(k)]^2 + \alpha_w \left\| \theta(n) - \theta(n-1) \right\|^2 \quad (7)$$

where λ is known as the "forgetting factor", and is used to discount old measurements (thereby allowing parameter estimates to change, based on recent data). λ is set to a value $0 < \lambda \leq 1$. Choosing λ close to zero corresponds to the greatest ability to track rapid changes, such as damages and failures, because only the latest data points significantly affect the estimate. Choosing λ close to 1 corresponds to a more slowly adapting algorithm, and to a greater robustness to noise. The second term of (7) is used to ensure the stability of the algorithm, which is needed when the requirement for persistent excitation is not met, such as during cruise conditions. It limits the deviation of the current estimate from the previous estimate, and its influence is adjusted with the factor α_w.

The recursive formulation of (7) is known as the *stabilized recursive least squares algorithm with forgetting factor* [1]. Discussions of this algorithm and related topics are given in [1], [2]. The algorithm was used for off-line identification on many flights with and without failures, and was tuned for the characteristics of the data by choosing the factors $\lambda = 0.998$ and $\alpha_w = 1000$.

B. Control Reconfiguration

Another objective of this project was to design a control system that would use the estimates of aircraft parameters to compensate for changes due to failures or damage. One approach to this problem consists of combining a failure detection scheme with multiple control laws associated with each of the failure cases. Such an approach assumes a finite set of pre-planned cases. In contrast, we consider here a continuously adaptive approach where the gains of the control law are computed from the estimated parameters, and no attempt is made to categorize the health or failure state of the aircraft.

The reconfigurable control algorithm discussed here can be viewed as a special case of *model reference adaptive control*. In general, model reference adaptive control attempts to modify the closed-loop dynamics of a system so that its responses track those of a "desirable" system called the reference model. In off-line identification experiments, it was found that the behavior of the R/C aircraft could be simplified to the point where the dynamics from the control variables (δ_e and δ_a) to the output variables (q and p) reduced to simple gains with some bias terms. In the presence of failures, as well as in some low speed conditions, these gains were found to change significantly. The reconfigurable control law discussed in this section consists in applying another set of gains to the pilot commands, so that the overall gains of the system remain constant and equal to some desirable values.

Two channels are discussed for real-time parameter identification and reconfigurable control. The primary parameter and a bias parameter were identified for the pitch rate and roll rate channels. The outputs, parameters, and regressors are

$$\begin{aligned} y &= q, \qquad \theta = \begin{bmatrix} \theta_{q,el} \\ \theta_{q,bias} \end{bmatrix}, \qquad w = \begin{bmatrix} v_n \delta_e \\ v_n \end{bmatrix}, \\ y &= p, \qquad \theta = \begin{bmatrix} \theta_{p,ail} \\ \theta_{p,bias} \end{bmatrix}, \qquad w = \begin{bmatrix} v_n \delta_a \\ v_n \end{bmatrix}, \end{aligned} \quad (8)$$

With these definitions, the control law consists in letting

$$\delta_e = q_{com} \cdot \frac{\theta_{d.q}}{\theta_{q.el}} \qquad \delta_a = p_{com} \cdot \frac{\theta_{d,p}}{\theta_{p,ail}} \quad (9)$$

where q_{com} is the pilot's pitch command, $\theta_{d,q}$ = -2.5 is the desired value for the elevator effectiveness (or "reference" value of the gain, from a model reference control point of view), $\theta_{q,el}$ is the estimated elevator effectiveness, p_{com} is the pilot's roll command, $\theta_{d,p}$ = 6.0 is the desired value for the aileron effectiveness, and $\theta_{p,ail}$ is the estimated aileron effectiveness. Since difficulties were encountered in reliably identifying its parameters, adaptive compensation was disconnected for the sideslip channel in the experiments reported in this paper.

A block diagram of the reconfigurable control system is shown in Fig. 2. Variable definitions are:

$$r = \begin{bmatrix} q_{com} \\ f_e \\ p_{com} \\ f_a \end{bmatrix} \quad u = \begin{bmatrix} \delta_{e,left} \\ \delta_{e,rt} \\ \delta_{a,left} \\ \delta_{a,rt} \end{bmatrix} \quad f = \begin{bmatrix} thr_{left} \\ thr_{rt} \\ taildrop \end{bmatrix} \quad y_{pid} = \begin{bmatrix} vt \\ q \\ p \end{bmatrix}$$

$$w = \begin{bmatrix} \delta_{e,rt} \\ \delta_{a,rt} \end{bmatrix} \qquad \theta_1 = \begin{bmatrix} \theta_{q.e;} \\ \theta_{p,ail} \end{bmatrix} \qquad \theta_2 = \begin{bmatrix} \theta_{q,bias} \\ \theta_{p,bias} \end{bmatrix} \quad (10)$$

where f_e is the input that triggers an elevator fault in the flight control program, f_a triggers an aileron fault, $\delta_{e,left}$ and

$\delta_{e,rt}$ are the commands to the separate elevators, $\delta_{a,left}$ and $\delta_{a,rt}$ are the commands to the separate ailerons, thr_{left} and thr_{rt} are independent throttle commands, and *taildrop* is the command to the tail-release servo. These last three variables are grouped into a separate input vector because they do not interact with the control system, but are sent from the pilot directly to the airplane. Normally, the elevator command in (9) was $\delta_e=\delta_{e,rt}=\delta_{e,left}$. During an elevator fault, however, $\delta_e=\delta_{e,rt}$ and $\delta_{e,left}=0$. Aileron faults were treated similarly. For engine faults, either thr_{left} or thr_{rt} was set to 25%, and the remaining engine received the *thr* command from the joystick. θ_2 and the trim system shown in Fig. 2 will be discussed later.

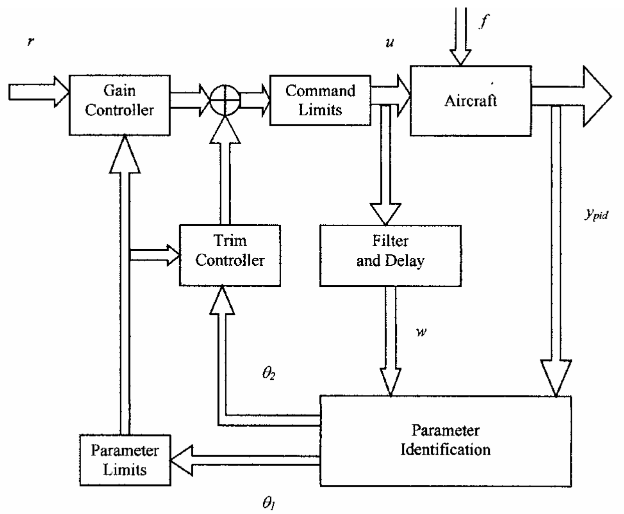

Fig. 2. Block diagram of reconfigurable control system.

The purpose of the "Filter and Delay" block was to align the control signals in u with the delayed output signals in y_{pid}. The control signals were measured at the input by the flight control program, but were delayed by PWM encoding/decoding and a mechanical delay in the actuators. The signal misalignment was further increased because the output signals were delayed during on-board data acquisition, transmission, and decoding in the flight control program. Overall, the Filter and Delay block applied a filter to the control variables to match the analog filters in the IMU, a filter that approximated the dynamics of the actuator response, and a delay of the control variables to account for the signal and mechanical delays.

For discussion of the results with the reconfigurable control system, the following terminology is used:

- the open-loop (OL) parameters are the gains from the actuator commands (u) to the output variables (y_{pid}), as determined by the identification algorithm. They are the gains that the pilot would feel if there were no reconfigurable control algorithm.
- the closed-loop (CL) parameters are the gains from the pilot's commands (r) to the output variables (y_{pid}), as determined by a similar recursive algorithm, but after the flight. These are the gains that the pilot perceives, and are the result of the actuator effectiveness multiplied by the control law gains.

Fig. 3 shows results of reconfigurable control from two flights with failures. In each plot, the upper curve is the aileron effectiveness and the lower curve is the elevator effectiveness. In the middle of each plot is the fault indicator line. The fault is in effect when the line is high. In the first flight, (a), an elevator failure occurred. In the second flight, (b), an aileron failure occurred. The algorithm attempted to stabilize the open-loop parameters (solid lines) at the desired values (dashed lines) of -2.5 and 6.0. The result was the closed-loop parameters (dotted lines). In both plots, the parameter associated with the failed actuator shows the expected change to half its no-fault value during the fault, and then a return to its normal value when the fault ends.

A notable feature of the plots is the large variation in parameters in normal conditions. Some of the variations, particularly the spikes, are thought to be caused by a dead zone (hysteresis) in the control surface responses. The variation in the estimates could be partly smoothed out by increasing the forgetting factor λ, or by increasing the stabilization factor α_w, but the cost would be slower adaptation to failures. The choices of λ and α_w used here resulted in convergence of $\theta_{q,el}$ in 6 to 12 seconds, and convergence of $\theta_{p,ail}$ in 3 to 9 seconds after a fault. The variations in convergence times are due to differences in excitation of the identified channels.

The results show a marked improvement of the reconfigurable system over the uncompensated system. In general, reconfiguration brought the effective gains close to the desired values (dashed lines), despite significant variations of the effectiveness of the actuators.

The pilot's evaluation of the reconfigurable system indicated that the airplane exhibited more consistent control responses, particularly after failures. Responses were also found to be more uniform during approaches to landing, an indication that the algorithm also compensated for changes associated with low speeds.

The stabilizer/elevator release fault was implemented during two flights. Fig. 4 shows photographs of one of these in-flight tail releases. A plot of the elevator effectiveness during this flight is shown in Fig. 5. The OL parameter is shown with solid line and the CL parameter is shown with dotted line. The desired value of $\theta_{q,el}$ for this flight was -3.0 (dashed line).

Surprisingly, the parameters did not exhibit any visible change in the elevator effectiveness after the release, although such changes were clearly visible in the previous, locked-elevator experiments. A possible explanation for this unexpected result is that the failure produced a reduction of the control authority of the pitch command *together with* a (comparable) reduction of the stabilizing effect of the horizontal tail. Therefore, while the dynamics of the system were altered by the failure, the steady-state gain was not. The pilot indeed reported that the airplane was less stable in pitch after the tail release, but was *not* less responsive to elevator commands.

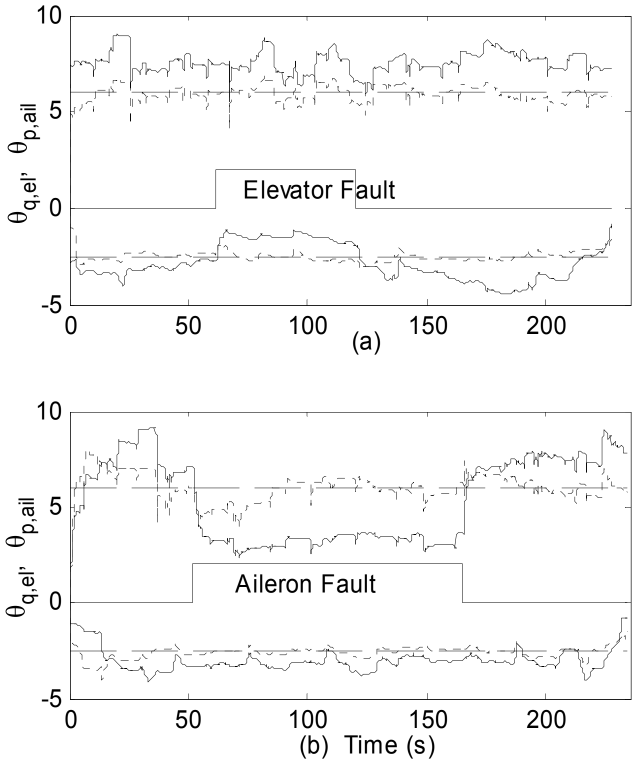

Fig. 3. Results of control reconfiguration during two flights.

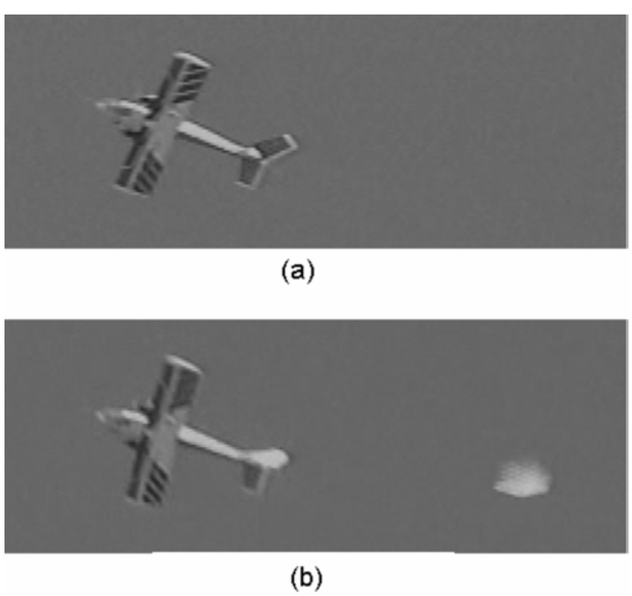

Fig. 4. The left stabilizer/elevator is beginning to separate from the tail in (a), and is well behind the airplane in (b).

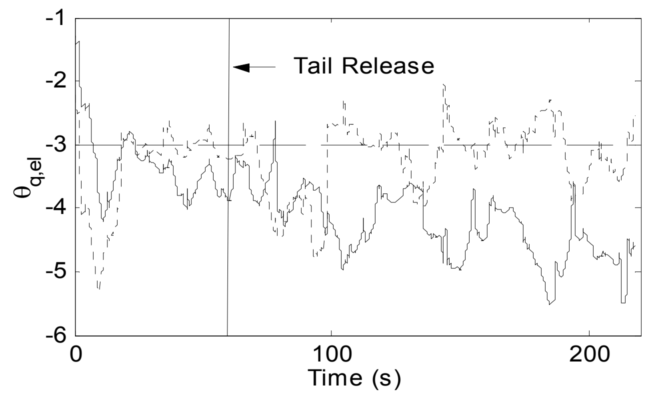

Fig. 5. Elevator effectiveness during a tail release fault.

V. Automatic Trim Compensation

The bias parameters (θ_2 in Fig 2) are the residuals from the identification of the primary parameters. They indicate the extent to which the aircraft is out of trim. The pitch bias term, $\theta_{q,bias}$, indicates the pitch rate with δ_e = 0. The roll bias term, $\theta_{p,bias}$, is a measure of the aircraft's roll rate with δ_a = 0. These biases can be eliminated by the pilot, using manual trim controls, but this task can distract the pilot from other duties, particularly after a mechanical failure. A twin engine aircraft can experience large yawing and rolling moments due to the asymmetric thrust that follows an engine failure. In this section, we discuss a reconfigurable control algorithm that not only adjusts the gains of the system, but also uses the estimates of the bias terms to relieve the pilot of the trimming workload.

The block diagram of Fig. 2 includes a trim controller. When the trims are added to (9) the commands in u become

$$\delta_e = q_{com} \cdot \frac{\theta_{d,q}}{\theta_{q,el}} - \frac{\theta_{q,bias}}{\theta_{q,el}}, \delta_a = p_{com} \cdot \frac{\theta_{d,p}}{\theta_{p,ail}} - \frac{\theta_{p,bias}}{\theta_{p,ail}}, \tag{11}$$

Flight tests were conducted in which automatic trim compensation was implemented on the flight control computer, and the aircraft's trim condition was adjusted by the control law (11). Fig. 6 shows the results of a flight test in which the right engine was brought to idle twice.

The OL bias is shown with solid lines, the CL bias with dotted lines. The OL roll bias is close to zero except during engine failures, where it increases to roughly 18 deg/sec. This is a roll to the right, and it is reduced significantly when the trim system is applied. A roll to the right is expected after a right engine failure because the asymmetric thrust causes the aircraft to yaw to the right. Both the yaw angle β and the lack of propeller wash from the right engine, which decreases the lift of the right wing, cause the aircraft to roll to the right. With automatic trim compensation, the aircraft handled much better for the entire flight, while the remaining engine continued operating at full power.

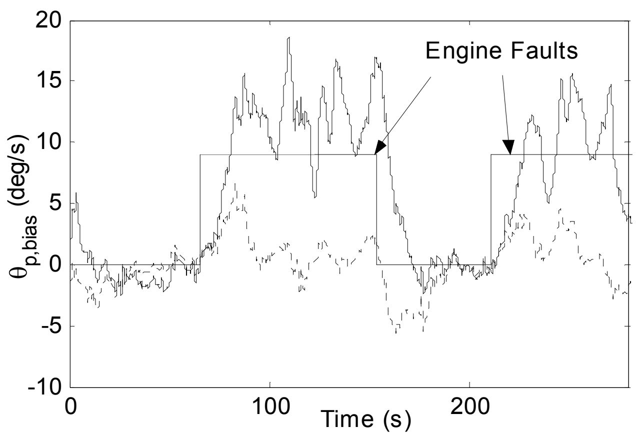

Fig. 6. Results of automatic trim compensation during two right engine faults.

The pilot's evaluation of this experiment indicated that the trim compensation system was effective in relieving the pilot of the need to manually trim the aircraft. This was beneficial during all phases of flight, and was crucial in compensating for the rolling effects of a failed engine (the R/C aircraft was found to be very difficult to control with only one engine, and a flight test with engine failure, but no reconfiguration, resulted in a crash). The automatic pitch trim did not help accommodate the engine failures, but *did* significantly improve the pitch behavior of the aircraft during all phases of flight. The yawing effects did not cause difficulty of control, and the automatic sideslip trim was used for some flights and was not used for others because of the difficulty of identifying the sideslip channel.

VI. CONCLUSIONS

The paper reported the results of parameter identification and reconfigurable control experiments performed on an R/C aircraft. An ambitious goal was to develop the test platform in such a manner that risky tests could be performed, including some that would not be imaginable in a piloted aircraft. Failure modes were implemented that resulted in a frozen elevator, a frozen aileron, an engine failure, and the separation of an entire tail surface.

Overall, the data showed that reliable identification of critical aircraft parameters could be obtained off-line and in real-time. Remarkably, this result was obtained despite the low cost of the test platform, which resulted in high sensor noise and biases, and strong actuator nonlinearities. The effect of actuator failures could be observed on the estimated parameters, which tracked variations of effectiveness and bias. Compared to experiments performed earlier with piloted aircraft, it was found useful to reduce the model of the R/C aircraft to a set of gains from the control surface deflections to the pitch rate, roll rate, and sideslip variables. Finer identification of the aircraft dynamics may not be feasible with low-cost sensors, and is possibly not useful for an aircraft with a high degree of internal stability.

Control reconfiguration experiments were also performed, and showed that a continuously adaptive control system was successful in compensating for parameter variations due to failures and changing flight conditions. Specifically, the algorithm computed the commands required to accommodate changes in the gains and biases of the system, reducing the workload of the pilot. The code for the adaptive algorithm, including the ground telemetry operations, was implemented on a personal computer at an update rate of 96 Hz. Therefore, one could easily imagine such a system implemented in piloted aircraft and unmanned air vehicles designed with fault tolerance in mind.

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