NON-LINEAR EQUATIONS COVERING THE FLIGHT ENVELOP pdf

1: Tuning of Gain-Scheduled Three-Loop Autopilot - MATLAB & Simulink

The complete six degree-of-freedom nonlinear equations of motion for F aircraft are considered directly to design the nonlinear Î¼ flight controller by treating the longitudinal and lateral.

 Abstract Design and synthesis of a nonlinear generic supersonic flight vehicle longitudinal dynamics control for angle-of-attack, AOA, output tracking in the atmospheric flight is presented based on sliding mode control. A sliding mode observer is invoked to estimate AOA which is difficult to measure in practice. Large parameter uncertainties accommodation envisaged by designing adaptive mechanisms for both the control and observer and high chattering authority due to large deviations of aerodynamic coefficients arising from wind-tunnel measurements are inhibited. The employed method enables the sliding mode control design to exhibit the desired dynamic properties during the entire output-tracking process. Simulations results are presented to demonstrate the performance, robustness, and stability. Introduction Tail-controlled skid-to-turn flight vehicles are commonly agile systems that should be equipped with control systems whose tasks are to provide stability, disturbance attenuation, and reference signal tracking, while their aerodynamic coefficients vary over a wide dynamic range due to large Mach-altitude fluctuations and due to aerodynamic coefficient uncertainties resulting from inaccurate wind-tunnel measurements. It is common practice, when designing a control system for a nonlinear flight vehicle, to represent the flight envelope by a grid of Mach-altitude operating points and then to perform a linearization of the nonlinear state equations at trim points of the gridded flight envelope. The plant in fact becomes a differential inclusion under continuously varying flight conditions. There are many possible ways of dealing with the control of such linear and time varying plants. In another method, methods are invoked to design a collection of controllers, where, for each operating point in the flight envelope grid, a controller with a fixed structure results [2]. The ensuing set of controllers is then transformed to a single gain scheduled controller by obtaining a least-square fit of its parameters with respect to angle of attack, or Mach number, and so forth. For highly agile air vehicles, these techniques would result in an extensive number of controller design points to be able to cope with the drastically changing dynamic behavior throughout the flight envelop. All of the aforementioned methods are linear design techniques that require either exact knowledge of system parameters or, alternatively, assumption of some uncertainty model such as norm boundedness, thus allowing for robust controller design. On the other hand, sliding mode control may, in principle, be implemented for dynamic systems having only a qualitative description and a number of inequality restrictions. As will be shown in sequel, this approach will provide a simple and high-performance controller on nonlinear system using straightforward design procedure. Sliding modes are the primary form of operational variable structure systems. A sliding mode is a motion on a discontinuity set of a dynamic system and is characterized by a suite of feedback control law and a decision rule known as switching function. Such modes are used to maintain the given constraints with utmost accuracy. The sliding mode controller is an attractive robust control algorithm because of its inherent insensitivity and robustness to plant uncertainties and external disturbances. The nonlinear pitch dynamics of a hypothetical tail-controlled flight vehicle is derived in Section 2. In Section 3 sliding manifold is employed to control pitch-axis dynamics of the air vehicle. Adaptive sliding mode control is designed in Section 4. Sliding mode observer is designed and utilized in control scheme in Section 5 on the basis of pitch rate measurement for the nonlinear system to estimate angle-of-attack which is difficult to measure in practice. Integration of sliding mode control with adaptation mechanism based on the designed observer is presented in Section 6. Problem Formulation The missile model employed in this analysis is based on the hypothetical tail-controlled air vehicle which has been used as a benchmark in a number of recent studies on nonlinear design techniques [1 â€” 3]. The control objective is to force the air vehicle to track a desired motion path generated by the guidance-navigation system as the reference angle-of attack. The problem is first formulated and performance objectives are specified. Dynamic Model of the Airframe The model assumes constant mass, no roll rate, zero roll angle, no side slip, and no yaw rate. Under these assumptions, the longitudinal nonlinear equation of motion is reduced to two forces and one moment. Using body axis coordinates, Figure 1 , these equations are Figure 1: Airframe and dynamic variables. The dynamic pressure is defined as Note that velocity and air density are not assumed to be constant or slow-varying, but a standard atmospheric model is assumed in simulation based on previously reported data [4]. Aerodynamic polynomials resulting from wind-tunnel measurements are given as [1] The numerical values of the various constant parameters in the dynamic equations are given in Table 1. Note that all coefficients are dimensionless and all angles are in radians. Consequently, the differential equations describing the body motion are The tail-fin actuator dynamics describing the tail deflection is where actuator fin deflection is limited to Table 2 gives the flight vehicle constants. As shown in the equations of motion, the time-varying aerodynamic parameters contribute heavily to the variation of dynamic forces and moments exerted on the vehicle airframe. It is assumed here as is often customary in air vehicle autopilot design that the vehicle velocity or Mach number is constant and the nonlinear state equation associated with is dropped from the design model [1]. In this case, the vehicle velocity becomes an independent external parameter upon which the state dynamics depend. By rewriting 4 and 5 , it is easy to show that where.

NON-LINEAR EQUATIONS COVERING THE FLIGHT ENVELOP pdf

2: Book: Flight Dynamics

A family of linear controller candidates for those flight conditions is constructed, then a gain-scheduled controller is obtained by blending controller candidates to cover the entire flight envelope,,,,,,. The advantage of this approach is that it is simple and practical.

 Abstract Energy efficiency plays important role in aeroelastic design of flying wing aircraft and may be attained by use of lightweight structures as well as solar energy. NATASHA Nonlinear Aeroelastic Trim And Stability of HALE Aircraft is a newly developed computer program which uses a nonlinear composite beam theory that eliminates the difficulties in aeroelastic simulations of flexible high-aspect-ratio wings which undergoes large deformation, as well as the singularities due to finite rotations. NATASHA has shown that proper engine placement could significantly increase the aeroelastic flight envelope which typically leads to more flexible and lighter aircraft. The areas of minimum kinetic energy for the lower frequency modes are in accordance with the zones with maximum flutter speed and have the potential to save computational effort. Another aspect of energy efficiency for High Altitude, Long Endurance HALE drones stems from needing to minimize energy consumption because of limitations on the source of energy, that is, solar power. Such aircraft typically have high-aspect-ratio wings with high flexibility, which leads to large deformation. Consequently, linear aeroelastic analyses are incapable of predicting the stability characteristics of such aircraft. They successfully proved that only nonlinear aeroelastic analysis provides correct information regarding the aeroelastic flight envelope of this class of aircraft [1 , 4 , 5]. Nonlinear aeroelastic trim and stability of HALE aircraft, NATASHA, is a computer program developed by the authors of references [1 , 4 , 5] that accommodates modeling of large deformation of high-aspect-ratio flying wings. The theory behind NATASHA is based on the geometrically exact, nonlinear, composite beam theory of Hodges [6], along with the finite-state induced flow model of Peters et al. Previous comparisons by [2] showed that results from NATASHA are in excellent agreement with well-known beam stability solutions [8 , 9], the flutter problem of [10], experimental data presented by [11], and results from well-established computer codes such as DYMORE [12 , 13] and RCAS [14]. A principal determinant of energy consumption in aircraft is drag, which must be opposed by engine thrust for the aircraft to fly. Flying wings may achieve significant drag reduction due to a smooth outer surface and the lack of a vertical tail [3]. Consequently, the performance of such aircraft may increase significantly, relative to conventional configurations of the same size. The potential increase of performance for this class of aircraft has inspired aeroelasticians to design new generation of aircraft based on a flying wing configuration [3]. Typical aeroelastic instability of these aircraft is body-freedom flutter when the short-period mode of the aircraft couples with the elastic bending-torsion modes [3 , 17 â€" 24]. In another context, a morphing solar-powered flying wing can maximize the energy absorption of solar panels on the wing surfaces by changing its configuration such that the panels have highest exposure to the sun. This change in the geometry of the flying wing is highly effective in energy absorption during times just before sunset and just after sunrise, and consequently the aircraft can sustain longer flight [25]. Use of solar energy is a novel method that eliminates one of the design constraints to a considerable extent by removing the limitation on the source of energy. The morphing concept could be based on either wing morphing systems or airfoil morphing systems, or a combination of both [26]. So far in the literature, several morphing concepts and systems have been developed based on altering various geometric parameters of the wing such as span, chord, camber, sweep, twist, and even airfoil thickness distribution to make the aircraft suitable for different missions and flight conditions [19 , 26 , 26 , 26 â€" 37]. The folding wing configuration has been analyzed using linear aeroelastic models [38 â€" 40] and nonlinear aeroelastic models [41 , 42]. In this paper, after a brief outline of the theory behind NATASHA, energy efficiency in aeroelastic design and simulation for flying wing configuration will be assessed as a a feature of the design i. Nonlinear Composite Beam Theory The fully intrinsic nonlinear composite beam theory [6] is based on geometrically exact first-order partial differential equations of motion for the beam that are independent of displacement and rotation variables. They contain variables that are expressed in the bases of the reference frames of the undeformed and deformed beams, and , respectively; see Figure 1. These geometrically exact equations are written in terms of force, moment, velocity, and angular velocity, and they contain no nonlinearities higher than second degree in the unknowns. The equations of motion are where the generalized strains and velocities are related to stress resultants and moments by the structural constitutive equations and the inertial constitutive equations Finally, strain- and velocity-displacement equations are used to derive the intrinsic kinematical partial differential equations [6], which are given as Figure 1: Sketch of beam kinematics. In this set of equations, and are column matrices of cross-sectional stress and moment resultant measures in the frame, respectively; are column matrices of cross-sectional frame velocity and angular velocity measures in the frame, respectively; are column matrices of cross-sectional linear and angular momentum measures in the frame, respectively;.

3: Designing a Guidance System in MATLAB and Simulink

a full-envelope aerodynamic model using regression together cover the complete flight envelope and in the nonlinear model equation.

The approach to solve this control problem is summarized in Figure 2. It consists of the aerodynamic data, the stability and control derivatives, the engine parameter as well as the geometrical data of aircraft, like the moment inertia, the mass, the wingspan, and the wing surface. The aerodynamic data from wind-tunnel test are compared with the calculated data. The both data closely match to each others. The data are expressed in the body fixed coordinate system that normally is used in the flight modeling and simulation. These six degree of freedom 6-DOF , non-linear equations of motion describe three translating motion force equation and three rotating motion moment equation of the UAV and can cover all flight conditions and flight maneuvers in the complete flight envelope, from the take-off until landing. To the equations of motion the kinematics equations and navigation equation below should be added. The following figure shows the graphical non-linear flight model for RPV Tamingsari. The trim flight condition is a condition in which the sums of forces and moments acting on the aircraft are equal zero. Since the equations of motion are non-linear and the dependence of the aerodynamic data is complex, the calculation of trim flight condition is performed with numerical trim algorithm using optimization method SIMPLEX. This trim algorithm will solve for required flight variables, control surfaces and throttle setting for a desired steady-state flight condition such as a given altitude and airspeed. The output variable y is critical variable such as accelerations and very important for controlling the aircraft motion. Finally, the linear state model matrices A, B,C, D are stored in a format suitable for the analysis software like MATLAB Based on the linear UAV model, the dynamic characteristics of UAV is analyzed, such as the trim, stability and control characteristics of the aircraft, the dynamic response of the aircraft to control input and external disturbance, the effect on the flight condition changes of the aircraft dynamics. The Analysis will be performed based on the linear UAV model as well as non-linear ones using flight simulation on the computer. This problem has been overcome by using feedback control to modify the UAV dynamics. The gain of this feedback must be adjusted according to the flight condition. The adjustment process is called gain scheduling technique. Here, the gains are designed for a large set of trim flight conditions and then are scheduled by interpolating them with respect to flight conditions: This analysis covers a wide range of velocities and altitudes and all possible UAV configurations. The second last step is so called hardware in-the-loop simulation or HILS. Well designed simulators allow the control laws and mission functionality of UAV to be tested without risking hardware in flight test. Although HILS can not replace flight testing, it measurably reduces the likelihood of failure by detecting bugs and deficiencies in the laboratory. To facilitate this vital and typically difficult function, an integrated autonomous onboard computer system real embedded controller that has been developed is connected to the real-time flight simulator computer to receive the measured flight variables from flight control simulator as well as to send the autonomous control surfaces signal to the flight control simulator via the external RS serial interface. At the same time, the integrated avionics system will receive send data from the ground control station, as showed in Figure System architecture of integrated avionics system for UAV Tamingsari. This embedded controller has more than enough CPU muscle to run complicated autocoded algorithms. Stevens and Franks L.

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